RTO-MP-079(II)
AC/323(AVT-085)TP/50

RTO-MP-079(II)

NORTH ATLANTIC TREATY ORGANISATION

RESEARCH AND TECHNOLOGY ORGANISATION

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

RTO MEETING PROCEEDINGS 79(II)

Ageing Mechanisms and Control

(Les mécanismes vieillissants et le contrôle)

Specialists' Meeting on Life Management Techniques for Ageing Air Vehicles

(Réunions des spécialistes des techniques de gestion du cycle de vie pour véhicules aériens vieillissants)

Papers presented at the RTO Applied Vehicle Technology Panel (AVT) Specialists' Meeting held in Manchester, United Kingdom, 8-11 October 2001.

Published February 2003

Distribution and Availability on Back Cover

This page has been deliberately left blank

Page intentionnellement blanche

RTO-MP-079(II)
AC/323(AVT-085)TP/50

NORTH ATLANTIC TREATY ORGANISATION

RESEARCH AND TECHNOLOGY ORGANISATION

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

RTO MEETING PROCEEDINGS 79(II)

Ageing Mechanisms and Control

(Les mécanismes vieillissants et le contrôle)

Specialists' Meeting on Life Management Techniques for Ageing Air Vehicles

(Réunions des spécialistes des techniques de gestion du cycle de vie pour véhicules aériens vieillissants)

Papers presented at the RTO Applied Vehicle Technology Panel (AVT) Specialists' Meeting held in Manchester, United Kingdom, 8-11 October 2001.

The Research and Technology Organisation (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 the following 7 bodies:

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

These bodies are made up of national representatives as well as generally recognised 'world class' scientists. They 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.

Published February 2003

ISBN 92-837-1089-4

Printed by St. Joseph Print Group Inc.
(A St. Joseph Corporation Company)
1165 Kenaston Street, Ottawa, Ontario, Canada K1G 6S1

Specialists' Meeting on Life Management Techniques for Ageing Air Vehicles

(RTO MP-079(II) / AVT-085)

Executive Summary

The military commanders in all NATO countries are seeing their aircraft operate well beyond the date that they were originally intended to retire. The costs of maintaining these ageing aircraft are draining the existing budgets to the point that money will not be available to modernise the fleets when obsolescence forces the retirement of the old aircraft. A possible path to reduce this economic burden is through targeted research and development. It is not always clear, however, what the best return will be when allocating funds for research and development for ageing problems.

The Specialist Meeting AVT-085 provided guidance on strategies for the development and implementation of new/existing technologies and logistic management processes, enabling the prioritisation of resources for fleet management and the research and development options. The emphasis was on military aircraft, but many of the principles could be applied to other defence systems. The papers covered the entire range of ageing problems including structural integrity, corrosion, avionics, mechanical subsystems, structures and wiring. There were also papers devoted to the role of information management as it applies to the ageing problem.

The programme contained forty-two papers addressing the needs of the manager charged with maintaining ageing systems. Papers provided an understanding of the safety and economic implications of ageing problems such as fatigue cracking, corrosion, wear and material degradation. In addition, the current status of key technologies was discussed, including non-destructive inspection, repair, modifications, prevention analysis, and health management. The shortcomings of current technology in effectively addressing the ageing problems were highlighted and the investment required was identified.

The need for research and development was clearly identified and it is recommended that these issues are pursued by the services to improve the availability of aircraft in service and to reduce the high costs associated with the maintenance of ageing aircraft. State-of-the-art technologies are available that could be adapted to ageing vehicles by additional research activity. The identified research areas should be pursued in order to reduce maintenance cost.

Under current regulations research and development resources are focused on the initial phases of new projects. This should be reviewed to facilitate the transfer and adaptation of existing technologies into ageing fleets. The services should take advantage of these possibilities to keep their fleets operational and cost effective, while maintaining a high level of safety.

It is recommended that the problems and technologies associated with ageing air vehicles should be pursued on a NATO wide basis and that RTO should continue to support the activities on ageing vehicles.

Mike Winstone

25/3/02

Réunions des spécialistes des techniques de gestion du cycle de vie pour véhicules aériens vieillissants

(RTO MP-079(II) / AVT-085)

Synthèse

Aujourd'hui, les chefs militaires de l'ensemble des pays membres de l'OTAN constatent que leurs flottes d'aéronefs sont maintenues en exploitation bien au-delà de leur durée de vie théorique. Les coûts de maintien de ces aéronefs vieillissants grèvent les budgets existants à un tel point qu'il est vraisemblable que les crédits nécessaires à la modernisation des flottes aériennes ne seront pas disponibles au moment où l'obsolescence imposera le retrait des vieux avions. La recherche et le développement ciblés sont l'un des moyens possibles d'alléger ce fardeau économique. Cependant, il n'est pas toujours facile de choisir le programme qui donnera le meilleur rendement lors de l'attribution des crédits de recherche et développement sur les problèmes de vieillissement.

La réunion de spécialistes AVT-085 a fourni des directives concernant des stratégies pour le développement et la mise en œuvre de technologies nouvelles/existantes, ainsi que de processus de gestion de la logistique, permettant l'établissement de priorités en matière de moyens de gestion des flottes aériennes, ainsi que les options de recherche et développement. L'accent a été mis sur les avions militaires, mais bon nombre des principes évoqués étaient applicables à d'autres systèmes de défense. Les communications couvraient l'éventail complet des problèmes du vieillissement, y compris l'intégrité des structures, la corrosion, l'avionique, les sous-systèmes mécaniques, les structures et le câblage. D'autres communications étaient consacrées au rôle de la gestion de l'information appliqué au problème du vieillissement.

Le programme était composé de 42 communications axées sur les besoins des responsables chargés de la maintenance de systèmes vieillissants. Les communications présentées ont permis de mieux comprendre les implications sécuritaires et économiques des problèmes de vieillissement tels que la fissuration par fatigue, la corrosion, l'usure et la dégradation matérielle. En plus, des discussions ont eu lieu sur l'état actuel des technologies clés, y compris l'inspection non destructive, la réparation, les modifications, l'analyse préventive et la gestion de l'intégrité des structures. Les lacunes des technologies actuelles vis-à-vis des problèmes de vieillissement ont été mises en évidence et l'investissement demandé a été identifié.

La nécessité de recherche et développement dans ce domaine a été clairement identifiée et il est recommandé aux armées de suivre ces questions de manière à rendre plus disponibles les aéronefs actuellement en service et de réduire les coûts élevés liés à la maintenance de flottes vieillissantes. Il existe des technologies de pointe susceptibles d'adaptation aux véhicules vieillissants, moyennant des activités de recherche supplémentaires. Il y a lieu de poursuivre les domaines de recherche identifiés afin de réduire le coût de la maintenance.

Le règlement actuellement en vigueur en matère de moyens de recherche et développement favorise les phases initiales de nouveaux projets. Cette situation est à revoir pour faciliter le transfert et l'adaptation des technologies existantes à des flottes vieillissantes. Les différents services devraient tirer profit de ces possibilités pour assurer la rentabilité et la disponibilité de leurs flottes, tout en maintenant un haut niveau de sécurité.

Il est recommandé de poursuivre les activités concernant les technologies et les problèmes des véhicules aériens vieillissants à l'échelle de l'OTAN. La RTO devrait continuer de soutenir ces activités.

Mike Winstone

25/3/02

Click inside the blue boxes or on the titles to view the corresponding section

Contents

(As presented in the AVT Meeting Announcement)

For your convenience a listing as presented at the meeting site has been added at the end of this list.

The conduct of this Meeting was marked by the "Events of 11 September 2001". A substantial number of authors were unable to present their paper at the meeting site, however these papers have been published in line with the schedule of the Meeting Announcement. Some replacement papers were submitted and are indicated after the "regular papers". The Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles" greatly suffered and an entirely new schedule was organised at short notice. Both the original schedule and the actual schedule have been included in the Proceedings.

AVT Executive

SESSION VIII: NONDESTRUCTIVE INSPECTION

SESSION IX: ANALYSIS AND LIFE EXTENSION

SESSION X: MODIFICATION AND REPAIRS

SESSION XI: INFORMATION MANAGEMENT

SESSION XII: FORCE MANAGEMENT

New Papers presented at the Meeting Site which were not announced in the Meeting Announcement

Theme

The military commanders in all NATO countries are seeing their aircraft operate well beyond the time they originally intended to retire them. The costs of maintaining these ageing aircraft are draining the existing budgets to the point that money will not be available to modernize their fleets when they will need to be retired because of obsolescence. A possible path to reduce this economic burden is through research and development. It is not always clear, however, what the best return will be when allocating funds for research and development for the ageing problem.

The specialists meeting will provide the attendees with guidance on strategies for the development and implementation of new and existing technologies, and logistic management processes. They will then be in a better position to prioritize resources for fleet management and fleet management research and development options. The emphasis will be on military aircraft, but many of the papers will apply to land vehicles as well. The topics will cover the entire range of ageing problems including avionics, mechanical subsystems, structures, and wiring. There will be papers devoted to the role of information management as it applies to the ageing problem. The forty-two papers on the agenda were written to address the needs of the manager charged with maintaining ageing systems. The technical details will be minimized in favor of providing a broad overview of the ageing issues.

The papers are designed to provide the attendees with an understanding of safety and, economic implications of ageing problems such as fatigue cracking, corrosion, wear, and material degradation. In addition, the current status of technologies such as nondestructive inspection, repair, modifications, prevention, analysis, and health management will be discussed. They will also describe the technology shortcomings for effectively addressing the ageing problems and, where possible, the investment required for attaining the needed capability.

Thème

Aujourd’hui, les chefs militaires de l’ensemble des pays membres de l’OTAN sont obligés de constater que leurs flottes d’aéronefs sont maintenues en exploitation bien au-delà de leur durée de vie théorique. Les coûts de maintenance de ces aéronefs vieillissants grèvent les budgets existants d’une telle façon que les budgets nécessaires à la modernisation des flottes pourraient être insuffisants au moment du retrait de service de ces appareils pour cause d’obsolescence. La recherche et le développement est un des moyens possibles de réduire ce fardeau économique. Cependant, en ce qui concerne le problème du vieillissement, la question de savoir, au moment de l’affectation des fonds, quel projet de recherche et développement sera le plus avantageux, est parfois délicate à résoudre.

La réunion de spécialistes fournira aux participants des orientations concernant les stratégies de développement et de mise en œuvre des technologies existantes et nouvelles, ainsi que des processus de gestion de la logistique. Ils seront alors plus à même de déterminer leurs priorités en matière de gestion des flottes, ainsi qu’en ce qui concerne les options de recherche et développement dans ce même domaine. L’accent sera mis sur les aéronefs militaires, mais de nombreuses communications porteront aussi sur les véhicules terrestres. Les sujets examinés couvriront toute la gamme des problèmes du vieillissement, avec notamment l’avionique, les sous-systèmes mécaniques, les structures et le câblage. Certaines communications seront consacrées au rôle de la gestion de l’information dans la mesure où elle s’applique au problème du vieillissement des véhicules. Les 42 communications inscrites à l’ordre du jour proposent des réponses aux besoins des personnes responsables de la maintenance des systèmes vieillissants. Les détails techniques seront réduits au minimum pour permettre un large tour d’horizon des problèmes de vieillissement.

Les participants pourront s’informer sur les implications économiques et de sécurité liées aux différents problèmes de vieillissement tels que les fissures, la corrosion, l’usure et la dégradation des matériels. De plus, la réunion fera le point de l’état actuel des connaissances dans le domaine des technologies d’inspection non-destructive, de la réparation, des modifications, de la prévention, de l’analyse et du contrôle de l’état des moteurs. Elle examinera également les lacunes technologiques dans l’approche des problèmes de vieillissement et, dans la mesure du possible, l’investissement qui serait nécessaire pour atteindre les capacités voulues.

Publications of the RTO
Applied Vehicle Technology Panel

MEETING PROCEEDINGS (MP)

Advanced Flow Management: Symposium Part A – Vortex Flows and High Angle of Attack for Military Vehicles / Part B – Heat Transfer and Cooling in Propulsion and Power Systems
MP-069(I), February 2003

Low Cost Composite Structures / Cost Effective Application of Titanium Alloys in Military Platforms
MP-069(II), February 2003

Ageing Mechanisms and Control: Symposium Part A – Developments in Computational Aero- and Hydro-Acoustics / Part B – Monitoring and Management of Gas Turbine Fleets for Extended Life and Reduced Costs
MP-079(I), February 2003

Ageing Mechanisms and Control: Specialists' Meeting on Life Management Techniques for Ageing Air Vehicles
MP-079(II), February 2003

Unmanned Vehicles (UV) for Aerial, Ground and Naval Military Operations
MP-052, January 2002

Active Control Technology for Enhanced Performance Operational Capabilities of Military Aircraft, Land Vehicles and Sea Vehicles
MP-051, June 2001

Design for Low Cost Operation and Support
MP-37, September 2000

Gas Turbine Operation and Technology for Land, Sea and Air Propulsion and Power Systems (Unclassified)
MP-34, September 2000

Aerodynamic Design and Optimization of Flight Vehicles in a Concurrent Multi-Disciplinary Environment
MP-35, June 2000

Structural Aspects of Flexible Aircraft Control
MP-36, May 2000

New Metallic Materials for the Structure of Aging Aircraft
MP-25, April 2000

Small Rocket Motors and Gas Generators for Land, Sea and Air Launched Weapons Systems
MP-23, April 2000

Application of Damage Tolerance Principles for Improved Airworthiness of Rotorcraft
MP-24, January 2000

Gas Turbine Engine Combustion, Emissions and Alternative Fuels
MP-14, June 1999

Fatigue in the Presence of Corrosion
MP-18, March 1999

Qualification of Life Extension Schemes for Engine Components
MP-17, March 1999

Fluid Dynamics Problems of Vehicles Operation Near or in the Air-Sea Interface
MP-15, February 1999

Design Principles and Methods for Aircraft Gas Turbine Engines
MP-8, February 1999

Airframe Inspection Reliability under Field/Depot Conditions
MP-10, November 1998

Intelligent Processing of High Performance Materials
MP-9, November 1998

Exploitation of Structural Loads/Health Data for Reduced Cycle Costs
MP-7, November 1998

EDUCATIONAL NOTES (EN)

Active Control of Engine Dynamics
EN-020, November 2002

Supercavitating Flows
EN-010, January 2002

Aging Aircraft Fleets: Structural and Other Subsystem Aspects
EN-015, March 2001

Aging Engines, Avionics, Subsystems and Helicopters
EN-14, October 2000

Measurement Techniques for High Enthalpy and Plasma Flows
EN-8, April 2000

Development and Operation of UAVs for Military and Civil Applications
EN-9, April 2000

Planar Optical Measurements Methods for Gas Turbine Engine Life
EN-6, September 1999

High Order Methods for Computational Physics, Published jointly with Springer-Verlag, Germany
EN-5, March 1999

Fluid Dynamics Research on Supersonic Aircraft
EN-4, November 1998

Integrated Multidisciplinary Design of High Pressure Multistage Compressor Systems
EN-1, September 1998

TECHNICAL REPORTS (TR)

Performance Prediction and Simulation of Gas Turbine Engine Operation
TR-044, April 2002

Evaluation of Methods for Solid Propellant Burning Rate Measurements
TR-043, February 2002

Design Loads for Future Aircraft
TR-045, February 2002

Ice Accretion Simulation Evaluation Test
TR-038, November 2001

NATO East-West Workshop on Magnetic Materials for Power Applications
TR-031, August 2001

Verification and Validation Data for Computational Unsteady Aerodynamics
TR-26, October 2000

Recommended Practices for Monitoring Gas Turbine Engine Life Consumption
TR-28, April 2000

A Feasibility Study of Collaborative Multi-facility Windtunnel Testing for CFD Validation
TR-27, December 1999

Programme Committee

Co-Chairman
Dr. J.W. Lincoln*
ASC/EN, 2530 Loop Road West
Wright-Patterson Air Force Base
OH 45433-7101, USA

Co-Chairman
Dr. M. Winstone
Structural Materials Centre
Room 2008, Griffith Building (A7), DERA, Ively Road
Farnborough, Hampshire, GU14 0LX
em: mrwinstone@dstl.gov.uk

BELGIUM
Prof. Dr. J. Vantomme
Royal Military Academy (RMA)
Dept. of Civil Engineering
avenue de la Renaissance, 30
B-1000 Brussels
em: jvt@cobo.rma.ac.be

CANADA
Dr. J-P. A Immarigeon
Institute for Aerospace Research
National Research Council of Canada
Montreal Road, Bldg M13
Ottawa, Ontario K1A 0R6
em: jean-pierre.immarigeon@nrc.ca

Mr. D. Rudnitski
Structures, Materials and
Propulsion Laboratory
Institute for Aerospace Research
1500 Montreal Road, Building M-7
National Research Council of Canada
Ottawa, Ontario K1A 0R6
em: don.rudnitski@nrc.ca

Mr. D.L. Simpson
Institute for Aerospace Research
National Research Council of Canada
1500 Montreal Road
Ottawa, Ontario K1A 0R6
david.simpson@nrc.ca

FRANCE
Mr. D. Chaumette
Direction Technique Aeronef
Dassault Aviation
78, Quai Marcel Dassault
92214 Saint Cloud
em: daniel.chaumette@dassault-aviation.fr

GERMANY
Mr. K. Fuellhas
EADS
Postfach 80 11 60
81663 Munich
em: konrad.fuellhas@m.eads.net

Dipl.Ing. G. Guenther
EADS GmbH
Military Aircraft, MT22
Postfach 80 11 60
81663 Munich
em: georg.guenther@m.eads.net

Mr. W. Koschel
DLR, German Aerospace Center
Space Propulsion
74239 Hardthausen
em: wolfgang.koschel@dlr.de

ITALY
Capt. M. Colavita
Italian Air Force-Chemistry
Dept. of the Flight Test Center
Aeroporto “M. de Bernardi”
00040 Pomezia (Roma)
em: mcolavita@tiscalinet.it

THE NETHERLANDS
Ir. J.J. Meijer
National Aerospace Laboratory (NLR)
Dept. AE
Anthony Fokkerweg 2
P.O. Box 90502
1059 CM Amsterdam
em: meijer@nlr.nl

Mr. H.J. Ten Hoeve
National Aerospace Laboratory (NLR)
P.O. Box 158
8300 Emmeloord
em: hjth@nlr.nl

UNITED STATES
Mr. J. Coyle
The Boeing Company
P.O. Box 516 Mail Stop S064-2809
St. Louis, Missouri 63166-0516
email: john.m.coyle@boeing.com

* Dr. J.W. Lincoln has since passed away.

Technical Evaluation Report

Günter Kagerbauer
EADS Deutschland GmbH Military Aircraft
P.O. Box 801160
81663 München, GERMANY

Introduction

The Fall 2001 Meeting of the Applied Vehicle Technology Panel (AVT) comprised two Symposia and one Specialists' Meeting on the subject of 'Life Management Techniques For Aging Air Vehicles'. This technical evaluation report addresses the above-mentioned Specialists' Meeting and reviews the papers, which have been presented in this meeting.

Objectives of the Specialists' Meeting

The programme for the Specialists' Meeting was structured into ten sessions covering the wide topic of aging air vehicle problems and in total thirty-one comprehensive papers were presented. Due to the non-availability of some authors and withdrawal of some papers on short notice the programme had to be continuously rearranged, but thanks to the effort by the Chairmen of the Specialists' Meeting Dr. J. W. Lincoln and Dr. M. Winstone the scheduled programme of the meeting could be performed.

The declared theme of the Specialists' Meeting was:

"The military commanders in all NATO countries are seeing their aircraft operate well beyond the time they originally intended to retire them. The cost of maintaining these aging aircraft are draining the existing budgets to the point that money will not be available to modernize their fleets when they will need to be retired because of obsolescence. A possible path to reduce this economic burden is through research and development. It is not always clear, however, what the best return will be when allocating funds for research and development for aging problems.

The specialists meeting will provide the attendees with guidance on strategies for the development and implementation of new and existing technologies, and logistic management processes. They will then be in a better position to prioritise resources for fleet management and fleet management research and development options. The emphasis will be on military aircraft, but many of the papers will be apply to land vehicles as well. The topics will cover the entire range of aging problems including avionics, mechanical subsystems, structures and wiring. There will be papers devoted to the role of information management as it applies to the aging problem. The forty-two papers on the agenda were written to address the needs of the manager charged with maintaining aging systems. The technical details will be minimised in favour of providing a broad overview of aging issues.

The papers are designed to provide the attendees with an understanding of safety and, economic implications of aging problems such as fatigue cracking, corrosion, wear and material degradation. In addition, the current status of technologies such as non-destructive inspection, repair, modifications, prevention analysis, and health management will be discussed, They will also describe the technology shortcomings for effectively addressing the aging problems and, where possible, the investment required for attaining the needed capability".

Session 1 – Overviews

This session consisted of three papers from the services covering the management aspects for aging aircraft fleets. The major challenges result from the facts that the average age of aircraft in service is increasing from year to year, that only small numbers of aircraft are replaced by new systems and that this results in reduced availability and reduced mission capability rates for the fleets and, therefore increased spending for maintenance action is required to overcome these problems. The papers highlighted that a management strategy is essential to overcome this “death spiral”.

The US Air Force has established a Programme Office for Aging Aircraft and is pursuing an integrated approach from analysis of the problems to integration of appropriate solutions in the form of roadmaps. A Depot System Capability Plan has been established to address future needs. Technology projects and technology demonstrations for aging aircraft are selected on the basis of return of invest (ROI).

The experience of the Italian Air Force (IAF) is that management tools, e.g. life cycle cost control are an essential element. Fleet monitoring for aging aircraft has been established and big emphasis is given to move to condition based maintenance and pro-active maintenance rather than to remain at corrective maintenance actions.

The German Air Force has learnt that the established maintenance databases are by far not sufficient to cover the problems of aging aircraft. The general approach followed is to change from a safe life to a damage tolerance philosophy. The German Air Force aims to improve usage monitoring by integrated systems.

Session 2 – Avionics

Unfortunately only one paper covered the issues of avionics in aging aircraft; however, the problems of obsolescence of avionic components are addressed in numerous other lectures. The paper showed that obsolescence even occurs on relative new projects and that the military market, due the small volume and the long periods between upgrades is no longer the technology driver for electronics. Operational capability upgrades of aging systems with respect to software and hardware updates are required to achieve long-term supportability of in-service weapon systems.

Session 3 – Strategy for Structures and Subsystems

Three papers have been presented on that subject. The papers outlined the following problem areas with respect to structures: Fatigue cracking and corrosion in structures, onset of widespread fatigue damage, substitution of materials and repair of airframes, lack of adequate non-destructive evaluation techniques for corrosion damage and widespread fatigue detection. The subsystems are one of the largest contributors to unscheduled maintenance and in general the subsystems become more and more unsupportable due to obsolescence problems, and a prediction of the remaining life of subsystem components is almost impossible. The papers highlighted that inadequate databases are available to determine damage and failure modes and the sources of damage/failure. A lack of non-intrusive techniques to access wiring health was identified. On subsystems, which in most cases do not comprise safety critical items, it was felt extremely difficult to obtain investment in research and development, as the appropriate return of invest could hardly be justified.

In September 1997 the United States National Research Council published a report identifying forty-nine research and development activities in aging aircraft. A System Programme Office for Aging Aircraft has been formed in 2001. A deterministic approach for ASIP and MECSIP has been adopted and a stepped approach from survey of the problems to transition of technology into aging systems is pursued. Research and development of technology has been initiated to assure safe and economic operation of military aircraft. The Air Logistic Centre has opted for the Functional Systems Integrity Programme (FSIP) in order to reduce availability problems.

Session 4 – Corrosion Management

This session was again covered by three papers, which showed the approaches in Australia, Italy and Canada. The papers addressed the different problems associated with corrosion management like impact of corrosion on airworthiness and flight safety, and readiness and efficiency of the fleet. A huge economic burden by corrosion was recognised in a "fix when found" maintenance environment, and a move to "assess and manage" corrosion problems should rather be adopted in future. The discovery of "fleet-threatening" corrosion has been identified as a major challenge. With respect to non-destructive inspection methods the suitability of NDI methods to detect corrosion in an early state and the probability of detection, respectively the detection of corrosion in different layers, in thick parts and in joints have been identified as areas where further research and development is required. In addition, to obtain deeper insight into the corrosion problem, appropriate modelling of the impact of corrosion on airworthiness and a dedicated monitoring of in-service corrosion data has to be established.

Australia has adopted the approach to incorporate corrosion into the Aircraft Structural Integrity Management Plans (ASIMP's) and to evaluate structural degradation management with important crossover of fatigue and corrosion effects. Emphasis is given to the analytical prediction of remaining life with exfoliation and pitting, and in maintenance procedures by the application of suitable corrosion preventives on aging aircraft.

In Italy a Corrosion Control Register Programme (CCR) has been established since six years, originally for the TORNADO aircraft, but in the meantime this CCR has been extended to six more aircraft. Indices for corrosion at different operational levels are applied down to part level. For the classification of corrosion damages four different classes have been selected.

Canada, in collaboration with USAF, has advocated a strict corrosion management approach. The Holistic Life Prediction Methodology (HPLM) includes environmental as well as fatigue mechanisms. The aim is to implement a HPLM based pro-active maintenance philosophy.

Session 5 – Modifications, Repairs, Analysis, and Life Extension

Due to the withdrawal of one paper from Germany three papers from the United States covered this session. The challenges with respect to aging aircraft addressed the changes in operational environment and incorporation of new stores, respectively role configurations, and the damaging environment, which could not be properly considered in the original design. Additionally major problems are found with regard to the life enhancement of airframe holes and the repair of holes and bushings, to fatigue cracks in structures requiring structural replacement and to the effects of widespread fatigue damage on residual strength. The economics of ongoing inspections, the correlation of analytical and test results and the complexity of joints has been addressed. For future modifications and repairs the application of new materials and fabrication methods, like the use of castings, should to be considered.

The papers presented different approaches to overcome the problems associated with repair and modifications. First, the application of the cold expansion process for holes and bushings for aging aircraft and new designs represents a low cost method to increase airworthiness by arresting the growth of small undetected cracks. Another cost-effective solution is the application of damped bonded patches into retrofit and repair actions, as a repair time of less than twenty-four hours could be achieved with room temperature curing adhesives. In addition it was recommended to develop a methodology for widespread fatigue to improve the correlation between actual test results and analytical methods.

Session 6 – Engines

The subject of aging engines was covered by four papers from the United Kingdom, from Canada, from Poland and from the United States. The challenges generally presented are the cost-effective management of engines, aging engine problems due to obsolescence, and the safe operation of engines at maximum life beyond the service life. A more detailed understanding of deterioration modes and the potential impact on engine performance, reliability and safety is required. An improvement of non-destructive inspection methods and a higher probability of detection of defects have to be achieved, and, based on the failure history, a validation of the inspection technologies has to be pursued.

The papers presented the incorporation of modern Health and Usage Monitoring System (HUMS), and with respect to maintenance actions the repair of components respectively substitution of materials to extend engine components lives. The use of innovative living techniques, and the use of databases to analyse the different operational phases have been proposed to cope with aging engine components, and, in addition a risk management methodology should be established to address the problems of aging engines.

Session 7 – Wiring and Electrical

Three papers addressed the problem areas of aging wiring and electrics. The deterioration of physical properties and the performance with time, the handling of electrical wiring, the environment, the usage, and in general installation and maintenance practices were identified as major sources for repair action. Conductors and connectors contribute to approximately forty-five percent of the failures and therefore have a significant impact on maintenance cost. The available diagnostic tools are considered not comprehensive to locate damage. Wiring problems are usually found by troubleshooting rather than by visual inspection Therefore a change from reactive to pro-active maintenance is required, as in older aircraft the risk due to wiring problems and arcing in circuit breakers is continuously increasing.

The solutions to this problems presented in the different papers addressed the requirements for additional wiring inspection with necessary portable inspection equipment to be developed, the identification of corrective action/repair and the urgent necessity for additional repair tools, techniques and materials. With respect to the electrical failures occurring in service appropriate databases for wiring codes and wiring systems have to be established, and wiring and electrical failure data/results need to be recorded and stored in these databases. In addition fault graph methodologies need to be established.

Session 8 – Non-Destructive Inspection

The two papers presented in this session discussed the pivotal role of non-destructive inspection for maintaining safety through early crack detection and for minimising corrosion maintenance cost. Corrosion becomes a significant driver for airframe maintenance planning with an high economic impact. Assessment methodologies for corrosion with appropriate probabilities of inspection have to be established. A major challenge exists in finding hidden corrosion economically and work should be focussed of those areas with high return of invest.

Further research is needed in modelling and metrics of corrosion in lap joints and thick sections, and to obtain decision guidelines whether to repair or not to repair parts affected by corrosion. More extensive use of automated inspection equipment to detect corrosion has to be considered in order to reduce the inspection burden.

Session 9 – Information Management

This session was covered by two papers from the United States. For aging fleets the tools and technologies to support obsolescence problems are required. In general the prediction of useful remaining life is a problem and a lack of information to identify and quantify the risks has been identified, especially as maintenance data are not consistently recorded and sometimes stored on different databases, which does not allow the fusion of the failure data. This information is considered essential for the change from a diagnostic to a prognostic maintenance approach. Current state of the art technologies, like health and usage monitoring systems need to be transitioned into aging systems to reduce the high cost of maintenance.

To overcome the problems with aging aircraft data gathering with new sensors has to be improved, and the fusion of available databases, together with the incorporation of expert knowledge has to occur. An extensive modelling of aging aircraft systems, together with an improvement of damage mechanism prediction, has to be initiated to determine when sustainment actions have to be performed. The available databases need to be established on a much broader base, e.g. on NATO level to collect the experience in the different services. Last not least the mental change form diagnostic to prognostic maintenance action has to be achieved.

Session 10 – Fleet Management

In the management of aging fleets the high standards of safety still have to be achieved, even when the supportability of in-service weapon systems is jeopardized by obsolescence and associated with continuously increasing maintenance cost. The DSTO and the Australian Air Force have addressed the special case of being the sole operator of F-111 aircraft.

As possible solutions to reduce the problems in fleet management the use and adaptation of state of the art technologies for modifications, repairs and part re-manufacture and additional investment in these areas to transfer available technologies, like health monitoring, repair actions and advanced corrosion treatments, to aging aircraft were presented. A necessity for better tools and simulation to forecast life cycle cost of aging fleets and to determine the economic service life has been identified. Within the modelling, methodologies to take into account corrosion, fatigue damage and widespread corrosion have to be developed to support a structural management approach for aging aircraft fleets. Extensive databases with historical data and fusion of these data are considered as key element to make this approach a success.

Concluding Remarks

As already expressed in the objectives of the specialists' meeting most papers addressed aging issues and the associated economic burden on military aircraft, and the further research and development work required. However, other vehicles, like land vehicles, could benefit substantially from the lessons learnt and the technologies incorporated in air vehicles, assuming that an exchange of information could be arranged.

Within the objectives of the specialists' meeting the papers presented were of very high quality and the content of the papers were in almost all cases highlighting the overall problems rather than to present too much technical detail. The necessary research and development issues were clearly identified and it is recommended that these issues are pursued by the services in order to improve the availability of aircraft in service and to reduce the high costs associated with aging aircraft maintenance.

It is recognised, that research and development funding can be more easily obtained for "new and advanced projects" than for research on existing and old aircraft and that it is sometimes hard to justify the return of invest of these research investments. However, as the available fleets become increasingly older from year to year and only small numbers of the existing fleet are replaced by new aircraft, basically a "mental" change is required with respect to aging aircraft problems. Due to the changing threat scenarios since the beginning of the 90's and the associated reduced defence budgets more aging aircraft will continue to be in service for even longer periods.

The specialists' meeting has shown, that state of the art technologies are available, which could be incorporated into aging air vehicles and that, in a lot of areas, adaptation of these new technologies to aging vehicles could be achieved by additional research activity. The research areas identified should be pursued in order to reduce maintenance cost.

It is the opinion of the author that current R&D regulations, according to which research and development spending can be obtained only in the initial phases of new projects, should be reviewed with respect to their rationale. The necessity to adapt and transfer existing technologies to aging fleets is without question and the areas where further research activities are required have been highlighted at the specialists' meeting. The services should take advantage of these possibilities to keep their fleets operational respectively and provide appropriate funding for research and development on aging air vehicles.

To RTO it is recommended that the subject and the problems associated with aging air vehicles should be pursued on a NATO wide basis and that RTO continues to support the activities on aging vehicles in the future.

Managing the Aging Aircraft Problem

John W. Lincoln
Aeronautical Systems Center
2530 Loop Road West
Wright-Patterson Air Force Base, Ohio 45433-7101
USA

SUMMARY

Aging aircraft face many challenges. Certainly, one of the most costly is corrosion. The United States Air Force (USAF) spends approximately $800 million dollars a year for corrosion detection, prevention, and repair. Another major challenge is structural fatigue cracking. This problem has significant safety implications as well as economic. Aging mechanical subsystems constitute a challenge in that they can have such a severe impact on aircraft availability. Aging wiring is also a safety and economic problem. It has not been subject to the research effort that corrosion and fatigue cracking has had in the past. Consequently, it has taken time to initiate remedial actions. Aging avionics is also a major economic burden. The problem is so severe that many of the aging aircraft are not supportable. The aging aircraft problem may be thought of as a spiral. The number of repairs increases. This causes the depot flow rates to decrease. Consequently, the maintenance burden increases. Since there is a lack of funding, the mission capable rate decreases. The problems grow larger through each turn in the spiral. This means that money for modernization of the fleet is not available. Consequently, the aging aircraft must be retained in the inventory longer than expected. Adequate funding combined with a well-conceived research and development program is essential to break the spiral.

BACKGROUND

Most aircraft operated commercially or in the military reach a state referred to as aging at sometime after entering operational service. Aging of an aircraft is not the same as it becoming obsolete. An aircraft may be obsolete before it reaches the aging state or, more typically, it reaches the state of aging before it is obsolete. A commercial aircraft is obsolete when it is no longer economically viable to keep it operational. A military aircraft is obsolete when its capabilities are no longer competitive with potential adversaries. The time when an aircraft reaches the aging state is usually much more difficult to determine. It is important to distinguish between the characteristics of the structure of a young aircraft and an aging aircraft. A young aircraft is one that continues to be airworthy with the maintenance program prescribed at the time of manufacture. The primary concern with a young aircraft is the potential for design errors that introduce unintentional high stresses in the structure that could lead to premature fatigue cracking incidents. When these are discovered the structure is modified to eliminate the problem. An aging aircraft may be characterized as one where the effects of corrosion and cracking from fatigue require modification of the maintenance program to retain adequate structural integrity. The word adequate here means that the expected number of failures would be less than one in a given fleet of aircraft. As an aircraft accumulates calendar time and flight time the effects of corrosion and cracking from fatigue, as well as accidental damage, leads to repairs on the aircraft. Cracking from fatigue can be so widespread that it degrades the integrity of the structure. When this occurs, the structure is said to be in a state of widespread fatigue damage or WFD and must undergo modifications to remove this problem. In addition, as an aircraft accumulates flight time, it may exceed its design life goal. Therefore, the maintenance program will require modification to include additional structural inspections. If the initial maintenance program requires modification from any of these events, then the aircraft may be considered to be in a state of aging.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

No one should be surprised there are aircraft all over the world today in a state of aging. Economic considerations demand that aircraft be operated long beyond originally identified retirement times. One reason for keeping aircraft in the inventory is that technological advances allow currently designed aircraft to effectively perform their mission for much longer than previously possible. An aircraft, even when sold by one airline, sees extended life in another airline's operations. In the commercial sector, new aircraft tend to be evolutionary in their designs. Consequently, they are maintained in service until they are not economically viable to operate. The cost of new aircraft, particularly for the military, is enormous. Each new military aircraft is a revolutionary change from the previous model since the services must maintain combat effectiveness in an environment of ever-changing threats. Therefore, military aircraft stay in the inventory until they are operationally obsolete or they are no longer economically viable to operate. The USAF retired many F-4 aircraft because they were obsolete as a weapon system rather than being economically nonviable. In the case of the KC-135, the USAF plans to keep these aircraft operational to the year 2040 since they believe it will be economical to operate them until that time. They would likely not be obsolete in the year 2040. If the USAF can maintain these aircraft operationally until 2040, their service life will be approximately 80 years. When the USAF procured these aircraft, they planned for a service life of about 20 years.

Sustainment of an aircraft is the act of keeping it operational (i.e., airworthy). Maintenance of an aircraft (that is, the work done by mechanics in keeping it airworthy) is one aspect of sustainment. However, sustainment also includes the engineering analyses and tests needed to determine an adequate maintenance plan for the aircraft. Sustainment is life management. One task of sustainment is the determination of structural inspections based on damage tolerance principles. These inspections protect against failure from defects that could be in the structure because of manufacturing or from operational service. The approach for developing a damage tolerance derived inspection program is discussed in Section 2. Another task of sustainment is the determination of the time of onset of fatigue cracking in the structure so widespread that the structural integrity of the aircraft could be compromised. Experimental evidence shows that fatigue cracks smaller than those that could be reasonably detected by current inspection methods could constitute WFD.

Today, the primary concern with aging aircraft is the cost of their sustainment. The commercial operators buy new aircraft when it becomes economically viable for them to do so. The aging aircraft problem, however, has often made itself known to both the commercial and military operators through failures of in-service aircraft. Both operators found the maintenance programs did not adequately protect aircraft as they progressed through their service lives. The failures in both commercial and military aircraft have been the primary factor that has changed rules and specifications that are the basis of their design. In many cases, the failures have identified threats to structural integrity that were not previously identified by the certification authorities. In many cases, the commercial failures have influenced the military specifications and the military failures have influenced the commercial aircraft rules. The new rules and specifications have lead to better maintenance programs that help alleviate many of the threats to failure that previously existed. However, the economic demand to fly these aircraft longer and longer has emphasized the need to re-examine these aircraft for the possibility of WFD, corrosion damage, and loss of damage tolerance capability through repairs.

It is difficult to determine the exact moment in time when an aircraft has reached the state of aging. However, all would agree that the costs associated with repairs or modifications from corrosion or cracking from fatigue would be an indicator of this condition. When these costs rise significantly, then the aircraft are certainly in that state. Chronological age is not always a good indicator of aging. However, since corrosion and fatigue are somewhat related to time in service, it does give some insight for the potential of this problem. A summary presented in July of 1997 detailed the reasons for the increase in chronological age of aircraft. In 1996, the total U.S. military fleet consisted of 20,400 aircraft. Reduced procurement budgets will prevent these aircraft from being retired since at the current rate of procurement, the time required to replace these aircraft would be approximately 100 years. The basis for this conclusion is that in 1996, the DoD budget procured only 159 new aircraft for $5.4 billion. The estimate of the cost of replacing the entire fleet of U.S. military aircraft is $530 billion in 1995 dollars. Many of the aircraft in the population today are currently more than 30 years old.

Many nations are now keeping aircraft in their inventories longer than ever before. In many cases, aircraft are left in the inventory longer because they are still operationally effective; however, in most cases, they remain in the inventory because the money is not available to replace them. Aircraft, which are seeing the effects of aging through corrosion and fatigue cracking, are causing their operators to bear a significant economic burden to keep them operational with the potential for degradation of flight safety of aging aircraft if they are not maintained properly.

The United States Air Force (USAF) has maintained safety of their aircraft for the last thirty years through the application of damage tolerance principles to determine inspection intervals. This approach has on occasion been modified because of the onset of widespread fatigue damage (WFD) or the loss of material because of corrosion. In the case of WFD, the USAF has developed a modification program to alleviate the problem. In the event of corrosion damage, both modification and reduced inspection intervals have been used.

The USAF has developed a strategy for the sustainment of their aircraft starting with the identification of user needs requiring research and development efforts. The strategy is based on identifying research and development opportunities that will have a favorable return on the investment through cost savings or cost avoidance and increased aircraft availability. This has presented problems since it is difficult to determine the cost of maintaining aircraft in enough detail to determine the return on the investment accurately. To date, identified activities include improvements in nondestructive inspection capability, corrosion tracking and prevention techniques, and advances in repair of metallic structures through composite patching. In addition, improved materials for substitution and environmentally compliant coatings have been identified. The purpose of this paper is to provide a discussion of the aging concerns found in the structure of USAF aircraft and the approach the USAF is pursuing to alleviate these concerns.

STRATEGY FOR AGING AIRCRAFT

The damage tolerance approach has led to a greatly improved understanding of aircraft structures and their performance. It, when properly applied, will essentially eliminate fatigue cracking as a threat to structural integrity. Therefore, damage tolerance should be the foundation on which the structural maintenance program should rest. During the 1970's and 1980's, the USAF performed an assessment on every major weapon system using the damage tolerance approach to develop appropriate inspection/modification programs to maintain operational safety. As a result, the USAF has maintained an excellent structural failure safety record despite the ever increasing age of its' fleets.

As the aircraft grow older, the potential for fatigue cracking and corrosion increases. Many of the aging aircraft in the USAF inventory are experiencing increased maintenance costs because one or more of these problems are present. To determine the research and development actions that could be pursued relating to these problems, the USAF formed the Aging Aircraft Technologies Team (AATT). At the outset, the AATT decided they would operate under the following guidance for the researcher:

- Research and development:
  - ♦ Must be directed towards needs of USAF aircraft
  - ♦ Must be oriented towards flight safety, maintenance cost reduction, and/or enhanced availability
  - ♦ Must be output-oriented and cost-focused
    - ➢ Researcher and customer must communicate on expectations from research
    - ➢ Researcher must be able to define cost and schedule for activity

- Develop technology that can be transitioned
- Augment highest level of capability in industry or government
- The USAF laboratories must maintain organic competencies in key areas related to aging aircraft

The most difficult challenge for the AATT was to determine the return on the research and development investment for aircraft. Although the USAF has usable cost information on total cost for depot or field operations, they do not know the cost in sufficient detail to judge whether an effort on a certain component of the aircraft is justified. For example, a wing section may be subject to corrosion, but without cost data on this component, it is difficult to determine if the USAF should replace this component using a more corrosion resistant material or keep repairing the corrosion damage.

The strategy for identifying research and development programs to reduce maintenance costs can be described by six steps. These steps are

- Conduct surveys to determine problems
- Identify and prioritize solutions requiring research and development
- Establish research and development roadmaps
- Obtain management and customer approval
- Execute research and development efforts
- Transition technology to the operator

The first three steps are designed to identify the problems and develop a plan for their solution. The final three steps engage management to implement the plans and carry the research and development through to technology transition.

The first step is to conduct surveys on aging aircraft in the inventory to identify their problems in as much detail as possible. This effort includes interviews with engineers and maintenance personnel that are directly responsible for the continuing integrity of the aircraft. In addition, the AATT discussed the aging aircraft issues with the operators of the aircraft. Identification of problem areas generally requires multiple meetings with these individuals in order to get a complete understanding of the problems that may have a solution through research and development. For example, for the USAF, surveys by the AATT of approximately thirty aircraft have taken place annually four times. In most cases, the reviews required approximately one day to complete. In some cases, however, there was interest in the review by the operator because of the potential for significant funds to be expended by them for modernization of the aircraft. In these cases, the reviews required two to three days to complete. The surveys, whenever possible included laboratory personnel so that they could see the nature of the problems for themselves. In addition, the surveys included the original equipment manufacturer (OEM) since their knowledge of the aircraft is essential for a clear definition of the problem. The relationship between the USAF and the OEM for maintaining the integrity of aging aircraft has been outstanding.

The second step is to determine the potential solutions to the aging problems. The AATT makes an initial screening of the problems and makes a preliminary determination of those problems that may have potential for a research and development solution. These problems are then given the widest dissemination possible to solicit possible solutions. The potential solutions are then categorized as basic research, exploratory development, and advanced development. The last category is for technology that is ready for transition. Usually, solutions categorized as basic research require development times that are extremely long for the technology to reach maturity. However, this area cannot be overlooked since it may, in time, have significant return on the investment.

The third step is to develop “roadmaps’ for the maturation of the technology for use in the aircraft. The roadmap identifies the problem, the technology to be used to solve the problem, the tasks to be performed for the solution, the schedule for completion of each of the tasks and the funding required for completion of each task. This step requires close adherence to the AATT guiding principles.

The fourth step, which is management approval, is likely the most important in that the success of the entire program rests on the agreement of the technologists and managers that the research and development program has the greatest return on the investment. This, of course, means that the manager has an understanding of the scope of the effort required to reach the desired goals. With this step the process changes from bottom up to top down. The first three steps could be identified as “bottoms up” activity. These steps started with identification of the problems and ended with a strategy for solution.

The fifth step is execution. The strategy is not complete, however, without the implementation step. This step is “top down” in that management charges the researchers to perform to the aging aircraft roadmaps. Essential to this step is the acceptance of the technology by the logistics managers and the operators. They must demonstrate willingness to implement the technology developed to reduce their maintenance burden.

The final or sixth step is technology transition. Actually, technology transition starts with the second step since there is no solution unless the technology can be transitioned to the logistics centers. Another name for this effort that is at times more descriptive is “industrialization of the process.” The first requirement for technology transition is adequate funding. The second requirement is that logistics personnel understand and are trained in the execution of the process. The final requirement is that logistics personnel are convinced that the new technology will enable them to do the job better than they are doing with existing technology.

AGING AIRCRAFT ISSUES

Funding

Funding for aging aircraft research and development activities is likely the number one problem faced by the manager. Funding of aging aircraft requirements is usually inadequate due to ever-increasing structural modification programs, safety issues, sustaining engineering needs, and responding to ever changing retirement dates. It is difficult, if not impossible; to support all identified technology areas. Consequently, there is a need to establish priorities.

In addition, justification is typically difficult for non-safety related problems since the available funding is usually consumed by safety related problems affecting the force. Care should be taken in the maintenance of non-safety related problems in that they may become safety issues through improper maintenance. An example of this is the use of inferior bonding techniques to repair honeycomb structures. Improper techniques can lead to moisture intrusion and an accelerated degradation that has the potential for loss of integrity of the component. Future funding requirements need a better understanding of return on the investment (ROI) where the ROI includes cost and availability in order to compete competitively with all the other identified requirements. Improved cost data collection procedures are needed in order to accomplish this.

In many cases, the budgets have not allowed the modernization of maintenance facilities or the upgrading of their information management systems. This has led to maintenance practices that are not state-of-the-art in that the use of information management has not become ingrained in the work force. This inadequacy is compounded by inaccurate and often-inadequate maintenance databases that lead to a misunderstanding of logistics requirements that raises costs and reduces aircraft availability. Retention of maintenance records for structural repair needs to be made a priority.

Fleet/ Depot Planning/Procedures

Another issue is the lack of support from operators for tail number tracking used to determine damage and sources of damage. Too often, since the recorder for flight loads is not flight essential equipment, the operator fails to adequately download data or service the recorder. This has resulted in many cases where the operators were not aware that some of their practices were causing damage to aircraft that could be have been avoided. The logistics community needs to communicate with operators to find operational techniques to reduce damage.

There is a concern that inadequate manning levels in both the field and at depots are causing lack of compliance with technical orders. In addition, the experience level of maintenance personnel has been steadily reducing for both the civilian and military population due to workforce downsizing. These problems are compounded by diminishing engineering resources in the manufacturing base result in increases in flow days and costs since the parts must often be fabricated through reverse engineering.

There are also many depot maintenance procedures/planning issues that cause increased costs. The maintenance planners should look at the frequency of depot maintenance visits, especially for aircraft experiencing moderate to high levels of corrosion. In some cases, the depot intervals are so long that many problems are discovered too late, resulting in more expensive and complex repairs that could have been caught earlier and remedied much easier with more depot visits. To make matters worse, many times within the same depot, there will be inconsistent maintenance practices being used from one product line to the next.

Finally, although experience shows that there are many surprises found in the maintenance of aging aircraft, there is usually a lack of planning and budgeting for these events. This leads to increased costs and a further lack of availability.

One of the most important aspects of an aging aircraft program is the quantification of the economic burden of systems in future years. This activity is necessary to support planning for retirement of existing aircraft and procurement new aircraft in the future. Without this information, the visibility is lacking to make sound judgements for aging aircraft. The process for doing this is well established for fatigue cracking. Unfortunately, for corrosion it is not as well understood.

FATIGUE CRACKING AND CORROSION

The USAF identified that the root causes for most aging related structural issues are fatigue cracking and corrosion. For each of these causes, the solutions could be obtained in one or more of the categories: nondestructive inspection (NDI), repair, modification, prevention, analysis, health, or information technology. Usually, the ultimate solution will be a combination of these categories. The discussion below covers the main efforts for both fatigue cracking and corrosion.

FATIGUE CRACKING

The introduction of damage tolerance principles by the USAF in their structural inspection program in the early seventies virtually eliminated fatigue as a safety problem in their aircraft. However, fatigue cracking of operational aircraft in the USAF is still a significant economic problem. The USAF estimates that this problem cost approximately $250 million in 1997. The USAF attributes much of this burden to operational usage being more severe than the usage assumed for design. This occurs because as the aircraft is fielded, the operators find unique and unanticipated ways to take full advantage of the capabilities of the aircraft. This often results in more severe usage due to weight growth for new capabilities or new operational mission profiles.

Based on design processes used today, fatigue cracking in an airframe should not be a significant factor for an aircraft whose operationally usage is approximately the same as its design usage. However, many of the older aging aircraft in use today were designed at a time when the effects of repeated loading was not a design consideration. Consequently, fatigue cracking is surely an economic problem and in most cases is a potential safety problem. Fatigue cracking found by inspections based on damage tolerance principles had resulted in many repairs on operational aircraft. In many cases, the cracks are repaired when found. The USAF has found that historically this is the most economical approach and consequently, this approach is most often used until it becomes evident that the structure needs to be modified.

The certification basis on which the structure was qualified also plays a critical role in the research and development for aging aircraft. The USAF guidance for structural certification includes both slow crack growth and fail-safe structures. Although, the slow crack growth approach is most often used today, the USAF strongly advocates the use of fail-safe designs whenever practical. Designs that are fail-safe can tolerate the failure of a structural member and still maintain adequate residual strength until the failed member is discovered through inspections.

Widespread Fatigue Damage

The certification basis for many aircraft is fail-safety because it provides a good overall approach to achieve both safe and economic operation. However, when the structure develops WFD, it can cause a loss of the fail-safe capability in the airframe and drastic action is needed to restore it. Perhaps, the most famous incident of WFD is the 1988 operational failure of a Boeing 737. This event provided the motivation for the considerable emphasis by the FAA on the structural issues associated with aging aircraft. This event occurred on Boeing 737 (N73711) on 28 April 1988. On this date, cracks in the fuselage lap-splices coalesced resulting in loss of the upper fuselage from just aft of the pilot’s cockpit to the wing leading edge. This was the start of the aging aircraft program for the FAA.

The effect of WFD on flight safety has long been a concern of many researchers. Most of the older USAF aircraft designs did not comply with the modern guidance for damage tolerance assessment (DTA). Consequently, there is a potential for the crack population to be so large in the structure that the application of the deterministic damage tolerance process may not protect safety. Large crack populations could also exist in monolithic structures such as the T-38 aircraft, which the USAF analyzed using probabilistic methods. The USAF refers to cracking as found in the T-38 aircraft as generalized cracking rather than WFD. The occurrence of WFD can significantly degrade the fail-safety of the structure. This problem has been evident on the KC-135, C-5A, C-141 and the E-8 aircraft. The USAF subjected these aircraft to teardown inspections. They incorporated the results of these inspections in a risk assessment to quantify the time when the probability of failure, conditioned by the fact there had been discrete source damage, becomes unacceptable.

CORROSION

Corrosion and fatigue separately have both led to serious safety as well as economic problems. Corrosion alone, in forms such as uniform corrosion (thinning) or exfoliation, may reduce the strength of aircraft and lead to failure. Both of these forms of corrosion may lead also to expensive component repair or replacement. There are many cases where corrosion alone is not significant from a safety consideration, but is a very significant economic problem. In the case of corrosion alone, one must judge the seriousness of this problem on an individual basis. Nondestructive inspections have found fatigue problems where there is essentially no influence from corrosion. Researchers have documented many cases over the years where the consequences were catastrophic. The results of fatigue cracking have caused many expensive repairs and modifications to the structure including component replacement. Fatigue often combines synergistically with corrosion. In these cases, the term corrosion fatigue is more appropriate. In most cases, corrosion, fatigue, or corrosion fatigue becomes a safety consideration only when either maintenance is not performed properly or the maintenance program is inappropriate. Experience derived from diligent maintenance has repeatedly shown that the operator need not compromise safety resulting from these problems. The purpose of this section is to describe some experiences with corrosion, fatigue, and corrosion and fatigue and to review some of the relative literature on this subject.

ASSESSMENT OF CURRENT SITUATION

Aeronautical Systems Center has created an Applied Technology Process that formalizes the relationship between the Air Force Research Laboratory, the Aging Aircraft System Program Office and the Air Logistics Centers for technology transition. This effort was enhanced when the Aging Aircraft Program Office Became the Aging Aircraft Systems Program Office on 25 Jan 2001.

The aging aircraft surveys conducted over the past four years have identified research and development needs for structures quantitatively. Solutions to these problems need to be found with cost and schedule determined. The roadmaps for solution of problems and transition of new technology should cover the next five to seven years. Any further out tends to be difficult to define adequately.

Aging aircraft surveys have identified needs for mechanical subsystems only qualitatively. There is still much work to be done to define these problems qualitatively. In the near term, emphasis needs to be placed on further development of the Functional Systems Integrity Program (FSIP) type approach. FSIP is a procedure for identifying, tracking, and taking remedial action for systems problems. In the early eighties, the USAF developed a proactive approach for subsystems called MECSIP. This concept, unfortunately, has not had needed laboratory development to make it practical for operational aircraft. Studies are needed to determine the future of MECSIP in the USAF. In its original form, MECSIP was developed using deterministic methods. Probabilistic methods in lieu of deterministic methods should be assessed to determine if they are better suited for this technology.

In an effort to understand the wiring problems in the USAF, field surveys at the Air Logistic Centers and field units served to document wiring and maintenance issues at each location as well as determining research and development needs. The Air Force Research Laboratory is currently working with the US Navy, Federal Aviation Administration, and the National Aeronautical and Space Administration to address needs. At this time, the research for this problem is immature. However, the consensus is that wiring problems can be managed.

For aging avionics, upgrading technology is the major influence. The problem is that the time period for technology change is short. Consequently, the older systems are not maintainable. Diminishing manufacturing resources is also a major problem in maintaining avionics equipment. The focus of the strategy is to develop policies/processes that create open architectures that allow future buys to be more affordable rather than by trying to maintain older technology. Many systems, such as the C-130 and the C-5 will benefit from this change in policy.

CONCLUSIONS

There is no question that research and development funding will remain a problem. It is very difficult to get adequate funding to make a real impact on the cost of operating aging aircraft. It is necessary; therefore, to place emphasis on government agency co-operation.

Quantification of maintenance costs is a problem that will continue to impede progress. There will need to be a major change in the current maintenance approaches to enable the researcher to quantify the cost of maintenance. Until this happens, it will be difficult to quantify return on research and development investment.

A related problem is parts management. The current approach leads to inefficiencies and unnecessary costs.

The USAF research and development program for aging aircraft have provided the technology base for safe and economic operation of military aircraft. This success, however, should not be used to indicate that there is no need for continued research on aging aircraft. The dangers from corrosion, fatigue or corrosion fatigue are ever present in operational aircraft. Presently, the largest danger by a considerable margin is economic rather than flight safety. All of the collective experience from both military and commercial operations indicates this to be true. No one can foretell with any degree of accuracy what to expect as both military and commercial aircraft push further into the uncharted waters of aging. It is incumbent; however, for the researcher, the engineer and the maintenance personnel to maintain a diligent approach to the problem. They must use all available techniques such as DTA scheduled inspections, special inspections, and assessments for the onset of WFD to help ensure that they maintain the safety of future aircraft operations. Diligent use of CPC's and research into better means of corrosion detection and prevention appear to be the most promising ways to reduce the economic burden of these problems in the future. The priority for corrosion research and development needs to be given to corrosion detection and the inhibition of corrosion when found.

One of the major problems found in operations with aging aircraft is the cost associated with corrosion damage. Unfortunately, the progress made in the recent past in the control of this problem does not bode well for the future. This is especially true when one considers the impact of new environmental laws that remove many of the corrosion fighting chemicals that are currently used. Continued emphasis on research in the area of corrosion control is certainly one area that could have significant benefit.

Another major problem is WFD in primary structural elements. There will be costs incurred to establish an estimate of the time of onset of this problem. This will need to be done through the analysis of data derived from teardown inspections of fatigue test articles and/or of operational aircraft. These estimates will need to be corroborated through the use of detail inspections of suspect structural elements. Once this onset time has been reached, then there will be costs incurred by the modification of the aircraft to remove this problem.

The severity of both of these problems is made worse today because of a lack of adequate nondestructive evaluation techniques to look for corrosion damage in structural joints and to find the small cracks that would be the indicator of the onset of WFD. It appears that the current efforts in research in nondestructive evaluation will produce the technology for these problems. It remains to be seen if there is an economic motivation to transition this technology from the laboratory to inspections of operational aircraft.

Many of the aging military aircraft problems find an exact parallel in aging commercial aircraft. It is prudent, therefore, that these problems be worked through the combined talents and resources of the responsible organizations. Efforts to date indicate that this approach will be successful and most efficient in solving these complex problems.

This page has been deliberately left blank

Page intentionnellement blanche

Setting up a Strategic Architecture for the Life Cycle Management of USAF Aging Aircraft

M. Carpenter
Aging Aircraft System Program Office, Aeronautical Systems Center
2145 Monahan Way, Wright-Patterson AFB, Ohio 45433-7017, USA

J. White
Aging Aircraft System Program Office, Aeronautical Systems Center
2145 Monahan Way, Wright-Patterson AFB, Ohio 45433-7017, USA

Abstract

The average age of United States Air Force (USAF) aircraft is over 22 years and increasing. The USAF is buying only a fraction of the new aircraft necessary to simply stop the declining age trend. The real effects of aging are seen in increased costs of ownership, and decreasing availability of aircraft to accomplish their mission. With thousands of aircraft in the USAF fleet, managed by numerous agencies, the job of managing the affects of aging is as much a managerial and leadership challenge as it is a technical one. Considering the immensity of the USAF and its breadth of locations around the world, the effort to develop and implement a strategy for managing the USAF's aging fleet is enormous.

This paper offers an overview of the management approach being taken by the USAF to manage it's aging fleet and to mitigate the unique effects of aging experienced by its aeronautical weapon systems.

First, we offer an understanding and scope of the problem posed by the aging of air vehicles and the systems that support them. Naturally, a strategy for management and technology is developed accompanied by an implementation plan. To carry out the strategy, an execution plan is developed and implemented. To assure continuous refinement of plans and strategies, methodologies for feedback and measurement of metrics must be put in place. Finally, adjustments must be made to strategies and plans to reflect corrective actions necessary in response to metrics, feedback, and environmental changes to assure we continue to lower the cost of ownership, and increase the availability of the aircraft and their supporting systems. This paper will describe these steps and how they were developed in support of a Comprehensive Aging Aircraft Strategy for the USAF.

Background

In 1998, the United States Air Force (USAF) established a funded program to begin addressing the issues peculiar to the aging aircraft in its inventory. The purpose of the program was to evaluate problems being experienced in the area of structures, and to develop technical solutions to those issues. The Aging Aircraft Office was established at Aeronautical Systems Center (ASC) at Wright-Patterson Air Force Base, Ohio to focus this effort.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Also in that year the Chief of Staff of the Air Force challenged ASC to address the growing problem of obsolete parts and diminishing manufacturing sources being experienced in the avionics of USAF aircraft. ASC reacted by establishing the Affordable Combat Avionics (ACA) initiative to address this problem. Because the viability of USAF avionics was the real goal of the initiative, in early 2001 it was renamed to the Viable Combat Avionics (VCA) initiative.

The Aging Aircraft Program's focus on structures and the inclusion of the VCA initiative were clear indications the USAF recognized a growing concern with its aging aircraft fleet, and the determination to concentrate on solving the aging issues. As additional aging systems issues were identified, it became apparent the small office, established as the Aging Aircraft Office, was not going to be up to the challenge.

On 25 January 2001, the USAF established the Aging Aircraft System Program Office (AA SPO) with the clear mission to extend the service life of its aging aircraft, and to significantly expand the scope of issues to be addressed.

Aging Aircraft System Program Office Focus

Develop Comprehensive Strategy and Plan to Extend Aircraft Service Life
- **Ensure Investments Systematically Address Aging Issues**
- **Drive Policy, Process Improvements**
- **Enable Optimal Fleet Sustainment**

Develop an Aging Fleet Management Framework that is:
- **Proactive**
- **Predictive**
- **Focused on Operational Capability**

Executing the Focus

First, we must understand and scope of the problem posed by the aging of air vehicles and the systems that support them. Naturally, a strategy for management and technology is developed accompanied by an implementation plan.

To carry out the strategy, an execution plan must be developed and implemented. To assure continuous updating of plans and strategies, methodologies for feedback and measurement of metrics must be put in place.

Finally, adjustments must be made to strategies and plans to reflect corrective actions necessary in response to metrics and feedback to assure we continue to lower the cost of ownership and increase the availability of the aircraft and their supporting systems.

Strategy Development

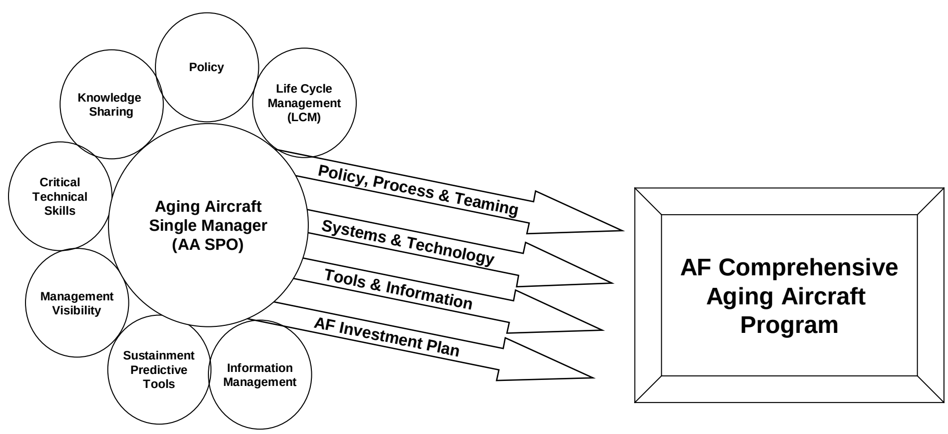
In 1998 the USAF commissioned a study to understand the issues that must be addressed as the aircraft fleet ages. The study group, called the Aging Aircraft Integrated Product Team (IPT), delivered eight recommendations.

1. Policy - Review/revise fleet management policy
2. Life Cycle Management - Reinvigorate mandatory weapon system Master Plans

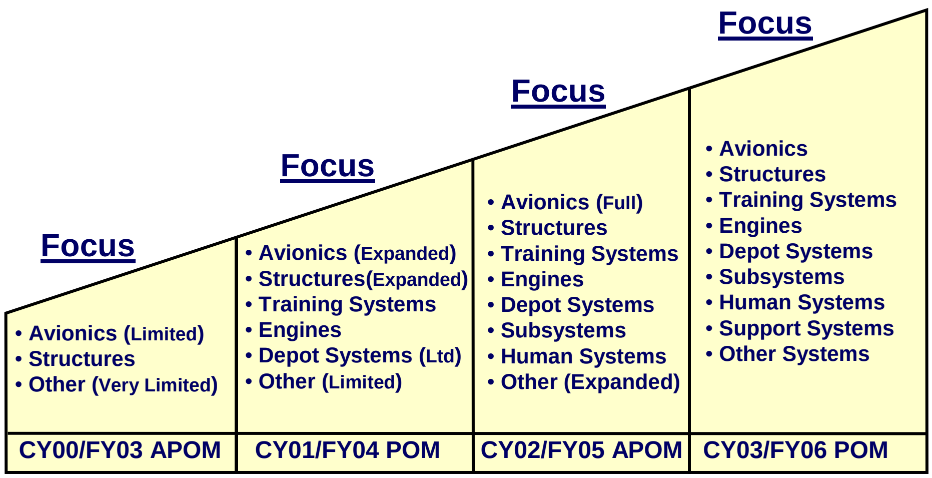
3. Information Management - Provide a knowledge management tool set
4. Sustainment Predictive Tools - Develop tools to quantify impact of aging aircraft, and task the scientific community to incorporate realistic sustainment factors in Simulation Based Acquisition
5. Management & Technical Skills - Re-baseline military and civilian career fields, and add Sustainment curriculum to current acquisition certification programs
6. Knowledge Sharing - Intensify level of shared information by creating an aging aircraft website, host an annual USAF Aging Aircraft Working Group, and disseminate all Aging Aircraft Meeting information and results
7. Management Visibility - Mandate inclusion of aging aircraft metrics in CSAF briefings for enduring, total system management visibility
8. Aging Aircraft Single Manager - Establish an Aging Aircraft Office as a Single Manager

The last recommendation was satisfied by establishing the AA SPO. This provided the USAF with the leadership necessary to tackle the other recommendations.

The AA SPO translated the eight recommendations into four thrust areas – Policy, Processes and Teaming; Systems and Technology; Tools and Information; and an AF Investment Plan. Concentrating on these thrust areas would lead to the USAF Comprehensive Aging Aircraft Program.



Because the AA SPO had an immense job in front of it but was emerging from being a small office, it needed to ramp up its efforts. The four thrust areas were converted to Ramp-up Plans. Because technical strategies for structures and avionics were the most mature, the Systems Ramp-up Plan is offered as an example.



The Systems Ramp-up Plan shows for each calendar year (CY) the subsystems that will be assessed for aging issues and a technical strategy developed to address those issues. As a result of developing the technical strategy, funding is sought to develop the technologies necessary to address the strategy. Each calendar year's strategy development areas drive budget inputs for each fiscal year (FY) program objective memorandum (POM). This connects the Systems thrust area back to the AF Investment Plan which is one of the thrust areas itself, thus providing the funding necessary to implement the Systems strategies.

Each thrust area has its own ramp-up plan and must be reflected in the each year's budget. With each thrust area reflected in the budget requests, the AF Comprehensive Plan has an full set of efforts to undertake and the funding needs identified for each.

Execution

The AA SPO has been tasked to look for cross-cutting issues and pursue common solutions for aging aircraft. In parallel most aircraft types have a weapon system manager, in the form of its own SPO. For instance, the KC-135 tanker aircraft has a SPO located at Tinker AFB, Oklahoma. It is the responsibility of that office to take care of the peculiar issues experienced on the KC-135 fleet. Some issues that the KC-135 SPO will encounter, such as corrosion, may be experienced by several aircraft types. Rather than ask the KC-135 SPO to solve all issues related to corrosion, it makes some sense to have a central office develop solutions developed that can be used on many aircraft types, to include the KC-135. That is the role of the AA SPO, to concentrate on shared problems and develop solutions that can be used on multiple type aircraft. These cross-cutting solutions can then be tailored for use on a particular type aircraft by their responsible SPO, such as the KC-135 SPO in the case of the KC-135 fleet.

The AA SPO has its own dedicated funding to address cross-cutting aging aircraft issues. Since 1998 the majority of that funding has been focused on aging structures issues. In particular, much of the AA SPO current funding is concentrated on four cross-cutting initiatives: Corrosion Effects on Structural Integrity, Advanced Non-Destructive Evaluation for Aging Structures, Bonded Repairs Enhanced Capabilities, and Advanced Aircraft Corrosion Protection.

These initiatives form the foundation of executing the USAF Aging Aircraft Program, but many other less focused efforts are underway, as well. At the writing of this paper, there are 24 active projects in the Aging Aircraft Program. For each dollar spent by the AA SPO on these type projects, the USAF avoids spending 27 dollars in its warfighting units to support their 6300 aging aircraft. Or the dollars saved could go toward modernizing the fleet to slow the overall aging trends.

It is obvious the USAF is not the only agency suffering from the issues of aging aircraft. Just as each individual SPO doesn't have the resources (funding, people and infrastructure) to solve all the aging aircraft issues, neither does any one agency. The AA SPO is attempting to develop collaborative efforts with several of those agencies affected by aging aircraft. While there is much more to do, some progress is being made.

Within the US Department of Defense, several agencies are suffering through the emergence of aging aircraft – the USAF, the US Navy, the US Army and the Defense Logistics Agency (DLA). ***Collectively these agencies have begun a DLA/Multi-Service Technical Cooperation Group with the initial focus of identifying technical areas to direct collaborative efforts.*** The initial thrust areas were corrosion, NDE and wiring. One result of this group has been the establishment of an annual Aging Wiring Strategy Working Group. Under the leadership of the AA SPO, the intent of the AWSWP is to identify problems with aging wiring and develop collaborative efforts that can collectively address those problems.

With the Air Force Research Laboratory (AFRL) providing the leadership, the USAF has established a Memorandum of Agreement (MOA) with the National Aeronautical and Space Administration (NASA) that targets four "common interest areas" to direct collaborative aging aircraft efforts. Those common interest areas are corrosion, NDE, aging wiring, and integrated vehicle health management. This MOA establishes specific projects that NASA and the USAF will fund which target problems each agency must solve on aging aircraft. By working together and coordinating efforts, the meager dollars each has to solve these problems is leveraged.

The AA SPO along with the North Atlantic Treaty Organization (NATO) and several aircraft SPOs (E-3, E-6 and E-8 SPOs) have decided to work together to address aging issues being experience on weapon systems that are based on the commercial Boeing 707 platform. As a sponsor of the B-707 Users' Group, the AA SPO hopes to help identify and solve specific technical issues shared by these aircraft, all of which are Low Density, High Demand aircraft supporting the US Navy and USAF, and several NATO and other foreign allies.

The potential exists for many other alliances and collaborative efforts. The largest challenges to establishing these alliances will be having the visibility of all the initiatives that could be focused into collaborative efforts, and the lack of willingness in some instances to eliminate barriers to cooperation.

Adjusting Strategies and Plans

As with any strategy or plan, it is appropriate to periodically reevaluate the direction the strategy or plan. The USAF Comprehensive Strategy and Plan is no exception. There are several reasons for reevaluation – results from those initiatives, changes in funding for identified initiatives, surprise problems, changed aircraft missions, etc.

Considering that the AA SPO has only been in existence since January 2001, and that the strategies have yet to provide substantive improvement in aging aircraft issues, it is a bit early to expect too much to be done to change the strategies. The exception is in the area of structures.

The USAF has had a Structures Strategy sionce 1997 when the National Research Council (NRC) delivered its study on Aging Aircraft. The USAF used the NRC report to establish a focus for technologies to address problems with aging structures. As a result of the study, the USAF also established the Aging Aircraft Technology Team (AATT). During the ensuing years, the AATT conducted Air Force-wide surveys to determine the health of USAF aircraft structures, evaluate the improvements expected from the structures technology thrusts the USAF had instituted via the Structures Strategy, and consider additional areas requiring focused research and development. In December 2000, the AATT concluded its third round of annual surveys and was prepared to adjust the USAF Structures Strategy. Under the direction of Dr. John "Jack" Lincoln, the USAF made modifications to the Structures Strategy. This change validated the need to continue some of the technology pursuits already underway. It also identified additional research and development necessary to proactively address emerging aging issues. Subsequently, the AA SPO adjusted its budget requests to reflect the new Structures Strategy.

The USAF structures community, effectively monitored by the AATT, has provided the feedback necessary to adjust the strategy it executes, which demonstrates in this one area of aging aircraft how the full cycle of Strategy-Execute-Adjust can be instituted in the entire aging aircraft community.

Conclusion

As the average age of the people of the industrialized nations of the world increases, peculiar ailments and conditions caused by age are becoming more prevalent. To improve the quality of life and productivity of the aged, the health and medical community is reacting by addressing those ailments and conditions with new and innovative technologies and care methodologies.

The USAF recognizes this same phenomenon in its aging aircraft fleet. It has bestowed the leadership of this effort on the Aging Aircraft System Program Office. The AA SPO has reacted by charting the path needed to tackle the challenge. It has established a strategy for managing the problem, and plans to target specific problem areas. Building upon the successes of the structural community, the AA SPO is expanding and institutionalizing in other disciplines, such as avionics, wiring, fuel tanks, depot repair, the processes and procedures already clearly understood within the structures community.

The existence of the AA SPO is a clear indication that the business practices and management philosophies which must be addressed as a result of aging aircraft are at least as much of a challenge as the technical issues. The journey to improve our technologies, business processes and management philosophies has only just begun.

USAF Viable Combat Avionics Initiative

Colonel Michael "Carp" Carpenter
Chief, Aging Aircraft Planning Division
Aging Aircraft System Program Office
Wright-Patterson Air Force Base, Ohio, USA
E-Mail: michael.carpenter@wpafb.af.mil

In the interests of readability and understandability, it is RTO policy to publish PowerPoint presentations only when accompanied by supporting text. There are instances however, when the provision of such supporting text is not possible hence at the time of publishing, no accompanying text was available for the following PowerPoint presentation.

Click here to view PowerPoint presentation; Press Esc to exit

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

This page has been deliberately left blank

Page intentionnellement blanche

USAF Strategy for Aging Aircraft Structures Research and Development

Frank M. Grimsley[1], John W. Lincoln[1], and Michael L. Zeigler[2]
Aeronautical Systems Center[1] and Air Force Research Laboratory[2]
2530 Loop Road West
Wright-Patterson Air Force Base, Ohio 45433-7101
USA

SUMMARY

Many nations are now keeping aircraft in their inventories longer than ever before. In many cases, aircraft are left in the inventory longer because they are still operationally effective; however, in most cases, they remain in the inventory because the money is not available to replace them. Aircraft, which are seeing the effects of aging through corrosion and fatigue cracking, are causing their operators to bear a significant economic burden to keep them operational with the potential for degradation of flight safety of aging aircraft if they are not maintained properly.

The United States Air Force (USAF) has maintained safety of their aircraft for the last thirty years through the application of damage tolerance principles to determine inspection intervals. This approach has on occasion been modified because of the onset of widespread fatigue damage (WFD) or the loss of material because of corrosion. In the case of WFD, the USAF has developed a modification program to alleviate the problem. In the event of corrosion damage, both modification and reduced inspection intervals have been used.

The USAF has developed a strategy for the sustainment of their aircraft starting with the identification of user needs requiring research and development efforts. The strategy is based on identifying research and development opportunities that will have a favorable return on the investment through cost savings or cost avoidance and increased aircraft availability. This has presented problems since it is difficult to determine the cost of maintaining aircraft in enough detail to determine the return on the investment accurately. To date, identified activities include improvements in nondestructive inspection capability, corrosion tracking and prevention techniques, and advances in repair of metallic structures through composite patching. In addition, improved materials for substitution and environmentally compliant coatings have been identified. The purpose of this paper is to provide a discussion of the aging concerns found in the structure of USAF aircraft and the approach the USAF is pursuing to alleviate these concerns.

BACKGROUND

The dawn of the aging aircraft program for the USAF began on 13 March 1958 with the structural fatigue failure of the wing of the B-47 aircraft [1]. Those events led directly to the initiation of the USAF Aircraft Structural Integrity Program (ASIP) [2]. The ASIP defines all of the structurally related activities on an aircraft from initial development until retirement; therefore, it can be considered an aging aircraft program. This program was significantly changed as a result of the failure of an F-111 on 22 December 1969,which ushered in the era of damage tolerance in the USAF [3], changing the technology basis of the program from fatigue to fracture. This change in approach prompted considerable research and development in area of fracture mechanics. In addition, since the damage tolerance approach forced the designer to better understand the stresses in the structure, emphasis was placed on the emerging finite element analysis methods.

In the early nineties, the then Wright Laboratory recognized the growing need for further research and development for aging aircraft and on 28 April 1993, they initiated the Aging Aircraft Structures Steering Group. This activity was designed to work hand-in-hand with both the Federal

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Aviation Administration (FAA) and the National Aeronautics and Space Administration (NASA). Both the FAA and NASA had ongoing aging aircraft programs as a result of the Boeing 737 failure on 28 April 1988 [4]. The Wright Laboratory activity identified many research and development activities, but the lack of sufficient funding prevented some of these initiatives from reaching fruition.

The climate changed on 1 June 1995 when the commander of the Air Mobility Command (AMC), growing concerned about the future of his aging aircraft fleet, initiated an Aging Aircraft Process Action Team (PAT) to identify actions needed to alleviate the impending crisis. Because of this action, on 26 February 1996 the commander of the Air Force Materiel Command (AFMC) established the Aging Aircraft Program Office in the Aeronautical Systems Center (ASC) to facilitate the transition of technologies from the laboratory to the USAF Air Logistics Centers.

On 28 June 1996 the Air Force Research Laboratory (AFRL) commander, because of concerns about the direction of his aging aircraft research and development program and potential for duplication across other agencies, initiated a National Research Council (NRC) study on this subject. The NRC report [5] was released in September of 1997and identified 49 research and development activities along with recommendations for some organizational changes to facilitate these recommendations. On 31 March 1998 the AFMC commander approved the formation of the Aging Aircraft Technologies Team (AATT) in response to the NRC report. The AATT consists of representatives from AFRL, ASC Engineering Directorate and the Aging Aircraft Program Office and its purpose is to orchestrate the Aging Aircraft Structures activities from research and development through transition into implementation.

Finally, on 25 January 2001, the Aging Aircraft SPO was formed out of the Aging Aircraft Program Office to further increase the awareness and importance of solving these aging aircraft issues within the USAF.

The problems with aging aircraft are not new and for many years have had a significant influence on the USAF research and development programs and have been a major driving influence on the elements of the USAF Aircraft Structural Integrity Program. The forty-year history of the development of this program provides insight on the how the USAF reacted to the problems with aging aircraft.

STRATEGY FOR AGING AIRCRAFT

The damage tolerance approach [6] has led to a greatly improved understanding of aircraft structures and their performance. It, when properly applied, will essentially eliminate fatigue cracking as a threat to structural integrity. Therefore, damage tolerance should be the foundation on which the structural maintenance program should rest. During the 1970's and 1980's, the USAF performed an assessment on every major weapon system using the damage tolerance approach to develop appropriate inspection/modification programs to maintain operational safety [7]. As a result, the USAF has maintained an excellent structural failure safety record despite the ever increasing age of its' fleets.

As the aircraft grow older, the potential for fatigue cracking and corrosion increases. Many of the aging aircraft in the USAF inventory are experiencing increased maintenance costs because one or more of these problems are present [8]. To determine the research and development actions that could be pursued relating to these problems, the USAF formed the AATT. At the outset, the AATT decided they would operate under the following guiding principles:

- Research and development:
  - ♦ Must be directed towards needs of USAF aircraft
  - ♦ Must be oriented towards flight safety, maintenance cost reduction, and/or enhanced availability

- ♦ Must be output-oriented and cost-focused
  - ➢ Researcher and customer must communicate on expectations from research
  - ➢ Researcher must be able to define cost and schedule for activity
- Develop technology that can be transitioned
- Augment highest level of capability in industry or government
- The USAF laboratories must maintain organic competencies in key areas related to aging aircraft

The most difficult challenge for the AATT was to determine the return on the research and development investment for aircraft. Although the USAF has usable cost information on total cost for depot or field operations, they do not know the cost in sufficient detail to judge whether an effort on a certain component of the aircraft is justified. For example, a wing section may be subject to corrosion, but without cost data on this component, it is difficult to determine if the USAF should replace this component using a more corrosion resistant material or keep repairing the corrosion damage.

The strategy for identifying research and development programs to reduce maintenance costs can be described by six steps. These steps are

- Conduct surveys to determine problems
- Identify and prioritize solutions requiring research and development
- Establish research and development roadmaps
- Obtain management and customer approval
- Execute research and development efforts
- Transition technology to the operator

The first three steps are designed to identify the problems and develop a plan for their solution. The final three steps engage management to implement the plans and carry the research and development through to technology transition.

The first step is to conduct surveys on aging aircraft in the inventory to identify their problems in as much detail as possible. This effort includes interviews with engineers and maintenance personnel that are directly responsible for the continuing integrity of the aircraft. In addition, the AATT discussed the aging aircraft issues with the operators of the aircraft. Identification of problem areas generally requires multiple meetings with these individuals in order to get a complete understanding of the problems that may have a solution through research and development. For example, for the USAF, surveys by the AATT of approximately thirty aircraft have taken place annually four times. In most cases, the reviews required approximately one day to complete. In some cases, however, there was interest in the review by the operator because of the potential for significant funds to be expended by them for modernization of the aircraft. In these cases, the reviews required two to three days to complete. The surveys, whenever possible included laboratory personnel so that they could see the nature of the problems for themselves. In addition, the surveys included the original equipment manufacturer (OEM) since their knowledge of the aircraft is essential for a clear definition of the problem. The relationship between the USAF and the OEM for maintaining the integrity of aging aircraft has been outstanding.

The second step is to determine the potential solutions to the aging problems. The AATT makes an initial screening of the problems and makes a preliminary determination of those problems that may have potential for a research and development solution. These problems are then given the widest dissemination possible to solicit possible solutions. The potential solutions are then categorized as basic research, exploratory development, and advanced development. The last category is for technology that is ready for transition. Usually, solutions categorized as basic research require development times that are extremely long for the technology to reach maturity. However, this area cannot be overlooked since it may, in time, have significant return on the investment.

The third step is to develop “roadmaps’ for the maturation of the technology for use in the aircraft. The roadmap identifies the problem, the technology to be used to solve the problem, the tasks to be performed for the solution, the schedule for completion of each of the tasks and the funding required for completion of each task. This step requires close adherence to the AATT guiding principles.

The fourth step, which is management approval, is likely the most important in that the success of the entire program rests on the agreement of the technologists and managers that the research and development program has the greatest return on the investment. This, of course, means that the manager has an understanding of the scope of the effort required to reach the desired goals. With this step the process changes from bottom up to top down. The first three steps could be identified as “bottoms up” activity. These steps started with identification of the problems and ended with a strategy for solution.

The fifth step is execution. The strategy is not complete, however, without the implementation step. This step is “top down” in that management charges the researchers to perform to the aging aircraft roadmaps. Essential to this step is the acceptance of the technology by the logistics managers and the operators. They must demonstrate willingness to implement the technology developed to reduce their maintenance burden.

The final or sixth step is technology transition. Actually, technology transition starts with the second step since there is no solution unless the technology can be transitioned to the logistics centers. Another name for this effort that is at times more descriptive is “industrialization of the process.” The first requirement for technology transition is adequate funding. The second requirement is that logistics personnel understand and are trained in the execution of the process. The final requirement is that logistics personnel are convinced that the new technology will enable them to do the job better than they are doing with existing technology.

AGING AIRCRAFT ISSUES

Funding

Funding for aging aircraft research and development activities is likely the number one problem faced by the manager. Funding of aging aircraft requirements is usually inadequate due to ever-increasing structural modification programs, safety issues, sustaining engineering needs, and responding to ever changing retirement dates. It is difficult, if not impossible; to support all identified technology areas. Consequently, there is a need to establish priorities.

In addition, justification is typically difficult for non-safety related problems since the available funding is usually consumed by safety related problems affecting the force. Care should be taken in the maintenance of non-safety related problems in that they may become safety issues through improper maintenance. An example of this is the use of inferior bonding techniques to repair honeycomb structures. Improper techniques can lead to moisture intrusion and an accelerated degradation that has the potential for loss of integrity of the component. Future funding requirements need a better understanding of return on the investment (ROI) where the ROI includes cost and availability in order to compete competitively with all the other identified requirements. Improved cost data collection procedures are needed in order to accomplish this

In many cases, the budgets have not allowed the modernization of maintenance facilities or the upgrading of their information management systems. This has led to maintenance practices that are not state-of-the-art in that the use of information management has not become ingrained in the work force. This inadequacy is compounded by inaccurate and often-inadequate maintenance databases that lead to a misunderstanding of logistics requirements that raises costs and reduces aircraft availability. Retention of maintenance records for structural repair needs to be made a priority.

Fleet/ Depot Planning/Procedures

Another issue is the lack of support from operators for tail number tracking used to determine damage and sources of damage. Too often, since the recorder for flight loads is not flight essential equipment, the operator fails to adequately download data or service the recorder. This has led to many cases where the operators were not aware that some of their practices were causing damage to aircraft that could be have been avoided. The logistics community needs to communicate with operators to find operational techniques to reduce damage.

There is a concern that inadequate manning levels in both the field and at depots are causing lack of compliance with technical orders. In addition, the experience level of maintenance personnel has been steadily reducing for both the civilian and military population due to workforce downsizing. These problems are compounded by diminishing engineering resources in the manufacturing base result in increases in flow days and costs since the parts must often be fabricated through reverse engineering.

There are also many depot maintenance procedures/planning issues that cause increased costs. The maintenance planners should look at the frequency of depot maintenance visits, especially for aircraft experiencing moderate to high levels of corrosion. In some cases, the depot intervals are so long that many problems are discovered too late, resulting in more expensive and complex repairs that could have been caught earlier and remedied much easier with more depot visits. To make matters worse, many times within the same depot, there will be inconsistent maintenance practices being used from one product line to the next.

Finally, although experience shows that there are many surprises found in the maintenance of aging aircraft, there is usually a lack of planning and budgeting for these events. This leads to increased costs and a further lack of availability.

One of the most important aspects of an aging aircraft program is the quantification of the economic burden of systems in future years. This activity is necessary to support planning for retirement of existing aircraft and procurement new aircraft in the future. Without this information, the visibility is lacking to make sound judgements for aging aircraft. The process for doing this is well established for fatigue cracking. Unfortunately, for corrosion it is not as well understood.

FATIGUE CRACKING AND CORROSION

The USAF identified that the root causes for most aging related structural issues are fatigue cracking and corrosion. For each of these causes, the solutions could be obtained in one or more of the categories: nondestructive inspection (NDI), repair, modification, prevention, analysis, health, or information technology. Usually, the ultimate solution will be a combination of these categories. The discussion below covers the main efforts for both fatigue cracking and corrosion.

FATIGUE CRACKING

The introduction of damage tolerance principles by the USAF in their structural inspection program in the early seventies virtually eliminated fatigue as a safety problem in their aircraft. However, fatigue cracking of operational aircraft in the USAF is still a significant economic problem. The USAF estimates that this problem cost approximately $250 million in 1997. The USAF attributes much of this burden to operational usage being more severe than the usage assumed for design. This occurs because as the aircraft is fielded, the operators find unique and unanticipated ways to take full advantage of the capabilities of the aircraft. This often results in more severe usage due to weight growth for new capabilities or new operational mission profiles.

Based on design processes used today, fatigue cracking in an airframe should not be a significant factor for an aircraft whose operationally usage is approximately the same as its design usage. However, many of the older aging aircraft in use today were designed at a time when the effects of repeated loading was not a design consideration. Consequently, fatigue cracking is surely an economic problem and in most cases is a potential safety problem. Fatigue cracking found by inspections based on damage tolerance principles had resulted in many repairs on operational aircraft. In many cases, the cracks are repaired when found. The USAF has found that historically this is the most economical approach and consequently, this approach is most often used until it becomes evident that the structure needs to be modified.

The certification basis on which the structure was qualified also plays a critical role in the research and development for aging aircraft. The USAF guidance for structural certification [9] includes both slow crack growth and fail-safe structures. Although, the slow crack growth approach is most often used today, the USAF strongly advocates the use of fail-safe designs whenever practical. Designs that are fail-safe can tolerate the failure of a structural member and still maintain adequate residual strength until the failed member is discovered through inspections.

Widespread Fatigue Damage

The certification basis for many aircraft is fail-safety because it provides a good overall approach to achieve both safe and economic operation. However, when the structure develops WFD, it can cause a loss of the fail-safe capability in the airframe and drastic action is needed to restore it. Perhaps, the most famous incident of WFD is the 1988 operational failure of a Boeing 737. This event provided the motivation for the considerable emphasis by the FAA on the structural issues associated with aging aircraft. This event occurred on Boeing 737 (N73711) on 28 April 1988. On this date, cracks in the fuselage lap-splices coalesced resulting in loss of the upper fuselage from just aft of the pilot’s cockpit to the wing leading edge. For the FAA, the aging aircraft program started on that day.

The Aloha incident was not the only operational failure caused by WFD. Another one in Japan was much more serious, but did not receive the notoriety in the United States that the dramatic Aloha incident received. WFD resulting from a faulty repair of the aft pressure bulkhead of a Japan Air Lines (JAL) 747 caused the failure of JAL Flight 123 from Tokyo to Osaka. The total casualties resulting from this accident was 520.

The effect of WFD on flight safety has long been a concern of many researchers. Most of the older USAF aircraft designs did not comply with the modern guidance for damage tolerance assessment (DTA) [9]. Consequently, there is a potential for the crack population to be so large in the structure that the application of the deterministic damage tolerance process may not protect safety. Large crack populations could also exist in monolithic structures such as the T-38 aircraft, which the USAF analyzed using probabilistic methods [10]. The USAF refers to cracking as found in the T-38 aircraft as generalized cracking rather than WFD. The occurrence of WFD can significantly degrade the fail-safety of the structure. This problem has been evident on the KC-135, C-5A, C-141 [11] and the E-8 aircraft. The USAF subjected these aircraft to teardown inspections. They incorporated the results of these inspections in a risk assessment to quantify the time when the probability of failure, conditioned by the fact there had been discrete source damage, becomes unacceptable.

Probabilistic Methods

For safety of flight structure, many of the problems, such as WFD need to be terminated by modification of the affected structure. There are other cases; however, where safety is not a concern, but there are significant economic implications that need considerable attention aimed at finding solutions for them. In some situations, cracking may occur at some point in the life of an aircraft where it may not be readily apparent whether a repair or a modification is the answer to a problem in fatigue cracking. In these cases, a probabilistic assessment may be useful to determine whether it is more economical to repair the aircraft when cracks are found or to perform a modification for the entire population at a certain time in their life [12].

One approach for accomplishing this is to determine an estimate of the expected number of future repairs from existing evidence of repairs made on the population. This requires that a probabilistic analysis be made to determine the distribution of fatigue cracking in the fleet. The essential element for performing this type of analysis is an accurate database of the cracking incidents. With this information, the analysts can use the powerful tools derived for the well-known distributions such as the Weibull and the Lognormal. With information derived from these analyses, the operator is able to make an informed decision on whether to keep repairing the structure or to make a modification to correct the problem.

Nondestructive Testing

The need for nondestructive inspection technology to enable the DTA driven inspections has been a major thrust of the USAF for many years. However, the AATT found inadequate NDI methods was a major concern with the ALC engineers. They stated that existing methods were too slow and sometimes unreliable. It was difficult for them to find cracks in fastener holes in multiple layered structure. In addition, the implementation times are too long for new techniques. There are two reasons for this. The first is that that the budgeting process has built into it long delays in getting funding for new technology. The second is that the problem with transitioning the technology. The approach for transitioning newly discovered methods it is not well understood and consequently many techniques that hold promise are never used. There is also a need for more efficient and reliable methods for depot and field level inspections. For many of the inspections, there is a lack of established limits for damage so that it may be repaired and replaced efficiently and economically.

For all aircraft, there is a need to develop crack detection in second layer for cracks approximately 1.25 millimeters in aluminum, titanium, and steel. In some cases, where there is a faying surface seal between the structural layers this capability could be obtained with ultrasonic methods. For other structures, this is much more difficult task, although low frequency eddy has enjoyed some success.

For fail-safe aircraft, there is a need to develop crack detection of 0.75 millimeters in fastener holes in fuselage splices that could degrade fail-safe capability of the structure. Because of the safety implications, it is important to be able to detect cracks that could be significant for determination of the onset of WFD. There is a need to make an estimate of this onset based on probabilistic assessment of cracking data derived from the teardown inspection of fatigue test articles or operational aircraft. It must be recognized, however, that this is only an estimate. The actual time may be either somewhat earlier or later than this estimate. It is important, therefore to be able to validate this prediction with nondestructive evaluation. This task is made difficult by the fact that the size of defect to be found is quite small. The experimental evidence to date indicates that cracks of the order of two millimeters can significantly lower the fail safety capability of certain structural configurations.

The reliability problems with some of the NDI techniques make it desirable to make the structure safe in the event of rapid fracture in a component. Fail-safe structure greatly simplifies the NDI problem in an aging aircraft. Consequently, there is a need to develop methods for enhancing fail-safety of all structures with safety of flight implications. One of the more promising approaches is the use of bonded composite straps.

Repairs/ Material Substitution

In the past, repairs placed on aircraft have been designed based only on static strength considerations only. One reason for this is there is no need to know the loads on the aircraft. However, on aging aircraft, the repairs to flight safety critical structure need to be assessed for their damage tolerance capability. This means that the stresses in the structure must be known. This knowledge is typically only known to the OEM; however, in many cases, the USAF has worked with the OEM to use their knowledge of the loads without violating their proprietary rights. In other cases, the USAF has contracted for the development of the loads by a source independent of the OEM. For new aircraft, the USAF has funded the effort for the OEM to design the standard repairs to be damage tolerant. This effort was a logical activity subsequent to the DTAs that were made for the intact structure.

There are numerous applications currently of composite repairs in the USAF, in addition to the applications in Australia and Canada and the United Kingdom. The USAF applications include the C-130, C-141, KC-135, and the B-1. The success of these applications has motivated the USAF to spend the resources to further exploit this technology. The procedures have been established for successfully preparing the surface for the bonding operation. Further research is needed to include other targets for composite repair, such as thick structures that are inherent in many bulkhead designs.

For composite repairs of metallic aircraft on safety of flight locations, it is essential to be able to determine the bond-line integrity of composite patches. Because of the limitations of NDI to determine the strength of the bond, the use of “smart patches” appears to be a viable alternative.

Metallic repairs to metallic structures will remain important for the aging aircraft problem. It is not uncommon to find hundreds of metallic repairs on a single aging aircraft. Many of them are on safety of flight structure. There is a need to develop the guidance for these repairs to ensure that they are damage tolerant and do not degrade the fail-safe capability of the structure.

Material substitution is an attractive approach for aging aircraft since it eliminates the mistakes in material selection made before the threats to their structural integrity were understood. Guidelines and criteria are needed for material substitution to enable this to be accomplished. These substitutions may include alternate product forms for obsolete forgings and extrusions.

The use of interference fit pins in fastener holes or fastener hole cold expansions are important for extending the life of aging aircraft. There is a need to determine stress intensities for interference fit and cold expanded hole locations so that fracture mechanics calculations are possible for these applications.

There are some problems associated with the dynamic response in aging aircraft from buffeting due to flow separation. In most cases, the technology to predict these problems is not well developed and likely will not be developed in the near term. Therefore, the researcher must rely on flight test data. Since these problems were not adequately solved in the design development phase, they become problems that usually never find a solution. The use of passive damping techniques has been successfully used and needs to be further evaluated for control of these problems. In the case of the F-15 vertical tail response to buffet, a partial solution was found by the use of composite patching to the surface of the vertical tail. This solution required that the dynamic response of the tail be extensively modeled.

CORROSION

Corrosion and fatigue separately have both led to serious safety as well as economic problems. Corrosion alone, in forms such as uniform corrosion (thinning) or exfoliation, may reduce the strength of aircraft and lead to failure. Both of these forms of corrosion may lead also to expensive component repair or replacement. There are many cases where corrosion alone is not significant from a safety consideration, but is a very significant economic problem. In the case of corrosion alone, one must judge the seriousness of this problem on an individual basis. Nondestructive inspections have found fatigue problems where there is essentially no influence from corrosion. Researchers have documented many cases over the years where the consequences were catastrophic. The results of fatigue cracking have caused many expensive repairs and modifications to the structure including component replacement. Fatigue often combines synergistically with corrosion. In these cases, the term corrosion fatigue is more appropriate. In most cases, corrosion, fatigue, or corrosion fatigue becomes a safety consideration only when either maintenance is not performed properly or the maintenance program is inappropriate. Experience derived from diligent maintenance has repeatedly shown that the operator need not compromise safety resulting from these problems. The purpose of this section is to describe some experiences with corrosion, fatigue, and corrosion and fatigue and to review some of the relative literature on this subject.

Economic Impact of Corrosion

There is considerable evidence that corrosion is a major economic problem with both military and commercial aircraft. The USAF sponsored a contract in 1997 to determine the cost of corrosion in USAF aircraft and found in this study as in a previous study performed in 1990 that the cost of corrosion prevention and repair is significant. They found that the total cost from corrosion in 1997 was approximately $795 million dollars. This was an increase of 4% over the 1990 costs although the USAF reduced the force structure by 28%. The C-5, KC-135, and the C-141 account for 50% of all direct corrosion maintenance costs. Of the $795 million spent, painting the aircraft cost $136 million. The USAF spent approximately $425 million of the $795 million specifically on corrosion repairs. The survey highlighted the A-10 as a success story in defeating a serious corrosion problem.

Many of the older military aircraft that are currently operating were constructed with corrosion prone materials and essentially no corrosion protection. One aircraft in this category is the KC-135 that initially used 7178-T6 for the lower wing skins. This material was selected rather than the 2024-T3 material used in the Boeing 707 to provide structural efficiency. That material exhibited low fracture toughness, poor crack growth rates, susceptibility to corrosion, and low resistance to stress corrosion cracking. To make matters worse, the USAF elected to save money by omitting corrosion protection of the material by faying surface sealing or wet-installation of fasteners. The result was that the USAF had to replace the lower wing skins at 8,500 flight hours [13]. Corrosion fatigue likely played an important role in the early cracking of this structure.

This same material, 7178-T6, is used in the upper wing skins of the KC-135. It is now causing a significant problem in that there is considerable exfoliation corrosion around the fastener holes. A process that may be able to find these problems is called search peening. In the performance of this process, maintenance personnel shotpeen the upper wing skins with glass beads that cause corroded areas to reveal themselves through local deformation around fastener holes. In some cases, the exfoliation is severe enough to cause panel replacement. When the Oklahoma Air Logistics Center (OC-ALC) replaces upper wing panels, they select a replacement material that is much more resistant to corrosion than the original material. The troublesome part of this problem is that the USAF does not have a solution that would preclude further replacements in the future. The use of peening; however, for new structure does appear to extend their lives. There is evidence of beneficial effects of shotpeening from the B-52H upper wing. These wings, which had their upper surface shotpeened at the time of fabrication, are showing only minor corrosion damage although they have been in service for many years. Another technology that appears to have application to corrosion inhibition is laser peening.

The AGARD Corrosion Handbook [14] discusses the problems found on these aircraft and many others through case studies. This document places the cost of corrosion in the United States alone in 1978 at $70 billion overall. The USAF can account for approximately one of those billions for their airplanes. The case studies in this handbook show the results of corrosion that remained undiscovered by maintenance personnel until it was a significant economic or safety problem.

These problems include both military and commercial aircraft. As evidenced by the currently documented cost of the major maintenance visits by large category transport aircraft, the cost of corrosion is a major factor in commercial maintenance budgets. However, the use of corrosion prevention compounds in commercial aircraft has significantly reduced the burden. It is encouraging that the airframe manufacturers are doing a much better job of applying corrosion protection during fabrication. One would anticipate that these improvements would relieve some of the maintenance cost burden when the operators bring these aircraft into the inventories.

Potential Safety Impacts of Corrosion

The fail-safe design augmented by DTA derived inspections is largely responsible for preventing the combined effects of corrosion and fatigue from becoming a major problem on large category transport aircraft. For the USAF, the inspection program derived from DTAs is certainly the primary means of maintaining safety from corrosion fatigue. The USAF also uses another inspection process called the Analytical Condition Inspection (ACI) to augment the DTA derived inspections. The USAF initiated the ACI program many years ago to help the maintenance personnel discover distressed areas of the aircraft not identified by the design analyses or testing. For this process, the USAF selects a small sample of aircraft from the inventory and inspects the entire aircraft thoroughly. To gain better access to concealed areas they remove fasteners and panels to interrogate the structure. This is done as completely as possible nondestructively to ensure that the structure is not experiencing corrosion or fatigue cracking that could jeopardize continued flight safety or economic operation. When they find an area that does not correlate with design experience, they make appropriate changes to the Force Structural Maintenance Plan (FSMP) [15] which is an integral part of the ASIP. The FSMP tells the maintenance personnel how, when and where to inspect the structure to maintain its safe operation.

A paper written in 1997 describes the forms of corrosion that could compromise flight safety. The authors of this paper list pitting corrosion, intergranular corrosion, exfoliation corrosion, stress corrosion cracking, corrosion fatigue and uniform corrosion as safety issues. In this document, they suggest a number of research and development activities that should enhance the state of knowledge about the effects of corrosion. They suggest many activities including teardown inspections of ex-service aircraft and scheduling of maintenance on a time basis as well as a usage basis.

The USAF Aircraft Structural Integrity Program makes an assumption about the effects of some of the forms of corrosion listed above. It presupposes that corrosion damage will not raise the stress sufficiently to change the inspection program to ensure flight safety. This infers that the maintenance program is able to control corrosion damage through inspections and preventative measures. Experience has shown that this assumption has not led to serious consequences. The USAF maintenance program is outstanding. One reason for its success is the advice given to them by the resident ASIP Managers and other engineers at the Air Logistics Centers. The USAF cannot attribute any catastrophic failure since the early seventies to corrosion or corrosion related phenomena. There have been; however, numerous local failures from corrosion, particularly stress corrosion cracking. This has been a chronic problem in many landing gear systems and in large bulkhead forgings.

Even though the USAF maintenance program is outstanding, many corrosion findings are surprises. Corrosion is difficult to predict, especially stress corrosion cracking. It is difficult to quantify the impact of corrosion on a specific aircraft because of maintenance variability within the population. The uncertainty in corrosion predictions appears to be large. Laboratory tests are difficult to correlate with operational experience. In addition to a lack of corrosion tracking and modeling, the corrosion database is immature

The applications of the 1975 version of ASIP addressed corrosion fatigue by modifying the crack growth rates in an attempt to account for the corrosion environment such as moisture, salt or sump tank water as appropriate. The validity of these corrections is subject to question because of such factors as loading frequency, temperature and the use of crack growth rate data from constant amplitude testing. However, the USAF has found few errors in the DTAs that they can directly attribute to the manner in which they accounted for the effects of the environment. Most often, the increase in crack growth rates in critical locations in the structure is associated with operational usage spectra that are more severe than the design spectrum of loading. The USAF accounts for these differences through individual aircraft tracking for loads.

Despite the USAF’s excellent track record, there are serious concerns about the impact of corrosion on structural integrity. The first and most obvious is the effect of lapses in proper maintenance that have led to significant loss of structural strength. The structure must be maintained in such a manner as to maintain the static margins at no less than zero. Thinning of the structure may increase the stress in the structure such that the structure inspection intervals required for flight safety will need to be decreased.

There is limited nondestructive inspection capability to accurately quantify the amount of material loss due to corrosion. Technology is needed to detect corrosion thinning at a level of three percent in multiple layers. A much easier task is to interrogate the structure for the existence of corrosion. Consequently, at this time, the safest approach is to remove corrosion indications when found. As the capability for nondestructive inspection is improved, then the opportunity for corrosion management is improved. Further, it is essential that those areas that are damage tolerance critical be an integral part of the corrosion detection program. In addition, those areas that are damage tolerance critical and the NDI demonstrates that the stresses have increased because of thinning of the structure, then suitable changes must be made to the FSMP.

The NDI methods for detection of hidden corrosion in the form of pitting, intergranular, exfoliation, and stress corrosion cracking are too slow and sometimes unreliable. As for NDI for fatigue cracking, the implementation times too long. There is a need for more efficient and reliable methods for depot and field level inspections since searching for corrosion is a major cost factor. There is a need for a NDI capability to reliably find corrosion pits with cross-sectional area of 0.04 square millimeters and pitting depth of approximately 0.125 millimeters.

Hidden corrosion is also a major concern for continuing structural integrity. It is essential that the maintenance personnel have a clear understanding where they may find corrosion in the structure so that they may use appropriate inspection procedures to find it when it is present. It is also incumbent on the laboratory to develop the inspection equipment to improve the likelihood of detecting corrosion. Since corrosion damage appears to reoccur in the same locations in a population of airplanes, the use of teardown inspections is helpful for locating potential damage. The OC-ALC performed a teardown inspection on a KC-135 on an aircraft retired to Davis Monthan Air Force Base in 1991.

This aircraft, delivered to the USAF in 1962, had spent twenty-nine years at Mildenhall Air Base in the UK. Therefore, the aircraft saw a severe corrosion environment during its life. The inspection interrogated the structure for cracking as well as corrosion. The USAF found little cracking since the aircraft had only 16,521 flight hours and 2,942 flights. They cleaned the parts and etched them approximately thirty micrometers to enhance corrosion and crack detection. The USAF, for this study, classified corrosion as light if it was less than 25 micrometers, moderate if it was between 25 and 250 micrometers, and severe if it was greater than 250 micrometers.

For the fuselage, there was extensive light corrosion in the skin and doubler faying surfaces. They found limited moderate and severe corrosion below the cargo door, lower bilge, and at the spotwelds. None of the fuselage corrosion was severe enough to affect flight safety. For the wing, there was extensive moderate and severe pitting at the steel fasteners in the upper surface. Most of these had not progressed to exfoliation and none was severe enough to affect safety of flight. There were several areas of severe corrosion at the upper wing skin and spar interface. The center section of the horizontal tail suffered from severe exfoliation on the lower spar caps. They also found stress corrosion cracking in the horizontal tail. This inspection is significant in that it provided considerable insight on the extent of corrosion. It also served as an excellent representative aircraft for identifying areas of hidden corrosion that the USAF did not address in previous depot maintenance activities. In addition, it assessed the ability of available nondestructive inspection procedures to locate corrosion. The goal is three percent for the detection of thinning in a structure. Even this detection capability may not be adequate to maintain zero or positive static margins in the structure.

Another major concern is that pitting corrosion may accelerate the onset of WFD. This could be a serious safety concern unless the operators take proper care to use teardown inspections and nondestructive inspections to reveal the problem. The damage tolerance initial flaws as adopted by the USAF in the early seventies are well in excess of the size of defects associated with pitting corrosion. The criterion used by the USAF for damage tolerance design of new aircraft is that the initial flaw will not grow to critical in two design lifetimes. Further, the damage tolerance guidance emphasizes the selection of ductile materials that are tolerant to defects. As a further safety measure, the USAF guidance to the contractor is to design the structure such that it is inspectable. This is an advantage, not only for crack detection, but also for corrosion detection. The effects of pitting corrosion; however, could affect the safety of older aircraft that were not designed to the current damage tolerance guidance. Therefore, for these aircraft it is likely that cracks derived from pitting corrosion will need to found by the inspection program.

The other concern with pitting corrosion is that it could result in a significant degradation of the durability of the structure and hence shorten its useful life in service. Experiments with specimens exhibiting pitting corrosion show that cracks appear much sooner than otherwise expected. These cracks could degrade the fail-safe capability of the aircraft and consequently precipitate the need for structural modifications. A teardown inspection of a high time Boeing 707 wing revealed many significant cracks [16]. These cracks appeared to be predominantly in holes where the manufacturer used steel fasteners. The steel was not protected from the aluminum wing skins and stringers. Consequently, it is likely that pitting corrosion did contribute to the cracking found in this structure.

Another concern about the affect of corrosion on structural integrity is the affect of corrosion products on stresses in structural joints. Mostly, maintenance personnel will find evidence of this problem in the longitudinal lap splices in the fuselage. The corrosion products are much less dense than the original material and consequently the trapped powder causes the joint to expand. The stresses derived from this expansion are significant. At the lower limit of NDI detection capability of joint thinning, the stresses may reach yield. There have been many cases found where the stresses are sufficiently high to cause cracking in the skins. There is no known case where this problem has caused a catastrophic failure or the onset of WFD. However, the potential is there for the cracks to turn into fatigue cracks and propagate. There is also the possibility that even without cracking the stresses may degrade the fail-safe capability of the structure. There is a need to investigate this problem. In addition, NDI needs development that can detect cracks from pillowing that are approximately 2.5 millimeters in length.

The potential for stress corrosion cracking to become a fatigue problem is another concern. This seldom happens, but the result can be serious if the maintenance does not find the crack. There is no known case where a stress corrosion crack in an airframe has resulted in a catastrophic failure. The use of 7079-T6, 7178-T6 and 7075-T6 in aircraft designed in the sixties and seventies has led to numerous cases of stress corrosion cracking. In the USAF the C-130, C-141, C-5A and the T-38 are aircraft where stress corrosion cracking is a significant economic burden. The Materials Information Analysis Center performed a study [17] in 1996 and found that 70 out of 115 or 60.9% of the corrosion failures (problems) in C-130 airplanes were attributed to stress corrosion. NDI needs development that can detect stress corrosion cracks that are approximately 2.5 millimeters in length

A paper given by a member of the AFRL of the USAF [18] begged the question on when a predictive model would be available for corrosion. Since then, AFRL has started research programs aimed at answering that question. Today, unfortunately, no one appears to posses such a tool. Fortunately, the results of these on-going AFRL research efforts will be available within the next few years and should go a long way toward answering that question.

Corrosion Prevention/Repair

One of the most effective corrosion inhibitors available today is the so-called corrosion preventative compound (CPC). However, the use of them is inconsistent among the aircraft in the USAF inventory. One of the reasons for this is that maintenance becomes more difficult after these chemicals are applied to the structure. Another reason is that there is inadequate research and development funding given to their development so that the benefits are clearly defined. They must be developed to provide protection of lap splices and interior surfaces. It would be desirable for them to suppress corrosion and stress corrosion cracking and have no significant impact on fatigue capability. Lastly, they should have no significant impact on maintenance practices

In addition, there is a need to develop coatings that have a long life and are environmentally compliant. They should provide corrosion protection for their entire life. In conjunction with this effort is the development of NDI tools that will be able to inspect without paint removal. There is no real benefit from a long life coating that will require removal at the depot every four or five years to enable the performance of an inspection.

The repair of corrosion damage is a problem in that the repair technology is immature. For example, the USAF has had poor experience in the purchase of older aircraft from the commercial market. Some of these were so badly corroded that they could not be economically repaired. The USAF believes the decision by the owners to sell the aircraft exacerbated the extent of corrosion damage in these aircraft. After they made that decision, the aircraft evidently received very little maintenance until the owners could sell them. Under these conditions, it did not take long before the damage from corrosion was so extensive that the USAF had to condemn the aircraft immediately after they were purchased. If the USAF had used the current nondestructive inspection capability at the time of purchase, they would not have purchased these aircraft. Examples such as this highlight the diligence needed in a maintenance program. The owners of these aircraft, up to few years before they sold them, maintained them properly and they were flying in an airworthy condition.

CONCLUSIONS

The USAF research and development program for aging aircraft have provided the technology base for safe and economic operation of military aircraft. As an indicator of this success, the failure rate for all systems designed to and/or maintained to the current policy, is one aircraft lost due to structural reasons in more than ten million flight hours. This is significantly less than the overall aircraft loss rate from all causes by two orders of magnitude. This success, however, should not be used to indicate that there is no need for continued research on aging aircraft. The dangers from corrosion, fatigue or corrosion fatigue are ever present in operational aircraft. Presently, the largest danger by a considerable margin is economic rather than flight safety. All of the collective experience from both military and commercial operations indicates this to be true. No one can foretell with any degree of accuracy what to expect as both military and commercial aircraft push further into the uncharted waters of aging. It is incumbent; however, for the researcher, the engineer and the maintenance personnel to maintain a diligent approach to the problem. They must use all available techniques such as DTA scheduled inspections, ACIs, and assessments for the onset of WFD to help ensure that they maintain the safety of future aircraft operations. Diligent use of CPC's and research into better means of corrosion detection and prevention appear to be the most promising ways to reduce the economic burden of these problems in the future. The priority for corrosion research and development needs to be given to corrosion detection and the inhibition of corrosion when found.

One of the major problems found in operations with aging aircraft is the cost associated with corrosion damage. Unfortunately, the progress made in the recent past in the control of this problem does not bode well for the future. This is especially true when one considers the impact of new environmental laws that remove many of the corrosion fighting chemicals that are currently used. Continued emphasis on research in the area of corrosion control is certainly one area that could have significant benefit.

Another major problem is WFD in primary structural elements. There will be costs incurred to establish an estimate of the time of onset of this problem. This will need to be done through the analysis of data derived from teardown inspections of fatigue test articles and/or of operational aircraft. These estimates will need to be corroborated through the use of detail inspections of suspect structural elements. Once this onset time has been reached, then there will be costs incurred by the modification of the aircraft to remove this problem.

The severity of both of these problems is made worse today because of a lack of adequate nondestructive evaluation techniques to look for corrosion damage in structural joints and to find the small cracks that would be the indicator of the onset of WFD. It appears that the current efforts in research in nondestructive evaluation will produce the technology for these problems. It remains to be seen if there is an economic motivation to transition this technology from the laboratory to inspections of operational aircraft.

Many of the aging military aircraft problems find an exact parallel in aging commercial aircraft. It is prudent, therefore, that these problems be worked through the combined talents and resources of the cognizant organizations. Efforts to date indicate that this approach will be successful and most efficient in solving these complex problems.

REFERENCES

1. Lincoln, J. W., "Significant Cracking Experiences in the USAF," Published in the Proceedings of the 22nd International Congress of Aeronautical Sciences, August 2000.

2. Negaard, G. R., "The History of the Aircraft Structural Integrity Program," Aerospace Structures Information and Analysis Center (ASIAC), Report No. 680.1B, 1980.

3. Coffin, M. D. and Tiffany, C. F., "New Air Force Requirements for Structural Safety, Durability, and Life Management," AIAA Journal of Aircraft, Volume 13, No. 2., 1976.

4. NTSB/AAR-89/03, National Transportation Safety Board Aircraft Accident Report, Aloha Airlines, Flight 243, Boeing 737-200, N73711, Near Maui, Hawaii, April 28, 1988.

5. National Research Council, "Aging of U.S. Air Force Aircraft," Publication NMAB-488-2, National Academy Press, Washington, D.C., 1997.

6. Department of the Air Force, 1974 "Airplane Damage Tolerance Requirements," Military Specification MIL-A-83444.

7. Lincoln, J.W. "Damage Tolerance - USAF Experience", Proceedings of the 13th Symposium of the International Committee on Aeronautical Fatigue, Pisa, Italy, 1985.

8. Noor, A. K., Structural Technology for Future Aerospace Systems, Volume 188, Progress in Astronautics and Aeronautics, American Institute of Aeronautics and Astronautics, 2000.

9. Department of Defense, Joint Specification Guide, Aircraft Structures, JSSG 2006, 30 October 1998.

10. Lincoln, J.W., "Risk Assessment of an Aging Military Aircraft," Journal of Aircraft, Volume 22, Number 8, 1985.

11. Lincoln, J. W. "Risk Assessments-USAF Experience", Proceedings of the International Workshop on Structural Integrity of Aging Airplanes, Atlanta, GA, March 31 - 2 April 1992.

12. Lincoln, J. W., "Risk Assessments of Aging Aircraft," Proceedings of the First Joint DOD/FAA/NASA Conference on Aging Aircraft, Ogden, UT, July 1997.

13. Lincoln, J. W. “Life Management Approach For USAF Aircraft”, Proceedings of the 72nd AGARD Structures and Materials Panel Meeting, AGARD-CP-506, Bath, United Kingdom, 1991.

14. Wallace, W., Hoeppner, D. W., Kandachar, P. V., “AGARD Corrosion Handbook, Volume 1, Aircraft Corrosion: Causes and Case Histories,” July 1985.

15. Department of the Air Force "Aircraft Structural Integrity Program, General Guidelines For,” MIL-HDBK-1530C, 30 August 1996.

16. Lincoln, J. W. “Aging Aircraft - USAF Experience And Actions,” Proceedings of the 19th Symposium of the International Committee on Aeronautical Fatigue, 16th Plantema Memorial Lecture, Edinburgh, Scotland, pp 3-38, 1997.

17. Mindlin, H., Gilp, B. F., Elliott, L. S., “Corrosion in DoD Systems: Data Collection and Analysis (Phase 1), February 1996.

18. Paul, C. A. and Mills, T, “Structural Integrity of Aging Aircraft,” presented at AeroMat ’98, June 1998.

This page has been deliberately left blank

Page intentionnellement blanche

USAF Strategy for Aging Aircraft Subsystem Research and Development

William C. Kinzig
ASC/ENFA
2530 Loop Road West
Wright-Patterson AFB, Ohio 45433-7101
USA

SUMMARY

Like many other nations today, the United States Air Force (USAF) is retaining their existing aircraft longer than planned. It is estimated that the current average age of aircraft in the USAF inventory today is 22 years old. By 2005, 75% of the USAF inventory will be over 20 years old. As the age of our fleet continues to rise, aircraft mission capable rates degrade and there is a potential for increased risk to safety of flight should the aircraft not be properly maintained. Maintenance data indicates that air vehicle subsystems are one of the largest contributors to aircraft downtime due to in-service failures. Fortunately aircraft typically are not lost due to the subsystem failures. However, if one is not careful, this aspect can tend to foster an attitude that we should accept these failure rates. What this approach fails to recognize is that we no longer have the budget or the number of available aircraft to support this level of maintenance. The purpose of this paper is to discuss aging aircraft concerns found in air vehicle subsystems and the approach that the USAF is using to alleviate these concerns.

BACKGROUND

Philosophically, the USAF aging aircraft program began in the late 1950's following an in-flight structural wing failure of a B-47 aircraft [1]. As a result, the USAF developed the Aircraft Structural Integrity Program (ASIP) [2]. The ASIP [3] is a disciplined engineering process that defines all of the tasks necessary to ensure the structural integrity of the air vehicle airframes. ASIP is a cradle-to-grave process and has become the basis for our aging aircraft programs.

ASIP has evolved over the last 40 years into a very effective and widely accepted process that is used extensively throughout the USAF. There is an Air Force Policy Directive [4] that requires implementation of the ASIP on all the USAF aircraft programs. The process has been very effective and we have not lost a USAF aircraft due to an inherent structural failure for over ten years [5].

As a result of the success of the ASIP, a similar integrity process has been established for the other elements of the air vehicle. In the 1970's, the Engine Structural Integrity Program (ENSIP) [6] was founded, and in the 1980's, the Avionic Integrity Program (AVIP) [7] and the Mechanical Equipment and Subsystem Program (MECSIP) [8] were also founded. ENSIP, AVIP and MECSIP are all patterned after the highly successful ASIP and all have similar type of tasks.

Even though the MECSIP has been in existence for about 13 years, the F-22 program was the first to implement it during the design/development phase. In 1995, due to the ever-increasing age of the USAF inventory, the Air Mobility Command became concerned with the future of their aging aircraft and initiated an Aging Aircraft Process Action Team (PAT). The purpose of the PAT

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

was to implement a disciplined engineering process for the sustainment of their fleet. The goal is to ensure safety, improve aircraft availability and assist in maintenance planning. Air vehicle subsystems were included in the process with the MECSIP being a key element for defining the overall disciplined engineering process. However, in reviewing the existing MECSIP MIL-HDBK-1798, it was concluded that the sustainment section of the handbook was lacking. Consequently, a team was formed to revise the MECSIP document and the document is in its final stages of coordination. The revised document should be released late this calendar year.

STRATEGY FOR AGING AIRCRAFT

J. Lincoln, Air Force Senior Leader for Aircraft Structural Integrity, described the strategy for using research and development to reduce maintenance costs as a six-step process [9]. The steps are:

1. Conduct surveys to determine problems
2. Identify solutions
3. Establish research and development roadmaps
4. Obtain management approval
5. Implement research and development
6. Transition technology to the operator

The major thrust of this paper is to discuss the results to date of Step 1 relative to aircraft subsystems – i.e. to report the findings of our surveys.

As Lincoln indicated [9], the USAF established an Aging Aircraft Technologies Team (AATT) to review the problems associated with our aging fleet and to make recommendations on how best to direct our research and development dollars. The team consists of members from the Engineering Directorate at the Aeronautical Systems Center, the Air Force Research Laboratory, and the Aging Aircraft Program Office. All three organizations are located at Wright-Patterson AFB, OH. The first survey was conducted in July through August of 1998 and addressed only structures. Additional surveys were conducted in June through December 1999 and August through September 2000. Both of these surveys were expanded to include air vehicle subsystems. A fourth survey is currently in progress and should be completed by 31 Oct 01.

The surveys consisted of visits to the three Air Logistics Centers responsible for the sustainment of the various USAF aircraft. Interviews were conducted with engineers and maintainers directly responsible for the integrity of the aircraft. Approximately 30 aging weapon systems were surveyed each year. In order to facilitate the review, a list of 13 questions relating to subsystems was submitted in advance. The questions were directed at identifying the problem areas, what actions were being taken to improve subsystem reliability, and where our development dollars could best be targeted to improve overall system safety and mission readiness. The systems reviewed included hydraulics, fuels, environmental control, landing gear, auxiliary power, flight controls, electrical power and wiring.

MAINTENANCE DRIVERS

Based on the survey of approximately thirty aircraft programs, the order of the “Top 10” maintenance drivers varied somewhat between aircraft but were very consistent between the two surveys conducted in 1999 and 2000. Based upon the qualitative results of the survey, the following subsystems in the order shown are estimated to be the leading maintenance drivers:

a. Landing Gear **-** Landing Gear was near the top of many program lists. Leading problems involve cracks in the main and nose landing gear assemblies and attachment points (e.g.

braces, trunions and torque arms), poor component service life due to wear, inadequate lubrication provisions, tire chunking and wheel speed transducer failures. The Landing Gear Team for the USAF overhaul facility located at Hill AFB stated that approximately 80% of the landing gear structural failures are due to stress corrosion cracking.

b. Flight Controls - The Flight Control System also appeared high on many program lists. The problems vary significantly between aircraft and include items such as wear and chafing of the control cables, actuator failures including rod end and servovalve failures, and inadequate lubrication provisions. Some of the actuator rod end failures were the result of corrosion.

c. Wiring - Wiring is a major concern with our aging aircraft. Wiring failures are typically attributed to chafing, deterioration of the insulation and corrosion at the connectors. Deterioration of the insulation is frequently attributed to the use of the Kapton material. Several aircraft System Programs Offices have initiated or are planning large-scale replacement of their wiring systems. Both the Aging Aircraft Technologies Team and the Air Force Research Laboratory have engineers dedicated to resolving the wiring issues.

d. Hydraulics - Even though many people view hydraulics as a mature technology, we still experience many problems with that system. Problems identified include tube chafing, fluid contamination, pump failures and hose failures. One problem recently noted on both an aging aircraft system as well as a new system under development involves chafing under clamps due to sand and dirt becoming lodged between the tube and the clamp liner. Both programs are exploring potential changes to eliminate the problem. The changes being considered include changes in liner material, changes in clamp design and the addition of a protective film around the tube itself.

e. Fuels - Component leakage, and particularly fuel tank leakage, is always a problem sometime during the life of an aircraft. The problem typically becomes amplified as the aircraft ages, particularly near the end of its life. Excessive component wear, in general, is also reported to be a problem.

f. Environmental Control System (ECS) - ECS problems tend to vary. Problems have been noted relative to bleed air overheat detectors, air cycle machines, temperature controls, pressurization, and bleed air ducts.

Periodically the question arises as to whether corrosion is a problem for subsystems. As indicated above, corrosion does not appear to be a widespread problem except for Landing Gear. Numerous corrosion problems have been noted with side braces, trunions and torque arms. The only other corrosion problems noted involved the rod end of a specific elevator actuator, the rain duct on another aircraft and several isolated cases of electrical connectors.

Even though subsystems are one of the largest contributors to unscheduled maintenance, System Program Offices previously tended not to perform pre-emptive maintenance unless the failures potentially impacted safety of flight. However some programs have started to change this philosophy. When queried on plans to correct poor performers, several System Program Offices reported planned actions involving the replacement of flight control actuators, stripping and resealing of fuel tanks, and replacing aircraft wiring. Likewise, six System Program Offices identified plans for subsystem upgrades, overhauls or modifications.

PROGRAMMATIC ISSUES

In reviewing the issues associated with each of the System Program Offices, two categories of subsystem issues were noted - programmatic issues and technical issues. The basic programmatic issues focused on inadequate funding, inaccurate databases for tracking unscheduled maintenance actions, lack of serialized part number tracking for mechanical system components, and failure to implement the Integrity Programs as a disciplined engineering process for sustainment. Below is a more detailed discussion of each issue:

a. Inadequate funding. Funding for aging aircraft research and development is probably the number one problem faced by the program managers. Funding of aging aircraft requirements for subsystems is even more challenging because the subsystem problems normally are not classified as immediate safety issues. Instead, the failures affect aircraft availability. Justification can be difficult for non-safety related problems because the available funding is normally consumed by the safety related problems. When funding does become available, it may be impossible to support all the necessary areas. Thus there is a need to establish priorities.

b. Inaccurate databases for tracking unscheduled maintenance actions. The most discussed issue during these reviews involved the need for an accurate component reliability and maintenance data tracking system. Such a database is truly the starting point in identifying the “bad actors” so that we can determine where best to direct our maintenance efforts and dollars. System Program Office that discussed component tracking indicated a concern with the current Reliability and Maintainability Information System (REMIS). The offices that seriously tracked the reliability and maintenance of their components stated that they would start with the REMIS database but would confirm the reliability of the data through detailed discussions with the maintainer. Frequently the maintainer could not confirm the failure rates or would identify new problem areas. Since the database is only as good as the data entered, tracking systems must be established that are more user friendly, and facilitate quick and accurate entry of the data. Attention must be given to making accurate inputs as foolproof as possible. Also a sufficiently complete set of component or part identifiers must be included in the database so that a failure or maintenance action can be accurately attributed to the correct part. For example, the Air Force Reliability and Maintainability Information System does not have part identifiers for wiring. Thus a wiring failure frequently gets erroneously entered against the electrical component it powers.

c. Lack of serialized part number tracking. Another need in establishing a proactive sustainment program is the existence of a serialized component tracking system. For example, one program office emphasized that the landing gear is not maintained as a matched set – i.e. parts are frequently mixed and matched during field maintenance. When such a gear is returned for overhaul, the depot has no idea of the condition or time accumulated on the various parts within the gear. Likewise, when a given part is overhauled, it is returned to stock. That part will then be used on the same type of aircraft (e.g. a C-5) or, in some cases, a totally different type of aircraft (e.g. F-15). A hydraulic accumulator is a prime example of a component that may be overhauled and returned to use on a totally different platform. Over time, the accumulator may encounter numerous overhauls with the structural fatigue life of the cylinder being exceeded. With no type of serialized tracking, the user has no indication when a specific unit has exceeded its design life. With today’s availability of computers and computer technology, there are numerous possibilities for serialized tracking. Bar codes can be embedded on each component nameplate and scanned into a central tracking system upon installation and removal. Chips can be embedded into a component that records operating times from an existing onboard computer. More detailed life related parameters can be calculated and either stored on a component computer chip or saved on a central aircraft computer and subsequently downloaded to a centralized ground data tracking system. This is similar in concept to the Individual Aircraft Tracking Program integral to the Aircraft Structural Integrity Program.

d. Failure to implement the Integrity Programs. As previously noted, the Aircraft Structural Integrity Program, ASIP, has been a very effective tool in maintaining our aging fleet. It was initiated in the late 1950's and is used for both development and sustainment. As a result, the Mechanical Equipment and Subsystems Integrity Program, MECSIP, was developed in the 1980's and patterned after the highly successful ASIP. The program offices have been slow in implementing the Integrity Program to subsystems, but we are now beginning to see some of the System Program Offices implement MECSIP as the disciplined engineering process for sustainment. As with ASIP, what MECSIP offers the System Program Offices is a proactive process that allows one to understand where the problems reside so that we can better focus our maintenance actions and dollars to improve aircraft safety, suitability and effectiveness. Consequently it is essential that our program managers understand the benefits of MECSIP and implement the disciplined engineering process into their weapon system programs.

TECHNICAL ISSUES

There were also several technical issues identified for subsystems throughout the survey. Specific needs noted include:

a. Capability to predict remaining life of components. As indicated previously, there has been a tendency to fly the subsystems to failure. Two of the driving reasons are (1) we have no method to determine remaining component life and (2) there is no component level serialized tracking system. However the need does exist to establish some method to predict remaining life so that we can plan the necessary maintenance actions to improve aircraft availability and reduce maintenance cost. One potential technique for electrically driven components would be to develop a correlation between electrical signature (e.g. current draw) and component wear. Some work has been done in this area with promising results. There may also be other techniques that could be developed with similar results.

b. Non-intrusive techniques to assess wiring health. Degradation of wiring insulation, coupled with wire chafing, is a problem noted on many of our aging aircraft. This creates not only a significant maintenance burden but also a potential safety issue should flammable fluids or explosive vapors be present. Attention is beginning to be focused on developing techniques to determine the integrity of wiring system. However, we must emphasize the need to develop quick, simple, non-intrusive methods that can pinpoint low-level shorts associated with a specific wire.

c. Non-intrusive techniques to assess condition of mechanical cables. Cable wear associated with flight controls, engine throttle, etc. is a significant problem with many of our older non-fly-by-wire aircraft. Unfortunately, the condition of the cable can frequently be determined only through removal of the cable. Several program offices noted a need for developing a non-intrusive method for assessing the health of installed cables.

d. Machine to measure tube configuration data. Tube failure is always an issue with our aging aircraft. When the tubing fails due to chafing, fatigue, etc., the new tubing is frequently fabricated at the base level. However, the tube bend data is not always available or the data does not always result in a newly fabricated tube exactly the same as the tube being replaced. Incorrectly configured tubing can result in tube pre-loading that, in turn, can result in premature failures. Since many of the fluids (e.g. hydraulic and fuel) contained within the tubing are flammable, tube failure can be a potential safety of flight concern. Consequently a need has been identified for a machine that can accurately measure the configuration of a tube that was just removed and convert it to set of data necessary for subsequent tube fabrication.

CONCLUSIONS

In the past, the USAF has tended to take an approach for subsystems whereby we would fly the systems to failure – i.e. little or no forced maintenance planning was done relative to subsystems. This was feasible because we seldom loose aircraft due to subsystem type of failures since our basic design philosophy is to provide sufficient subsystem redundancy to preclude the loss of aircraft. However, the USAF has now begun to recognize the impact of subsystems on aircraft availability and downtime. Consequently several specific actions have been initiated:

a. The Air Mobility Command has initiated an Aging Aircraft Process Action Team to address the issues associated with their aging fleet.

b. The Aging Aircraft System Program Office now includes air vehicle subsystems in their yearly surveys of over thirty aircraft systems.

c. The results of the surveys are being communicated to government and industry organizations responsible for research and development.

d. The Air Mobility Command is adopting the Mechanical Equipment and Subsystem Integrity Program as a method for providing a disciplined engineering process to the sustainment of their weapon systems.

Even though the Air Mobility Command is still in the early stages of implementing the integrity program for subsystems, it is an initial and necessary step towards characterizing the problems that require further research and development.

With an initial understanding of the current problem areas, we now have a starting point for identifying where best to invest our research and development efforts. Similar to what Lincoln indicated [9], many of these same aging military aircraft problems have similar or exact parallelism in the aging commercial aircraft world. It is therefore prudent that these problems be worked through the combined talents and resources of both the military and commercial worlds so that we can more effectively channel the limited research and development dollars in the most efficient manner possible.

REFERENCES

1. Lincoln, J. W., "Significant Cracking Experiences in the USAF," Published in the Proceedings of the 22nd International Congress of Aeronautical Sciences, August 2000.

2. Negaard, G.R., "The History of the Aircraft Structural Integrity Program," Aerospace Structures Information and Analysis Center (ASIAC), Report No. 680.1B, 1980.

3. Department of the Air Force "Aircraft Structural Integrity Program, General Guidelines For," MIL-HDBK-1530C, 30 August 1996.

4. Department of the Air Force "Aircraft Structural Integrity," Air Force Policy Directive 63-10, 1 November 1997.

5. Gideon, F. C., Major General, "Air Force Safety Center Mission Briefing," Presented at the Third Joint FAA/DoD/NASA Conference on Aging Aircraft, September 1999.

6. Department of the Air Force "Engine Structural Integrity Program (ENSIP)," MIL-HDBK-1783A, 22 March 1999.

7. Department of the Air Force "Avionics/Electronics Integrity," MIL-HDBK-87244, 30 January 1995.

8. Department of the Air Force "Mechanical and Subsystems Integrity Program," MIL-HDBK-1798, 17 April 2001.

9. Lincoln, J. W., "USAF Strategy for Aging Aircraft Structures Research and Development," Proceedings of the North Atlantic Treaty Organization Symposium on Aging Mechanisms and Control, Manchester, England, October 2001.

This page has been deliberately left blank

Page intentionnellement blanche

Ageless Love Aging Fleets:
A User's View

Brig. Gen. Piertro Antonio Giuliani
Maj. Francesco Marradi
Italian Air Force
Logistics Command
Comando Logistico, 2a Divisione, 1 Reparto
viale dellÚniversita'4
00100 Roma, Italy

Introduction

Today around the world there is great interesest in old fleet and in ageing aircraft.

The reason for this is due mainly to budget constraints but also to other reasons as safety, and enviromental laws. Interesting is to note that today one of the more sparkling area in aerospace industry is related to the MRO activities.

But this type of industry requires today and much more in the future the setting of rules and the development of a huge know-how, as well the a/c manufacturer wich will introduce as design parameter the "age" and not only the flight cycles or fligth hours, as it is today, during the desing phase.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Italian Air Force - Logistic Command

First question: what kind of "system" am I talking about?

We may define a system as a group of many interlocking parts operating together with a common goal...

Italian Air Force - Logistic Command

Not strictly a technological system …
but a "broad-defined" system composed by

- Weapon system
- Support factors
- Operative and geopolitical factors
- Environment

…and this definition leads me to a second question

Italian Air Force - Logistic Command

Second question: what is “aging”?

Not a single aging phenomena…
but a “three-leg table” composed by

metaphysical

- Operational Aging
  - Absolute
  - Relative
  - Community
- Economical Aging (mantinaibility)

physical

- Technological Aging (classic)

Italian Air Force - Logistic Command

But for a Logistician what Aging does mean?

MONEY, AS ALWAYS!

Italian Air Force - Logistic Command

Third question: Where is the money ?

These charts are the theory about ILS…

But where are the costs of
aging

?

Italian Air Force - Logistic Command

TIMES ARE CHANGING

- At the beginning, systems were developed aiming only at performance.
- Then, the Integrated Logistic Support approach was adopted, taking care of the Life-Cycle Cost of the system.
- Now it's time to adopt the ILS+Aging Awareness approach

(physical and metaphysical)

?

Italian Air Force - Logistic Command

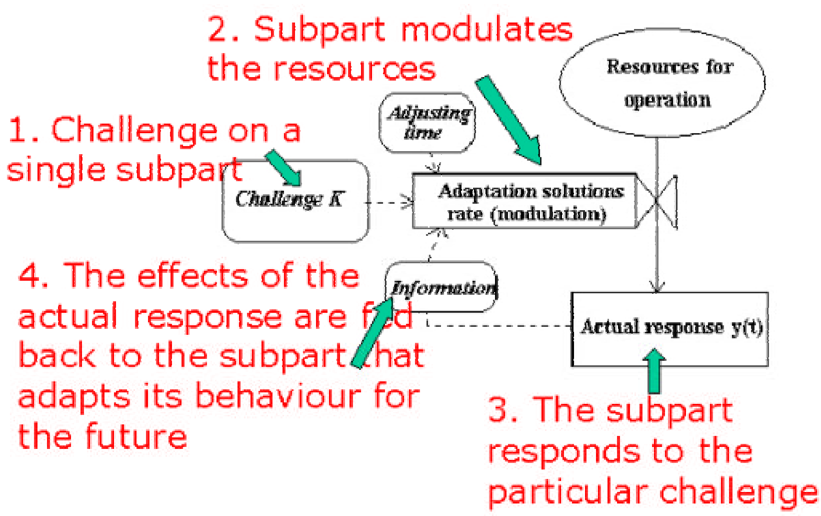
The resulting ageing of the system can be modelled by the combination of many positive and negative first order feedback loops which occur in the subparts of the system when time elapses.

Note that ageing is taken in the sense of an evolution or progress in time, without giving any qualitative value to these words: it is just the displacement of the system along the positive time axis in a quadri-dimensional space-time.

Italian Air Force - Logistic Command

Negative feedback loop

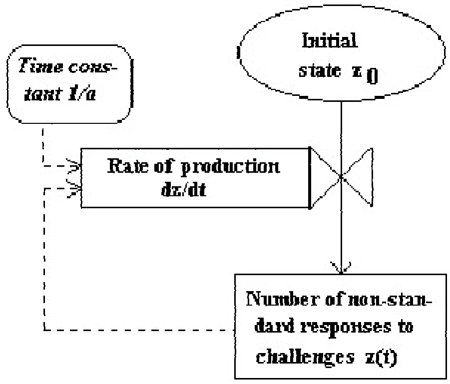
The operating of the system results in internal challenges on the subparts which may jeopardize the system's integrity. The subparts have to adapt to the challenges in due time in order to allow further operation of the system as a whole.



Italian Air Force - Logistic Command

Positive feedback loop

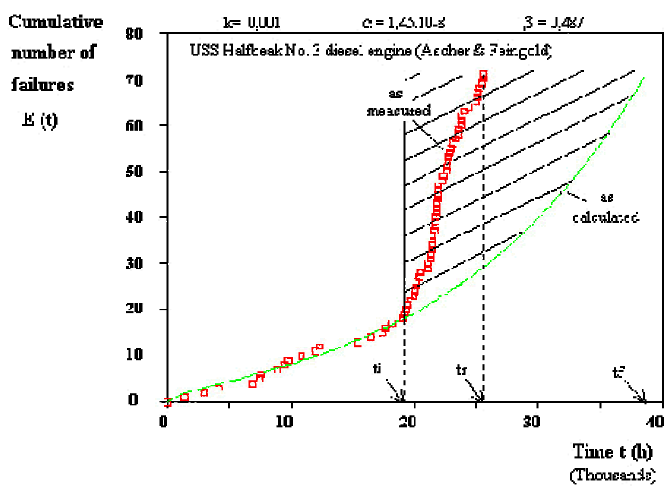
During operation, the challenges appear again and again, sometimes on a steady base sometimes not. It may happen that the adaptations in the subparts be sometimes non-standard even if satisfactory for the further operation of the system. The word "non-standard" means "different from what could be expected from the past behaviour". This may result in changes of information for neighbouring subparts about what their challenges are. Their responses could then also become non-standard. There will be a snowball effect: non-standard responses will induce differing assessments of the challenges which in turn will induce other non-standard responses.



Italian Air Force - Logistic Command

The Ageing Zone

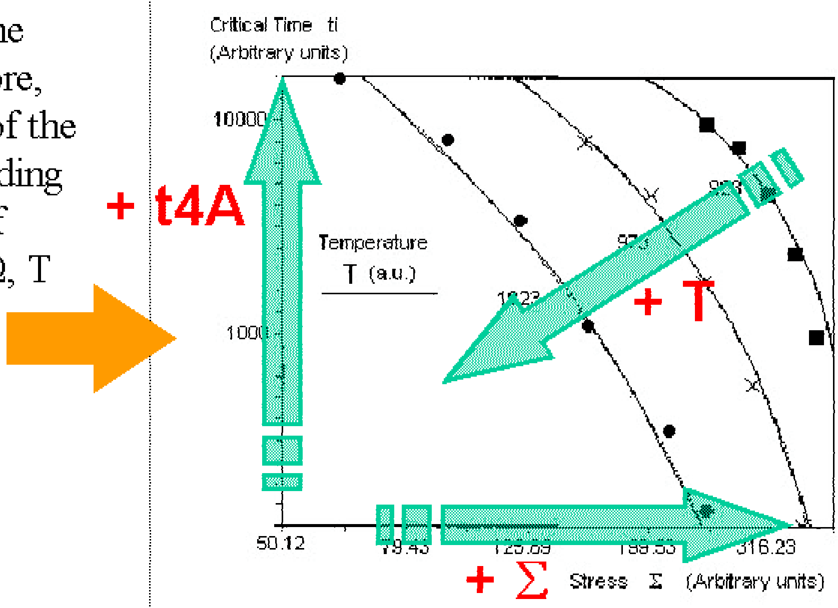
After t_i, the system will follow a path between sudden collapse and the continuation of the curve described by the equation y(t) (or its normalized form). This zone is the shaded in one (the Ageing Zone).



Italian Air Force - Logistic Command

The “Time Ageing Limit”

So it is possible to predict the “Time Ageing Limit” t_i. More, using the normalized form of the equation y(t) (β-only depending form), we can trace a sort of trade off chert between Σ, Ω, T and t_i, like this example



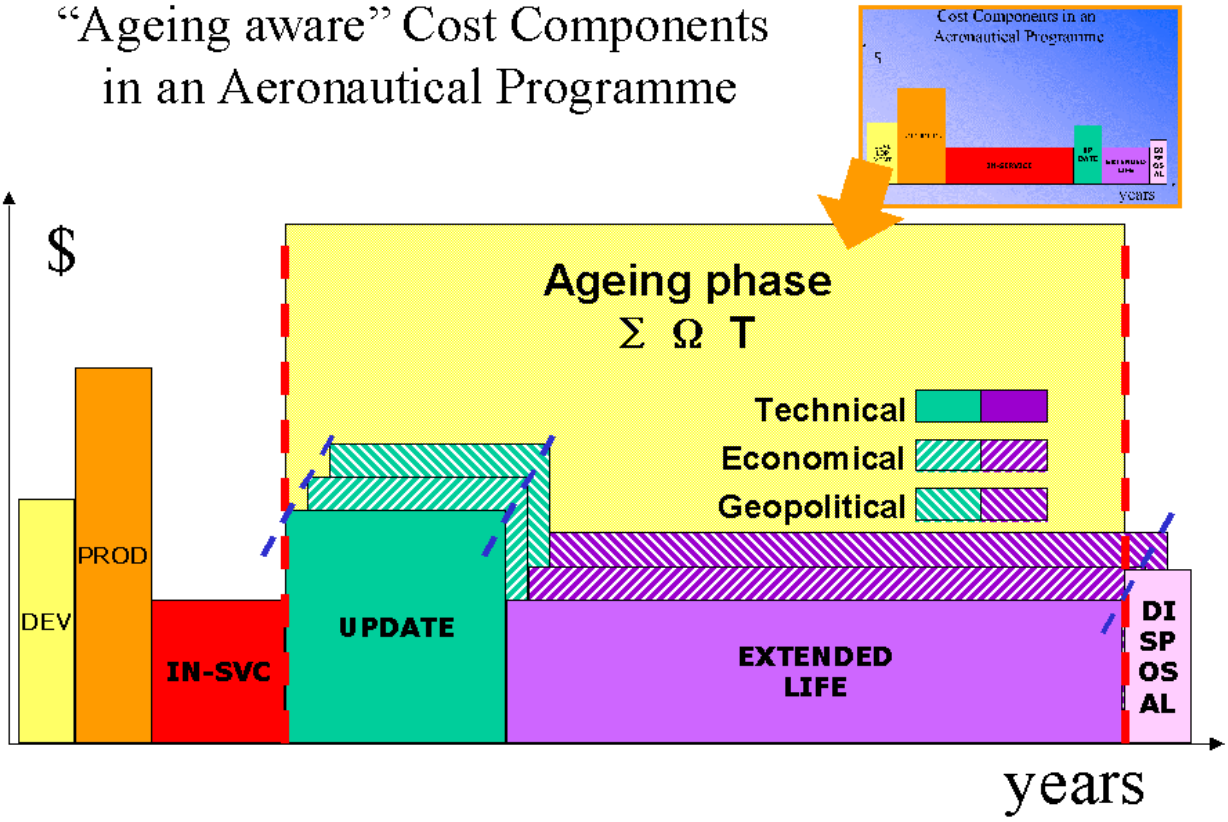
Italian Air Force - Logistic Command

Σ, Ω, and T are broad-meaning terms.

We can use them in all kind of phenomena: technical, medical, economical, geopolitical, logistical, etc.

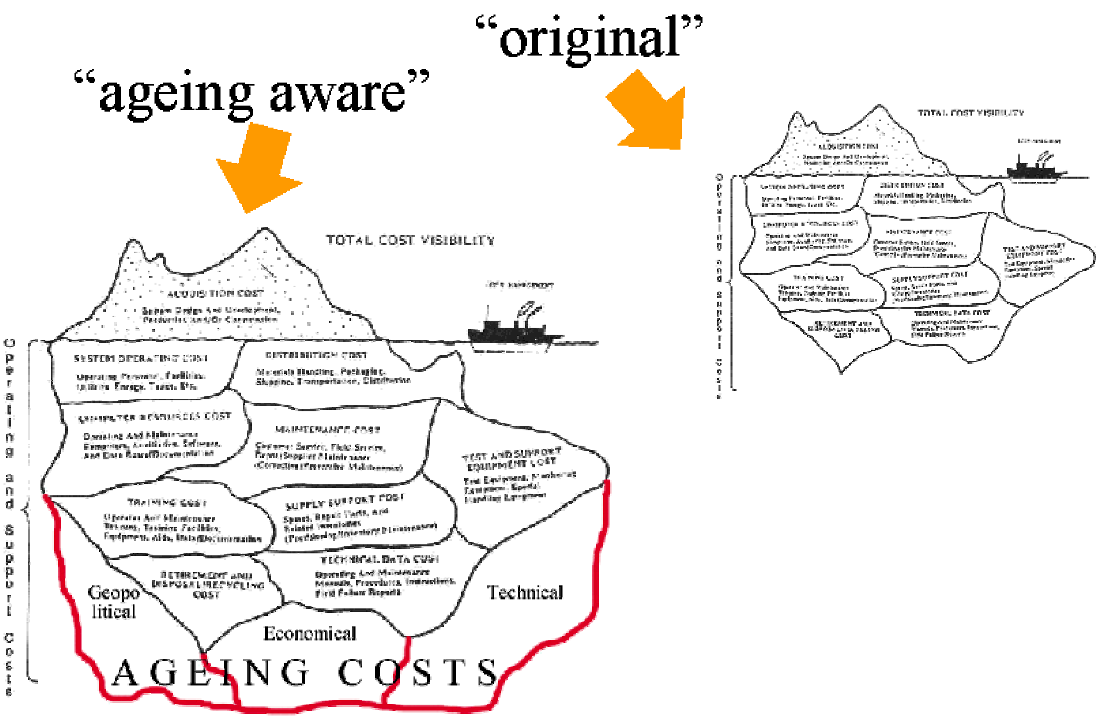
Italian Air Force - Logistic Command

"Ageing aware" Cost Components in an Aeronautical Programme



Italian Air Force - Logistic Command

The "Ageing awared" Iceberg of Costs

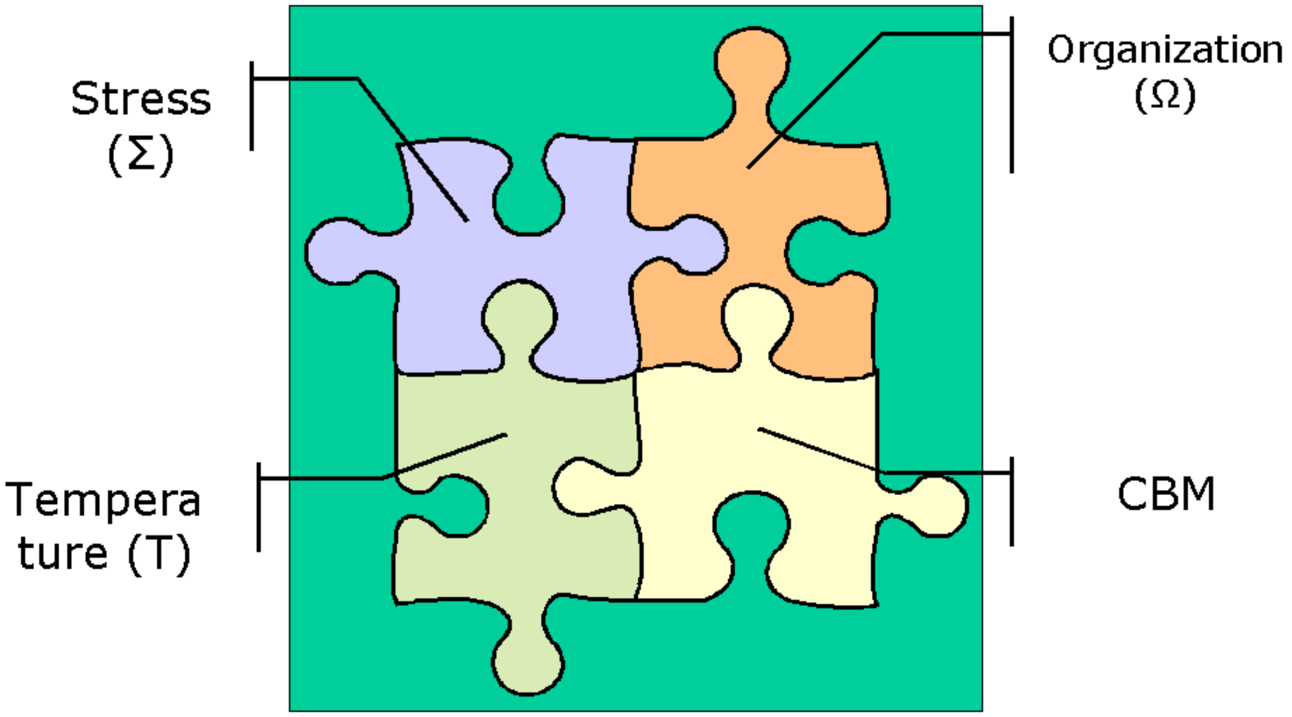


Italian Air Force - Logistic Command

To control Σ, Ω and T across technical, geopolitical and economical factors, requires the use of IT. Dropping politics and econimics stuff, for the technical ones what is a key to cope with ageing? …CBM !

Italian Air Force - Logistic Command

The Aging Puzzle



Italian Air Force - Logistic Command

For Western Countries it is mandatory to keep a stable geopolitical situation in the World (T). And it is mandatory to keep military organizations lean and well oiled (Ω).

Italian Air Force - Logistic Command

And at the end of the day the use of the CBM approach to maintenance we expect to cut LCC better than the other maintenance strategies monitoring technical ageing of the fleet.

Conclusions

Based upon these considerations, mathematics, coupled with CBM, can help us in this "chaotic" situation.

It is necessary for aerogeriatrics, as we are, to start following a "local" approach, but in accordance with our slogan:

...Act locally, but think globally.

This page has been deliberately left blank

Page intentionnellement blanche

Common Understanding of Life Management Techniques for Ageing Air Vehicles

Dipl.Ing. Thomas Vogelfänger
EADS Deutschland GmbH
Military Aircraft
Reliability Safety
Dept. MB 414, P.O. Box 80 11 60
81663 Munich
Germany
E-Mail: thomas.vogelfaenger@m.eads.net

Abstract

Ageing aircraft are a growing problem in both military and commercial aviation. With the economic constraints of keeping current military and civilian aircraft in service, and the growing demand for air travel worldwide, the problems of ageing aircraft will continue to worsen.

The service life extension of equipment or a system over the initial design period is indeed a question of safety however, as the subsequent explanations show, this a question of efficiency and economy.

In a general sense, ageing aircraft are characterised by the deterioration of structural strength properties and the related problems and the increasing maintenance costs. Some of these problems are time dependent, such as corrosion, which also depends heavily on the usage environment. Others are usage dependent, such as in fatigue cracking, which is naturally caused by the mechanical loads that are introduced into the structure and also in electronic devices. Often the damage state of an aircraft is the result of both time (calendar years) and usage (operating hours). To maintain structural integrity, steps must be taken toward the prevention, detection, repair and prediction of the initiation and growth of aircraft structural damage.

What are now the valuation criteria for a decision for or against the further extension of the service life of an ageing aircraft?

The paper concludes with a view of future technologies, which could contribute to an expense-optimised useful life extension.

1. Introduction

The combination of shrinking post Cold War Military Budgets and escalating costs for development and acquisition of new military aircraft have lead to increasing efforts world-wide to extend the operation of military aircraft far beyond their original design lives. Defence departments of even the wealthiest nations and nowadays also the poorer nations have begun to invest heavily in ageing aircraft programs. The intent of such programs is to preserve the integrity of aircraft structure and critical subsystems. The primary objective is to develop and transition technologies to further extend the life and/or reduce the cost of a weapon system forced to remain in service beyond its original design life. Sustainment refers to all activities necessary to keep the system operating except major modifications, modernisation projects or upgrades of existing subsystems.

One of the biggest challenges at the current time is the fleet management of a much bigger number of ageing aircraft. In 1993 already 51% of the US Air Force inventory are more than 15 years old and more than 20% had already crossed a life of more than 20 years. A look at the worldwide civilian aircraft makes clear the problem.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

A/C Type	> 15 year [%]	> 20 year [%]	> 25 year [%]
A 300	43	9	-
B747 "Classic"	68	34	20
L-1011	80	49	6
DC-10	76	50	15
DC-8	51	51	51
B 727	80	53	32
B 727-100/200	71	35	21
DC-9	83	73	58
Total	**72**	**47**	**30**

Table 1: Civilian aircraft in service in 1997

Therefore more than *⅔ of the aircraft are older than 15 years and nearly half more than 20 years old. It* is astonishing that still half of all the DC-8 ever built are still in service. We can expect similar results for the type DC-9 and the Boeing 727.

The purpose of this document is to provide an overview of the complexities of life extension, to discuss the general process involved in a life extension analysis, and to provide the reader with some guidelines to follow when trying to make the decision of whether or not life extension is a viable alternative to system replacement. The information presented is based on a review of the current literature, a consolidation of various system specific life extension procedures and discussions with individuals who are knowledgeable about or who have experience relating to life extension.

2. Life extension

Life extension, sometimes called service or system life extension, life cycle management or life optimisation has become an important topic of analysis in many of today's industries.
While life extension is of paramount importance to the utility industry, other industries also must and do consider life extension as an alternative to develop new systems. Even the military services are placing more emphasis on life extension, in light of declining budgets and reprioritised federal spending. No matter which industry is being considered, however, the incentives to save money and delay replacement of older systems are compelling. Yet, the risks and benefits must be carefully evaluated or the short-term cost savings will be offset by greater long-term costs, safety problems, or decreased system performance.

The primary focus is to maintain operational fighting capability by directing resources toward high payback technologies.

- That can identify structural deterioration
- That could threaten aircraft safety or degrade performance, prevent or minimise structural deterioration.
- That could become an excessive economic burden or adversely affect force readiness, assist in replacement of components.
- That can no longer be procured and assist in development of failure analysis and life predictive tools for such problems as fatigue damage, corrosion and stress corrosion cracking.

Systems and equipment are designed to have a certain service life. They are designed to perform one or more specific functions over a specified length of time or, in the case of one-shot devices, be capable of performing a function after storage or within a specific interval of time after manufacture, when operated and maintained according to some stated plan.

The oldest German Tornado will reach very soon the design life of 20 years in-service, where the aircraft runs out of certification. Considering a further planned usage up to the year 2028 for the Tornado IDS variant some IDS aircraft will stay in-service for nearly 50 years.

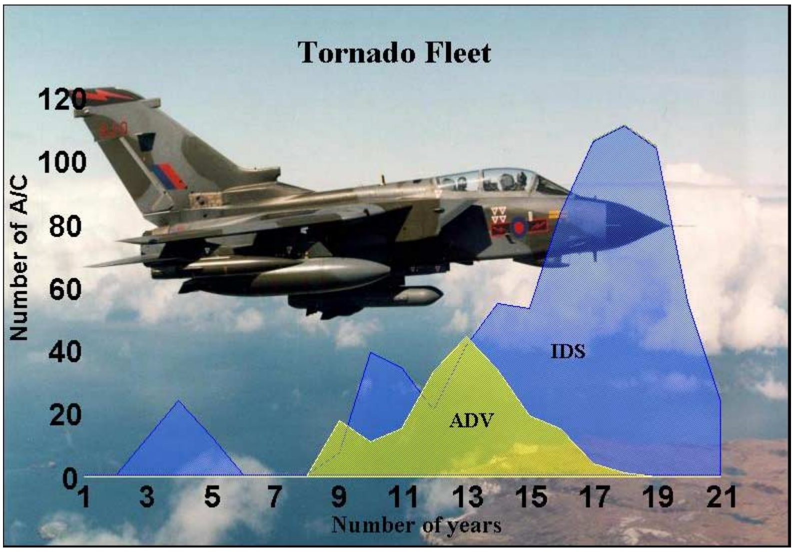


Figure 1: German Tornado Fleet

This does not mean that flight system components will remain in-service such a long time. Many equipment and components of systems are usually already been replaced during scheduled or unscheduled maintenance by overhauled, repaired or new items. A uniform in-service life for all system components in an aged aircraft does not exist. It is therefore important for critical equipment to record in-service operational and maintenance data in order to be able to assess the usage history and life consumption and define the point of retirement.

3. Reliability / Safety of the concerned items

The Reliability of the considered equipment decreases since the rear field of the bathtub curve is achieved because the ageing and/or wear caused malfunctions increases. In this way, the spare part demand and the number of the maintenance actions increases. These result in higher operating costs and reduced Availability.

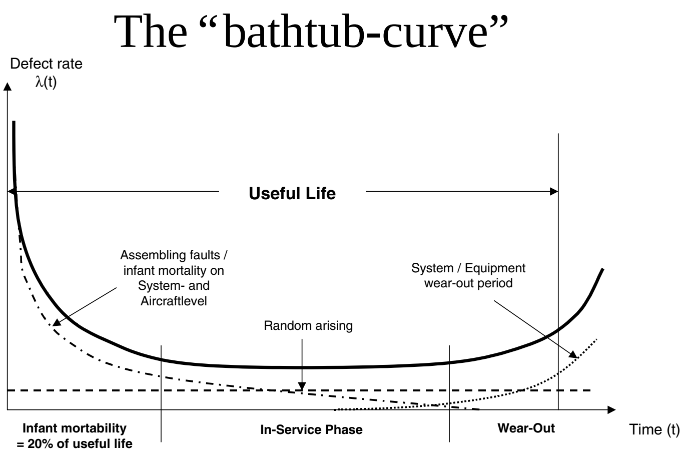


Figure 2: Bathtub curve

A measure within a life extension program would have to be determining the optimal point of time, when it is more economical to replace an item by a new part, a redesign or a modernised variant (upgrade) instead of to repair the old one again and again.

Therefore take into account, that so-called “old” equipment is often younger, than the aircraft cell/frame in which they are installed. Equipment will certainly fail during an aircraft life and one exchange is to be anticipated sometimes. To what extent-repaired equipment can then be classified, e.g. as a new value part, is also to be clarified. "Mixed equipments" are often to be found in an aircraft, where some single components have the original ageing, although other components have a younger date of manufacturer and/or days in use.
Decreasing Reliability also concerns the other aspect, Flight Safety. The use of a device after the specified life does not mean that the device immediately becomes not airworthy. Only the probability with which a failure can happen is increases and therefore the probability that a safety critical failure occurs!

However, in general the specification required the avoidance of safety critical failure, e.g. by bringing in redundancy within the equipment or on system level. Furthermore, the supplier has to provide proof that safety critical or safety relevant failure does not exceed a defined probability.
The proof is guaranteed normally by the preparation of Fault Tree Analyses (FTA), Failure, Modes, Effects and Critically Analyses (FMECA) and/or Defect Rate Prediction Reports.

4. Verifications of the trends at the example Tornado

To give an overview of the ageing-conditional failure rate of the different basic systems in a period of 10 years, an analysis of the documented defect data was carried out at the example Tornado.

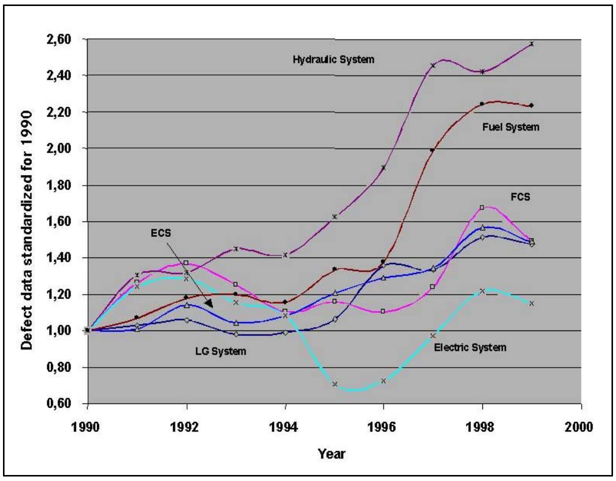


Figure 3: Document ageing-conditional in-service defect data standardized for 1990

The trend curves were constructed with the help of the in-service defect data for which corrosion, leakage, non-sealing and wear was identified as a failure cause.
The analysis was executed for the following basic systems:

- Landing Gear System (LGS)
- Flight Control System (FCS)
- Electric Systems
- Fuel System
- Hydraulic System
- Environmental Control System (ECS)

At this point should be mentioned that the analysis of the defect data is complex. Frequently no secured and sufficient descriptions of the failure cause are available. Therefore assumptions must often be met in the evaluation. A trend can nevertheless be derived from the curves, which confirm the increase of the failures in relation to the useful life.
A significant increase of the failure rate comes out from the examinations within the last 10 years.

So the failure rate increased for Hydraulics System by 160%, for the Fuel System about 120% and for LGS, FCS and ECS about approx 50%. For the structure / cell arises a similar picture.

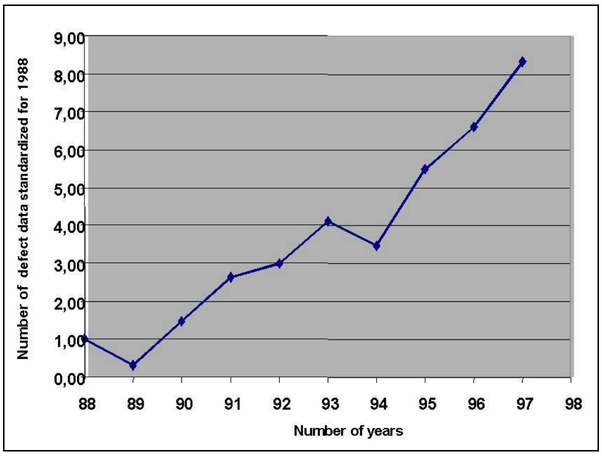


Figure 4: Result of the defect data with regard to the cell/airframe for the period 1988 to 1997 standardized for 1988

An accompanying analysis of the inspection distribution expenditure shows Figure 5. Therefore the biggest interest approx 61% fell in 1980 to the visual inspection, while the scheduled and unscheduled life extension measures contributed to 8% respectively 31% to the whole inspection expenditure. The unscheduled life extension measures are measures, which are respectively on the examination of repaired components parts or components parts with supposed failures.

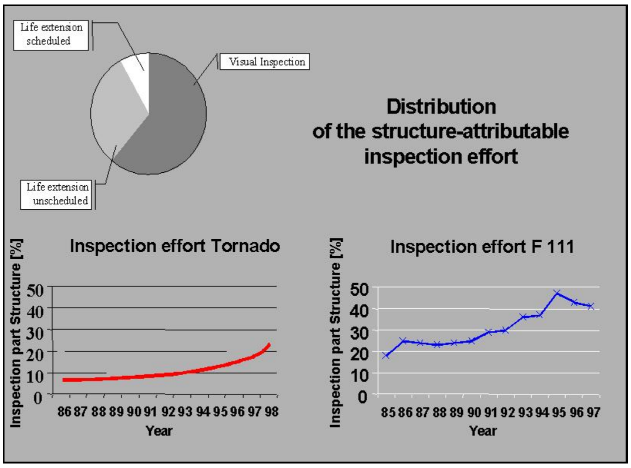


Figure 5: Distribution of the structure-attributable inspection effort

An examination of the distribution, which reflects the period of the last 5 to 10 years, shows an unambiguous shift. Therefore not only the quantity of inspection instructions has increased, but the distributions of the inspection demands have changed. This stands in the direct connection to the represented increase of the basis failures in Figure 3 and 4.

5. Ageing Mechanisms

Ageing is understood as a process, where the structural and/or functional integrity of equipment/components will be continuously degraded by the exposure to environmental conditions, under which the equipment is operated.

This could lead in the worst case to a situation where the aged equipment cannot fulfil any more its design function; even before the design life of the equipment is reached. In this case, system functions, which the equipment has fulfilled or supported, could be lost or degraded, which may also affect Flight Safety.

There are various mechanisms, which alone or in combination are responsible for equipment/component ageing.

- The exposure to normal or salty atmosphere, heat, water, oil, fuel, etc., which could lead to corrosion, overheating, melting or other material degradation, electrical interruptions, short circuits, etc..
- The exposure to vibration and acoustic environment, which could lead to fatigue damages, wear and tear, etc..
- The endurance, which lead to leakage, wear and tear, etc..
- The maintenance activities, which can induce accidental damages, could be a particular problem for wiring.

It is likely that Reliability and Availability of the equipment will be impaired by these influences.

Additionally, incorrect installation of equipment could also have a detrimental effect and support the ageing process, e.g. enable scouring of wire bundles on surrounding structure (this can be a result of poor design but also of poor maintenance).

Aged equipment/components will be removed from service, if further use is not recommended for Safety reasons or if further safe operation is uneconomical due to required inspection and maintenance activities.

Life expired equipment/components can be defined by the fact that any specified and certified life limitations are reached by the in-service usage and therefore need to be replaced.

Life extension might be possible and needs to be investigated. In the event that the design and certification authority permit further on aircraft operation under clearly defined conditions, further use of the equipment can be tolerated without formal re-qualification and for a limited time. As the aircraft runs out of certification (e.g. Tornado > 4000 FH) and the formal certification process up to 8000 FH has not been completed yet, the definition of such conditions and requirements for further usage up to 5000 FH is e.g. part of the Tornado life extension program.

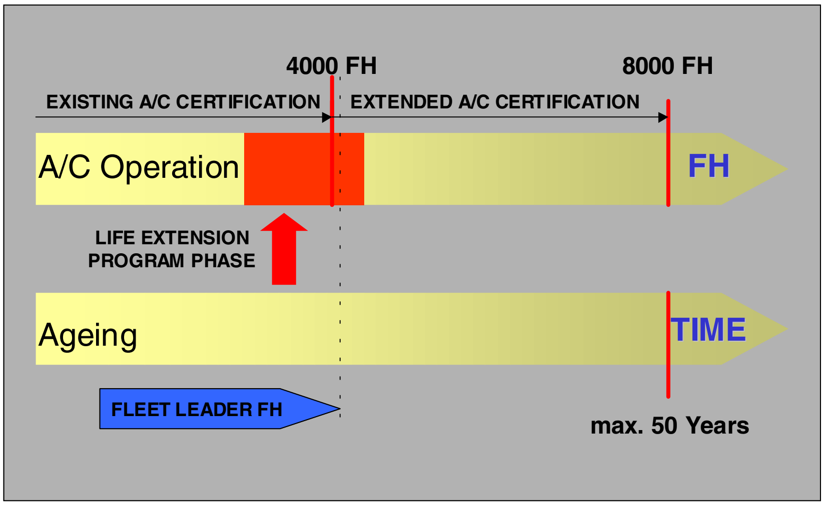


Figure 6: Life extension demand

6. Influence of maintenance on ageing

As already mentioned, ageing of equipment/components cannot be prevented but slowed down. It is reasonable to say that if maintenance is poor or not timely conducted certain ageing effects (e.g. corrosion, contamination, wear and tear, etc.) could be accelerated or even induced.
This means that the physical condition of aged equipment/components in different aircraft can vary due to different quality of maintenance carried out, which could result in earlier retirement of components than assigned. When investigating the condition of wiring in different aircraft from different Nations this aspect became obvious.

Aircraft maintenance programs will always be a compromise as beside other logistic costs and overall fleet management are important issues. It is important to include preventive maintenance actions for critical equipment and areas in the existing maintenance procedures to ensure that ageing problems will be detected at an early state, where the effort for repair is still acceptable and an airworthiness critical situation avoided. This may lead to shorter intervals in periodic servicing schedules or in special inspections.
So within equipment Life Extension Program ageing effects on equipment/components should be investigated as far as they are obvious and known. The manufacturer, to establish revised maintenance procedures for his equipment, where necessary, will use the results of that investigation.

7. Problems with increased aircraft age

There are some factors that have an effect on the problems associated with aircraft ageing, e.g. the usage of the aircraft, the environment the aircraft is subjected to and the inspection and maintenance practices and tools.
Depending on how an aircraft is used, the aircraft may have an expended life significantly different from what is predicted for that aircraft at that time. The simple fact is that aircraft are often not used the way they were intended to be use when they were designed and commissioned into the fleet. The effect is that an aircraft that is used harder than expected will have higher cumulative equivalent flight hours than expected. As a result, that aircraft may have a higher damage state than is predicted and may have a correspondingly higher probability of failure.

Depending on where an aircraft is used, an aircraft will have corrosion problems from the environmental conditions. This is most prevalent for military aircraft. Military aircraft that are stationed near saltwater environments experience a higher degree of corrosion than other military aircraft.

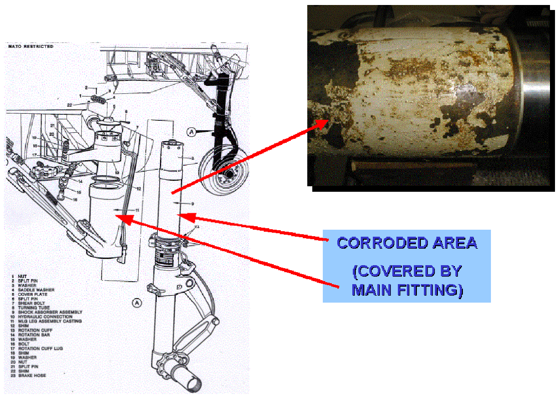


Figure 7: Corrosion Main Landing Gear Shock Absorber

The predisposition to corrosion depends, additional to environmental conditions, on further factors:
- Corrosion resistance of the material
- Combination of material
- Surface protection and sealing material

The problems, which can hang together with corrosion, have the following causes:
- Penetrations of humidity in the structure in case of faulty coat of varnish and sealing materials.
- Penetrations of humidity in (dry) hollow cavities
- Lack of adequate ventilation and drainage
- Choice of a not optimal surface protection and sealing material
- Contaminated fuel

However, these military aircraft are probably the most carefully monitored and cared for aircraft as far as corrosion is concerned.

Besides the ageing problematic other topics, which are not related to ageing but cause problems in an aged aircraft, need to be considered and investigated:

- Usage history, life consumption

As required by formal certification rules, equipment, which exceeds stated life limits are formally out of certification. Therefore, the usage history needs to be established and this requires the Availability of complete in-service documentation (FRACAS, Maintenance documentation).
Unfortunately, it has been experienced that it is sometimes very difficult to get all the information, which is required to assess service life and life consumption. For Safety relevant and life limited components this causes many problems, as it might be not possible in these cases to establish whether equipment is already life expired or not.

- Repairs and concessions

The influence of repairs and concessions on life limitations needs to be assessed.
Normally, minor concessions should not affect life limits, but if the operational conditions have changed in the past, the classification of concessions needs to be reassessed.

- Obsolescence

Obsolescence is a problem, which becomes more problematic with increasing age of the aircraft.
Materials and components, which have been used in the original design, could be no more available and this could require re-design and re-qualification activities if no proofed and certified alternative exists. Especially for electronic parts obsolescence is a critical and costly issue. The technology progress over the last 20 years was so enormous that certain items are only still be available in a limited quantity. A strategy for future support of that equipment needs to be established.

- Availability of original supplier

For most equipment there was only one supplier selected, who has designed and qualified the item. This supplier might have disappeared or no more able to produce the required equipment. Introduction of alternative supplier will cause problems in terms of time and costs. Dependency on one supplier is critical but usual.

- Costs for procurement of equipment

The procurement of equipment for replacing aged or life expired equipment after 20 years in-service could be very costly as e.g. original suppliers are no more existent, design and qualification of new components are necessary or the required number of equipment is very small.

- Costs for maintenance actions

The expenditure for maintenance rises continuously for the operation lifespan. S.G. Sampath published an examination about the maintenance expenditure of the F 111A. Approx 2200 Man Hours (MH) per aircraft were required in 1985 for the execution of inspections and repairs. In 1996 the expenditure was estimated for the maintenance the F-111A already at 8000 Man Hours (MH).

- Failure and maintenance reporting systems

Most aircraft users maintain comprehensive databases to collect and evaluate failure and maintenance reports for the whole aircraft down to equipment and component level in order to identify unreliable equipment and other problem areas. In the German Forces we have established the tool WIDAV (Maintenance / Inspection In-Service Defect Data Acquisition and Evaluation System). In our Reliability department we have established a special Failure Reporting Analysis and Corrective Action System (FRACAS) tool called DEMON (Deviation Monitoring).

However, in-service data collection systems are mainly defined by the aircraft users and reflect their specific needs and points of interest. The definition of data elements and the level of detail may significantly vary, even between the different users of the same aircraft type.
For this reason, the probability, that failure and maintenance data collected during service is compatible with design analysis and predictions is fairly low.
Moreover, in-service data are not always available to the system design authority at a required level.
The value of an in-service database can only be as good as the input data. It is therefore important to consider that any analysis or curves derived from databases needs specific interpretation and should be taken with care.
The major " lessons learnt" for future data collection systems are therefore continuous availability to both user and manufacturer and the need for compatibility with design analyses, in order to gain maximum benefit over the aircraft lifetime.

8. Concept for ageing aircraft

For a total concept the problems appearing during the qualification (service life test), as well as the effect of the combination by material fatigue and corrosion must be considered.

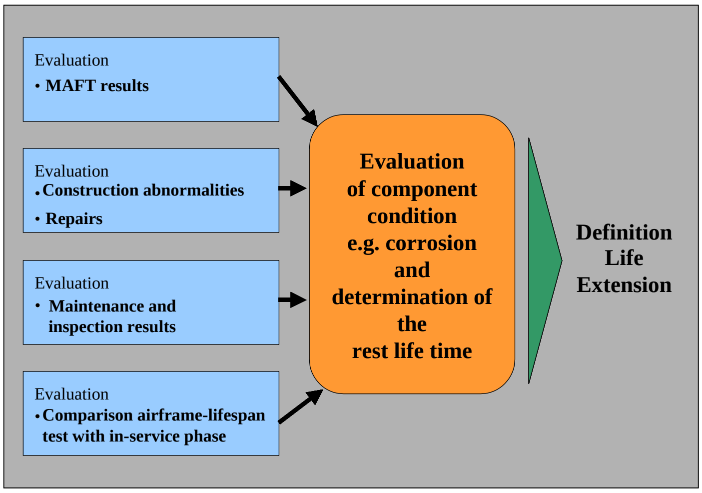


Figure 8: Concept to the preservation of the structural integrity beyond the original design lifetime.

The continuation of the extension of in-service life can be reached in principle by different measures:
- Exchange of parts to a fixed point of time
- Incorporation of a simple lifespan increasing modifications (how cold worked of drillings and surfaces)
- Combination of inspection and exchange to minimize downtimes.

In principle some of these life extension measures can be executed with the regular small and big inspections.

8.1 Evaluation of repairs and construction abnormalities

During the in-service time of an aircraft individual repairs are inevitable.

Reasons for this are e.g.:
- Damages cover in the flying operation by wear, hard landings, accidents...
- Bird strikes
- Damages during the maintenance
- Re-equipments
- Corrosion

Repairs are defined or are judged normally for the full design loads in regard to the demanded lifespan, just as the construction abnormalities appeared during the production. Otherwise an entry would be required in the logbook of the aircraft.

8.2 Evaluation of inspection and maintenance results

During the aircraft utilization phase a database for structural defect data should be established. With the help of the database recurring defects can be identified and thru measures reduced or removed.

Special measures should be considered:
- Corrosion including tank-space corrosion
- Wear
- Edge erosion in fibre construction units,
- Damages in small parts
- Damages in maintenance-intense areas

as well as:
- Damages in connecting elements
- Damages in hardly detectable structure areas

8.3 Comparison of the airframe-lifespan test (MAFT) with in-service phase

A re-valuation of the results of the Major Airframe Fatigue Test (MAFT) is executed by comparison of the introduced test loads with the real in-service spectra.
So it is possible, e.g. that spectra are lower in the flying operation than in the MAFT-test, but the individual component part damage is higher.

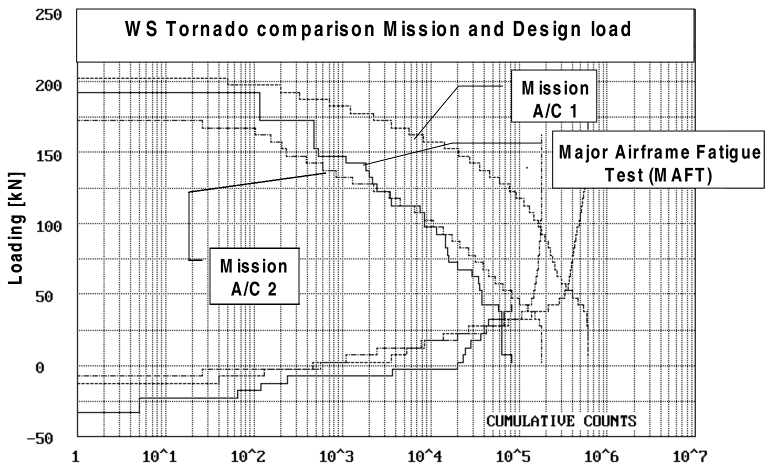


Figure 9: Comparison Mission and Design load at the example Tornado

9. Technological inception to life extension

A life extension requires a high expenditure in Man Hours (MH), in the definition, retrofit as well as in the future maintenance. The use of new technologies can reduce the expenditure by an aimed operation and herewith carries to an expense-optimised life extension for ageing aircraft. At this point only three technologies are named which possess a high potential to optimise life extension and maintenance cost.

These are:

- Automation of inspections
- Introduction of a Health Monitoring System
- Monitoring of critical component parts by the use of sensors (Damage Monitoring)

9.1 Automatically inspections

How already in chapter 4 represented the inspection expenditure rises continuously for the maintenance of ageing aircraft. The automation of inspections, by means of inspection robots, allows the reduction of the test expenses by the substituting man for machine and the direct data processing. This technology is also advantageous from a safety point of view where a danger to the technicians might exist.
However, at the moment the acquisition of an inspection robot is still expense-intensive, on account of the high expenditure to the navigation, control and image processing. But in the future cost might sink. The subject of inspection automation is mentioned only of the completeness. No following information are available at that point.

9.2 New Philosophy

A new philosophy will be the installation of the Damage Tolerance concept.

Distinction:

- "Safe Life" means that a structure sustains a specified period of lifetime without cracks (no inspections).
- "Damage Tolerance" is the property of a structure to sustain defects or cracks over a specified period of lifetime.

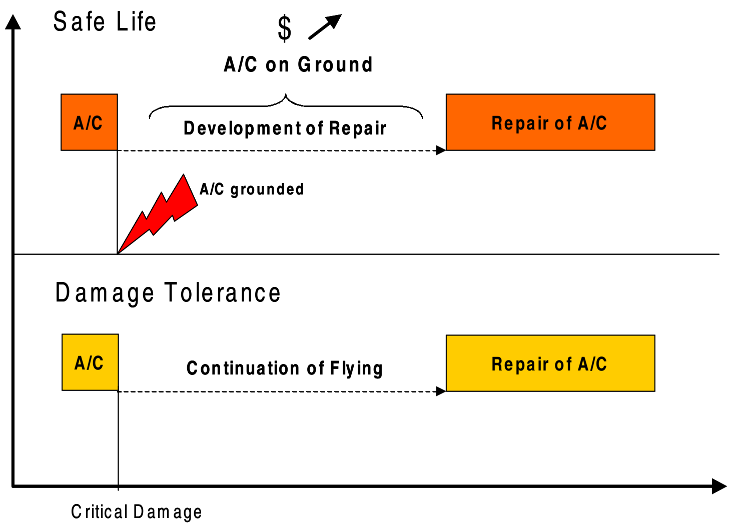


Figure 10: Comparison of Safe Life and Damage Tolerance Philosophy

Which are the advantages of Damage Tolerance Philosophy in respect to Safe Life Philosophy?

- Increased service life, "Retirement for Cause," (significant cost savings for customer)
- Increased ratio between Safety and Structural Measures
- Better planning of structural maintenance, better Fleet Management and individual aircraft tracking (significant cost savings)
- Increased Operational Readiness
- Better handling of upgrades and increased usage scenarios

9.3 Usage Monitoring

Usage Monitoring stands for:

- A monitoring of the structure and systems with the aim to steer the scheduled material-preservation-measures and to guarantee therefore an optimised fleet management.
- For the monitoring of the events which can lead to unscheduled measures, how e.g. Over-g or hard landings.

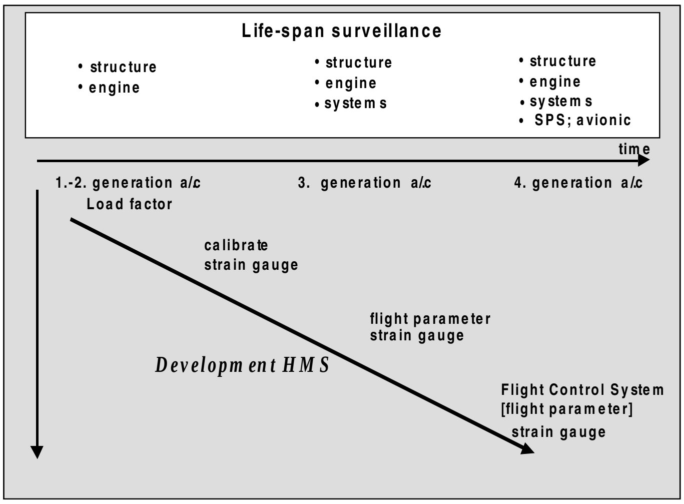


Figure 11: Overview Usage / Health Monitoring development

There are different options to realize Usage Monitoring.

- Development and integration of a reasonable Usage Monitoring which offers the possibility for trend analysis or
- Development and integration of an efficient Usage Monitoring as an integral component of the avionics structure [on aircraft or off aircraft].

9.4 Future Usage Monitoring systems

The trend in the development of Usage Monitoring systems goes unambiguously to the integration in the avionics structure of an aircraft. In this connection, one uses the possibility that the IPU (Interface Processing Unit) can provide all flight data. Therefore all relevant flight parameters of the data buses (FCS: Flight Control System, UCS: Utility Control System, AVS: Avionic System) are available for the calculations.

For example, the Eurofighter Usage Monitoring is subdivided:
- Engine Usage Monitoring
- Structural Usage Monitoring
- SPS (Secondary Power System) Usage Monitoring
- Aircraft-System Usage Monitoring

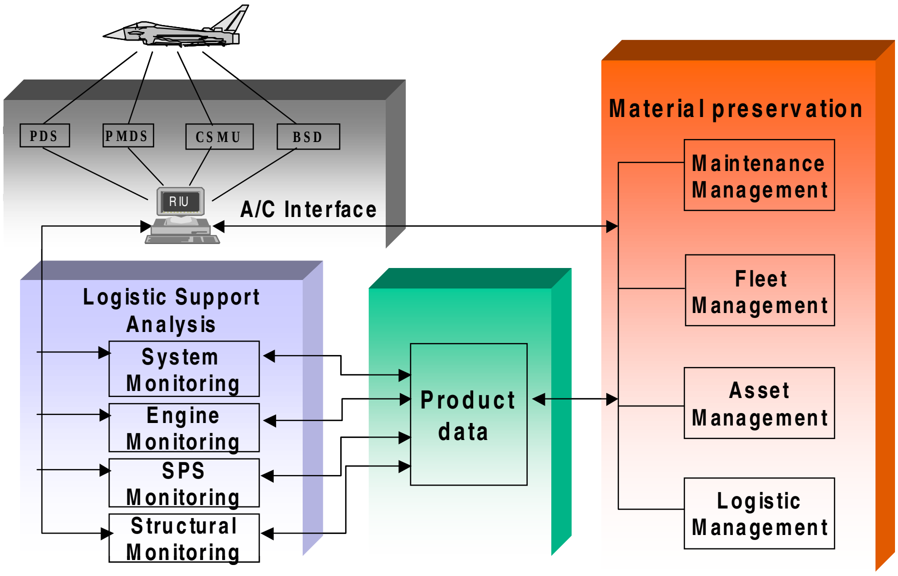


Figure 12: Data processing at the example Eurofighter.

9.5 Health Monitoring

With the development of microelectronics new ways are presented to monitor structures and systems by means of sensors. Today we are in the position to be formed the dimensions and the Reliability of the sensors so that they can be used at reasonable price in the application for the monitoring of weight-optimised structures. Also increased computer-performance offers the possibility of fast signal processing. Therefore the possibility insists to specify the on-condition inspection as an integral part of a Health Monitoring system and therefore the maintenance additionally optimised, i.e. reduction of the Life Cycle Costs. In the technical literature one speaks of Damage Monitoring. Within an integrated Health Monitoring system the on-condition inspections would automatically be executed. The resulting advantages being e.g.:

- Avoidance of expense-intense conventional inspections life extension in hardly accessible areas
- Increase of the inspection intervals
- Increase of the Availability of aircraft
- Increase of the useful life without direct repair measures

The principle of a Damage Monitoring system consists of sensors, signal-amplifier, filter and signal processing. This concept allows the collection of signals, which are caused by the damage directly (comparison of the loads before and after the damage event).

Concerning the collection of the signals all sensors are suitable which are able to measure frequency, which are produced by loads or damages. Three types of sensors are suitable particularly for an integrated Damage Monitoring.
These are:

- Fibre-optical sensors
- Piezoelectric sensors
- Sensory folios

Fibre-optical sensors are interesting because of their low weight, low power demand, long lifetime and low expense. Fibre-optical sensors have particularly proved themselves in damage identification by means of acoustic emission. The possibility to imbed in fibre composite materials is a special advantage.

Piezoelectric sensors are used traditionally to measure acceleration, which result from low or high frequency (vibrations). The piezoelectric sensors are producible in almost arbitrary size. With the help of this technology new applications and possibilities of structural monitoring arise. A very promising application in the aircraft construction is the use of sensory foils. In this connection, thin piezoelectric sensors are embedded together with the wiring between thin foils. These foils can be bonded onto the structure or be embedded in a fibre group structure.

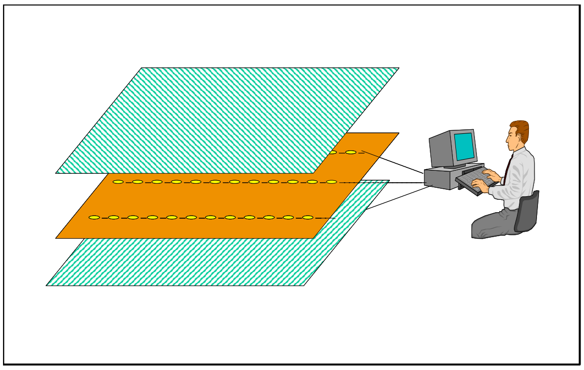


Figure 13: Construction of a sensory foil.

With the use of sensory foils a new level of on-condition monitoring is exhibited.

The damage monitoring could then become an integral component of lifespan-monitoring. It would be possible to monitor CFC (Carbon Fibre Composite) repairs. Therefore life extension and scheduled maintenance measures could be defined. It would be possible to maximize the lifespan and to optimise the retrofit point of time.

10. Conclusion

With a high probability the use of an aircraft changes during the whole operation lifetime. Therefore, it is necessary that the flight operation is monitored continuously by individual measurements and on-condition inspections. Only thus can Mission Reliability of the aircraft within the qualification or certification be guaranteed.

The Life Cycle Costs (LCC) prediction considers the increase at the end of the design lifetime. But no LCC development predictions existed when an aircraft oversteps the defined design lifetime. As already mentioned an increase of the life cycle cost is expected, which is caused by increased inspections, scheduled and unscheduled repair measures. A definition of the life extension measures must consider these aspects. So the aircraft can be further operated economically.

Life extension measures are inevitable with ageing aircraft. In the definition of the life extension measures the maximization of the remaining lifespan and the optimisation of the life extension retrofit point of time is the aim. Only then can a flexible fleet management be offered. A Health Monitoring System could perform here an essential contribution. With an HMS the maintenance measures could directly help support the material preservation. On-condition inspections become an integral part of the lifespan-monitoring.

In conclusion it should be mentioned that a Usage Monitoring is necessary during the utilization phase. A Health Monitoring System need not be employed at the beginning of the in-service phase. The damage monitoring should nevertheless be a modular part for the development of a new aircraft. That is the basis for an economic integration of the damage monitoring as supplement to the Usage Monitoring of the aircraft. The retrofit of a Health Monitoring System in existing ageing aircraft should also be taken into consideration. But it requires a cost-benefit analysis to justify investment in a Health Monitoring system.

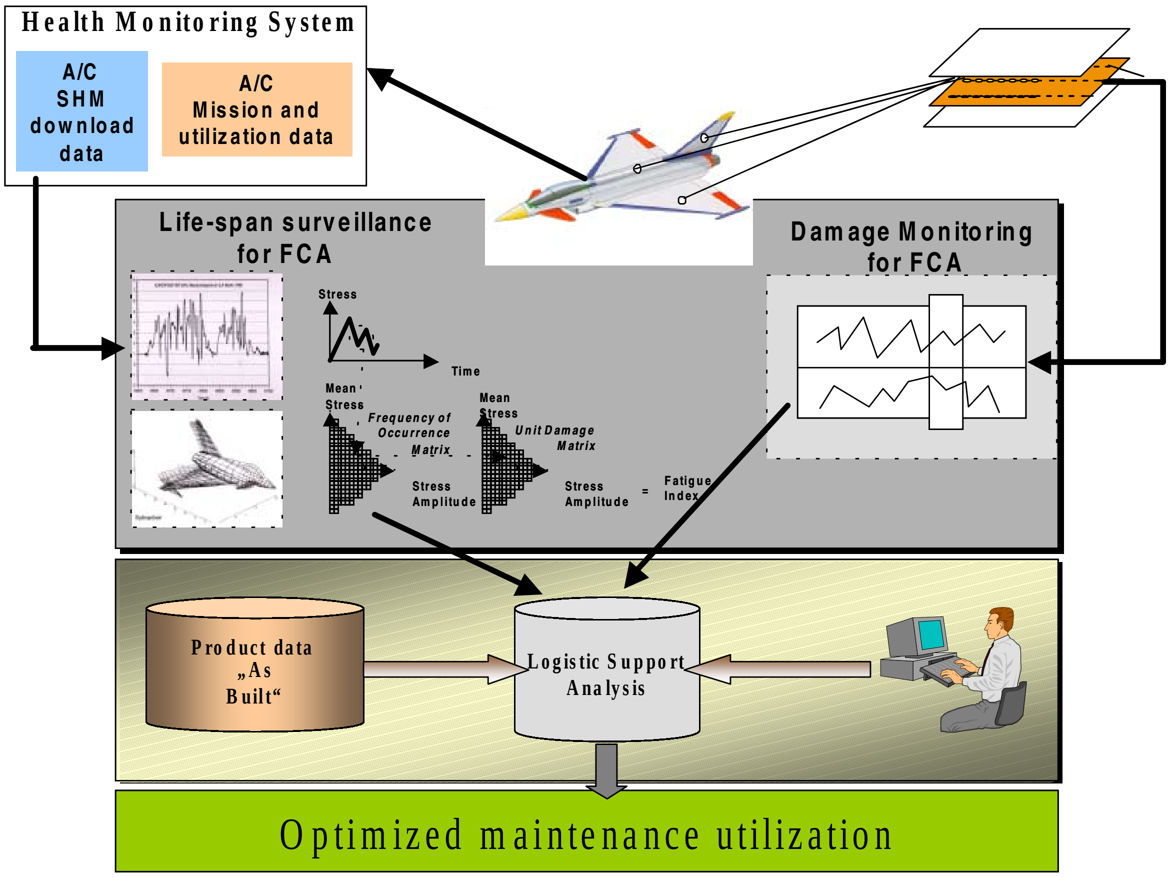


Figure 14: Total concept of the load and damage monitoring at the example Eurofighter

References

"Service Life Extension Assessment", Ned Criscimagna, Richard Unkle, Reliability Analysis Center, Rome, NY, USA, September 1995.

"Aging Aircraft Subsystems Equipment Life Extension within the Tornado Program", G. Blech, PANAVIA Aircraft GmbH, Hallbergmoos, Germany

"Transall C-160 Life Extension and Avionics Upgrade Programs", P. Blumschein, EADS Germany GmbH, Manching, Germany, April 1999

"Aging Combat Aircraft Fleets – Long Term Applications", S.G. Sampath, 1996

"Philosophien in der Entwicklung und Definition von Nutzungsdauerverlängerung am Beispiel Tornado und die daraus folgenden Auswirkungen auf die Instandhaltung", M. Buderath, EADS Germany GmbH, Ottobrunn, Germany, June 2001

This page has been deliberately left blank

Page intentionnellement blanche

Management of Corrosion in Aging Military Systems

Graham Clark
Defence Science and Technology Organisation
Aeronautical and Maritime Research Laboratory
Fishermans Bend
Victoria, Australia

Abstract

This paper discusses research in Australia into the impact of corrosion on structural integrity, and reviews the steps being taken to improve the capability of the Australian Defence Force to manage corrosion in aircraft. The aim of the research discussed is to develop useful methodologies which are similar to those already in place for fatigue management and which can therefore be introduced relatively easily, and the paper discusses some of the implications of pursuing this goal of absorbing corrosion into our structural integrity management approaches. The research has already achieved some useful developments in assessing the impact of some types of corrosion, and the paper will discuss these highlights briefly to illustrate the methodology being used.

1. DSTO approach to corrosion management.

1.1 Background

Increasing economic pressure has encouraged extension of the service lives of many military aircraft fleets well beyond their original design goals. Since the incidence of corrosion tends to increase with aircraft age, its importance as a life limiting form of degradation has increased in these fleets.

The Defence Science and Technology Organisation (DSTO) has provided scientific and technical support to Australian Defence Force (ADF) aviation for many decades, and as part of this periodically undertakes reviews of the engineering problems affecting military aviation in Australia. The author and a colleague Dr Bruce Hinton visited all ADF aviation bases in the early 1990's as part of this program [1], and observed a common thread to the discussions with base personnel - their concern about the rising cost of corrosion repairs, and the increasing impact of corrosion-related maintenance on aircraft availability. Obviously, this would become a more severe burden with aging of the ADF fleet, directing attention to the need for a strategy for management of corrosion. Examples of the cost burden of corrosion, for detection, repair and repainting of some aircraft types in the ADF fleet, are shown in Figure 1; these are broadly comparable with the experience of other military operators.

Obviously, the introduction of more effective corrosion preventive measures would be a cornerstone of any strategy; key measures identified were:

(a) improved training in corrosion recognition and treatment.,
(b) wider use of corrosion preventive compounds (CPCs) during regular maintenance, and
(c) washing of aircraft with water containing inhibitors.

DSTO initiated research to develop each of these areas [2], examining the effectiveness of CPCs, for example, and preparing a handbook for ADF personnel on corrosion recognition [3]. Introducing these approaches to service is still a key target for DSTO and the ADF. The corrosion control aspects of the approach – a program led by Dr Bruce Hinton – have been particularly successful in reducing the progression of corrosion using Corrosion Preventive Compounds

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

(CPC's);. Since 1993, there have been many cases of corrosion which have been arrested by the one-time or repeated application of CPC's; examples include tailplane corrosion and centre section corrosion in the Macchi MB326H, and the stress corrosion cracking (SCC) in BL20 longerons and main landing gear vertical beams in C130. The latter case has been managed for approximately six years by inspection and CPC application, with no evidence of significant additional SCC growth. Concern still remains, however, about our ability to adequately assess the impact of the corrosion which, while inactivated, remains in the component and will impact the structural integrity management of the aircraft.

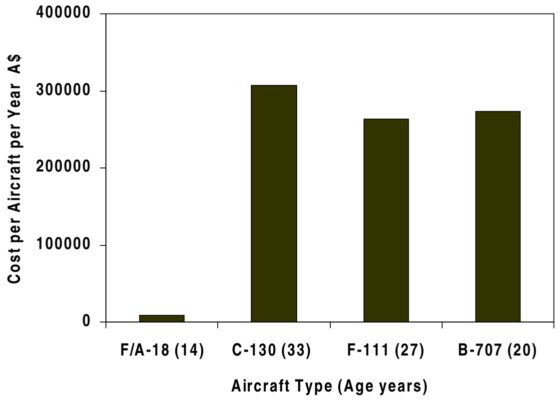


Figure 1. Cost of corrosion for examples of aircraft types from the ADF fleet.

1.2 Corrosion impact.

The ADF has experienced several cases where the presence of corrosion raised uncertainties over the continued airworthiness of some RAAF aircraft. These cases were resolved, but they showed how structural integrity concerns associated with the detection of corrosion can lead to reduced aircraft availability and substantial increases in maintenance and support costs. Examples which were particularly significant in terms of reduced availability of aircraft included the replacement of spar caps following discovery of stress corrosion cracking in P-3C wing rear spar caps, and in Macchi MB326H tailplane spar caps [4]. In both cases, it had been observed that the replacement of components was necessary because no suitable methods existed for analysing the impact of the corrosion damage on static strength and fatigue performance. A further example – stress corrosion in BL20 longerons in C130 aircraft – led to attempts to assess the corrosion by representing it as cracking, but with limited success because of uncertainty about the exact configuration of the damage and about the local stressing.

1.3 Decision-making tools for corrosion impact assessment

DSTO undertook a detailed review [5] of the possible approaches to managing the impact of corrosion on structural integrity. This review observed that the inefficiency in the "fix-when-found" approach to corrosion management arose from a perceived need to remove most corrosion whenever it is detected, a conservative approach required because corrosion lies outside the parameters normally forming the basis of structural integrity management. The problem of assessing corrosion impact was not confined to major fleet-threatening cases; methods for avoiding unwarranted repairs could dramatically reduce maintenance times, simply by allowing continued operation of aircraft with identified corrosion defects until a more convenient maintenance opportunity. The key proposal from the review was that the ADF adopt an "inspect and manage" philosophy, and that research focus on improved methods for corrosion assessment, to allow decisions to be made concerning the effect of corrosion on structural integrity, and hence to determine the need for repair.

The development of decision-making tools for assessing the structural integrity impact of corrosion formed the second part of the DSTO corrosion management strategy.

Only a small proportion of aircraft accidents and incidents are attributed directly to the presence of corrosion, but the potential for corrosion damage and corrosive environments to reduce structural integrity in aircraft cannot be assumed to be negligible. The review concluded that since the ADF already had structural integrity management plans in place for each platform, it was appropriate to incorporate corrosion into these structural integrity management approaches. It identified a research program which has been pursued by DSTO, and is developing tools to assist RAAF assess corrosion in a manner compatible with the current management processes for cracking.

1.4 Conditions addressed.

Aircraft design and structural integrity management approaches in current use are customarily validated using benign environments; the effects of either (a) corrosion as an initiator of fatigue cracking (corrosion/fatigue), or (b) corrosive environments accelerating fatigue crack growth (corrosion fatigue), are not sufficiently understood to permit them to be incorporated with confidence, and on a routine basis, into structural integrity assessments.

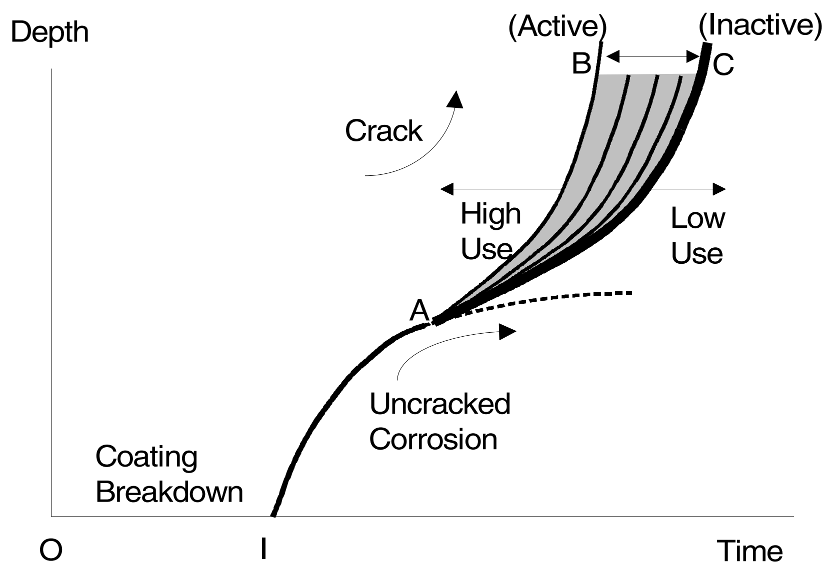


Figure 2. Schematic of the growth of corrosion and fatigue cracking with time, illustrating the significant parts of the life cycle.

A whole-of-life prediction capability for corrosion, and any fatigue cracking which develops from it, is clearly desirable, and research into the various life phases could well lead to valuable corrosion management methods. Unfortunately, reliable ab initio prediction will require resolution of many complex and uncertain factors [6], for example, coating breakdown time OI in Figure 2, and its dependence on coating condition and maintenance intervention is clearly significant, as is the complex interaction between fatigue cracking, environment and materials parameters (ie. the shift from crack growth AC in an inert environment to AB in a corrosive environment). Consequently, effective prediction capability will probably rely on data from fleet experience. The ADF fleet is relatively small in world terms, and in itself is unlikely to provide sufficient data to allow reliable prediction, although data from Australian experience could add significantly to other data. A key factor is the development of data management strategies.

Two approaches identified were (i) improved data collection and management to support empirical predictive approaches, and (ii) DSTO research into selected parts of the corrosion life cycle. This research, while generally supporting the longer term goal of a prediction capability, would focus on aspects which would yield useful tools for corrosion management. Accordingly, the DSTO approach was to focus on a limited scenario.

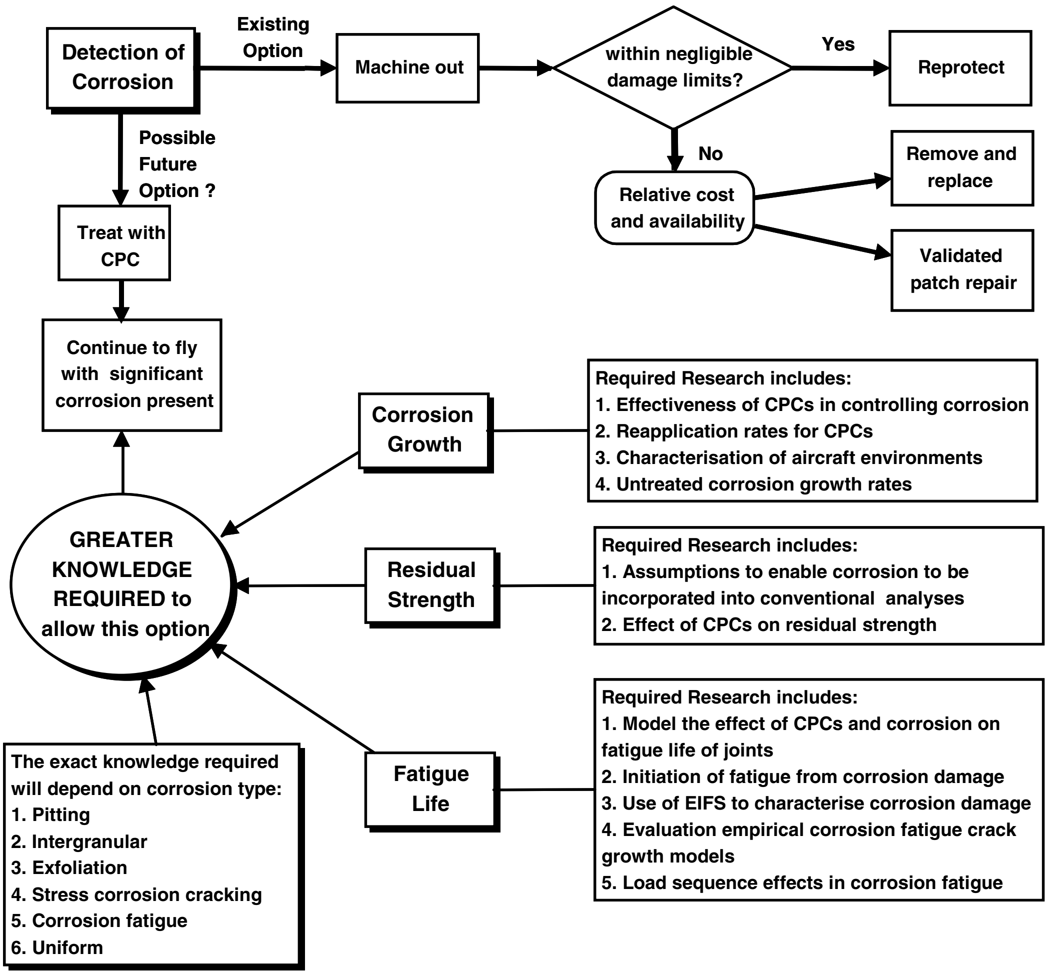


Figure 3. From [5]. Research required to allow continued component service after discovery of corrosion.

The first simplification was to consider the situation where corrosion is discovered in a structurally significant location, and to address potential fixes Figure 3 [5]. A key assumption and second simplification (made initially on the basis of limited experience with Macchi and C130 stress corrosion, but supported by subsequent experience) is that a program of application of suitable corrosion preventives can reduce the problem to an inert condition ie. no active corrosion. Hence the problem becomes one of growth of fatigue from a geometrical feature (line AC in Figure 2). Figure 4 represents the decision-making required to allow continued service of a component in such a situation. DSTO has been addressing this part of the life cycle by developing predictive approaches for the remaining life of parts containing such damage.

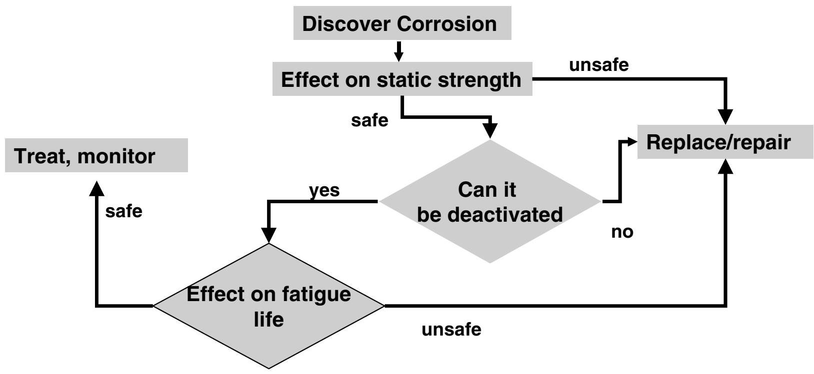


Figure 4. Simple decision tree for continued operation. [5]

2. Progress in tool development

The DSTO research program uses an Equivalent Precrack Size (EPS) approach, correlating features of the corrosion damage to notional fatigue crack sizes, and allowing simple prediction of remaining fatigue life. The different systems on which the DSTO program has focused are:
Fatigue from pitting corrosion, high Kt aluminium alloy
Fatigue from pitting corrosion, low Kt aluminium alloy
Fatigue from pitting corrosion in high-strength steel
Fatigue from exfoliation corrosion
Fatigue from laminar stress-corrosion
Impact of CPCs and environment on joint failure.

Prediction of remaining life for condition (c) ie. pitting in D6ac high strength steel has been particularly successful, with a good match between experimentally-determined lives of pitted specimens and the lives predicted from geometries deduced from measurements on pits. System (a), however, does not produce results which are as consistent; correlation of pit depth and the EPS is better for the steel (Figure 5) since the pits have a more regular morphology. In the aluminium alloys investigated, pit shape shows a higher level of variability.

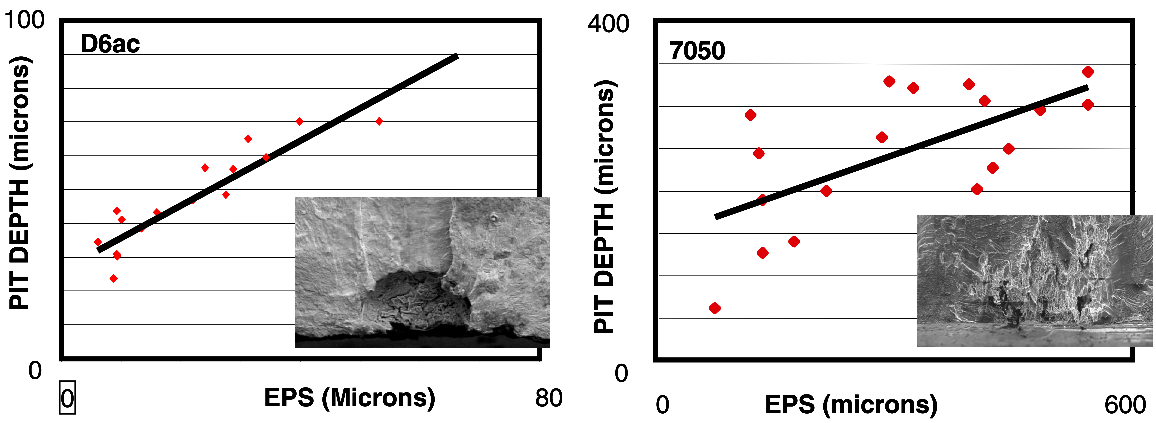


Figure 5. Correlation between pit depth and Equivalent Precrack Size for high-strength steel and 7050 aluminium alloy.

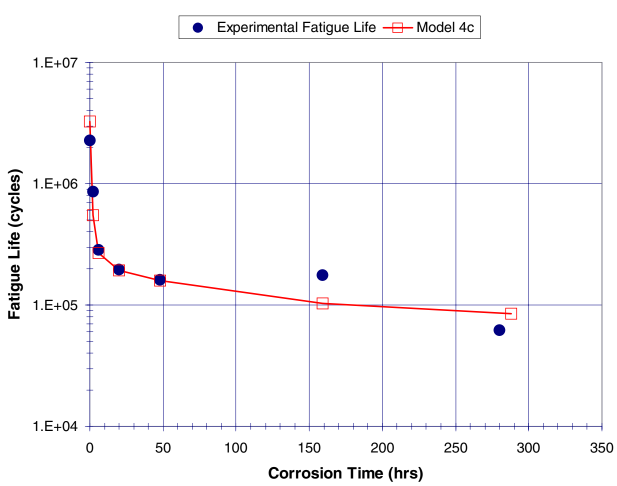


Figure 6. Prediction of remaining life for 2024-T351 aluminium alloy specimens with exfoliation and pitting.

DSTO modelling of crack growth from exfoliation [7] adopts a similar approach to the modelling for pits, by modelling the progression of exfoliation into the body of the material as a pitting-type process. Observation of the initiation sites for fatigue cracks supported this model, and allowed construction of a notch+crack representation of the process zone at the base of the exfoliation, giving excellent correlation with the experimental fatigue lives of corroded specimens (Figure 6).

3. Transfer to ADF aircraft.

To date, the modelling has assessed simple loading and simple geometries, but shows promise for extension to variable-amplitude loading, and the next stage, a transition to component and assembly testing. The extensive effort expended on protocols for generating corrosion similar to that observed in service is being used to introduce pits into a full-scale wing fatigue article in support of the RAAF F-111 Sole Operator Program.

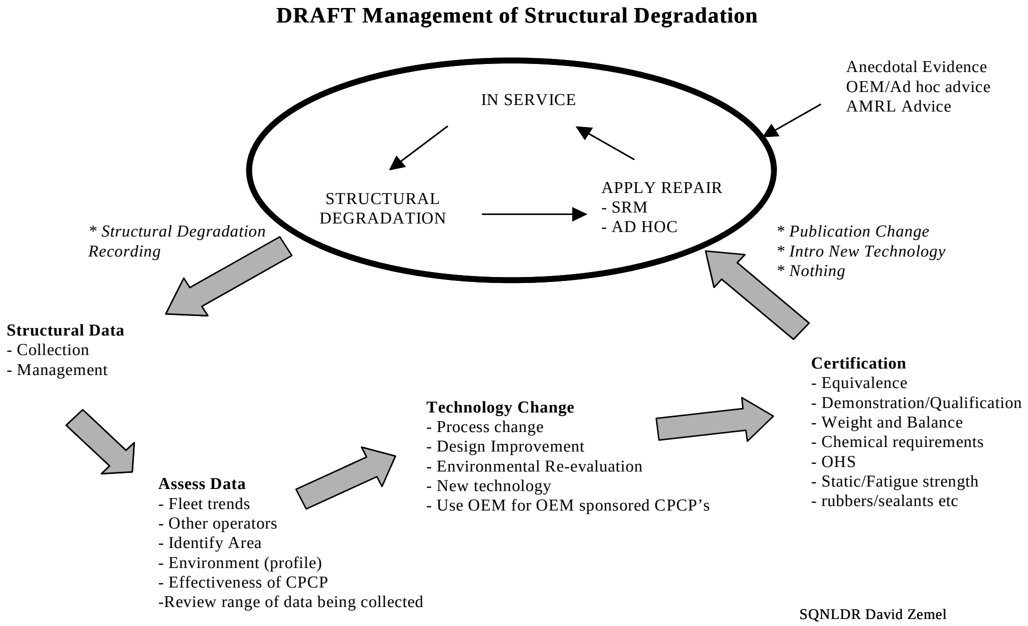


Figure 7. Draft representation of RAAF approach to management of structural degradation [8]. Key elements related to this paper include Structural Degradation Recording, and Technological Inputs from DSTO.

The RAAF manages the airworthiness of ADF aircraft, using Aircraft Structural Integrity Management Plans (ASIMPs) for each system, and is considering a use of the approach shown in Figure 7[8] for managing structural degradation. This approach will address the higher-level aspects of structural damage reporting, and the data storage and processing required to allow tracking of fleet condition, as well as identifying the technological inputs, such as those described in this paper, which are required as inputs to the management approach.

A key element in this approach is the need for structural condition monitoring. The RAAF recognises two main areas for data collection: usage and condition, each of which will provide input to the two key management systems; Fatigue Degradation Management (FDM) and Environmental Degradation Management (EDM). The two management systems will generate input for the fleet processes which control structural integrity and corrosion, and will also allow continuous improvement of the data acquisition and assessment processes being used as part of long term Service Life Assessments (SLA's). The annual work (whether fatigue management or condition monitoring) will be conducted under the ASIP cycle. Figure 8 shows more detail of the data flow through data collection and verification, followed by analysis under the FDM or EDM processes. It is important to note that the same condition and usage data, will be used as input to both management processes; the usage monitoring data will allow improved assessment of corrosion condition, and the identification of structurally significant corrosion will assist in managing structural integrity (by validating, refining or changing SBI and safe-life management programs). The environmental degradation (corrosion, etc.) and changes to the operating environment (usage) will be assessed under the EDM process, for the purpose of developing progressive changes to corrosion management of the fleet.

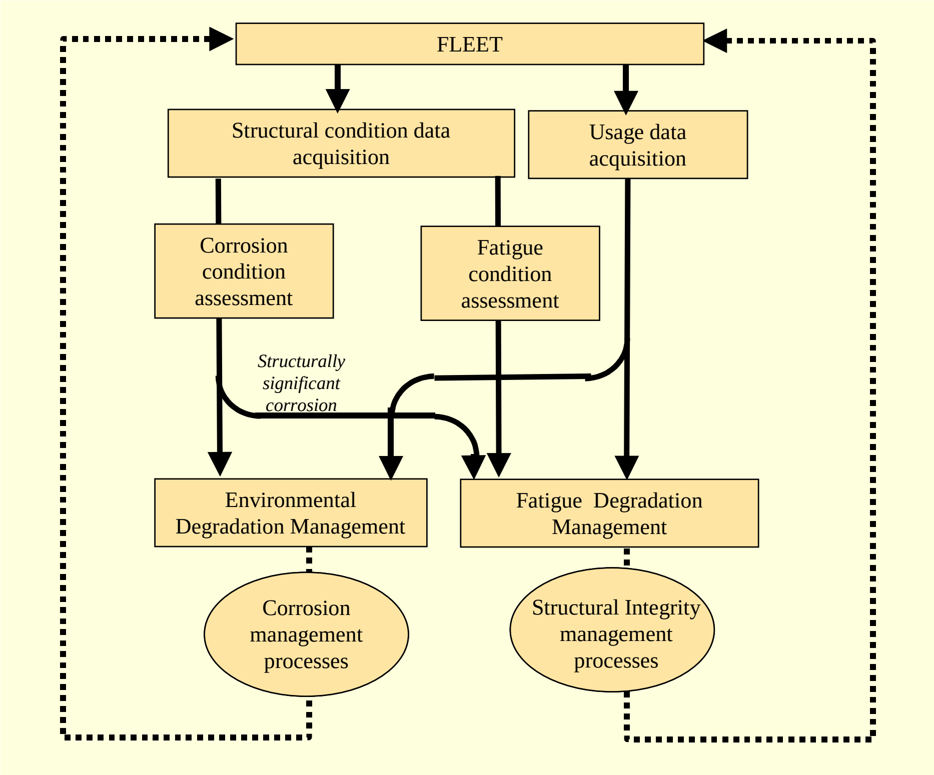


Figure 8. Structural degradation management in RAAF [8].

The RAAF approach is identifying a suite of technological tools for analysis, modification and repair, whose use for fleet operations will be managed under relevant Aircraft Technology Management Programs. The management of corrosion, more specifically, revolves around development of a guide for ASIP managers; much of this will involve adoption of OEM Corrosion Prevention and Control Programs (CPCPs) with modification, where relevant, together with the proactive aspects of corrosion prevention and control discussed earlier in this paper. The decision-making tools discussed in this paper will provide a basis for advice to the system managers who will still need to determine appropriate reactions to corrosion occurrences. It is intended that these tools will be used to address the (hopefully) infrequent occurrences which have high economic/safety impact and which cannot easily be managed by CPCPs or on-condition maintenance programs.

4. Acknowledgments

The author wishes to acknowledge the contributions of several individuals in DSTO and RAAF, to this program of research. Khan Sharp, in particular, was the major contributor to the pitting and exfoliation program referred to in this paper. Bruce Hinton provided information on application and use of CPCs, and SQNLDR David Zemel provided valuable input on the developing RAAF approach to corrosion management.

5. References

[1] "Survey of ADF Aviation Materials-Related Problems" Clark, G and Hinton, B R W, General Document 35, DSTO Aeronautical Research Laboratory, Dec 1992.
[2] "Research into Materials Aspects of Aircraft Maintenance and Life Extension-I (Corrosion Control)", Chester, R .J., Clark, G., Hinton, B. R. W., and Baker, A. A., Aircraft Engineering 65(1) Jan 1993

[3] “Aircraft Materials And Corrosion Protection Handbook” Document DI(AF) AAP7021.014-2 1992, Royal Australian Air Force.
[4] “Recovery of the RAAF MB326H Fleet; the Tale of an Aging Trainer Fleet” Clark, G, Jost, G.S. and Young, G., in “Fatigue in New and Ageing Aircraft” (ed. Poole, P and Cook, R) EMAS, Warley 1997
[5] “The Implications of Corrosion with respect to Structural Integrity” Cole, G.K. , Clark, G. and Sharp, P.K., Research Report, DSTO-RR-0102, Defence Science and Technology Organisation, April 1997.
[6] "Corrosion and the management of Structural Integrity", Clark, G., in " Structural Integrity for the Next Millennium " (ed. J L Rudd) EMAS, Warley 1999.
[7] "The Effects of Exfoliation Corrosion on the Fatigue Life of Two High-Strength Aluminium Alloys." Russo, S., Sharp, P.K., Mills, T.B. and Clark, G., Fourth DoD/FAA/NASA Conference on Aging Aircraft, St Louis, May 2000.
[8] SQNLDR D Zemel (ASI4), personal communication 2001.

Corrosion Management of the Italian Air Force Fleet

Capt. Mario Colavita, Capt. Fabrizio De Paolis
Chemistry Dept. of CSV - Italian Air Force
"M. De Bernardi" Airport
00040, Pomezia (Rome), Italy

aermatcnd@tiscali.it

ABSTRACT

Economic, safety and logistic issues are strongly affected by military aircraft corrosion, more when it regards an aging fleet[1-3].
Italian Air Force manages this matter by a Corrosion Control Register Program (CCR), a flexible and integrated support for making decisions both on prevention and operational measures.
The Program was born in 1994 as a necessary instrument to enhance partnership, in Tornado maintenance[4], with the German and the Royal Air Forces, but was immediately extended to six more aircraft (AM-X, MB.339, C-130, F.104, G.222, and Br.1150) and to two helicopters fleets (HH-3F and AB.212).
Some results of the activity carried out along these years, expressed in terms of useful information for the decision-makers, are shown.

Keywords: Corrosion Control Register Program, CCR, Italian Air Force, aging aircraft, Tornado.

1. INTRODUCTION

For too many years aircraft corrosion has been considered just as a technical matter that could be solved by means of a "found and fix" policy, while not enough attention has been given to the economic and safety factors related with it.
Nowadays, the need experienced all over the world to manage an ever more aged fleet gave back to corrosion a new interest, because of its strong connection with the increasing maintenance cost on one side and the difficulties in understanding the actual residual mechanical properties of a corroded part on the other.
This is the case of the Italian Air Force, that for a long time had not conceived any specific program about corrosion control except for a uniform training program for people working in maintenance, started in the eighties, and organized in two different levels depending on responsibility.
When, in 1994, Italian Air Force had to create the Corrosion Control Register in order to manage the Tornado maintenance in agreement with German and Royal Air Forces procedures, this instrument was made as flexible as possible in order to be extended to many more fleets. The Italian Corrosion Control Register Program (CCR) was started at the same time for six more aircraft and two helicopters fleets: since then a corrosion data bank managed at the Chemistry Department of Italian Air Force is updated each time corrosion is found at any step of the maintenance inspection carried out by the Air Force.
This Program doesn't cover the maintenance performed by any depot outer the Air Force.

2. BACKGROUND

The aim of the data collection is to produce a detailed picture of the IAF corrosion situation that once a year is discussed with all the depot maintenance center and provided to the Logistic Command to enhance the knowledge on the state of the fleet, where special attention is focused on the aging phenomena, and their effective needs.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Because of the high level of interdisciplinary which is characteristic for corrosion science and engineering, the Corrosion Control Register Program overlaps with many different fields first of all with NDE, helping in creating and developing new and more dedicated inspection procedures as far as new inspection techniques. In this sense for example, the use of a monitoring system based on electrochemical corrosion sensors, when located on previously investigated areas already detected as prone to suffer corrosion problems, seems a very promising integrated system to promote an early detection and, as a consequence of this, to enhance an effective corrosion protection and control[5].

The CCR Program actually moves from the report of every detected corrosion to the data bank by means of a standard procedure: the corrosion specialists must fill in a double sheet form divided in different sections where they report much useful information.

The data requested include:

- information about the aircraft (total flight hours, flight hours since last inspection, the base of operations where corrosion is found and the base where the aircraft more often stood in the past six months;
- information about the corroded part (name, Identification Code, Part Number, Serial Number, Work Unit Code, superior unit and its Part Number, etc..);
- information about the type of corrosion, using a classification code arranged by Italian Air Force to simplify the description and consequently to minimize false messages;
- schematic information about the maintenance operation carried out and its cost.
- a map of the corroded area (mandatory just for the parts where corrosion is already expected).

Corrosion classification, as said, was simplified by dividing corrosion distribution into only three different groups: general corrosion, selective corrosion and localized corrosion[6]. To each corrosion distribution some morphological aspects was then univocally linked.

2. RESULTS

Up to last year more than 2600 corrosions were recorded and much information was acquired. The most important of them can be summarized in:

a) Fleet corroding index

It has been calculated as an average on the last five years of the ratio *“corrosion per year”/”number of aircraft constituting the fleet”*. As expected, corrosion must be considered much more problematic for some fleets (fig. 1), in particular for the anti-submarine Br.1150, the cargo aircraft C.130 and helicopters used in Search and Rescue operations as AB.212.

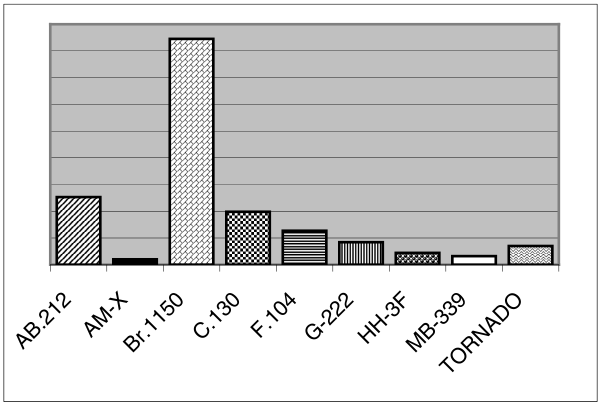


FIGURE 1 – FLEET CORRODING INDEX

This basic information doesn't take into account the differences in danger among the different types of corrosion neither the importance of the affected parts; nevertheless in a glance it gives a useful idea when, in occasion of immediate needs to fix priorities, will be reasonable to spend money and people in prevention first of all to face problems related to these fleets.

b) Index of fleet detectable corrosion

It has been calculated as an average of the last five years of the ratio *"corrosion per year"/"number of aircraft inspected in the Air Force"*. The information given in Figure 2 shows more details about the present engagement addressed to corrosion for the different fleets.

In fact, in the case of a perfect balance of the available resources, this graph should be superimposed on figure 1. In case of some discrepancy, assuming that the information about corrosion maintenance had correctly flown, it means it is necessary to adjust something, primarily in maintenance scheduling or in maintenance people training program, because too much or too little attention has been given to some or some other fleet.

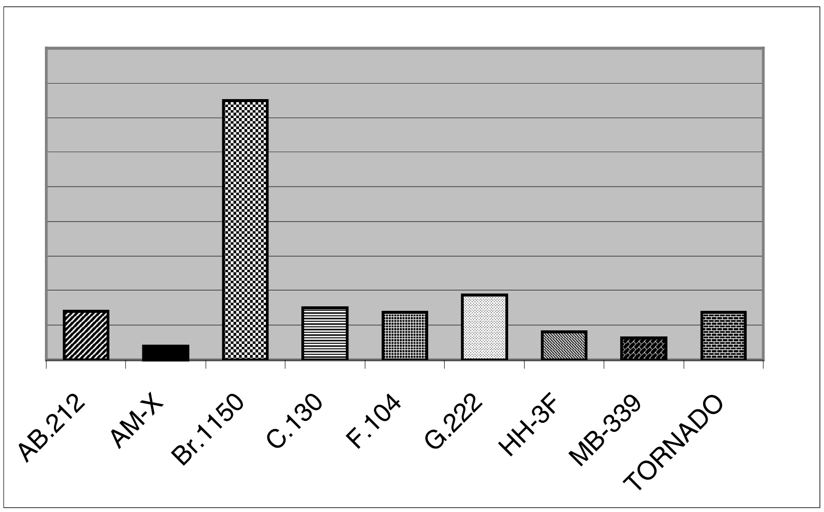


FIGURE 2 – INDEX OF FLEET DETECTABLE CORROSION

c) Corrosion detection reliability of the low inspection levels

In the IAF maintenance is generally conceived on three different levels the first two being devoted to the bases where aircraft operate. As the only difference is in the scheduled maintenance operations to carry out but not in the staff and its skill, here any difference will be considered between them.
The highest maintenance level is a depot inspection devoted to a Squadron operating for the entire fleet.
Previous GAF experiences of in-service corrosion on military aircrafts[7], pointed out the corrosion detectability before paint removal compared with corrosion detected after it: on average approximately 60% was considered as a typical value. As paint removal and components take down occurs only during the depot inspections, this value was also regarded as a reference term at the beginning of our data collection.
The model to be used, if we want to take into account that during the minor inspections corrosion can't ever be completely detected and considering a linear relationship between time and corrosion development, is that reported in Fig. 3. In this case I, II and III are three different scheduled minor inspections carried out by the *i* base, where each one allowed to detect the corrosions developed in the period of time elapsed between two inspections; IV is the major scheduled maintenance where more corrosions were found on the inspected aircraft proceeding from the same *i*-base.

In accordance with this model the corrosion detection reliability of the low inspection levels compared with the major one will be given by the following expression:

$$\frac{\dfrac{c_i}{n_i^{.\,min.\,insp.}}}{\dfrac{c_i + C_i^{maj.\,insp.}}{n_i^{min.\,insp} + N_i^{maj.\,insp.}}} \cdot 100$$

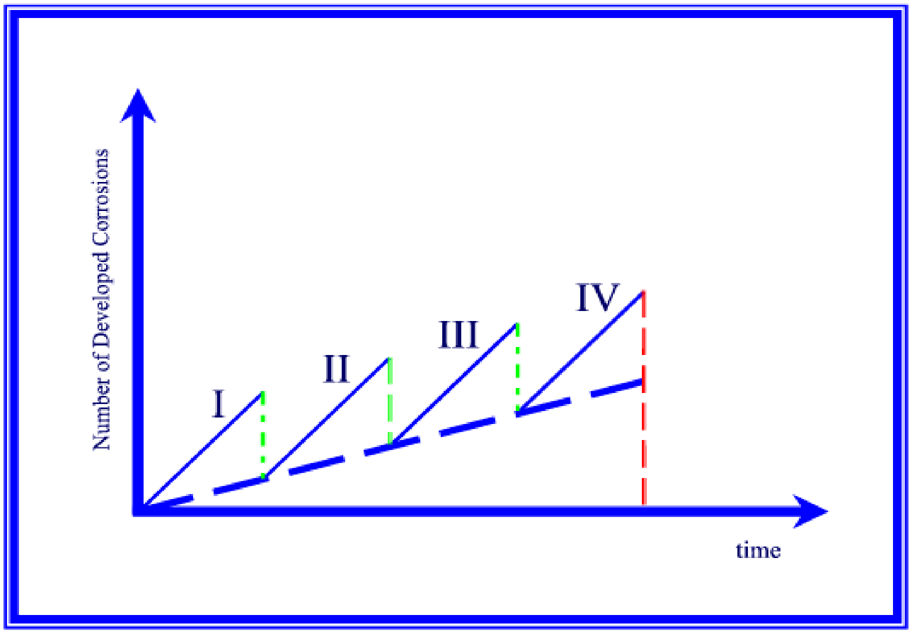


FIGURE 3 – SCHEME FOR CORROSION DETECTION RELIABILITY

By summing all the *i*-index maintenance results, the average on the corrosion detection reliability for the minor inspection levels will be obtained.

Of course such analysis needs a statistical significance of collected data, but previous results obtained on different fleets having very different mission profiles from each other give, for now, approximately the same average values.

If we consider for example the corrosion data acquired up to last year for the helicopter Agusta AB.212, an 80.6% corrosion maintenance detectability during the minor inspections was calculated on the 474 data collected (see Fig. 4).

The average value must be obviously considered as the most reliable, but the statistical analysis carried out until now already gives some reliable results for some bases (the colored ones); in some other cases (the white values) the data, themselves, have not yet a statistical significance because few helicopters proceeding from these bases have gone through the major inspection. Of course for the bases whose results are considered reliable (it means just on the colored ones) their maintenance work can be evaluate: in this real case we can positively judge the inspections carried out in this last three years at the bases *e, g, h* and *i* while big problems appear on the maintenance carried out at the base *f*.

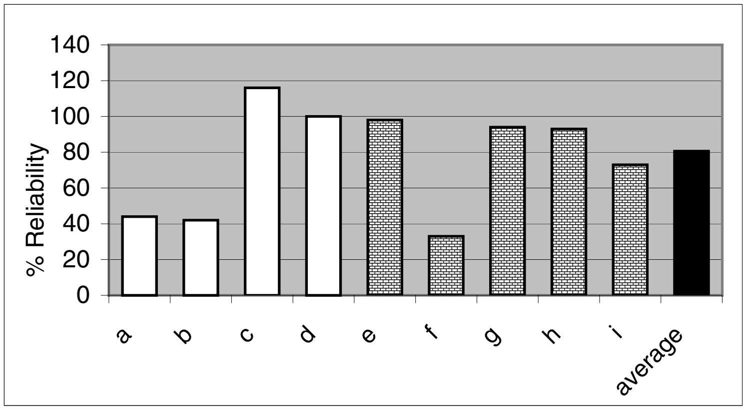


FIGURE 4 – CORROSION DETECTION RELIABILITY OF THE LOW INSPECTION LEVELS

For the anti-submarine Br-1150 operating in only two Italian bases, of which one is also the depot maintenance center, a similar result has been obtained: the corrosion detection reliability during the minor inspections, calculated on 555 data, was 70% in comparison to the corrosion found at the major ones.

d) Corrosion index distribution for each Squadron

This information allows us to understand the contribution of environmental and operative factors on corrosion. More frequently it is an environmental effect that has been observed, concentrated on those bases operating on the sea especially in the south, where high chloride content, high temperature and humidity are often combined.

e) Corrosion index for different parts

It has been considered for every fleet in order to evaluate for each one those items more often and more easily corroded. As the Program is working on very different aircraft they can be divided into the following categories:

- for cargoes, the majority of corrosion occurs on the wings and secondarily on the taileron and the fin;
- for helicopters, the most susceptible items are the honeycomb structures on the floor, followed by the main rotor and the blades;
- for fighters, corrosion is most often located on the body fuselage, the fuel cells, the air intakes and the landing gear;
- for anti-submarines corrosion is very spread, especially on the wings and on the bomb compartment.

f) Corrosion index as a function of aircraft flight hours

In the past, corrosion maintenance was scheduled in accordance with prescriptions fixed at the beginning of the aircraft life. Unfortunately many factors, among them the most important being time and usage, will make each aircraft increasingly unique[8]. To be proactive it needs to be able to understand the differences promoted by the age and in this sense this index allows us to monitor the corrosion susceptibility depending on the usage of the fleet and, consequently, to modulate the maintenance to aging.
This decision is demanded to the Logistics Command that is the organization responsible both for the fleet efficiency and for the way to spent the available resources to achieve it.

g) Maintenance procedures and costs for the different fleets

This kind of information is fundamental to address the maintenance policy, because it is the most direct instrument to know what is really happening with corrosion costs.
In effect, the economic parameter not only affects directly readiness and efficiency of the fleets but is also the starting point to balance any risk assessment.

At the moment is only showing the direct costs, the most important of them being the man hours spent, but it will be probably extended in the next future to much more elements.

h) Classification of corrosion danger for the different fleets

Safety can be certainly enhanced when damage danger, strictly related to damage tolerance, is known. In this sense, corrosion classification has been very useful in detecting the produced damage danger.
In fact, not every observed corrosion must be considered on the same danger level and particular attention must be paid to those selective attacks or some specific localized corrosion detected on principal items.
Four classes have been individuated (the most dangerous being the class 1) and the overall percentage of them is shown in figure 5. Such information must be considered as a powerful quantitative analysis instrument to fix the objective of an enhancement in corrosion prevention in control, because gives the opportunity to easily evaluate the effectiveness of any change produced in the medium-long term on the corrosion maintenance inspection and maintenance procedures which should be accomplished with a reduction in the class 1 phenomena.

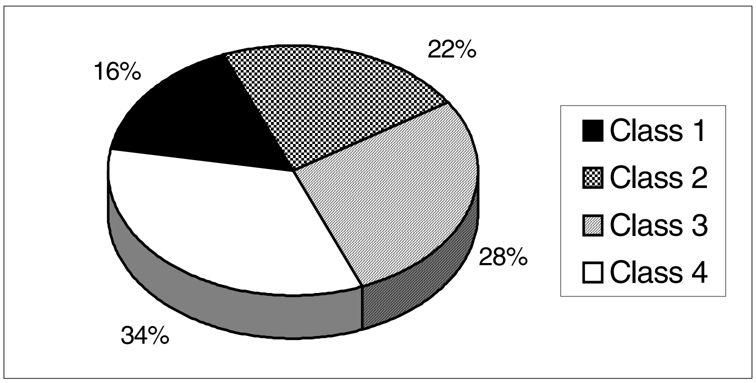


FIGURE 5 – PERCENTAGE OF CORROSION DANGER

When this information is split in the different fleets it is possible to evaluate for each one of them the risk factor associated with corrosion phenomena.

4. CONCLUSIONS

The Corrosion Control Program moved six years ago from a standardization of the training of people working in maintenance and it was originally born in order to manage the Tornado maintenance in agreement with the German and Royal Air Forces.

Actually, the Italian Air Force Corrosion Management can be summarized as a multitasking strategy where a specific engagement consists in the evaluation of the relationships occurring between corrosion and residual strength of aged structures.
Since it was running the Program has shown to be powerful in understanding problems and offering solutions in many different areas. Some of the most important information acquired up to now has been discussed as far as some actions already taken on this base.

BIBLIOGRAPHY

1. S.G. Sampath, “Aging Combat Aircraft Fleets – Long Term Implications”, AGARD Lecture Series 206, Introduction, (Neuilly-Sur-Seine, France: AGARD Lecture Series 206, 1996-1997).
2. V.S. Agarwala, Canadian Aeronautics and Space Journal 42, 2 (1996): p. 68.

3. M. Colavita, G. Trivisonno, E. Dati, "Aging Aircraft: In-Service Experience on MB.326", RTO Meeting Proceedings 18, paper n° 4 (Neuilly-Sur-Seine, France: 18th Meeting RTO held in Corfù -GR, October 1998).
4. M.J. Marlow-Spalding, "Ageing Aircraft – Managing the Tornado Fleet" AGARD Lecture Series 206, paper n° 2, (Neuilly-Sur-Seine, France: AGARD Lecture Series 206, 1996-1997).
5. V.S. Agarwala, P.K. Bhagat, G.L. Hardy, "Corrosion Detection and Monitoring of Aircraft Structures: an Overview" AGARD Conference proceedings 565, paper n° 19, (Neuilly-Sur-Seine, France: 79th AGARD Meeting held in Seville – SP, November 1994)
7. M. Colavita, "Occurrence of Corrosion on Aircraft", RTO Lecture Series 218 bis, paper n° 11, (Neuilly-Sur-Seine, France: AGARD Lecture Series 208 bis held in Sofia –BG 6-10 November 2000).
8. H.J. Voss, "Experience of In-Service Corrosion on Military Aircraft", AGARD Conference proceedings 565, paper n° 21, (Neuilly-Sur-Seine, France: 79th AGARD Meeting held in Seville -SP, November 1994).
9. R.D. Giese, G.D. Herring, J.F. Bockman, "Tracking Aircraft Structural Repairs from a Fleet Risk Management and Economic Standpoint", The Second Joint NASA/FAA/DoD Conference on Aging Aircraft (NASA/CP-1999-208982/PART 1, p. 21).

This page has been deliberately left blank

Page intentionnellement blanche

Corrosion Management – A Statistical Approach

Robert Bell and Don Shelton
Lockheed Martin Aeronautics Company
86 South Cobb Drive
D 6ESM, GA 30063-0569
United States

Mike Falugi
AFRL/VA
Building 45, 2130 8th Street, Suite 1
Wright Patterson AFB, OH 45433-7542
United States

SUMMARY

One of the major cost drivers for any aging aircraft is the mitigation of corrosion damage. This problem has been exacerbated as aircraft are being kept in service beyond their original design service life in terms of both flight hours and calendar years. As this trend continues, the need to understand the impact corrosion has on reliability, maintainability, and the cost to maintain an aircraft becomes increasingly important. This paper will focus on these issues and the proposed methodologies by which they can be addressed in the Aircraft Structural Integrity Program (ASIP) today.

BACKGROUND

Table 1 below provides a brief summary of the average age of some United States Air Force weapons systems.

Table 1. Average age of USAF Aging Fleets

Aircraft	Average Fleet Age (years)
C-141	35
KC-135	38
C-5A	30
C-130	36
B-52	39

Corrosion maintenance cost were found to be the highest for the heaviest aircraft. Table 2 summarizes the annual maintenance cost due to corrosion per aircraft for each fleet. This data was obtained from an NCI Information Systems report [1].

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Table 2. 1997 Annual Corrosion Cost per Aircraft

Aircraft	$ Cost Per Aircraft
C-141	$341,258
KC-135	$324,425
C-5A	$806,500
C-130	$63,338
B-52	$363,739

Again, these dollar amounts reflect maintenance cost for corrosion. They do not include the lost revenue for the aircraft due to unscheduled downtime caused by corrosion. All of this is mentioned to emphasize the significant impact corrosion has on the overall cost of ownership for an aging aircraft. For the purposes of this paper the C-141 will be discussed and used as an example.

The need for a different approach exists in regards to corrosion maintenance for an aging fleet. The current ASIP methodology is "Find it Fix it". While this does address the issue, it leaves much to be desired. Ideally, it would be beneficial to manage the fleet for corrosion much in the same way it is for fatigue. The current methodologies in ASIP for fatigue are as follows:

- Where to inspect (FSMP)
- When to inspect (FSMP, Individual Aircraft Tracking)
- How to Inspect (FSMP, NDE)
- What to Look for (FSMP, NDE)
- Inspection results (Repair Databases; i.e. G081, CARR, OSCAR, 202's)
- What to do next; inspect?, repair?, replace? (FSMP)
- Cost to inspect, repair, or replace (needed)
- Individual Aircraft Tracking

These same issues are relevant to corrosion. Therefore, corrosion maintenance methodologies could be incorporated into the existing ASIP framework including the Force Structural Maintenance Plan (FSMP), Individual Aircraft Tracking, Repair databases, and Non-Destructive Evaluation (NDE) requirements. Some of the proposed corrosion methodologies that could be incorporated right now to improve reliability, maintainability, and cost of maintenance for corrosion related problems are as follows:

Reliability can be improved by reviewing maintenance data on aircraft. Costs analysis can be performed using data gathered from planners at Warner Robins Air Logistic Center (ALC). Dollar amounts would be associated with each maintenance option (repair, replace, leave as is and inspect, apply CPC's). The result would be improved maintainability for the aircraft. Our approach will be to statistically analyze the maintenance data related to corrosion at the Air Force depots and determine the probability of occurrence. Cost data is available for the C-141 and KC-135 in the Crevice Corrosion computer program[2] developed under prior Air Force contracts. Using this data a matrix of maintenance options can be developed and the probabilities of experiencing different costs obtained. This will allow maintenance managers to manage their aircraft in the most efficient manner. Obviously, it is essential to maintain safety of flight while reviewing any maintenance options. Programs such as "Environmental & Cyclic Life Interaction Prediction Software" (ECLIPSE)[3,4] can provide inspection intervals to be used in the final cost benefit determination. The Corrosion Maintenance Improvement (CMI) program is also developing data which can be used in assessing options such as the use of corrosion preventive compounds (CPCs), optimum finishes, etc. Another ongoing program, "Corrosion Fatigue Structural Demonstration", like the CMI program, is studying the effect corrosion has on aircraft structures. As these programs and others like them come to a conclusion, any significant findings and technologies provided by them could potentially be incorporated into the existing ASIP framework as deemed appropriate by the aircraft ASIP manager. This would provide additional corrosion management tools for the aircraft maintainer. These programs and any subsequent technologies are mentioned only as potential future additions to the aircraft management framework.

This paper will focus primarily on statistical methodologies for corrosion maintenance that could be incorporated into ASIP now. The first part of this paper is dedicated to discussing the assessment of the aircraft structure. Below is an outline of the Statistical Methodology for the structure to be presented in this paper followed by a detailed description of the process. The second part of the paper will address the cost issue. Obviously, the key to success for the statistical approach discussed in this paper is the amount, and accuracy, of data that is available for an aircraft. Because of this criteria, the C-141 aircraft was chosen to demonstrate the proposed methodology.

STRUCTURAL ASSESSMENT METHODOLOGY

1. Calculate corrosion growth rate per wetted month
   a) Review historical data for reported corrosion maintenance actions
   b) Generate a Normal, Log-Normal, and Weibull plot to determine the corrosion growth rate probabilities per wetted month
   c) Determine which plot to use
2. Calculate corrosion growth by tail number for the period from when corrosion was found to the prior paint date
   a) Determine time of wetness over this time period
   b) Use corrosion growth rate from Weibull and Log-Normal plots
3. Determine the mean and standard deviation for the allowable grind-out limits
4. Calculate the probability that corrosion growth will exceed the mean allowable grind-out depth for the specified time period
   a) Perform a interaction analysis using the grind-out limits and the projected corrosion growth
   b) Calculate the probability that one corrosion finding in the area of interest will exceed the mean allowable grind-out depth thus necessitating the need for a major repair

Calculate Corrosion Growth Rate per Wetted Month

The first step is to calculate a corrosion growth rate per wetted month. In order to do this maintenance records for twenty-six C-141 aircraft were reviewed. From these records fifty-nine maintenance actions for corrosion were found. The depth of corrosion for each instance was determined by measuring the depth of the grind-out. Next, the Time of Wetness (TOW) [5] in months was calculated for the time span from when the corrosion was found till the immediately preceding aircraft repaint at Program Depot Maintenance (PDM). The TOW is based on ISO 9223, "Corrosivity Classification". A brief description can be found on the world wide web at http://www.physics.odu.edu/~cmmp/corrosion/ISO.html. The depth of corrosion was then divided by the corresponding time of wetness for all fifty-nine corrosion entries. Log-Normal, Weibull, and Normal plots were made to determine the corrosion growth rate probabilities per wetted month. These plots are shown in Figures 1, 2, and 3 respectively.

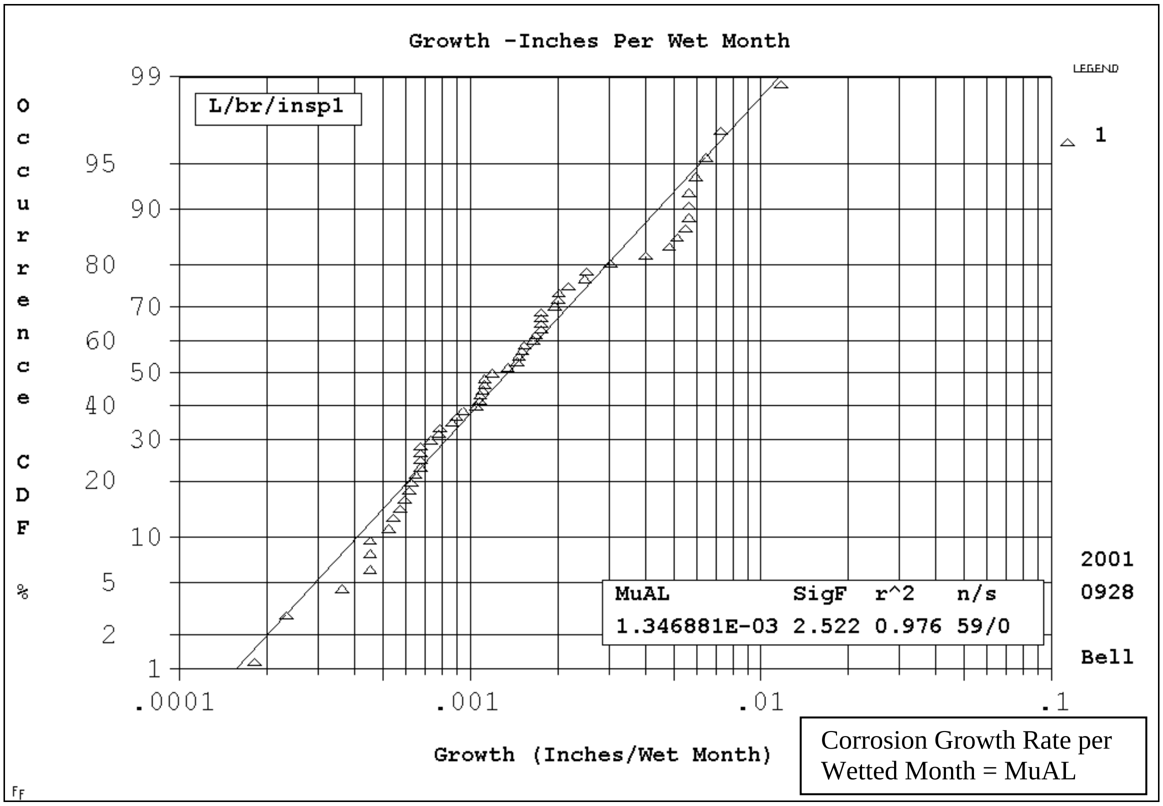


Figure 1. Log-Normal Plot of Corrosion Depth/TOW for 59 Corrosion Records

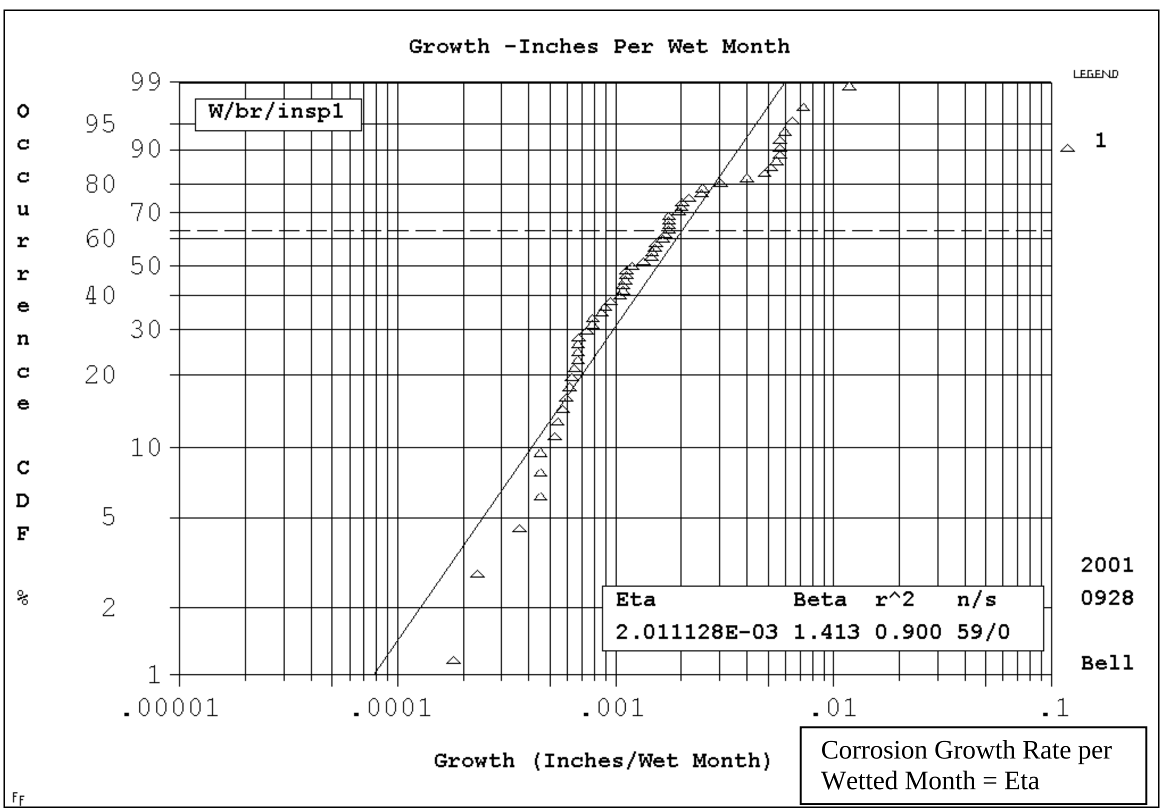


Figure 2. Weibull Plot of Corrosion Depth/TOW for 59 Corrosion Records

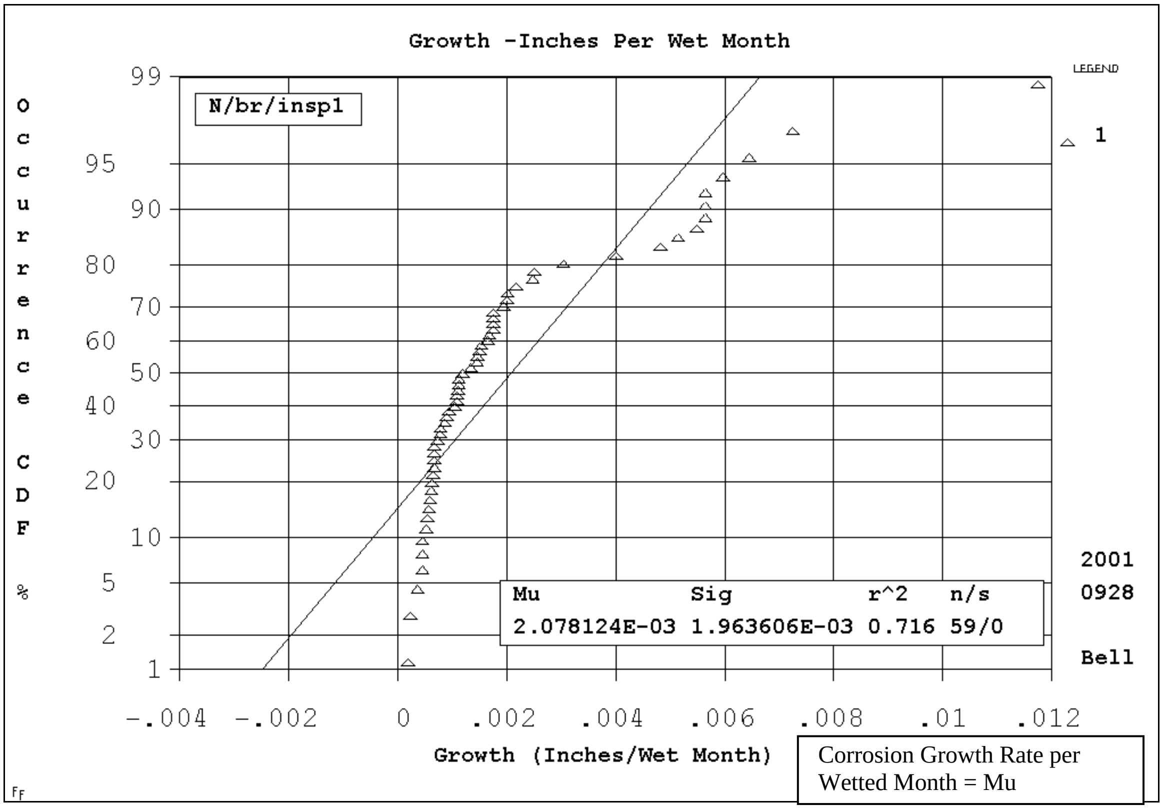


Figure 3. Normal Plot of Corrosion Depth/TOW for 59 Corrosion Records

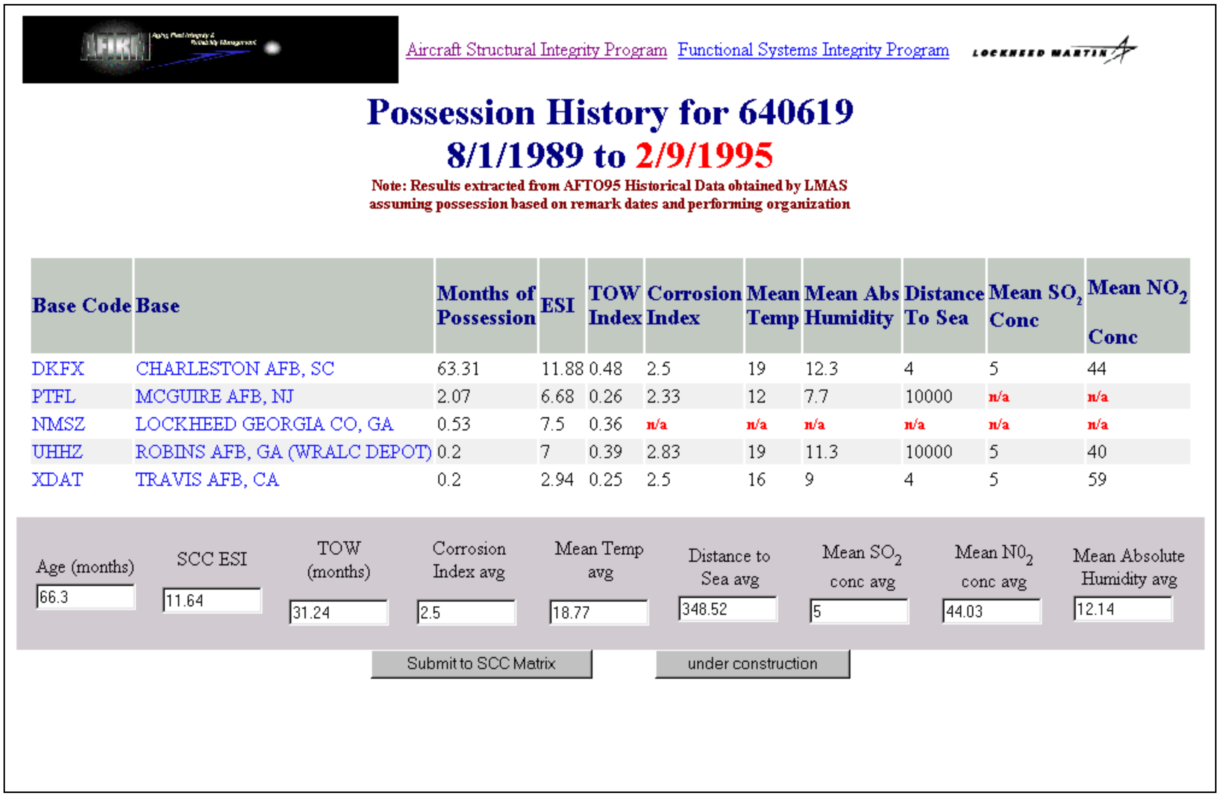
From the plots it appears the Log-Normal Distribution had the best fit. The Weibull plot had the second best fit. However, due to the limited number of data points it is recommended to use the results from the Weibull plot. Another observation made is that it appears that Corrosion is not a random process according to the Variance from the Normal Distribution plot. Using the results from the Weibull plot the following mean corrosion growth rate was obtained.

Mean corrosion growth rate per wetted month = Eta =2.011128E-03 inches/wetted month

It should be noted that the fifty-nine (59) data points used are from various locations on the aircraft. Some are from the fuselage, inner wing, outer wing, center wing, vertical stabilizer, and horizontal stabilizer. Therefore, the corrosion growth rate derived is a generic one encompassing the entire aircraft. Ideally, it would have been preferred to derive a corrosion growth rate per wetted month for each separate component of the aircraft (i.e. one for the fuselage, one for the inner wing, one for the outer wing, etc.). However, at the present time this was not possible. There were approximately sixteen thousand Air Force repair records documented on paper 202's to review. These records span the years from 1989 to 1997. Out of these 16,000, about eight hundred of them were found to be related to corrosion. Compilation of an electronic database detailing the findings from these 800 records has begun. Unfortunately, due to time constraints it was not possible to input all of the available records prior to completion of this paper. At the point in time when all of this data has been input into the electronic database, a mean corrosion growth rate per wetted month will be calculated for each component of the aircraft to determine how much, if any, variation exists. All of this is mentioned to reference where the data came from and why a single mean corrosion growth rate that encompasses the entire aircraft structure was used for the purposes of this paper.

Calculate Corrosion Growth for Individual Aircraft (by Tail Number)

Aircraft 640619, 660151, and 640646 were chosen for this demonstration. Using the Mean Corrosion Growth Rate per wetted month from the Weibull plot, calculate the corrosion growth (depth of corrosion) for each of the three aircraft above. In order to do this we need to calculate the time of wetness for each individual aircraft for the time period from when the corrosion was reported to the PDM paint date prior to when the corrosion was found. The calculated TOW for each aircraft is shown in Figures 4, 5, and 6.


Aircraft Structural Integrity Program Functional Systems Integrity Program LOCKHEED MARTIN

Possession History for 640619
8/1/1989 to 2/9/1995

Note: Results extracted from AFTO95 Historical Data obtained by LMAS assuming possession based on remark dates and performing organization

Base Code	Base	Months of Possession	ESI	TOW Index	Corrosion Index	Mean Temp	Mean Abs Humidity	Distance To Sea	Mean SO_2 Conc	Mean NO_2 Conc
DKFX	CHARLESTON AFB, SC	63.31	11.88	0.48	2.5	19	12.3	4	5	44
PTFL	MCGUIRE AFB, NJ	2.07	6.68	0.26	2.33	12	7.7	10000	n/a	n/a
NMSZ	LOCKHEED GEORGIA CO, GA	0.53	7.5	0.36	n/a	n/a	n/a	n/a	n/a	n/a
UHHZ	ROBINS AFB, GA (WRALC DEPOT)	0.2	7	0.39	2.83	19	11.3	10000	5	40
XDAT	TRAVIS AFB, CA	0.2	2.94	0.25	2.5	16	9	4	5	59

Age (months)	SCC ESI	TOW (months)	Corrosion Index avg	Mean Temp avg	Distance to Sea avg	Mean SO_2 conc avg	Mean NO_2 conc avg	Mean Absolute Humidity avg
66.3	11.64	31.24	2.5	18.77	348.52	5	44.03	12.14

Submit to SCC Matrix under construction

Figure 4. Calculated corrosion growth for aircraft 640619 from 8/1/1989 to 2/9/1995

Aircraft Structural Integrity Program Functional Systems Integrity Program LOCKHEED MARTIN

Possession History for 660151
7/26/1988 to 5/11/1994

Note: Results extracted from AFTO95 Historical Data obtained by LMAS assuming possession based on remark dates and performing organization

Base Code	Base	Months of Possession	ESI	TOW Index	Corrosion Index	Mean Temp	Mean Abs Humidity	Distance To Sea	Mean SO_2 Conc	Mean NO_2 Conc
XDAT	TRAVIS AFB, CA	35.84	2.94	0.25	2.5	16	9	4	5	59
SCEY	NORTON AFB, CA	21.26	2.63	0.25	2.5	18	9.1	10000	23	85
PCZP	MARCH AFB, RIVERSIDE, CA	10.02	2.63	0.25	n/a	n/a	n/a	n/a	n/a	n/a
UHHZ	ROBINS AFB, GA (WRALC DEPOT)	2.37	7	0.39	2.83	19	11.3	10000	5	40

Age (months)	SCC ESI	TOW (months)	Corrosion Index avg	Mean Temp avg	Distance to Sea avg	Mean SO_2 conc avg	Mean NO_2 conc avg	Mean Absolute Humidity avg
69.49	2.94	17.7	2.51	16.83	3974.79	11.43	67.54	9.13

Submit to SCC Matrix | under construction

Figure 5. Calculated corrosion growth for aircraft 660151 from 7/26/1988 to 5/11/1994

Aircraft Structural Integrity Program Functional Systems Integrity Program LOCKHEED MARTIN

Possession History for 640646
8/22/1985 to 7/19/1995

Note: Results extracted from AFTO95 Historical Data obtained by LMAS assuming possession based on remark dates and performing organization

Base Code	Base	Months of Possession	ESI	TOW Index	Corrosion Index	Mean Temp	Mean Abs Humidity	Distance To Sea	Mean SO_2 Conc	Mean NO_2 Conc
DKFX	CHARLESTON AFB, SC	74.09	11.88	0.48	2.5	19	12.3	4	5	44
PQWY	MCCHORD AFB, WA	15.74	11.9	0.4	2	11	7.9	10000	17	n/a
PTFL	MCGUIRE AFB, NJ	14.88	6.68	0.26	2.33	12	7.7	10000	n/a	n/a
UHHZ	ROBINS AFB, GA (WRALC DEPOT)	10.68	7	0.39	2.83	19	11.3	10000	5	40
YEQM	WACO, TX CHRYSLER TECHNOLOGY CORP.	3.48	NO ESI	0.25	n/a	n/a	n/a	n/a	n/a	n/a

Age (months)	SCC ESI	TOW (months)	Corrosion Index avg	Mean Temp avg	Distance to Sea avg	Mean SO_2 conc avg	Mean NO_2 conc avg	Mean Absolute Humidity avg
118.87	10.76	50.76	2.44	17.01	3581.73	6.88	43.5	11.01

Submit to SCC Matrix | under construction

Figure 6. Calculated corrosion growth for aircraft 640646 from 8/22/1985 to 7/19/1995

Table 3 summarizes the TOW for the three aircraft along with their respective calculated Corrosion growth.

Table 3. Summary of Calculated Corrosion Growth for Three C-141 Aircraft

Tail Number	TOW (wetted months)	Eta (Weibull) (inches/wetted month)	Corrosion Growth (TOW x Eta) (inches)
640619	31.24	2.011128E-03	.0628
660151	17.7	2.011128E-03	.0356
640646	50.76	2.011128E-03	.1021

With this information in hand we want to determine the probability that the projected corrosion growth will exceed the mean allowable grind-out limit for the fuselage skins for each of these aircraft. In order to do this the mean and standard deviation must be determined for the fuselage skin allowable grind-out limits.

Determine the Mean Allowable Grind-out Limit for the Fuselage Skins

The allowable corrosion grind-out depths for all of the fuselage skins from T.O. 1C-141B-23 were reviewed. In all, there were one hundred and eight (108) locations with varying grind-out depths allowed. A Log-Normal distribution plot of the 108 allowable grind-out depths was generated. This plot is shown in Figure 7.

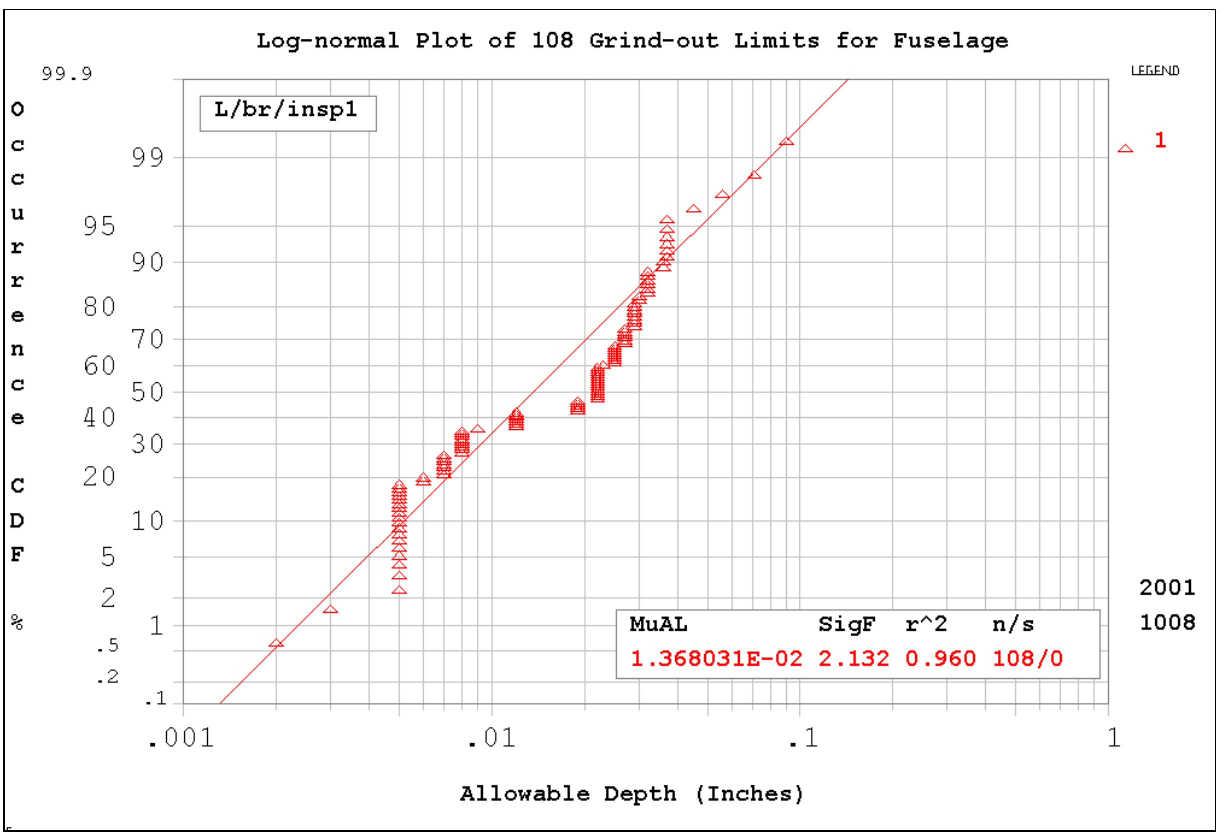


Figure 7. Log-Normal Plot of Allowable Fuselage Skin Grind-out Depths

As can be seen from the plot, the data correlated fairly well. With this in mind, the mean allowable grind-out depth of MuAL = 1.368031E-02 inches, and the standard deviation of 0.00733 will be used for the interaction analysis discussed below.

Interaction Analysis

With the mean allowable fuselage grind-out depth (MuAL), the corresponding standard deviation, and the calculated Weibull corrosion growths shown in Table 3, an interaction analysis using WinSMITH [6] can be performed. This will calculate the probability for each of the three C-141 aircraft that the anticipated corrosion growth for the fuselage skins will exceed the Mean Allowable fuselage skin grind-out depth. The results of the Interaction analysis are shown in Table 4.

Table 4. Summary of WinSmith Interaction Analysis for Three C-141 Aircraft

Tail Number	Grind-out Mean & Standard Deviation	Calculated Corrosion Growth & Weibull Slope	Probability (calculated corrosion growth > Mean allowable grind-out)
640619	.01368, 0.00733	.0628, 1.413	84.20%
660151	.01368, 0.00733	.0356, 1.413	71.05%
640646	.01368, 0.00733	.1021, 1.413	91.20%

The next step is to calculate the probability that a found corrosion spot would exceed the mean allowable grind-out depth. This will be obtained by using the results from the interaction analysis.

Probability that a found corrosion spot would exceed the mean allowable grind-out depth

Recall that the area of interest for each of the three aircraft was the fuselage skins. In order to calculate the probability that a found corrosion spot would exceed the mean allowable grind-out depth, the effect of each individual zone on the fuselage skins must be included. During the Structural and Systems Assessment Program (SSAP) for the C-141, the fuselage skins were divided into one thousand and seventy (1070) zones. Each zone represents an area twenty inches by twenty inches (20"x20"). The calculated probability is also dependent on the number of corrosion hits found. For demonstration purposes we will assume one corrosion spot was found on aircraft 640619. Therefore, the probability that this one corrosion finding on the fuselage skins will exceed the mean allowable grind-out depth is:

Probability = "# of corrosion hits" x "the inverse of the number of zones on the fuselage skins" x "the probability the Log-normal corrosion growth > mean grind-out limit for the fuselage skins".

Probability = 1 x (1/1070) x 0.8281 = 7.739x10E-04 (probability one corrosion hit on the fuselage skins of 640619 will exceed the mean allowable grind-out depth)

From this assessment it appears that the effects of corrosion based on the calculated corrosion growth rate for tail number 640619 reduces the structural integrity of the aircraft. However, keep in mind the purpose of this paper was to demonstrate a proposed corrosion maintenance methodology. Not to determine structural integrity. The calculation was done simply to show that corrosion growth may have an impact on the structural integrity of the airframe. Therefore, the simplified approach shown above was deemed appropriate for the intended purposes of this paper. A much more detailed strength assessment, one that is outside of the scope of this paper, would need to be conducted before any conclusions could be drawn regarding the true impact of corrosion on the fuselage skins.

This concludes the structural portion of the proposed statistical approach to corrosion maintenance. In the following section, the cost issue will be discussed.

COST ASSESSMENT

The first step in determining the cost assessment is to plot the time of wetness (TOW) verses the number of corrosion hits on the fuselage skins by aircraft tail number. The time of wetness was calculated for the time span from when the corrosion was reported to the prior aircraft repaint at PDM. This in essence was the time between consecutive repaints at PDM. The electronic database documenting some of the eight hundred corrosion records mentioned above were reviewed and three aircraft were selected. They were C-141 tail numbers 660151, 640619, and 640646. It should be noted that none of the corrosion records from these three aircraft were included when calculating the generic corrosion growth rate per wetted month discussed at the beginning of this paper (refer to Figures 1, 2, and 3). This was done intentionally.

Table 5 summarizes the findings for aircraft 640619, 660151, and 640646. Included in the Table is the aircraft tail number, associated number of corrosion hits found on the fuselage skins, date the corrosion was found, date of the aircraft repaint preceding the corrosion being found, and the calculated TOW. The time of wetness was calculated using the time from when the corrosion was reported to the prior PDM repaint, and the corresponding base of assignment for the aircraft during that time period.

Table 5. Summary of TOW verses Number of Corrosion Hits on Fuselage Skins

Tail Number	TOW	# of Corrosion Hits on Fuselage Skins	Date Corrosion Reported (at Repaint)	Prior PDM Repaint
640619	31.24	4	2/9/1995	8/1/1989
660151	17.7	2	5/11/1994	7/26/1988
640646	50.76	12	7/19/1995	8/22/1985

From the limited data shown in Table 5 it appears there is some correlation between the time of wetness and the number of reported corrosion findings for these three aircraft. These results represent the first step in validating the statistical approach to corrosion maintenance discussed in this paper. In order to continue the validation of this method more aircraft tail numbers need to be included. Then an accurate "TOW" vs. "number of reported corrosion hits on fuselage skins" could be generated and used. That is providing a correlation analysis was performed that showed good results. However, this was not feasible at the time of this paper due to the amount of time and effort required to process and input all of the accumulated eight hundred corrosion maintenance records into an electronic database. Therefore, only the corrosion data for the three aircraft shown in Table 5 was used at this time.

Once a "TOW" vs. "number of corrosion hits on the fuselage skins" plot is created from existing data and validated, the plot could be used to forecast the anticipated number of corrosion findings by tail number at the aircraft's next scheduled PDM repaint. This will be illustrated using aircraft 640619. The last PDM repaint for this aircraft was February 2^{nd}, 1995. The next scheduled repaint is July 16^{th}, 2001. With these two dates and the base of assignment data for the aforementioned timeframe, the TOW can be calculated. In this instance, the TOW for aircraft 640619 from February 2^{nd}, 1995 to July 16^{th}, 2001 was 19.97 wetted months. Then using a TOW = 19.97 and the "TOW" vs. "number of reported corrosion hits on fuselage skins" plot mentioned above, an anticipated number of corrosion findings on the fuselage skins at the next scheduled PDM repaint can be obtained. Recalling that this plot was not generated because of the limited amount of data that had been processed, a different approach was taken for this illustration. Since the calculated TOW = 19.97 was relatively close to the TOW = 17.7 shown in Table 5, it was assumed that the same number of corrosion hits would be found for either one. In this case, two. With this in mind, it is now possible to estimate the corrosion maintenance cost for the fuselage skins on aircraft 640619 when it arrives at PDM for repaint on July 16^{th}, 2001.

Calculate Estimated Corrosion Maintenance Costs for Fuselage Skins

In order to calculate the estimated corrosion maintenance costs, an average cost at PDM for a major repair due to corrosion on the fuselage skins was needed. It was assumed that a major repair would be a panel segment replacement. This would be done when the grind-out depth due to corrosion exceeds the allowable grind-out depth for the fuselage skin shown in T.O. 1C-141B-23 manual, and the damage could not be repaired using a flush "dime" and "dollar" repair because of size limitations. Based strictly on manpower hours and a billing rate it was estimated that a fuselage skin panel segment replacement costs $50,000. Again, this amount is only for labor.

Now, to calculate estimated corrosion maintenance cost for the fuselage skins on aircraft 640619 at the next scheduled PDM repaint multiply the estimated panel segment replacement cost by the estimated number of corrosion hits at the next scheduled PDM repaint and by the probability that the Log-Normal corrosion growth will exceed the mean allowable grind-out (refer to Table 4.). The probability that the Log-Normal corrosion growth will exceed the mean allowable grind-out is included because this is the probability that a major repair (i.e. panel segment replacement) would be required. Therefore, the estimated corrosion maintenance cost for the fuselage skins on aircraft 640619 at the next scheduled PDM repaint is:

Estimated corrosion cost for 640619 fuselage skins = $50,000 x 2 x .8281 = $82,810
at next scheduled PDM repaint on July 16^{th}, 2001

CONCLUSIONS

From the preliminary findings documented in this paper it appears the proposed statistical methodology shows promise for forecasting corrosion damage and corrosion cost. Again, this approach hinges on the amount of corrosion data that is available for a particular weapons system, as well as its accuracy. Following is a list of some of the other conclusions that were drawn.

- Mitigation of corrosion is a major cost driver for an aging aircraft
- For most weapons systems the approach taken for corrosion maintenance is reactive, such as "Find it" "Fix it". It is desired to have a proactive approach for corrosion maintenance similar to that used for fatigue.

- The proposed Statistical Methodology could be incorporated into the existing ASIP framework with minimal effort.
- From the variance for the Normal distribution plot shown in Figure 3, it appears that corrosion is not a random process.
- Corrosion locations for same material type appear to be a random process
- More data needs to be processed before determining if the number of corrosion occurrences based on material type is a random process.
- From the calculated corrosion growth rates shown in Table 3, and the interaction analysis, it appears the effects of corrosion may reduce the structural integrity of the aircraft. However, a much more detailed strength assessment would need to be conducted before any conclusions could be drawn regarding the impact corrosion has on the structural integrity of an aircraft.
- Cost of corrosion damage can be calculated as long as accurate cost data is supplied by the maintenance provider.
- The statistical corrosion maintenance methodology presented in this paper was demonstrated on the C-141 but is applicable to any weapons system provided the historical data exists documenting the base of assignment and repair records for each tail number.

RECOMMENDATIONS

The following are recommendations for further areas of study for the proposed corrosion maintenance methodology.

- Continue to analyze and compile an electronic database containing all of the available historical PDM corrosion data to verify and modify the growth rates as required.
- After a sufficient amount of data has been entered into the electronic database, calculate a corrosion growth rate for each separate component of the aircraft (i.e. one for the fuselage, inner wing, outer wing, etc.) and determine how much, if any, variation exists. If variations do exist, use a growth rate for each component as opposed to one that encompasses the entire aircraft structure.
- Calculate a more accurate probability for a major repair being required by using improved methods to obtain an assessment of the structural strength.
- Break out the repair severity probabilities to determine more accurate costs. This would be done in lieu of using an average cost for a repair as was done for this paper.

REFERENCES

1. Cooke, G. Vore, P.J., Gumienny, C. Cooke, G. Jr., “A Study to Determine the Annual Direct Cost of Corrosion Maintenance for Weapon Systems and Equipment in the United States Air Force”, Final Report, Contract Number F09603-89-C-3016.
2. Final report for corrosion / fatigue effects on structural integrity. Contract No. F09603-97-C-0349, task order 9128, CDRL No. A001, NCI Information Systems Inc, Fairborn Ohio, 45324, November 1998
3. K. Honeycutt, C.L. Brooks, and S. Prost-Domasky. ECLIPSE-Environmental and Cyclic Life Interaction Prediction Software. Presented at the ASM AEROMAT Conference, Dayton, Ohio, June 1999.

4. C.L. Brooks, S. Prost-Domasky, and K. Honeycutt. Corrosion is a structural and economic problem: transforming metrics to a life prediction method. Presented at the NATO RTO's Workshop 2 on Fatigue in the Presence of Corrosion, Corfu, Greece, October, 1998.
5. ISO 9223, "Corrosivity Classification, Time of Wetness", http://www.physics.odu.edu/~cmmp/corrosion/ISO.html
6. Fulton Findings, "WinSMITH Weibull (WSW) by Fulton Findings", 1251 W. Sepulveda Blvd #800, Torrance, CA 90502.

This page has been deliberately left blank

Page intentionnellement blanche

Improved Corrosion Maintenance Practices

R. Kinzie
USAF Corrosion Prevention and Control Office
Robins AFB, GA 31098, USA

The USAF, and much of the aerospace industry, currently manage corrosion by providing clear engineering direction that it will be found and fixed prior to becoming a structural or safety concern. New procurements have been reduced and current fleets are now at, or projected to be beyond, their original design lives. While there is significant fatigue life left, corrosion maintenance costs are escalating rapidly. Initial protection systems have broken down and corrosion is becoming the dominant factor in the life of the aircraft.

Under the current engineering policy, often much of the corrosion cost is associated with the dismantling and reassembly of aircraft structure and not the repair itself. Where the corrosion is superficial many of these repairs could be done at a more opportunistic time were there the tools to assure there would be no compromises to the structural integrity. Likewise, there are currently no tools with which to quantify the structural impact of benefits from corrosion abatement technologies. The "find and fix" approach supports better prevention, detection, and repair technology development. However, it does not quantify the beneficial impacts nor facilitate the needed changes in maintenance practices to significantly reduce the rapidly growing corrosion repair costs.

Current engineering philosophy requires fleet management vs. management by individual tail number. There are no tools to quantify the structural significance of a given level of corrosion nor determine alternative required inspections. Likewise, there is no tool to define exactly how good NDI for corrosion must be, nor are the needed parameters (pitting, thickness loss, etc.) specified. NDI techniques that identify nuisance corrosion can be counterproductive driving up both costs and extending schedules.

The lack of tools to analyze the mechanical impacts of corrosion on aircraft structure also result in less robust maintenance practices because the impacts of improvements can only be measured subjectively. Benefits from specific maintenance changes cannot be quantified. Forging replacements are machined from thick plate or bar stock of the same alloy but with very different grain structure. The relative lives and corrosion characteristics of the different material forms are not determined or tracked. Corrosion Prevention Compound (CPC) use is authorized but not mandated nor are CPC programs focused and tailored to the overall maintenance program since the mechanical impacts cannot be quantified. Likewise, there are no tools to objectively guide in the choice of repair options, which are now chosen based on the judgment of an individual engineer.

To facilitate improved corrosion maintenance practices the necessary tools are being developed in a multifaceted and comprehensive corrosion program. Many aspects of this program will be covered in other papers being presented here. However, an integral part

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

of such an effort is the definition of the severity of the environment to which the aircraft is exposed. For many years, attempts have been made to measure the corrosion severity of the natural environment in order to anticipate the levels of corrosion damage that may be expected or to define a reasonable level of corrosion protection to be required.

In the 1970's, the USAF developed an algorithm using readily available weather parameters for predicting corrosion damage under the Pacer Lime Program. The parameters included chloride deposition levels based on distance from the sea, average annual rainfall, average annual humidity, SO2 levels, total suspended particulates, UV and 03 levels. This algorithm proved useful as a tool for determining aircraft wash and rinse frequencies and gave a rough approximation of the severity of the environment (Reference 1). However, this did not provide a close correlation with actual corrosion maintenance costs or hours nor did it provide insight into the type of corrosion damage to be expected in specific structure on a specific aircraft. A much larger study was done in the 1980's under the National Acid Precipitation Assessment Program. Incorporation of this information into an improved algorithm gave some improvement but still there were many instances where the ranking did not track with the corrosion being encountered. Factors were then incorporated for extraneous sources of chlorides, for instance road salts in the winter, and roughness of the seas, prevailing winds, etc. (Reference 2). Still the results were not satisfactory for use with any predictive modeling efforts. Subsequent analysis indicated that the revised algorithm might be adequate if there were sufficiently detailed data to allow integration of the parameters over time. However, such detailed data simply does not exist for all of the locations of concern to the USAF. Comparison of the calculated severity to that measured showed reasonably good correlation in the milder environments but a significant under estimate in the more severe environments (Reference 3).

The alternative of conventional exposure testing at all locations seemed impossible given the costs and time required. However, conventional exposure racks were placed at 6 severely corrosive environments where KC-135 aircraft were stationed. Various panels and lap joint configurations were exposed for 1 to 5 years giving some indications of the expected type and extent of corrosion damage in these joints. These racks also allow correlations to be made with corrosion maintenance experience on the aircraft stationed at these locations.

A unique corrosion exposure system, developed and refined by Dr. William Abbott, was also employed (Reference 4). This exposure system, which is quite simple, provided amazingly accurate results fairly quickly. The modular rack measuring approximately 6" x 6" x14" can be mounted on any supporting structure including fences, poles, etc. (Figure 1).
In this rack there are 4 cards, each containing 5 strips of bare metal including silver, copper, 2 aluminum alloys, and mild steel (Figure 2).

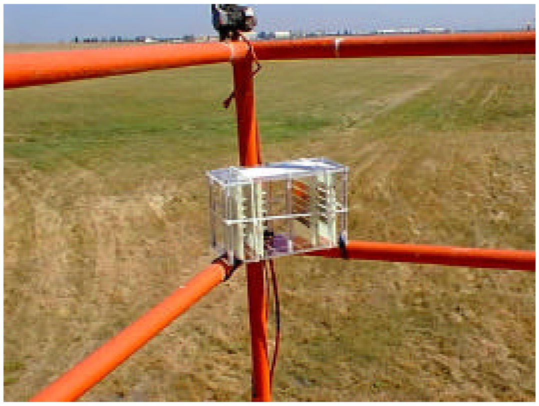


Figure 1. Example Of Typical Sample Installations In Proximity To Runways Or Aircraft Ramps

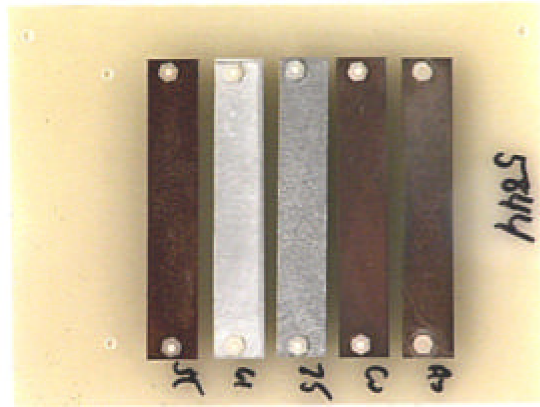


Figure 2. Typical Corrosion Test Card Containing 5 Metallic Coupons; 4 Cards Per Test Site

One card is removed every 3 months and various analyses are done to ascertain the relative corrosivity. The analysis of the silver coupon yields intelligence as to the chloride element of the corrosion process with additional indications of the affects of sulfur containing components when analyzed in conjunction with the copper coupon. These metals allow some detailed insight into the severity of a specific environment to specific types of metals. Likewise, precise measurement of weight losses and gains of the aluminum and steel coupons allows determination of the severity of the specific environment for those materials (Reference 4). Exposure testing has been done at over 80 USAF sites, from Antarctica to Saudi Arabia, and from seacoast to desert. These exposures were done on uncoated and openly exposed coupons. Significant new information was obtained.

This testing yielded several surprises. Chloride species are found at virtually all locations, with close proximity to the seacoast not a prerequisite (Figure 3). This result may be due to entrainment of oxidized sea salt aerosols in upper atmosphere wind currents. There is very limited evidence of corrosion resulting from sulfur containing compounds though this cannot be eliminated as a possibility. With the cleanup of the environment, SO2 levels, and the associated acid rain, have been reduced by orders of magnitude in many areas but corrosion levels have not changed proportionally.

Other information extracted from this testing indicates that there is minimal seasonal variation in the corrosion rates and that rates calculated after 3 months are reasonable accurate and correlate closely with those obtained after a full year of exposure (Figure 4). Likewise, relative values of chlorides obtained from the silver coupon alone provide a reasonably good indication of the severity of that environment for most locations though there are exceptions. Use of the silver sensors alone for shorter periods thus allows a much faster and inexpensive means of screening new and/or temporary locations though the information is not necessarily sufficient for robust predictions.

The relative severity of most environments to the various metals is roughly the same although there are some exceptions and minor shifts in relative rankings. On boldly exposed coupons the rates for some aluminum alloys varied almost 300:1 over the range of locations (Figure 5). Absolute rates of corrosion of 2024, 7075, and 6061 aluminum alloys spread fairly consistently over nearly a 10:1 range with the corrosion rate increasing with the amount of alloyed copper, as expected (Figure 6). Across the locations, the rates for mild steel covered a nearly 200:1 range (Figure 7).

These ranges may become significant when analyzing specific types of structure or components. For instance, an environment specifically severe for copper could yield a disproportionate number of avionics failures. There is some indication from maintenance data that this might be the case. This experience is more clearly shown in locations specifically damaging to steel. Here support equipment and vehicles exhibit much more damage proportionally than do aircraft with primarily aluminum structure.

Some of the unexpected results include those for the Saudi Arabian desert, which has a very low humidity but a fairly high corrosion rate. This might be explained by the fact that the chloride content of the sand is approximately many times that found, for instance, on an US mainland beach. Kunsan AB Korea, located on the Yellow Sea, showed significantly higher corrosion rates than would have been projected based on the chloride levels alone. This may be the result of higher humidity levels and or other pollutants. Guam and Hawaiian locations had much lower rates than would have been projected based on the chlorides (Figure 3). Clearly other factors or synergistic effects exist.

This information is useful but insufficient to provide a basis for rates for corrosion predictive modeling. First, it cannot be concluded that corrosion rates occurring within a lap joint, for instance, will be the same as those on a boldly exposed surface. Likewise, just because an aircraft is based at a specific ground based location, does not mean that the severity of that location is the primary driver of the corrosion rates. This is particularly true of large transport aircraft, which routinely are exposed to a variety of environments.

Aging Aircraft Corrosion efforts have most recently focused on anticipating and managing lap joint corrosion. Thus, meaningful environmental severity data was required. In addition to the simulated and actual lap joint samples exposed on the racks at the KC-135 bases, small lap joint coupons were exposed on these racks at multiple locations. Not only are corrosion rates required for this effort but also the damage profiles to allow the mechanical impacts of the corrosion damage to be determined. This profiling has been done using both laser profilometry and other methods. Characterization of this damage and its impact is the subject of several other papers being presented at this conference.

The lap joint exposures have resulted in further refinement of the initial rankings. The corrosion rates of aluminum lap joints are at least 5 times larger than those of the openly exposed samples with values as high as 20 times larger. However, the 300:1 spread in openly exposed rates over the various locations shrinks to less than 20:1 for the laps. Furthermore, the damage profiles are vastly different than that of the openly exposed coupons with most of the damage occurring just inside the occluded area of the lap. From this and other laboratory testing, corrosion rates appear to be high even in areas judged to have lower chlorides and otherwise appearing to be less severe. Ostensibly it would seem a threshold of chloride is required to nucleate and sustain corrosion after which times of wetness etc. may be the determining factors for rate. While the reasons for the observed behavior are open to speculation, this corrosion damage is being characterized and quantified for a broad range of environmental severities.

Associated laboratory studies have also shown that contaminants can easily be wicked into unprotected lap joints but a drying out of these joints does not occur except under relative extreme conditions. Thus, lap joints previously exposed in a severe environment are being moved to various more benign environments and the corrosion rates subsequently determined for both the previous exposed and newly exposed joints in the more benign environments.

To ascertain the validity of these indices for use in determining actual damage requires that these exposure racks be flown on actual aircraft so that severities could be established and matched to actual corrosion damage. Both interior and exterior locations of the aircraft must be measured for the data to be specifically beneficial. These and other types of exposure racks are currently being flown and data extracted. The racks are placed in interior cargo areas as well as wheel well areas of both USAF and US Coast Guard aircraft.

Such Environmental Severity Indices do not necessarily have to determine the actual rates used in the modeling, but rather may tie a particular exposure to a separately determined rate derived from other work. As with fracture mechanics models, this work is focused on rates at which existing corrosion in specific materials and specific structure will grow. This then may serve as a basis for maintenance and inspection frequencies. It may also provide a quantitative basis for opportunistic repair of corrosion rather than the currently mandated immediate action. The effects of exposure and mechanical stress on the breakdown of protective systems is not a critical aspect of this effort which focuses primarily on the growth rates of preexisting corrosion. Complimentary work is being done in other research and development programs to ascertain the time to nucleation of such corrosion.

While extensive exposure testing continues, both the quality and quantity of data should allow anticipatory approaches to corrosion inspection and repairs. These indices will soon be included in USAF technical data as the basis for some prevention activities. As the tools are developed, improved corrosion detection, analysis, and repair options can be optimized and focused. This should facilitate management of corrosion by specific tail number, with opportunistic corrosion inspection and repair.

REFERENCES

1. Final Report for Air Force Environmental Severity Ranking: Contract Number F09603-95-D-0053; NCI Information Systems; November 1998.

2. National Acid Precipitation Assessment Program, (Benarie 1983; Lipfert et al 1985; Benarie and Lipfert 1986).

3. Kinzie, R. C.; Peeler, D. T., "Corrosion Modeling of KC-135 Lap Joints", Proceedings of the 20th Symposium of the International Committee on Aerospace Fatigue, 14-16 July 1999, Bellevue WA.

4. Abbott, W. H., Final Report for Field Site Reactivity Monitoring: Subcontract Number NCI-USAF-9138-001; 29 February 2000.

FIGURES

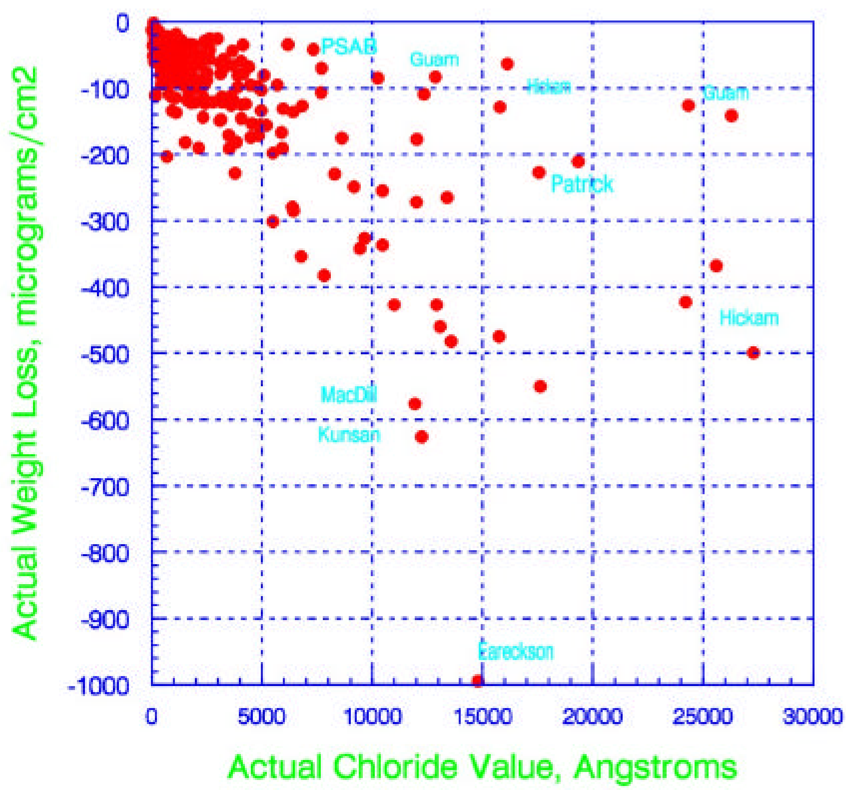


FIGURE 3. CORRELATION OF 6061 ALUMINUM WITH CHLORIDE SENSOR VALUES

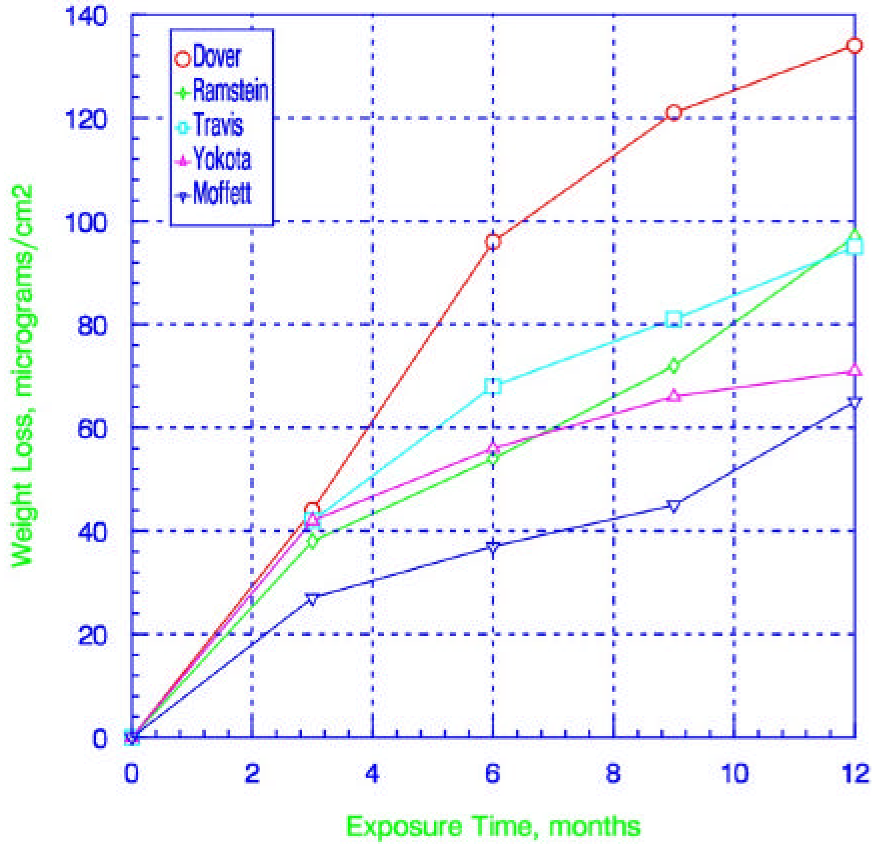


FIGURE 4. CORROSION OF 6061 ALUMINUM DURING OUTDOOR EXPOSURES

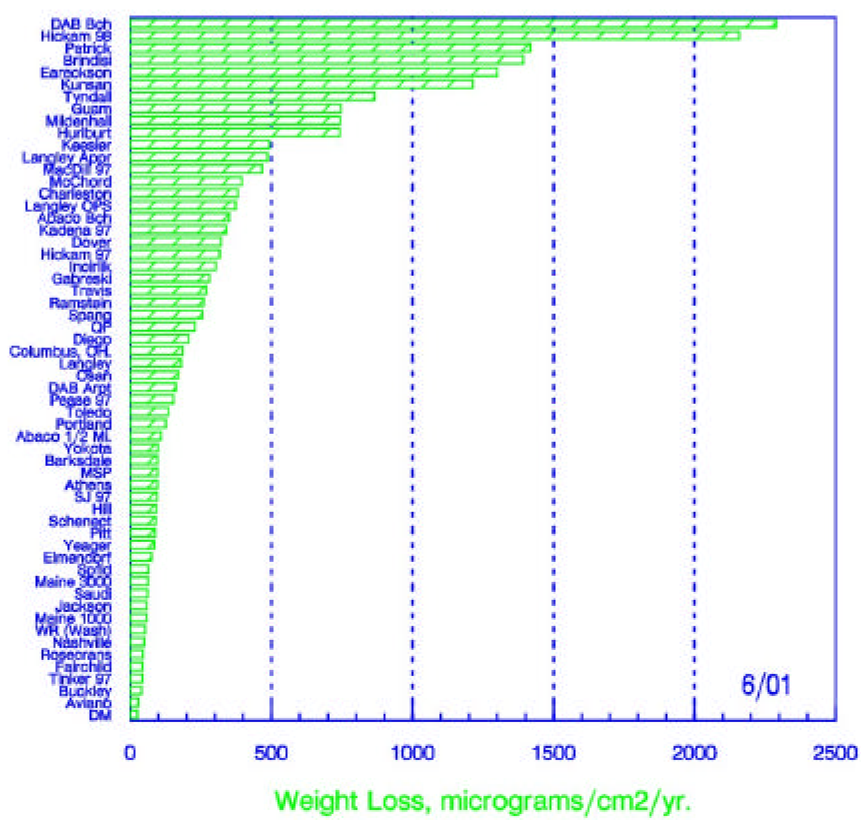


FIGURE 5. WEIGHT LOSS OF 7075 ALUMINUM AT OUTDOOR FIELD SITES

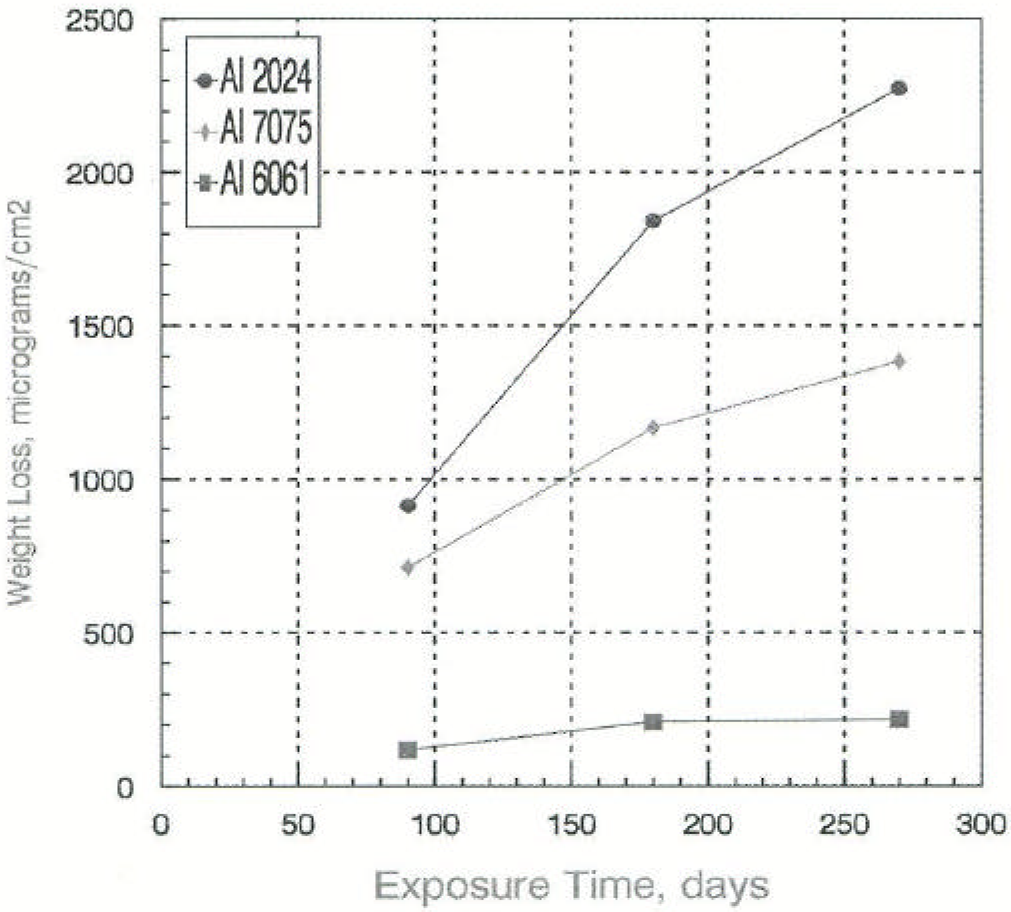


FIGURE 6. EFFECTS OF ALLOY TYPE ON CORROSION OF ALUMINUM

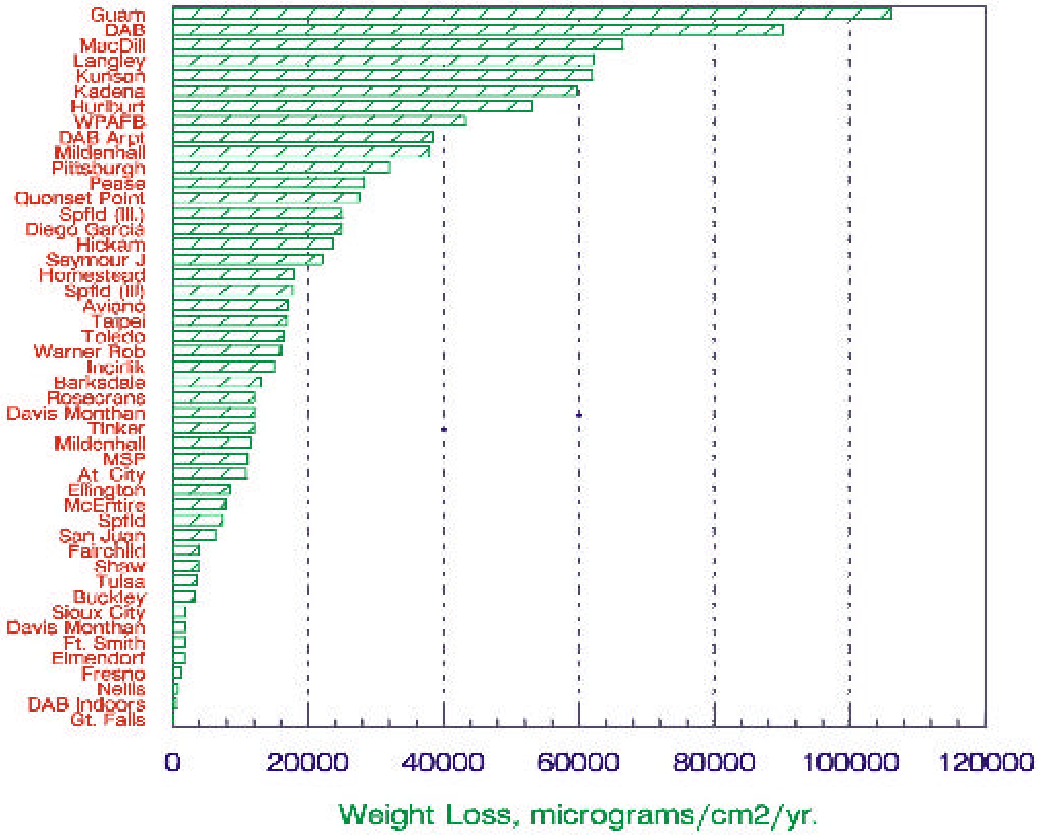


FIGURE 7. WEIGHT LOSS OF 1010 STEEL IN OUTDOOR BASE ENVIRONMENTS

This page has been deliberately left blank

Page intentionnellement blanche

Life and Damage Monitoring-Using NDI Data Interpretation for Corrosion Damage and Remaining Life Assessments

Jerzy P. Komorowski, David S. Forsyth, Nicholas C. Bellinger
Institute for Aerospace Research, National Research Council Canada
Building M-14, 1200 Montreal Road
Ottawa, ON, Canada K1A 0R6
email jerzy.komorowski@nrc.ca

David W. Hoeppner, P.Eng., Ph.D.
Department of Mechanical Engineering, University of Utah
50 S. Central Campus Drive, Room 2202
Salt Lake City, UT 84112-9208, USA

Introduction

The progress achieved in the understanding and modelling of fatigue and fracture in aircraft structures has led to the development of Damage Tolerance as the framework for ensuring continued airworthiness. Success of Damage Tolerance and Fail-Safe concepts combined with general economic conditions led to the current situation where an increasing number of aircraft are operated beyond their original design goal.

When today's ageing aircraft were designed and built most manufacturers made little allowance for corrosion. Corrosion protection was given low priority as the structures were typically not expected to remain in service beyond 20 years. While all elements of the aircraft industry (the operators and maintainers, certification authorities and manufacturers) recognised that corrosion could have a negative impact on the structural integrity, this impact could not be quantified until recently. Corrosion continued to be viewed as an economic burden since all corrosion found in an aircraft structure had to be "fixed". This maintenance practice usually included removal of the corrosion and when the net section was found to be reduced by a certain amount (typically 10%), the affected part had to be repaired. Often the term "fix" represented the replacement of an expensive component.

Research programs in the United States and Canada are developing a new corrosion management approach, based on that proposed by Kinzie and Peeler [1]. The Holistic Life Prediction Methodology (HLPM) being developed works within the existing aircraft structural integrity framework, and includes environmental as well as fatigue mechanisms for degradation. HLPM places new requirements on non-destructive inspections (NDI) for corrosion beyond simple material thinning. Quantitative measures of characteristic corrosion effects are required for structural analysis. The United States Air Force and the Canadian Department of National Defence are sponsoring work at (among others) the National Research Council Canada (NRC) to develop the data analysis tools and infrastructure to provide quantitative NDI results for HLPM in a maintenance environment.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Corrosion in Airframes

There are many types of corrosion that occur in aircraft structures (see Wallace and Hoeppner [2] for detailed definitions). This paper will concentrate on splice joints. The most common type of corrosion in splice joints is general attack on the faying surfaces. This can occur when adhesive bond or sealant between the layers breaks down, allowing moisture ingress. General attack is often accompanied by pitting corrosion. In severe cases, general attack evolves into exfoliation corrosion. Each of these types of corrosion can be characterized in terms of thickness loss of the original material. However, the effects of the different modes of corrosion on splice joint fatigue life are not well understood. Brooks et al.[3] have shown that the topography of the corroded faying surface can severely reduce fatigue life of a joint.

Another key mode of damage caused by corrosion in splice joints is pillowing. Pillowing is the deformation of the layers of the joint between the fasteners. This deformation is due to the fact that the corrosion products, aluminum oxides, have a much greater volume than the original material. The changes in the stress state of the splice joint due to pillowing change the locations, orientations, and aspect ratios of cracks emanating from the rivets [4], which are traditionally believed to be the life-limiting damage mode. Pillowing stresses may be large enough to be the cause of pillowing cracks, which are environmentally assisted cracks under sustained stress [5]. Wanhill [6] through very meticulous microscopic study has shown that nearly every fastener hole in a corroded splice joint may be subject to corrosion pillowing crack damage. He has in effect identified corrosion as a source of multi-site damage that might have significant implications to structural integrity.

Finally, the pillowing stresses can cause failure of the rivets themselves [7]. During disassembly of service-retired aircraft specimens at NRC, many examples of cracked and failed rivets have been found. Often, the failed rivets are still in place; held by corrosion, mechanical interference, or paint. This makes detection of these failed rivets by visual inspection impossible.

Pre-corroded (5% average thickness loss) splice joint coupon specimens have demonstrated close to 50% reduction in life to visible crack [8]. A holistic life prediction model including all identified corrosion effects in a splice joint at 10% average thickness loss has predicted over 80% reduction in life to critical crack size as compared to traditional analyses involving an 0.05” flaw growing to a 1 or 2-bay failure [9].

Nondestructive Inspection for Corrosion in Airframes: Current Practices

Despite years of increased funding for the development of nondestructive inspection (NDI) techniques for corrosion in airframes, little has changed in military or commercial aviation maintenance practices. In the commercial world, operators in general must repair corrosion damage upon discovery: the so-called “find and fix” practice. Thus NDI techniques sensitive to small amounts of corrosion damage are undesirable: operators feel current practices maintain an acceptable level of safety, and more sensitive NDI will end up increasing maintenance costs by provoking early repairs. Although military operators may not have the same regulatory requirements, the same conditions prevail.

In specific cases where flight safety is threatened by corrosion, such as the DC-9 spar cap or on landing gear, NDI techniques have been developed and successfully implemented in commercial operation. In other cases where imminent danger is deemed unlikely, such as splice joints and wing planks, NDI is in general restricted to visual inspection only. For commercial operators who achieve design service objectives in a much shorter time frame than military operators, this mode of operation has been somewhat successful from a safety point of view. Notable exceptions are the decompression accidents of a Boeing 737 operated by Far Eastern Air Transport in 1981, and another Boeing 737 operated by Aloha Airlines in 1988. Other incidents are documented in a report by Hoeppner et al. [10]. A key conclusion of this report was that the lack of consistent and stringent reporting requirements makes it difficult to determine when corrosion and/or fretting were important factors in the cause of incidents.

In the military environment, operation of ageing airframes beyond original design goals has in many cases come with increased maintenance costs and reduced availability due to corrosion. Detection of low levels of corrosion, even if below Structural Repair Manual (SRM) limits, provokes repair actions that may not have been planned. Thus sensitive NDI equipment and techniques are not welcome in the absence of supporting structural analysis tools.

The K/C-135 fleet operated by the United States Air Force (USAF) was previously targeted for full inspection of splice joints using the MAUS system. Recently it appears that the new NDI tools will not be implemented for this inspection, and visual inspection will continue to be used. Without any rigorous data to support the level of thickness loss detectable by visual means, this decision implies the operator is willing to accept a significant amount of section loss in these splice joints.

Holistic Life Prediction Methodology

Engineering is a profession based in science, but in the face of limited data or resources, the application of engineering judgement becomes an art. Life predictions of engineering designs are usually a compromise between what is practically achievable and scientifically rigorous. Years of progress in the physical sciences and rapid advances in information technology have opened new possibilities to engineers. Holistic life predictions, or in other words: “emphasizing the importance of the whole and the interdependence of its parts” [11] can now be considered.

Hoeppner has discussed the concepts of holistic life prediction methodologies as far back as 1971 [12]. At the time he and others searching for fundamental concepts had not yet used the word “holistic”, however they had advocated the use of systems approach to structural integrity. Hoeppner used the terms “holistic lifing” and “holistic structural integrity based design” in his FAA workshop on Aircraft Structural Fatigue in 1979 which he first gave at the University of Toronto. At the time some of the focus of his work was on engine components [13]. In 1994 Hoeppner [14] presented an comparison between Safe Life, Damage Tolerance (as employed with starting “flaw” size specified) and Holistic Structural Integrity Design.

Some of the most fundamental aspects of HLPM, as it is referred to currently, that Hoeppner listed are:

- Design is "closed loop" and concepts of failure processes and "pathology" of structures pervade all phases of design and operation.
- The material is characterised as manufactured, considering intrinsic variability and process variability.
- HLPM considers fatigue as a multi-faceted process with extensive internal and external interactions in all stages of the process. (see Table 1).
- HLPM uses continuum mechanics but defines limits of applicability - is material and process specific.
- HLPM defines "defects" in relation to representation, variability, probability of detection (POD), and fitness for purpose. Introduces Discontinuity, Heterogeneity concepts.

- Surfaces are recognised as Discontinuity sources - thus they are characterised, modelled, evaluated, and controlled related to requirements and variability.
- Nondestructive inspection and destructive characterisation are pervasive throughout the design process. Probability of detection is intrinsic to activities. Inspectability is part of the design requirement.
- Variability in cyclic load response, variability in material behaviour, and variability in processes are intrinsic parts of design. Determinism is not used except for simplistic explanations.
- Probabilistic Life Estimation Techniques always are used. Fatigue, fracture, and related activities are recognised as intrinsic parts of the design process. Testing technology development is an ongoing activity.
- The load spectrum is recognised to be structure specific, and thus extensive effort is expended to develop standardised load sequences. Dwell effects are considered.
- Extensive effort is expended to understand the failure processes and methodology is developed for specific physically based degradation processes. Initiation concept used only to refer to start of a specific failure process. Physical characterisation of the degradation process is pervasive throughout, with control of material and process related to the specific failure mechanism. Inspection is pervasive throughout related to the specific degradation/failure process.

Table 1. Stages in fatigue life (after Hoeppner[14]). Total life L=L1+L2+L3+L4.

L1	L2	L3	L4
NUCLEATION	"SMALL CRACK" GROWTH	STRESS DOMINATED CRACK GROWTH	FAILURE (FRACTURE)
Material failure mechanism with appropriate stress/strain life data Nucleated discontinuity (not inherent) type, size, location Presence of malignant D*, H* Possibility of extraneous effects: • Corrosion • Fretting • Creep • Mechanical Damage	Crack Prop. Threshold related to structure (micro) Structure dominated crack growth Mechanisms, rate Onset of stress dominated crack growth Effects of: • R ratio • Stress state • Environment (t, chemical, T) • Spectrum ⇒ waveform	Fracture mechanics: • similitude • boundary condition (LEFM – EPFM?) Data base** Appropriate stress intensity factor Initial D*, H* size, location, type Effects of: • R ratio • Stress state • Environment (t, chemical, T) • Spectrum ⇒ waveform	K_{Ic} etc. C.O.D. Tensile/compressive buckling

*Discontinuity, Heterogeneity, **i.e. Mil. Handbook 5

- Probabilities of occurrence of specific corrosion, wear, fretting, and thermal degradation mechanisms acting singly or conjointly with fatigue are acknowledged as part of the failure process. Constant evaluation, model development, test method development, and design methodology development are intrinsic to design process.
- Attempts to characterise details of the failure process. ACTIONS based on understanding and recognition (of knowledge gaps) of failure PROCESSES.
- Recognises probability of localised discontinuity formation in relation to failure processes. Recognises high probability of multiple discontinuity sites (e.g. fatigue cracks, corrosion pits, etc.) Recognises need for assessment of Principal Structural Elements, Structurally Significant Items, and Structurally Significant Locations on the basis of failure process interaction effects. Defines "damage" growth in trackable inspection parameters or recognises need to limit lives of components.
- Microstructural control through process control always used to control and optimise response of materials for specific failure mechanisms.
- Design is viewed in terms of response brought about by extrinsic factors. Develops response parameters. Approach is holistic.
- All life assurance personnel are trained in failure processes and structural pathology. A proactive methodology to provide immediate feedback into the design system of "lessons learned" is a part of the design system.

In 1998 Brooks et al.[3] have shown that the HLPM can be embodied in a practical analytical process that can account for corrosion fatigue effects on aircraft structural integrity. At the same time Simpson and Brooks [15,16] proposed the integration of corrosion into the aircraft structural integrity program (ASIP) using USAF ASIP tasks to illustrate how a revised program would be structured. While in the light of the conservatism of many in the structural integrity community the concept they have proposed might seem revolutionary, the changes can be introduced selectively.

Brooks and co-workers, partly under the support of USAF Corrosion Maintenance Initiative program, have developed a computer code ECLIPSE [17] - Environmental and Cyclic Life Interaction Prediction Software. ECLIPSE is based on HLPM, and the software has been exercised extensively on specific lap splice joint geometries and on early-pitting-to-fracture specimens and components. The software is being modified so that it can assist in making holistic life predictions for other geometries that may or may not have different driving mechanisms for the discontinuity progressions. ECLIPSE is expected to evolve to incorporate all fundamental aspects of HLPM listed above. It is currently subject to an intense verification effort under the USAF Corrosion Fatigue Structural Demonstration (CFSD) program, which also has been tasked with development of some of the data needed for HLPM. Phase 2 of the CFSD is currently in progress[18].

While models of discontinuity state evolution are fundamental to HLPM (ECLIPSE), practical application of the methodology to account for corrosion fatigue degradation requires the knowledge of corrosion growth rates in response to known environmental spectra and quantified nondestructive assessment of current (‘as-is’) condition. Significant progress in the field of corrosion rate assessment has been published by Kinzie [19]. The progress in the quantification of ‘as-is’ state is described in the following section.

NDI in Support of the HLPM

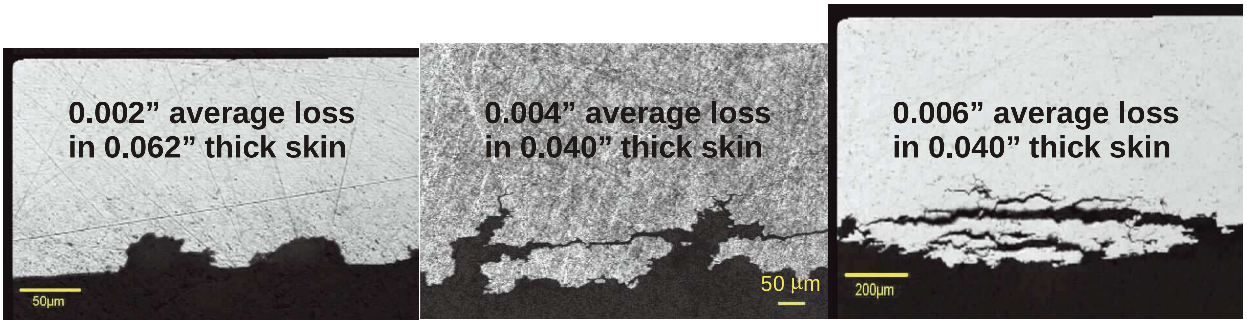
The first applications of HLPM were directed at the fuselage splice joint. This is a common element of construction for transport aircraft, including large ageing fleets such as the K/C-135, C-141, C-130, B-52, and P3/CP-140 fleets operated in the U.S.A. and Canada. Although there are many different joint configurations used, there are common properties which make NDI development applicable across a broad range of applications. The splice joint was also the target of many NDI development programs sponsored by the Federal Aviation Administration (FAA) in the USA in the early 1990's, thus there are a number of nearly mature NDI techniques available for this application.

HLPM is not only concerned with corrosion but also fatigue including MSD. Since fatigue has been the focus of intense study and concern in aircraft structural integrity, NDI methods used for crack detection are considered mature. There is a concern, however, that corrosion and modifications (repairs) to ageing structures will result in cracks in inaccessible surfaces and deeper layers which are less amenable to inspection. Repairs where flush rivets are replaced with button head rivets, or bonded or fastened doublers are installed, are examples of situations where crack detection is adversely affected. For the integration of NDI with maintenance and analysis tools, better recording of NDI for cracks is also a concern. Even data from manual inspections can be entered in databases, with a resulting improvement in damage reporting, tracking, and integration with analysis tools.

Previous NDI for corrosion in splice joints was aimed at measuring "general" thickness loss, that is, a spatially averaged measure of thickness of the individual layers of the joint. These developments were aimed at satisfying current FAA and industry practices. The analysis models developed to support HLPM require this measure or metric of corrosion, but also require information about faying surface topography and, in riveted joints, pillowing deformation and cracking.

Many NDI techniques have been developed to determine "general" thickness loss. Thickness measurement of the first layer can be carried out using ultrasonic, thermographic, or single frequency eddy current techniques. There are some difficulties in inferring general thickness loss from thickness measurements: First, sheet tolerances in the as-manufactured state are in the order of 5% for common skin thicknesses. Second, in older aircraft, variations in manufacturing and repairs are much more common, and often poorly documented. Second and third layer thickness measurements are much more difficult, and it is widely believed that multi-frequency and pulsed eddy current techniques hold the most promise for practical use in this situation.

For the determination of faying surface topography, it has been shown by Smith and Bruce [20] that the topography of the top faying surface can be measured using ultrasonic methods. However, the interlayer gap in corroded specimens is poorly defined, often without mechanical bonding, and containing varying amounts of corrosion product. Thus information about the second and deeper layers is almost impossible to obtain directly. It is believed that faying surface topography can be correlated with general thickness loss, and experimental work to date supports this [21,22] (see Figure 1 and Figure 2). This means that many of the currently available NDI techniques can be used for this metric.



(a) (b) (c)

Figure 1. Changing faying surface topography with thickness loss in sections from service-retired aircraft splice joints (from Bellinger et al. [22]).

NDI techniques also exist for measuring pillowing deformation, ranging from simple mechanical devices to automated optically-based systems [23]. The combination of a general thickness measurement indicating less than nominal thickness with an indication of pillowing is often the best indicator of the existence of material loss due to corrosion.

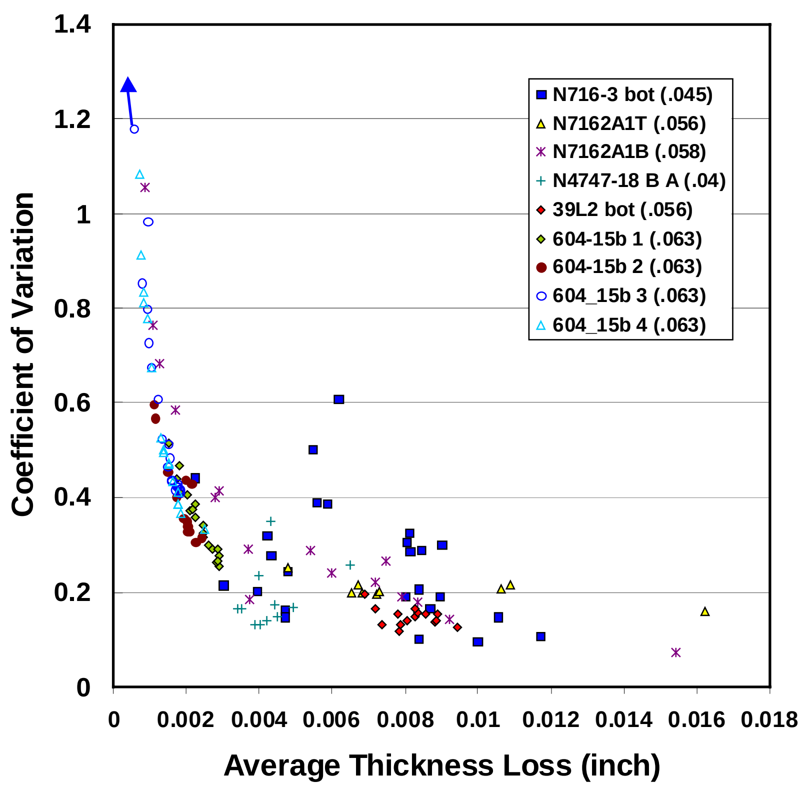


Figure 2. Correlation of faying surface topography feature with amount of material loss (from Bellinger et al. [22]).

NDI Infrastructure Requirements in Support of HLPM

In order to achieve better NDI, and make better use of NDI in performing a structural integrity assessment, there are “infrastructure” issues which need to be addressed. A conceptual approach to depot maintenance for corrosion in airframe components is shown in Figure 3. Inspection planning and recording become a significant task when faced with the large areas of splice joints on transport aircraft. Analysis of data also needs to be improved and automated so that quantitative NDI results on the metrics of interest can be directly entered into the structural analysis tools. NDI for fatigue cracks will likely have to undergo the same application of more rigorous data recording and analysis in order to achieve accurate assessment of MSD/WFD scenarios in the presence of corrosion.

The USAF is sponsoring the Corrosion Quantification (CQ) program at S&K Technologies, the Institute for Aerospace Research of NRC, and LMI Automotive in an effort to address these concerns. An existing proprietary inspection planning tool is being enhanced to allow for any type of NDI system and for the input of wireframe diagrams of airframes for data registration purposes (see Figure 4 for an example). The inspection planning software is also being enhanced to allow for integration with existing maintenance databases. Existing and new data registration and fusion tools are being evaluated for their ability to generate reliable and quantitative data from NDI, and for the export of this data directly into structural analysis tools.

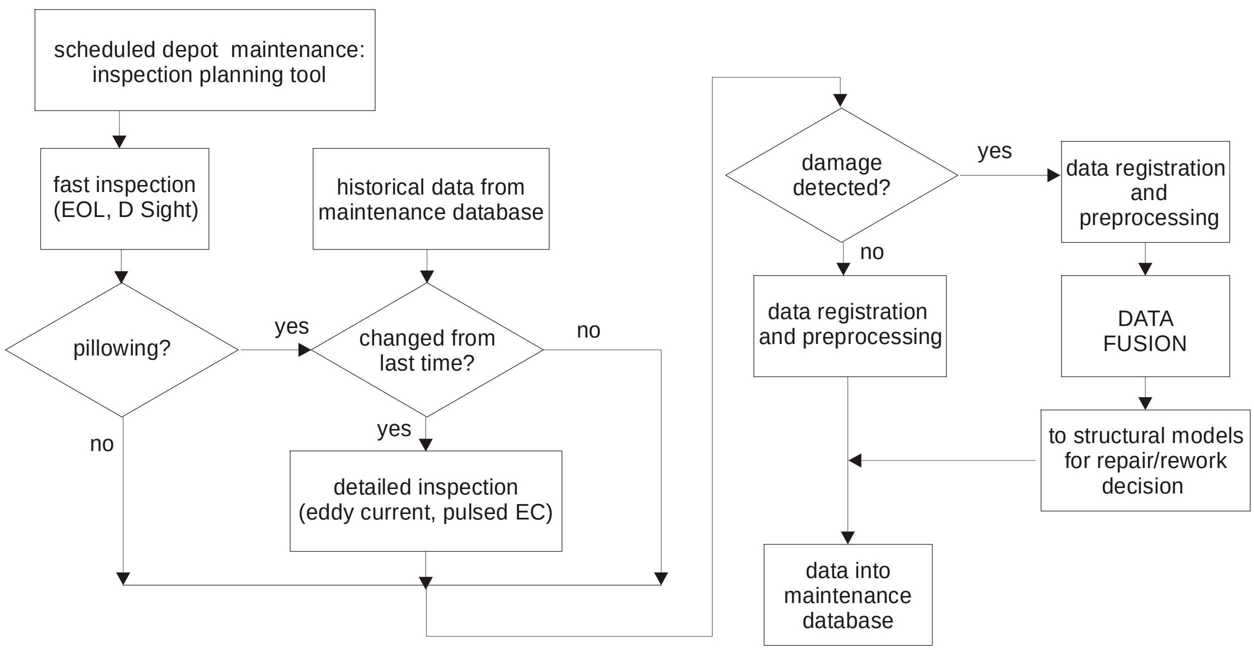


Figure 3. A conceptual flowchart for splice joint corrosion NDI during depot maintenance.

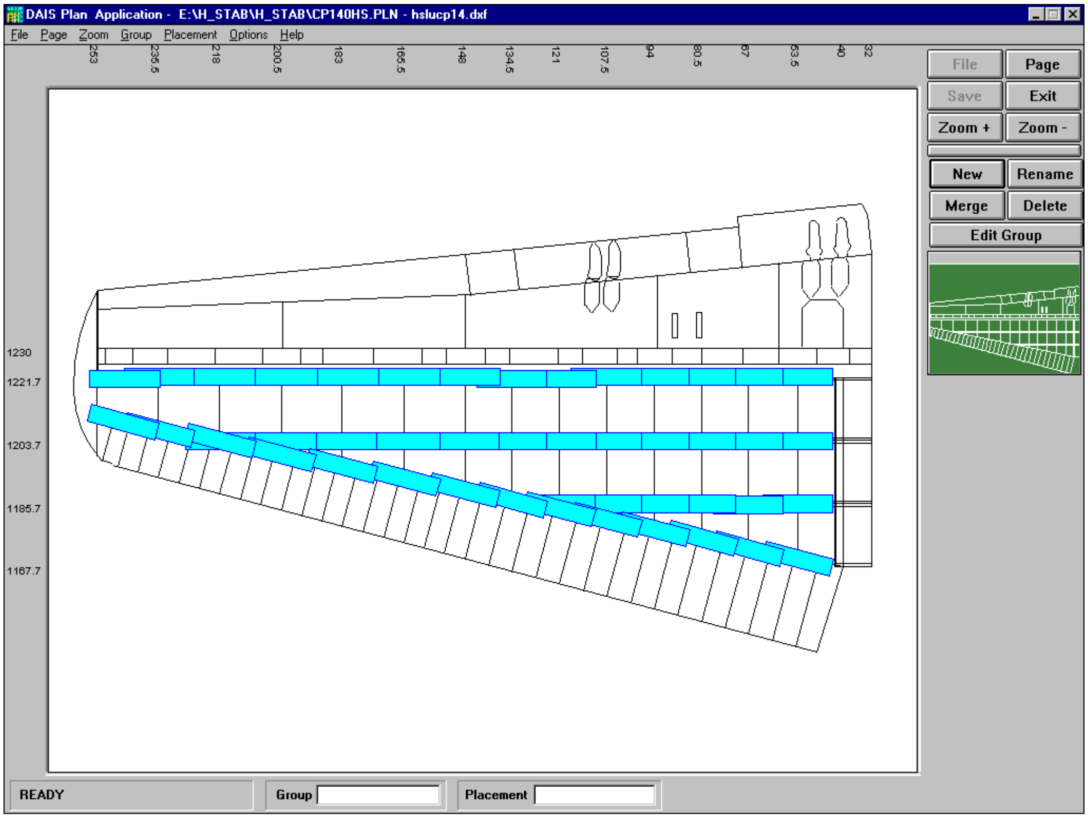


Figure 4. A view of inspection planning software being enhanced under the CQ program.

An example of data fusion applied to NDI of a lap splice joint from a retired Boeing 727 aircraft is shown in Figure 5 (see reference [24] for details). The data shown is for the second layer thickness. The image on the top shows thickness loss from nominal, obtained from the fusion of pulsed eddy current and Edge of Light inspections. This data has been registered on a co-ordinate system, and then can be compared to the image on the bottom, which is a thickness map obtained after disassembly of the joint. While these are preliminary results, they show promise for the use of modern data analysis in obtaining quantitative results from NDI. The CQ program is building on these results and seeking further improvements.

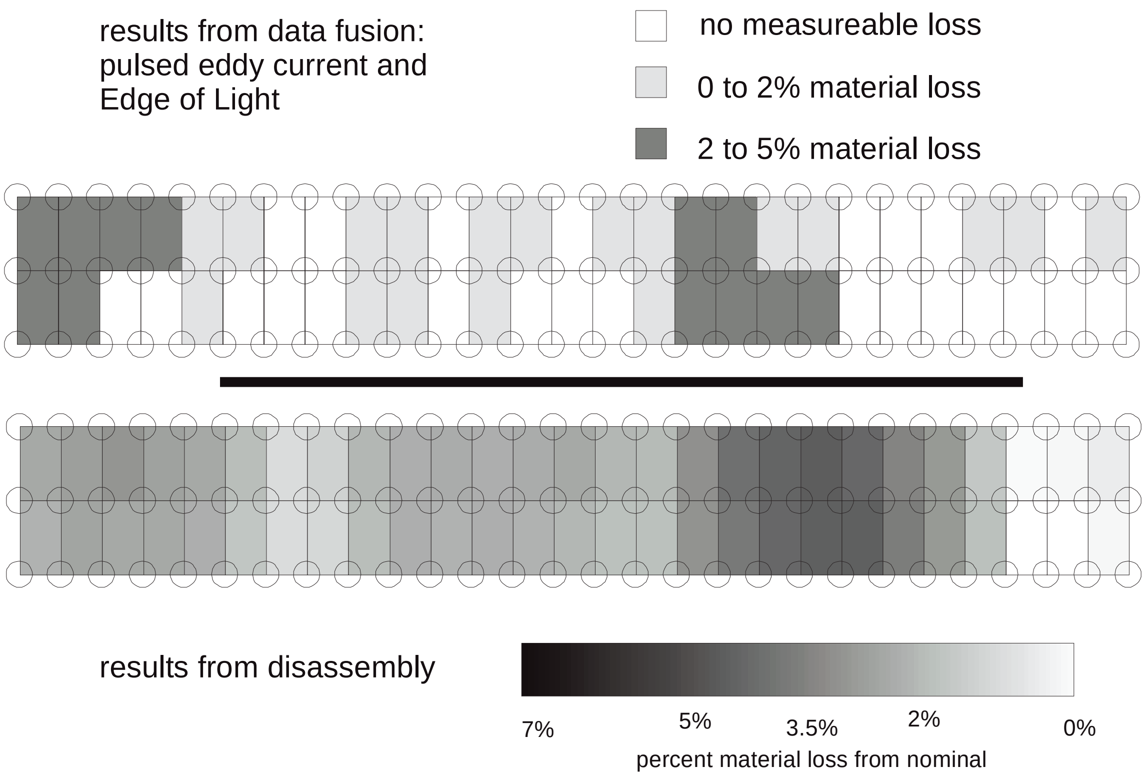


Figure 5. An example of data fusion used to obtain thickness data for the second layer of an actual lap splice joint (from Forsyth and Komorowski [24]).

Integration of HLPM Into the Maintenance Environment

A typical maintenance cycle time-line is shown in Figure 6. Between the end of the previous planned depot maintenance cycle, and the entry of the aircraft into the next PDM, no focused corrosion inspections are carried out. There are two reasons for this:

– Corrosion NDI tools are not deployed, and unless corrosion damage becomes severe enough to be evident under visual inspection, no corrective action is undertaken.

– In the absence of tools which account for the structural impact of corrosion, focused inspections before entry into PDM are discouraged as no rationale for a decision to continue to operate an aircraft with known corrosion damage can be found.

Aircraft thus enter PDM in an unknown condition, and the extent of damage and thus time required to repair (fix) corrosion can not be predicted, spares can not be ordered, special repairs can not be designed etc.

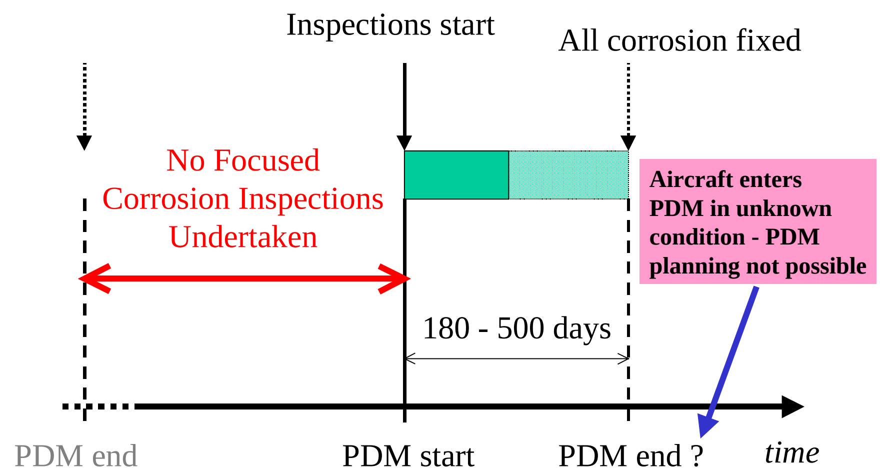


Figure 6. Current aircraft maintenance cycle time-line. PDM – planned depot maintenance.

The application of HLPM to splice joints has reached the maturity level that, along with the NDI tools described above, allows fleet managers to consider transitioning the technology. This process should occur gradually such that the necessary infrastructure can be put in-place along with staff training. Recent completion of the USAF Corrosion Maintenance Initiative (CMI) program and availability of dedicated corrosion splice joint inspection equipment allow some changes to be introduced today towards a more planned approach to PDM, as shown in Figure 7.

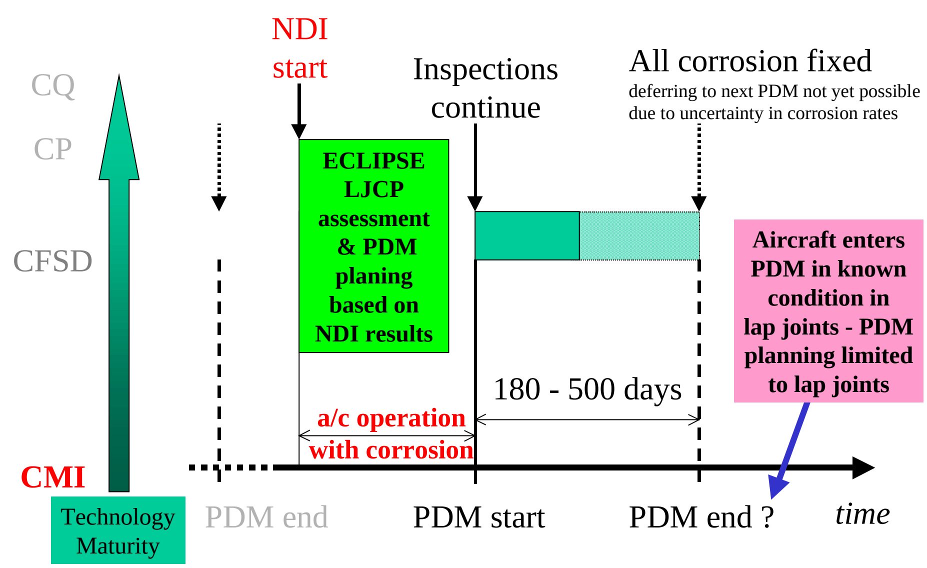


Figure 7. First step in the introduction of HLPM and NDI to Predict and Manage approach to aircraft maintenance. CFSD, CP and CQ are USAF corrosion technologies programs. LJCP – lap joint corrosion prediction program.

NDI inspections for corrosion could be carried out several months before an aircraft's entry into the PDM. Inspection data processed using the tools developed so far will help plan corrosion repairs, as the extent of damage would be known. Since ECLIPSE performs calculations from 'as-is' to 'to-be', the safety of operation of the aircraft with known corrosion would be assured. It is possible that some damage identified will require immediate attention. However, this is unlikely as such findings would indicate that current practices (as shown in Figure 6) are unsafe.

The benefits of the introduction of HLPM based code like ECLIPSE and NDI into the aircraft maintenance cycle will have to be limited to splice joints until the completion of CP – corrosion prevention and CQ – corrosion quantification programs which aim to expand the HLPM (ECLIPSE) to other aircraft structures. More data is also required to gain confidence in the ability to predict the rates of corrosion damage accumulation. Figure 8 shows the PDM cycle time line with fully implemented mature HLPM and NDI technologies. The PDM cycle will be significantly shortened, as few surprises will be uncovered in maintenance.

It is also possible that aircraft slated for near-term retirement will not require all corrosion to be fixed, or that less difficult repair schemes such as the application of corrosion prevention compounds (CPC) could be used instead of grindouts or part replacement. Data supporting confidence in corrosion growth rates and the efficacy of CPC's is being obtained to support this. Aircraft structures will be safely operated with known levels of corrosion until they are either retired or repaired at most opportune time.

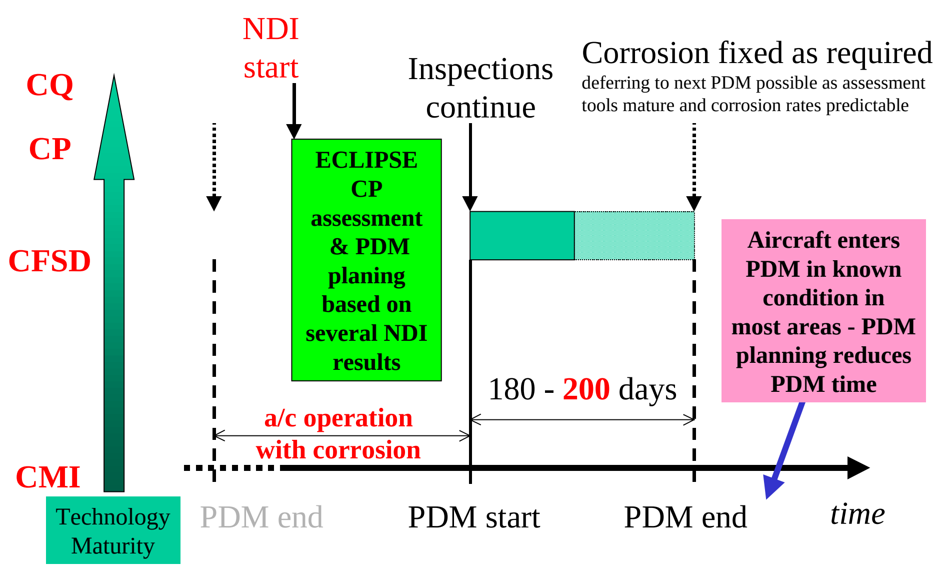


Figure 8. Full implementation of Predict and Manage HLPM based aircraft maintenance.

Conclusions

Holistic life prediction methodologies have been under development for at least the last 30 years. The ever-increasing availability of cheap computing power has now allowed HLPM to be implemented in practical analytical procedures. In the case of aircraft splice joints, the NDI technology required to implement HLPM is available today. The benefits of the implementation of HLPM into fleet management are immediate with current capabilities, and will increase as existing research programs come to fruition.

Acknowledgements

The authors are grateful to several organisations that have supported elements of the work that contributed to this paper. In particular we are grateful to USAF for their support through NCI Inc. and more recently S&K Technologies, and to the Department of National Defence through Defence Research and Development Canada.

Discussions and open exchange of ideas with Mr. Craig Brooks of Analytical Processes and Engineering Solutions have been truly invaluable.

References

1. Kinzie, R. and Peeler, D., "Managing Corrosion in the Aging Fleet: A New Approach to Corrosion Maintenance", in the *Proceedings of the Third Joint FAA/DoD/NASA Conference on Aging Aircraft*, Albuquerque, USA, September 1999.
2. Wallace, W. and Hoeppner, D. W., *AGARD Corrosion Handbook Volume 1 Aircraft Corrosion: Causes and Case Histories*, AGARD-AG-278 Volume 1, 1985.
3. Brooks, C. L., Prost-Domasky, S., and Honeycutt, K., "Corrosion is a Structural and Economic Problem: Transforming Metrics to a Life Prediction Method", *Fatigue in the Presence of Corrosion*, Report AC/323(AVT)TP/8; RTO-MP-18, North Atlantic Treaty Organization, Brussels, Belgium, 1999.
4. Bellinger, N. C., Komorowski, J. P., and Gould, R. W., "Damage Tolerance Implications of Corrosion Pillowing on Fuselage Lap Joints", *Journal of Aircraft*, **35**, No. 3, May-June 1998, pp 487-491.
5. Bellinger, N. C., and Komorowski, J. P., "Environmentally Assisted Cracks in 2024-T3 Fuselage Lap Joints", in the *Proceedings of the Third Joint FAA/DoD/NASA Conference on Aging Aircraft*, Albuquerque, USA, September 1999.
6. Wanhill, R. J. H., "Corrosion and Fatigue Assessment of Aircraft Pressure Cabin Longitudinal Lap Splices", presented at the *5th International Aerospace Corrosion Control Symposium*, Amsterdam, Netherlands, November 1999.
7. Forsyth, D. S., Fahr, A., Chapman, C. E., Bellinger, N. C., and Komorowski, J. P., "NDI Capabilities for Airframe Corrosion Damage Tolerance", in the proceedings of the *2000 USAF Aircraft Structural Integrity Program Conference*, San Antonio, USA, December 2000.
8. Eastaugh, G. F., Straznicky, P. V., Krizan, D. V., Merati, A. A., and Cook, J., "Experimental study of the effects of corrosion on the fatigue durability and crack growth characteristics of longitudinal fuselage splices", in the *Proceedings of the Fourth Joint DoD/FAA/NASA Conference on Aging Aircraft*, St. Louis, USA, September 2000.
9. Brooks, C. L., Honeycutt, K., and Prost-Domasky, S., "Case Studies for Life Assessments With Age Degradation" in the *Proceedings of the Fourth Joint DoD/FAA/NASA Conference on Aging Aircraft*, St. Louis, USA, September 2000.
10. Hoeppner, D., Grimes, L., Hoeppner, A., Ledesma, J., Mills, T., and Shah, A., "Corrosion And Fretting As Critical Aviation Safety Issues: Case Studies, Facts, And Figures From U. S. Aircraft Accidents And Incidents", ICAF DOC 2055, EMAS, U. K., 1995, pp. 87- 106.
11. The American Heritage Dictionary of the English Language, Third Edition Copyright © 1992 by Houghton Mifflin Company.
12. Hoeppner, D. W., "Corrosion Fatigue Considerations in Materials Selections and Engineering Design", *Corrosion Fatigue: Chemistry, Mechanics, and Microstructure*, NACE, 1972, pp. 3-11.
13. Hoeppner, D. W., "Parameters that Input to Application of Damage Tolerance Concepts to Critical Engine Components", in *Damage Tolerance Concepts for Critical Engine Components*, NATO-AGARD AGARD-CP 393, France, 1985, pp. 4-1 to 4-16.
14. Hoeppner, D. W., "The Role of Inherent Discontinuities (D) and Heterogeneities (H) in the Process of Nucleation of Fatigue Cracks in Aircraft Structural Materials", presented at the *Third DND-Canada Conference on Structural Integrity*, September 1994, Ottawa, Canada.

15. Simpson, D. L., and Brooks, C. L., “Tailoring the structural integrity process to meet the challenges of aging aircraft”, *International Journal of Fatigue*, **21**, 1999 pp. S1-S14.
16. Brooks, C. L., and Simpson, D. L., “Integrating Real Time Age Degradation Into the Structural Integrity Process”, *Fatigue in the Presence of Corrosion*, Report AC/323(AVT)TP/8; RTO-MP-18, North Atlantic Treaty Organization, Brussels, Belgium, 1999.
17. Brooks, C. L., Honeycutt, K., and Prost-Domasky, S., “ECLIPSE - Environmental and Cyclic Life Interaction Prediction Software”, presented at the *Aeromat Conference*, Dayton, USA, May 1999.
18. Falugi, M., Hoeppner, D. W., Eastaugh, G. E., Brooks, C. L., Prost-Domasky, S., Honeycutt, K., Ginter, T. J., Bell, R. P., “Application of the Lockheed Martin Corrosion Fatigue Structural Demonstration Program to Corrosion Induced Damage on Aging Aircraft-Phase I and Phase II Plans”, in the *Proceedings of the Fourth Joint DoD/FAA/NASA Conference on Aging Aircraft*, St. Louis, USA, September 2000.
19. Kinzie, R. C., “Anticipating Damage in the Fleet: Development of a Robust Environmental Severity Index”, in the *Proceedings of the Fifth Joint NASA/FAA/DoD Conference on Aging Aircraft*, Orlando, USA, September 2001.
20. Smith, R. A., and Bruce, D. A., “Pulsed Ultrasonic Spectroscopy for roughness measurement of hidden corrosion surfaces”, *Insight* **43**, 2001, pp. 168-172.
21. Bellinger, N. C., Cook, J. C., Forsyth, D. S., and Komorowski, J. P. “The Role of Surface Topography in Corroded Fuselage Lap Joints”, in the proceedings of the *2000 USAF Aircraft Structural Integrity Program Conference*, San Antonio, USA, December 2000.
22. Bellinger, N. C., Cook, J. C., Forsyth, D. S., and Komorowski, J. P., “Damage Characterization of Corroded 2024-T3 Fuselage Lap Joints”, in the *Proceedings of the Fifth Joint NASA/FAA/DoD Conference on Aging Aircraft*, Orlando, USA, September 2001.
23. Forsyth, D. S., Gould, R. W., Komorowski, J. P., “Correlation of Enhanced Visual Inspection Image Features with Corrosion Loss Measurements”, in *TONE Volume 3: III International Workshop - Advances in Signal Processing for Non Destructive Evaluation of Materials*, edited by X. P. V. Maldague, ASNT, Columbus, 1998, pp 365 - 372.
24. Forsyth, D. S., Komorowski, J. P., “The Role of Data Fusion in NDE for Aging Aircraft”, in *Nondestructive Evaluation of Aging Aircraft, Airports, and Aerospace Hardware IV*, Ajit K. Mal editor, Proceedings of SPIE Vol. 3994, 2000, pp 47 - 58.

F-15 Structural Life Enhancement

Jeff L. McFarland
The Boeing Company
MC S1066420
PO Box 516
St. Louis, Missouri 63166, USA

Dr. Rigoberto Perez
The Boeing Company
MC S1021322
PO Box 516
St. Louis, Missouri 63166, USA

Abstract

This paper summarizes the effort funded by the United States Air Force Research Laboratory at Wright Patterson Air Force Base to identify the problem structural areas on the F-15 and recommend appropriate solutions with the development of new technology. Recent modifications to the F-15 airframe structure have taken place or are under consideration to reduce honeycomb water corrosion, reduce maintenance costs, quickly produce spares, provide technology demonstration for future aircraft, and eliminating/ reducing maintenance, including NDI inspections and problem fatigue cracking issues. The recommendations in the plan address solutions that can be integrated into an overall life extension plan for fighter aircraft.

1.0 INTRODUCTION

Fatigue and corrosion damages are issues in the USAF aging fighter aircraft fleet. Periodic inspections and replacements of the damaged components have solved many of these problems, but this approach is expensive and significantly reduces aircraft availability. In addition, damage can reoccur requiring the same repair to be performed several times. Alternative structural life enhancement technologies exist which offer more efficient solutions.

The F-15 has provided a reliable and robust airframe, even though first flight was nearly thirty years ago. The material and design technology used for this airframe was based on technology available in the late sixties, where the extensive use of titanium was considered a major technological change. The addition of the F-15E into the USAF inventory in the mid-eighties incorporated the first new structural technology, BLATS, Built-up Low-cost Advanced Titanium Structure. And, while no major structural fatigue issues have surfaced in thirty years of operation, the F-15 aircraft has experienced secondary structural cracking due to buffet and sonic fatigue at various locations along with secondary structure cracking due to maneuver loading. Additionally, corrosion in secondary and some aluminum primary structure has begun to surface, creating maintenance and logistics issues. Several recent structural modifications to the F-15 have taken place, or are under consideration: reducing honeycomb water corrosion; reducing the cost of spare parts, substitutions to quickly produce spares; providing technology demonstrators for future aircraft; and eliminating/reducing maintenance, including NDI inspections and problem fatigue cracking issues. To assess the remaining airframe issues, a survey was taken and those components that could benefit from structural life enhancement technologies were identified for further study.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Included in this paper are the results of the study conducted for the United States Air Force Research Laboratory at Wright-Patterson Air Force Base, to identify the problem areas on the F-15 and determine appropriate solutions. This study provided an assessment of available structural life enhancement technologies and was focused on technologies that can be applied on existing structure. In addition to this study, recent technological upgrades to the F-15 as spares, technology demonstrators, corrosion prevention enhancements, or resolutions to fatigue areas of concern have been included.

Each of the identified problem areas were ranked to determine which enhancement techniques offer the most potential in cost savings and increased aircraft availability. The high ranking critical locations on the F-15 (that have not been recently upgraded) include:

- Inner Wing, Lower Wing Skin at Shoulder Rib and Intermediate Spar, Edge of Skin
- Fuel Cell No. 1, Lower Keel Longeron
- Vertical Tail, Main Torque Box Composite Skin
- Vertical Tail, Main Torque Box Honeycomb Assembly
- Horizontal Stabilator, Main Torque Box Honeycomb Assembly
- Horizontal Stabilator, Main Torque Box Composite Skin
- Aft Fuselage, FS 712 Bulkhead Outboard Section
- Inner Wing, Pylon Rib
- Outer Wing, Upper Torque Box Skin at Ribs and Ribs at XW 172, XW 188 and XW 206.

From the list of high-ranking locations, several enhancement techniques were recommended for additional research and development. These included: composite patches, health monitoring, and damped bonded patch. The report was completed with plans to develop, demonstrate and transition these techniques to the F-15 aircraft. The recommendations in the plan address solutions that can be integrated into an overall enhancement plan for the F-15.

2.0 F-15 STRUCTURAL PROBLEM AREAS

2.1 Critical Locations

A survey of available sources was conducted to identify problem areas on the F-15C/D and E structure. In general, the F-15 was designed and is inspected based on crack growth methodology, or the time it takes for an initial assumed flaw to grow to failure. The analysis assumes that a "rogue" flaw exists at the most critical location on a part. The size of the initial flaw assumed was dictated by what current NDI instrumentation can reliably detect. A rogue flaw can exist for any number of reasons; i.e., material voids or inclusions, mechanically made notches or scratches, etc. In addition, the F-15 has been through numerous full-scale fatigue tests and has had extensive in-service time. Data documenting damage found during test article teardowns and fleet inspections was collected. The results of this documentation has not been included in this paper for brevity, but included descriptions of the items, how the location was determined to be a candidate, what the issue is with the location, and what is the most likely enhancement technique to eliminate the concern.

2.2 Rating System for the F-15 Critical Locations

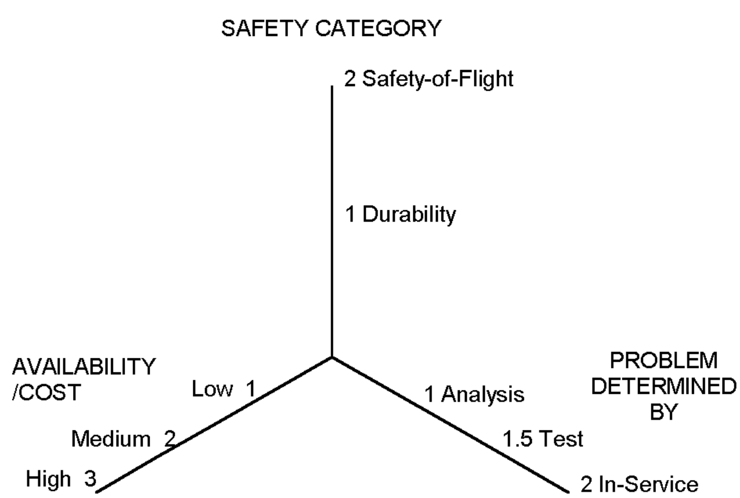
It was necessary to develop a numerical rating system for the locations found to require a maintenance or inspection action on the F-15. A three dimensional rating system was developed to weigh important aspects of each location. The three parameters used to evaluate the critical locations were safety, aircraft availability/cost of maintenance, and how the problem was determined. The product of the three parameters provides an overall rating. Figure 1 graphically illustrates the rating system.

2.2.1 Safety Category – Safety-of-flight criticality is the single most important aspect of all weighting parameters. The F-15 bases inspections on crack growth methodology, or the time it takes an initial flaw to grow to failure. For this category a rating of “1” was assigned to durability critical structure and a value of “2” was assigned to safety-of-flight critical structure. The safety-of-flight criterion, with initial flaws placed at the most critical locations and oriented in the most critical orientations, was defined to prevent failure from a rogue flaw.

2.2.2 How the Problem was Determined – All critical locations fell into one of three categories based on how the problems were determined; by analysis, by full scale fatigue testing, or by in-service failures. Analysis is the most common method of determining critical locations. This was used as the baseline and a value of one was assigned to critical locations found in this manner. Full scale fatigue testing was assigned a value of one and a half since failures found during testing or in the subsequent tear down simulate cracking that could occur in service. Loading tends to be more realistic than by analysis since loading will distribute differently than a coarse Finite Element Model might indicate. In-service failures were assigned a value of two. Failure during actual usage provides the best and clearest picture of how aircraft are used operationally.

2.2.3 Aircraft Availability/Cost – Three basic ratings (low=1, medium=2 and high=3) were assigned to all critical locations based on aircraft availability and cost. Aircraft availability refers to the impact of part maintenance on down time or mission capability and represents the effect of repeat repairs on the same component. Cost not only includes the part, but all maintenance operations. Often, replacing a “buried” part will result in a cost to the customer that far exceeds the basic part cost if major disassembly is required. Availability and cost were combined into a single parameter because of the close correlation between them. An expensive and difficult repair can lead to longer downtime. However, lower cost repairs can also cause significant downtime if parts or equipment are not available. In general, the effect of maintenance on aircraft availability is greater during field repairs.

FIGURE 1
RATING PARAMETERS FOR F-15 CRITICAL LOCATIONS



3.0 STRUCTURAL LIFE ENHANCEMENT TECHNIQUES

3.1 Life Enhancement Techniques - State of the Art

The following subsections describe the current state of the art of various enhancement techniques available. Particular aspects of these techniques are classified as “mature”, “near general application”, or “needs additional development”. The criteria for classification are as follows.

3.1.1 Mature – Techniques under this category are in general use on operational aircraft. Procedures are well documented in Tech Order Manuals, for example. Analysis methods associated with the technology are in general use within industry. A brief listing includes:

3.1.1.1 Cold Working - Through this process, a hole is radially expanded resulting in circumferential compressive residual stresses. The compressive stresses delay crack initiation and can retard crack growth. Structural enhancements applications of parts installed on the aircraft have been improved by the development of one-sided cold working methods and rivetless nutplates [3.1]. Additional work is necessary to provide full analytical benefits from cold working.

3.1.1.2 Interference Fit Fasteners - These types of fasteners increase fatigue life by reducing the stress concentration of the fastener hole and by introducing beneficial compressive residual stresses around the hole wall. There are three types of interference fit fasteners that can be used to improve fatigue life: solid rivets, pin and collar fasteners (hi-loks and lockbolts) and tapered fasteners (taper-loks). As with cold working additional work is necessary to provide full analytical benefits from cold working.

3.1.1.3 Shot Peening – The shot peening. This process adds beneficial residual stresses for increased fatigue resistance. The material being peened must be considered in choosing shot peening intensity, shot size, shot hardness, and coverage. Shot peening is primarily used in retarding crack initiation because the depth of the residual stresses is too shallow (up to 0.03 in.) to affect crack growth. Care must be taken not to cause surface damage from over peening. High strength steel landing gears are a major application in aircraft. Recent developments in this area include the use of shot peening to search for inter-granular corrosion and exfoliation [3.2].

3.1.1.4 Fleet Monitoring - Fatigue tracking and usage monitoring programs have evolved with each new weapon system and have ranged from tracking of flight hours to multi-channel recorders and strain gages. Because of conservatism, tracking flight hours can result in premature retirement or in maintenance performed more frequently than necessary. Aircraft flown under very severe missions can exceed the design usage and accumulate damage more rapidly than expected. To improve the accuracy of aircraft tracking programs, counting accelerometers were installed in various aircraft models. These counting accelerometer-based systems can be classified as a second generation tracking system, since they follow after the simpler flight hour based method. The need for further improvements in accuracy led to the development of the third generation tracking systems based on multi-channel recorders to track additional parameters. Fourth generation systems have incorporated strain gages to more directly compute stresses in the tracked components.

It is possible to upgrade fatigue-tracking systems to obtain more accurate data. As mentioned earlier, more accuracy reduces conservatism and can result in longer inspections intervals or longer operational life. However, the cost of a fatigue tracking system upgrade can be prohibitive for some fleets. One of the reasons for the high expense is the cost of integrating a new system with the existing hardware and software. A stand-alone health monitoring system that can be installed without affecting other aircraft system offers many advantages.

3.1.2 Near general application – A technique in this classification has been used on a limited number of operational aircraft or has been applied on only specific aircraft types, but is not in general use on the USAF fleet. Process is documented, but requires engineering supervision. Analyses methods associated with the technology have been verified, but are in use by a limited segment of industry (one contractor, for example).

3.1.2.1 Corrosion Prevention - Grinding to remove damaged material or part replacements are standard methods for repairing corrosion. Coatings are needed to protect the part from subsequent corrosion with the goal of restoring the required operational life or in some cases extend the life. One of the major issues currently affecting corrosion control is the effect of non-chromated primers and conversion coats. These materials will see increased use as conventional coatings containing hexavalent chromium are replaced.

3.1.2.2 Damped Bonded Patch - Fatigue from vibratory loads is a common cause of structural failure in fighter aircraft. The F-15 lower fuselage skin has been affected by acoustic fatigue, while the outer wing upper skins have experienced buffet. Constrained viscoelastic layer damping treatments are one method to reduce vibration.

A more effective method being evaluated combines damping within a composite material patch. The patch can be bonded very effectively to the structure while containing the necessary damping. Under a company sponsored IRAD project, Boeing developed damped composite patches to reduce acoustic response on the F-15 [3.3]. These damping patches have flown on lower fuselage skins and doors of two F-15s resulting in up to a 69% reduction in vibration.

3.1.2.3 Composite Patches - Composite patch repairs of cracked metallic structure have been applied extensively by the United States Air Force Warner Robins-Air Logistics Center on the C-141 and C-130 fleet. Several patches have also been installed on B-52 and F-16 aircraft. Under United States Air Force sponsorship, Boeing is currently performing the Composite Repair of Aircraft Structures (CRAS) program. The program objectives are to: (1) fill in gaps in the existing composite repair patch technology, and (2) promote the technology with new users such as the F-15.

An application for composite patches is on the front wing spar on the F-15 aircraft. Fatigue cracks have been found in the web of this spar emanating from a conduit hole in earlier F-15 models. Bolted repairs have been developed for this problem, and the component was redesigned in the F-15E. However, inspections are still required to detect if cracks are present.

Under the CRAS program, Boeing has developed a pre-emptive patch for this location. The patch restores the required crack growth life, reduces inspections and avoids drilling of holes required for bolted repairs. Laboratory tests are in work using two machined specimens representing the F-15 spar conduit hole location. The first specimen will contain an initial crack and will be tested without a patch in order to obtain baseline data. The second specimen will also have a crack started, but this time the specimen will be repaired with a composite patch. Both specimens will be fatigue tested to failure to verify the life improvement expected with the composite patch. Conventional strain gages will be used in both tests. A demonstration of the patch on an operational F-15 is also planned.

Once these tests are completed the technology will be ready for additional applications on the F-15. One area where additional development is needed is in health monitoring systems that result in a “smart patch” capable of detecting disbonds. Such a system would promote applications in areas difficult to inspect.

3.1.2.4 Grid-Lock* – Grid-lock*, patent held by BF Goodrich, is on the verge of becoming a mature technology. The F-15 is in the process of eliminating all honeycomb structure in preference of Grid-lock* to eliminate the inherent corrosion issues of honeycomb. Grid-lock* utilizes a unique method of construction by precise machining of tongue-in-groove channels, adhesive bonding, and designing structure such that each joint is reacted in shear. The F-15 has completed fatigue testing of the aileron (subjected to buffet) and is in the process of testing a wing tip to validate usage in all locations.

3.1.2.5 Laser Formed Titanium – Laser formed titanium, developed by AeroMet Inc. (Eaden Prairie Minn., USA) is another process that is on the verge of becoming a mature technology. The F-15 is in the process of manufacturing spares developed from laser formed titanium to lower overall part cost and reduce overall cycle time. Laser formed titanium is developed by depositing a layer of pre-alloyed titanium powder on a target plate with a CO2 laser. From this target plate flanges, stiffeners, etc. are built up. Final machining can eliminate this target plate if the target plate properties are not desired.

* Grid-lock is a registered trademark

3.1.2.6 Material Substitution – The F-15 is currently under consideration to replace aluminum parts that are found corroded with the newly developed aluminum 7055, produced by Alcoa. Lower longerons and skins under the fuel cells and the flap hinge beams are regularly found and replaced for corrosion. Unfortunately, the replacement material used is the same as was the original design. The new material is now under investigation for verification of material properties and is hoped to replace all 7000 series parts in the future.

3.1.3 Needs additional development – Technologies in this category are still in the laboratory stage or have been demonstrated on aircraft as part of an R&D project. This category may also include more established methods where technology gaps exist. Analysis methods may exist, but need verification.

3.1.3.1 - Advanced Riveting Technology - Force controlled riveting can result in longer fatigue life than with conventional displacement controlled riveting. The benefit is caused by greater compressive residual stresses surrounding the hole. The residual stresses lead to crack growth retardation at the riveted hole [3.4].

3.1.3.3 Active Vibration Suppression - Buffet is a major source of fatigue damage on fighter aircraft empennage structure. Two methods used in developing a solution to this problem include "smart" materials and active rudder control.

"Smart" materials distributed over vertical tail structure have been demonstrated to control structural vibration modes [3.5 – 3.9]. The results from these programs show that piezoelectric driven skins can reduce buffet loads at all flight conditions. However, power requirements and durability have limited piezoelectric system applications.

The RANN Corporation evaluated the use of active rudder control to mitigate buffet response on the F-15 and F/A-18 [3.5]. Boeing completed a wind tunnel experiment using a 15% scaled F/A-18 model to demonstrate the use of the rudder to control vertical tail buffet response [3.10]. The vertical tail was designed with a movable rudder that was controlled using a hydraulic actuator. RMS bending moment reduction of up to 42% were achieved. These reduced loads can improve the dynamic fatigue lives by factors of 4 to 10.

3.1.3.4 Laser Shock Processing - Laser shock processing, also known as laser shot peening, is a method for inducing surface beneficial residual stresses [3.11]. Energy absorbed from a laser pulse generates high shock pressure over the surface of the metal part. Energy absorption is enhanced with black paint on the metal surface. The energy is concentrated near the surface with a layer of water. Laser shock can induce residual stresses deeper than with regular shot peening (up to 0.04 in.), and it avoids surface damage. Applications of this method include high value components such as jet engine blades.

3.1.3.5 Friction Stir Welding - Friction stir welding is a process of welding by the rotation of a threaded cylindrical pin tool [3.12]. First, the rotating pin tool penetrates the material at a location where a weld is required. The material reaches the plastic state because of the frictional heat generated by the rotating tool. Finally, the rotating tool moves along the weld line, stirring the material on both sides of the weld. This results in a bond between the parts being joined. The process is in production use and new applications in complex airframe structure are in development [3.13]. One potential Service Life Enhancement application is in repair of cracked structure.

3.1.3.6 Health Monitoring – Two areas of health monitoring are the active detection of cracking and the active detection of corrosion.

For example, during a F-15 full-scale fatigue test, a strain gage was located on a wing spar cap near a crack location. Changes in strain values recorded during the test were correlated to the growth of the crack. Other methods use acoustic emissions for monitoring cracks and smart fasteners for detecting damage in bolted joints.

Corrosivity measurement is another area of health monitoring. A sensor unit has been developed by USN to record moisture and temperature [3.13]. The unit is self contained and small (4.1 cm x 4.7 cm x 1.7 cm excluding the sensor element). Data is downloaded remotely with a hand held data-gathering unit at a range of 40 m. These types of corrosivity sensors just measure the environment, but not actual corrosion.

Composite patches are an effective method of repairing cracked or corroded structure, but the patches are only good as long as they remain bonded to the structure. If done properly, modern surface preparation techniques result in very strong bonds. However, patches installed in substructure cannot be easily inspected for disbonds. Repaired primary structure must be capable of carrying limit load in the event that the patch falls off.

To address these issues, health monitoring systems that result in a "smart patch" capable of detecting disbonds are in development. One project uses strain gages and piezoelectric sensors installed on the patch to monitor the ratio of patch strain to component strain [3.14].

The damage dosimeter is an example of a stand-alone system for measuring the environment at a specific location. The dosimeter was designed to record time, temperature and dynamic data from three strain gages. It is compact (7.5 in. x 4.5 in. x 1.25 in. plus a battery pack), and does not require aircraft power or cooling. A dosimeter can be installed easily in an aircraft to record data in areas affected by dynamic fatigue caused by high acoustics or buffet, for example. Data can be downloaded from the aircraft by connecting a laptop computer to the dosimeter. Once the needed data is obtained, the dosimeter is removed.

3.2 Enhancement Techniques - F-15 Usage

3.2.1 Fastener Holes - The basic choice for critical fastener holes, if no previous enhancement had been incorporated, was to use interference fit fasteners or cold working. These improvements are mature and have shown definite improvements over standard holes. Installation of either procedure is straightforward and the both processes are well documented. Analysis of either enhancement is conservative and development of more sophisticated analytical tools can provide for increased inspection intervals.

3.2.2 Composites - Composite patches were chosen in several situations with different goals in mind. For locations where buffet, noise, or vibrations are problems, "Active Vibration Suppression" techniques or "Damped Bonded Patch" techniques are appropriate. In many cases some form of these techniques have been applied with varying amounts of success. In many early applications the suppressants have disbonded and been of limited use. Application of more robust bonding is required to build confidence throughout the fleet. For locations with high stresses, a composite patch to reduce localized stress peaks is desired. Currently, use of bonded patches is limited in nature and has not been used as an active fighter technique to eliminate "hot" spots due to unknowns associated with bonded patches. The need for this confidence is necessary as bolted repairs can create as many problems as they solve with the addition of fastener holes. "Smart" patches may be one method of obtaining confidence in bonded patches. The USAF and industry are making major break throughs in this area and practical application of bonded patches in key locations is needed to spread this technology.

3.2.3 Peening - Shot Peening and Laser Shock Processing were mentioned in a very limited fashion for critical locations. Since the F-15 inspection system is based on rogue flaw theory, these two methods have limited practical influence, as the improvements are so surface localized. In practicality either techniques would improve the economic life of the aircraft. Rogue flaws generally do not exist and cracking initiates, in almost all cases, on the surface of the part. A layer of beneficial residual stresses would retard the initial crack growth in the part.

3.2.4 Health Monitoring - Health monitoring was listed fairly extensively in the Wright Patterson study. Corrosion determination is interdependent with moisture collection. Therefore, a specific schedule cannot be defined since it is impossible to know when moisture will be trapped. With the use of a system to detect trapped moisture, corrosion can be prevented by a maintenance action to remove the accumulated water (for example, cleaning of plugged drain holes). The second health monitoring system needed is the full development of instrumentation to detect disbonds in composite structure or detection of cracks growing in areas of high stresses. As mentioned earlier, these systems exist but have not been implemented to any degree.

3.2.5 Spare and Replacement Parts –

3.2.5.1 Material Substitution - For corrosion, material substitution appears to be the optimum choice. For the F-15 lower longerons and skins, the use of 7055 may potentially provide enhanced corrosion protection. For the F-15 wing pylon, a material substitution from aluminum to titanium not only eliminates corrosion, but also eliminates issues of fatigue cracking and static overloads. Simple coatings have not offered substantial and sufficient protection from corrosion since they can be worn off or scraped away during normal operation and maintenance.

3.2.5.2 Laser Formed titanium - Laser formed titanium offers availability of titanium spare/replacement parts in a much more rapid procurement fashion. Lead times to obtain forgings can be cut in half with a significant control on costs.

3.2.5.3 Grid-Lock* - Grid-lock* helps to eliminate the water entrapment and subsequent corrosion problems seen in honeycomb structure. Use of corrosion resistant materials, i.e., 7055 aluminum, could be used to enhance these benefits. Honeycomb is extremely efficient, but not at the cost of the current maintenance burden.

4.0 SLE TECHNOLOGY DEVELOPMENT, DEMONSTRATION AND TRANSITION

For each “Near General Application” and “Needs Additional Development” technology, development steps to fill technology gaps are recommended. These are followed by demonstration plans to ensure the technology is ready for transition into the fleet. In most cases, a demonstration program should scale up the test specimens to represent actual aircraft structure.

The demonstrations should include flight tests with the corresponding aircraft to increase the maturity level and confidence in the new methods.

The following subsections describe a strategy for defining, developing, and demonstrating each enhancement technology and transitioning theses technologies to the USAF fleet.

4.1 Composite Patches

Additional work should focus on expanding technologies developed under CRAS and related programs. Specific areas include thick structure, complex geometry, repair of stress corrosion cracking and pre-emptive patches. Pre-emptive patches, installed before cracks initiate, do not need to be as robust as patches used in repair of cracked structure. They can have fewer plies, and because less stiffness is required, alternatives to boron/epoxy can be used. Analysis of these patches is needed to determine the stresses in the un-cracked structure, and predict the number of flight hours to initiate a crack. The calculated stresses are also used to perform damage tolerance assessments with an assumed initial crack (typically 0.05 inch long). New analysis methods should be verified with coupon and element tests prior to demonstration in actual aircraft parts. The tests should also evaluate the effect of temperature on the adhesive.

Another area where additional development is needed is in health monitoring systems that result in a "smart patch" capable of detecting disbonds. Because several programs are already focusing on sensors, further work on new sensor development is not necessary. Instead, additional work is needed on the integration and demonstration of this technology.

Therefore, a demonstration program on "smart pre-emptive patches" is recommended. Laboratory specimens should be instrumented with strain gages to verify stress analyses and should represent actual structure. In addition to strain, patch temperature measurements need to be demonstrated. Patches with disbonds of various sizes will help verify the ability of the sensors to detect different levels of damage. Flight tests following the laboratory demonstrations are suggested.

4.2 Health Monitoring

In addition to the smart patch mentioned in the previous section, several high-ranking locations on the F-15 were identified as candidates for health monitoring. The main source of these problems is water entrapment. Clogged drain holes can lead to water entrapment and resulting corrosion. Freezing of water within honeycomb cells can cause disbonds between the skin and the honeycomb. Sensors that signal when water is trapped could help prevent corrosion and disbonds in the identified locations by alerting maintenance personnel.

Sensor development has been conducted in other programs, and any additional work should focus on the integration and demonstration of this technology. This work should define how to use the sensors. For example, an indication of water may signal nothing more than normal drainage and would result in a false alarm. However, water present for an extended period may be caused by an actual clogged drain hole. The demo should establish the criteria to identify real problems using specimens representing actual structural configurations. The F-15 locations for sensing when water is contained in honeycomb structures are:

Vertical Tail, Main Torque Box Composite Skin
Vertical Tail, Main Torque Box Honeycomb Assembly
Horizontal Stab, Main Torque Box Honeycomb Assembly
Horizontal Stab, Main Torque Box Composite Skin

Sensors should be installed on operational aircraft to test their capacity to detect water in real conditions. The sensors should be used to measure drainage time periods. In some cases aircraft drain holes should be plugged to simulate clogged drainage. Tests should include temperature changes to evaluate the sensors in the presence of frozen water.

4.3 Interference Fit Fasteners/Cold working

One of the main technology gaps is the lack of a robust analysis methodology that accounts for the beneficial residual stresses resulting from this technique. This is especially the case with interference fit fasteners and hole propping.

NASA Langley Research Center has been developing analysis methods in this area. In cooperation with NASA, the analysis methodology should be completed and a test program defined to verify the analysis. Vendors associated with this technology should help with test plan definition. The test plan needs to cover the following:

Baseline open hole specimens
Cold worked holes
Tight fit fasteners with no interference
Interference fit fasteners (hi-loks and lockbolts)
Tapered fasteners (taper-loks)

Sleeve-bolts
Combined cold worked hole with fasteners
Titanium and steel fasteners
Single and double lap specimens with titanium, aluminum and combined layers
Constant amplitude and spectrum loading

During fabrication of the specimens, procedures for verifying the level of interference need to be demonstrated. The vendors should help in developing these procedures. Once the analysis methods are developed and verification tests completed, companies associated with this technology should help in the evaluation of results. The demonstration program should include tests with specimens representing actual aircraft structure.

4.4 Damped Bonded Patch

Damped bonded patches have flown in lower fuselage skins of the F-15. However, additional R&D is needed in this area to cover two technology gaps.

First, up to now the structure patched has been relatively thin. Additional development is needed to treat thicker structure affected by dynamic fatigue.

The second area involves the analysis. Crack initiation analysis methods based on RMS strain vs. life data exist. These methods are adequate for the analysis of thin durability critical structure, but if thicker structure needs to be repaired, crack growth analyses need to be developed.

The recommended project shall develop analysis methods and verify these with vibration (shaker) tests. The verification test specimens should include baseline un-patched panels and repaired panels containing various levels of damage and different patch configurations. Strain gages and accelerometers mounted on the specimens are necessary to measure the level of vibration suppression obtained with the patches. The propagation of cracks should be measured until the cracks reach a significant length or until failure of the specimen occurs.

A second possible set of tests can be completed with similar panels tested in an acoustic chamber. Variations in test specimen configuration include different boundary conditions and the presence of stiffeners.

Once the analysis methods are verified a demonstration program should be undertaken with test components representing actual structure. Options include acoustic chamber tests and flight demonstration of patches.

One of the solutions for buffet on the F-15 vertical tails is the installation of graphite epoxy doublers on the boron epoxy main torque box skins. Damping material, in addition to improved bonding technology, may improve the durability of these doublers.

Another F-15 location where damping patches are needed is on the Outer Wing, Upper Torque Box Skin and Ribs at XW 172, XW 188 and XW 206. Previously used damping tape at this location has been known to fall off the structure.

Although the specific candidates mentioned have focused on the F-15, the technologies should cover additional aircraft to maximize the payoff for the USAF.

4.5 Next Generation Materials and Manufacturing

Both 7055 and laser formed titanium are on the verge of becoming production capable materials. Laboratory testing is needed in each case to verify the static ad fatigue capabilities of both. Degradation of material properties over the original chosen materials would create critical issues for spares and replacement parts. Geometrical concerns generally preclude the use of increasing thickness to compensate for material property deficiencies due to other existing mating parts. New production will be capable of compensating for specific material deficiencies if the main virtue is of new material if of primary importance.

Material testing for 7055 is currently under investigation with Wright Patterson to provide a corrosion resistant aluminum. A program to substitute various candidate parts on aircraft in operation is underway. The primary intent of this investigation is to verify form, fit, and function. The F-15 ASIP office is interested in applying this technology to eliminate recurring corrosion issues.

Once material validation for 7055 are obtained, actual part selection should be made across a wide spectrum of platforms, i.e., transports, bombers, and fighters to provide maximum exposure and experience under various loading conditions and scenarios. Upon completion of all testing, spares, replacement parts, and new production should be visited for use.

Laser formed titanium has been under investigation on the F-18 and is beginning initial verification testing on the F-15. Warner Robins is experiencing shortages in parts traditionally manufactured from forgings. Lead times and high costs are making the traditional acquisition of forgings an ineffective proposition that increases the potential for aircraft groundings. The current test plans are to verify the crack growth capabilities of a spar made from laser formed titanium and to produce spars if the properties are deemed acceptable. Use of laser formed titanium is now in work to replace aluminum pylon ribs that are experiencing corrosion and fatigue cracking.

Grid-lock* testing is more mature. A complete component test for the F-15 aileron has passed successfully. Testing of the F-15 wing tip is just now been initiated. The aileron testing was sufficient to validate manufacturing of six F-15 grid-lock* replacement spares for honeycomb on the F-15. The wing tip will require additional validation due to spectrum severity and material properties.

Upon completion of the wing tip fatigue testing, all original F-15 honeycomb parts will be cleared for substitution with grid-lock*. One concern under consideration is grid-lock weight increases over honeycomb. These are small, but flutter driven locations need to be managed carefully.

Friction stir welding has had limited testing for static and fatigue properties. This process holds promise due to low heat affected zones, but requires special consideration if a final machining does not remove the inherent “live crack” left in the operation.

Testing to accurately quantify the static and fatigue characteristics is essential to validate this process. Repair of cracked structure is a potential benefit from this process and needs to be explored since conventional welding leaves large residuals stresses and requires careful consideration of potential contaminants in the materials.

5.0 Summary

The F-15 has become involved in a large number of state-of-the-art technologies to enhance future structural capabilities. Many of these technologies are intended for new production, but careful adaptation indicates that replacement of parts with new technology for existing aircraft is a fully viable solution to current concerns. These processes are needed, and will be investigated further, as maintenance and operational budgets grow tighter in future years.

REFERENCES

[3.1] Fatigue Technology, Inc., www.fatiguetechnology.com.

[3.2] J.R. Harrison and E.R. Thompson, "Exfoliation Corrosion Located by Search Peening," presented at the 1994 USAF Aircraft Structural Integrity Program Conference, San Antonio, TX, December 1994.

[3.3] S. Liguore, K. Hunter, R. Perez, T. Beier, "Flight Test Evaluation of Damped Composite Repairs for Sonic Fatigue," presented at AIAA/ASME/ASCE/ASC Structures, Structural Dynamics, and Materials Conference, 1999.

[3.4] R.P.G. Muller and L.J. Hart-Smith, "Making Riveted Lap Splices with 200-Year Crack Free Fatigue Lives," ICAF 97, Fatigue in New and Aging Aircraft, Volume I, Proceedings of the 19th Symposium of the International Committee on Aeronautical Fatigue, Edinburgh, Scotland, 18-20 June 1997.

[3.5] S.M. Rock, H. Ashley, et. al. "Active control for fin buffet alleviation," Proceedings from AIAA Guidance, Navigation and Control Conference, 1993, pp. 1051-1056.

[3.6] R.W. Moses, "Vertical Tail Buffeting Alleviation Using Piezoelectric Actuators - Some Results of the Actively Controlled Response of Buffet-Affected Tail (ACROBAT) Program", *Proceeding of the SPIE Smart Structures and Materials Conference*, 1997.

[3.7] R.W. Moses, "Vertical Tail Buffeting Alleviation Using Piezoelectric Actuators And Rudder", *High Angle of Attack Technology Conference* September 17-19, 1996 NASA LaRC, Hampton VA.

[3.8] L.E. Pado, S.L. Liguore, P.F. Lichtenwalner**,** and D. Drouin, "Neural Predictive Control for Active Buffet Alleviation", Proceedings, SPIE Symposium on Smart Structures and Materials, Vol. 3326, San Diego, CA, 1998.

[3.9] L.E. Pado and P.F. Lichtenwalner**,** "Neural Predictive Control for Active Buffet Alleviation", AIAA SDM Conference, St. Louis, MO. 1999, AIAA- 99-1319.

[3.10] S. Liguore, D., Drouin, T. Beier, **"**Active Buffet Load Alleviation for an F/A-18 Vertical Tail," Seventh International Conference on Recent Advances in Structural Dynamics, The Institute of Sound and Vibration Research University of Southampton, Southampton England, 2000.

[3.11] L. Hackel, J. Halpin, C.B. Dane, J. Daly, F. Harris, J. Harrison, "Lasershot Peening of Metals for High Throughput, Low Cost Processing," presented at the 1998 USAF Aircraft Structural Integrity Program Conference, San Antonio, TX, December 1998.

[3.12] The Welding Institute, www.twi.co.uk.

[3.13] R. Talwar, D. Bolser, R. Lederich, and J. Baumann, "Friction Stir Welding of Airframe Structures," presented at the 2nd International Symposium on Friction Stir Welding, Gothenberg, Sweden, 26-28 June 2000.

[3.13] V. Agarwala and R. Wood, " Corrosion and Corrosivity Monitoring, An Update," presented at the 4th International Aircraft Corrosion Workshop, Solomons Island, MD, 22-25 August 2000.

[3.14] A.A. Baker, S.C. Galea, I.G. Powlesland, "A Smart Patch Approach for Bonded Composite Repairs to Primary Airframe Structures," presented at the Second Joint FAA/DoD/NASA Conference on Aging Aircraft, Williamsburg, VA, 31 August – 3 September 1998.

Life Extension Methodologies and Risk-Based Inspection in the Management of Fracture Critical Aeroengine Components

A.D. Boyd-Lee, D. Painter and G.F. Harrison
Structures and Materials Centre, QinetiQ, Farnborough, Hants GU14 0LX, U.K.
Phone: +44-1252-392000
Fax: +44-1252-397298
Email: adblee@qinetiq.com, dipainter@qinetiq.com and gfharrison@qinetiq.com

1 ABSTRACT

This paper briefly describes the main methodologies used in the assessment of fracture critical parts. The development of a procedure for the quantitative assessment of non-finite results is discussed and typical life extension levels are illustrated. Risk assessment is considered within the context of the safe life methodology. It is used to quantify the potential risks associated with the short-term continued operation of life-expired parts and to allow their managed withdrawal. The paper then considers risk assessment applied to the damage tolerance and retirement for cause life assessment. The significance benefits of risk based inspections intervals over standard fixed inspections are illustrated. Finally with regard to retirement for cause, it is shown that a risk based approach automatically sets an economical limit to retirement for cause but avoids the progressive risk levels associated with the current embodiment.

2 BACKGROUND

Aeroengine components are classified into fracture 'critical' or 'non-critical' depending on the consequences that a malfunction might have on the integrity of the aircraft. Turbine and compressor discs and shafts are identified as the major fracture critical components. They experience extreme thermo-mechanical transient loadings, which, in the absence of an adequate life prediction policy, eventually lead to low cycle fatigue (LCF), creep or corrosion failures. Since it is not practical to design engine casings capable of containing such events, it is essential to ensure that their occurrence in service is an extremely remote possibility. Hence a fundamental requirement of both military and civil for engine certification is therefore to ensure the continued airworthiness and integrity of fracture components such as discs and shafts throughout their service operation. Such events are non-random and hence as service operation progresses, both the risk of failure and the rates at which risks accumulate rise rapidly.

In earlier times, prior to aircraft reaching their full declared lives, the introduction of next generation aircraft led to fleet withdrawal prior to many major components becoming life expired. This is certainly no longer the case and lifing methodologies must ensure that thoughout the declared service lives of the components the likelihood of failure is extremely remote. Within the U.K., the inclusion of a requirement to fatigue test ex-service discs has significantly reduced the risk of the unforeseen. The current paper will review briefly the major methodologies applied to meet these requirements. Additionally, it will illustrate the concepts and application of several methods which allow safe life extension. Within the general context of Damage Tolerance, the benefits of risk based inspection intervals over conventional procedures will be demonstrated.

3 LIFE-TO-FIRST-CRACK

Under current UK Military Defence Standards (Def. Stan. 00-971) [1] and European Civil Joint Airworthiness Requirements (JAR-E) [2], the service lives for fracture-critical components are derived from spin-rig test results of actual engine discs under cyclic loading at stress and temperature conditions similar to those experienced in engine operation. These regulations are based on a safe-life policy wherein component fatigue 'failure' is defined as the occurrence of an 'engineering crack' of 0.38mm radius.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

If the total fleet dysfunction distribution were known i.e. all components were tested to failure, the mean dysfunction life would be known exactly. In practice a sample is selected from this distribution and tested until significant crack growth or burst. The mean life for the sample provides an estimate of the mean life of the rest of the fleet.

A consequence of the high cost of disc spin testing is that it is only practical to test a very small sample of the discs. Hence such tests are extremely expensive and few results are available, dealing with the probabilistic nature of fatigue is an essential part of any life assessment. To address the statistical difficulties associated with small sample sizes, the lifing regulations define the procedures for the safe interpretation of these results and for the calculation of the declared service lives. Certain assumptions have therefore to be made concerning the statistical form of the dysfunction population distribution. From past UK experience the failure distribution is assumed to be lognormal and significantly the assumption is also made that the scatter associated with the distribution is also known. It is therefore assumed that the ratio in life (scatter) between the +3σ and - 3σ quantiles (749/750 and 1/750) is known. This assumption has major implications regarding the analysis performed on the test sample. Hence the mean life from a small sample used as an estimate of the population mean life will only be approximate, provided that the population scatter factor is known, for any test sample taken from the population the sampling error must be established absolutely. For example, if the whole of the population was randomly split-up into samples of any specified size (e.g. 5), tested, and the mean lives from all the samples plotted, the distribution that would be obtained can be predicted statistically without any testing. Hence by associating the known distributions to the actual test sample, the magnitude of the sampling error is also known. In other words from the mean life of the test sample the population mean life can be established to any required level of confidence e.g. 95%. Also if the mean life of the population is known, the mean life to any other quantile is also known.

This is the basis of the UK statistical models applied to both (Life to First Crack) ltfc and to the dysfunction distribution.

Having established the burst distribution, risk assessment is about identifying where, at any specific time, individual discs in service lie relative to the failure distribution. In practice, it is easier to attach the error function to these points rather than the full burst distribution but in statistical terms the effects are identical. The statistical model is the same as that applied in ltfc and 2/3 dysfunction approaches. That is, a predicted safe cyclic life (PSCL) is calculated by applying statistically-derived safety factors to the geometric mean (GM) of the respective test results, such that at this life, not more than 1 in 750 service discs would be expected to contain an 'engineering crack', to 95% confidence. It is from this base that the life-extension procedures discussed in the paper are quantitatively assessed.

Experience has shown that typically disc fatigue lives are distributed according to a lognormal density function. Also the assumption is made that the ratio of the fatigue lives at the +3σ and -3σ points on the life-to-first-crack (ltfc) distributions is 6. Given an assumed scatter factor of 6, the life corresponding to the lower 1/750 quartile is automatically located at a factor of 3 standard deviations (i.e.√6 (= 2.449)) below the geometric mean (GM) ltfc obtained from the sample. A 95% confidence level allows for the effect of component test sampling error on the safety level inherent in the calculation. This lower confidence bound corresponds to a safety factor in life of:

$$6^{\left(1.645/6\sqrt{n}\right)} \qquad (1)$$

where the 95% confidence corresponds to 1.645 standard deviations. The combined safety factor y is given by the expression

$$y = 6^{1/6\left(\frac{1.645}{\sqrt{n}}+3\right)} \qquad (2)$$

Finally, a predicted safe cyclic life (PSCL) or Ar is obtained by dividing the log mean test cycles (converted to equivalent reference cycles) by the factor y.

$$Ar = \frac{\sqrt[n]{\prod_{i=1}^{n} N_i}}{y} \qquad (3)$$

where 'Ni,' are the individual ltfc test results. It can be seen that equation 2 is an increasing function with respect to decreasing sample size. Application of factors involved in determining 'Ar' is illustrated in Figure 1.

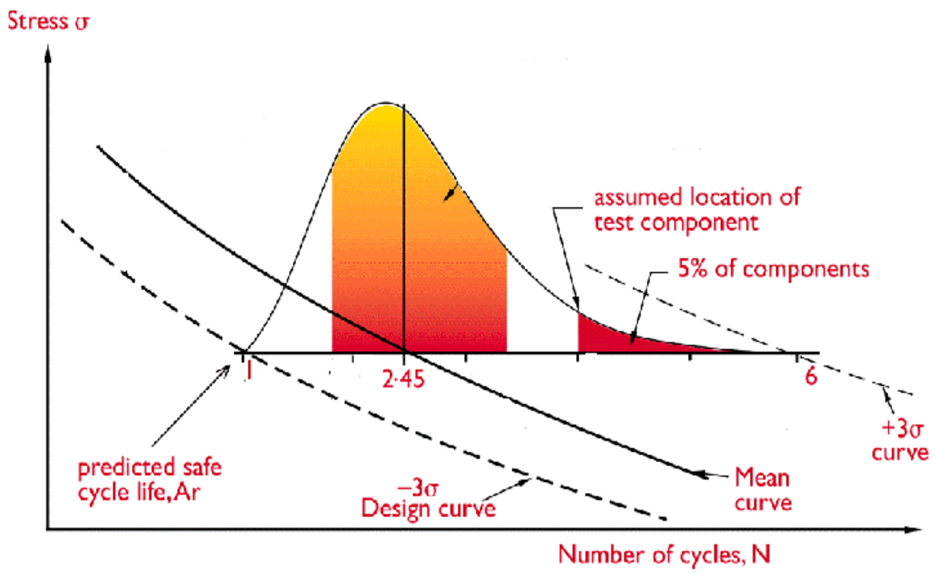


Figure 1. Derivation of the material design curve and predicted safe cyclic life.

Through using an 'engineering crack' as the basis for the calculation, the additional life taken to grow to the critical size associated with dysfunction (failure) acts as a further margin of safety. Ultimately, it is this, in combination with the factors in the denominator of equation 2 which define the actual safety level inherent in the application of this method.

4 LIFE REVISION OF AGEING ENGINES

A consequence of the high cost of disc spinning tests is that once a specified (required) design life has been demonstrated, there is great pressure to discontinue the respective test programme, even though the discs have not reached 'failure' (defined as the occurrence of an engineering crack). Results of this kind are called 'non-finite'. Non-finite results can also arise for several other reasons. Amongst these, a common cause is a change in the identified failure location within a component as the result of either service experience or reanalysis of the basic design. The current lifing regulations were originally derived to accept only finite results and this has caused non-finite results to be either rejected unnecessarily or accepted overconservatively by assuming 'failure on the next load cycle'. The corollary is, that where the newly established failure location is now identified as having experienced little or no overstress, the service life that can now be declared would need to be reduced by a factor of up to 4, depending on the number of test results available.

4.1 Improved statistical analysis of non-finite results

In the current paper an analysis methodology for samples comprising of only non-finite results, is illustrated. However, in addition, the general methodology can also handle samples comprising of mixed finite and non-finite data. To illustrate the principles, in the following analysis it is assumed that the results belong to a lognormal ltfc distribution having a known scatter factor. As is the case for samples of finite results, the geometric mean (GM) of the population ltfc distribution is estimated to 95% confidence. Once this value is obtained, the 1/750 (-3σ) quantile is selected to provide a safety factor to give a value for the required safe-service life, Ar, (cycles). To identify a conservative confidence interval, the probability of obtaining a non-finite result at N_i^- cycles is established based on a given value of the GM (the minus superscript indicates that the value is non-finite).

A 95% lower confidence bound for the true GM can be obtained by identifying that mean value for which the probability of a component not surviving N^- cycles is 5%. That is, there is a 95% chance that the defined likelihood lies within the interval 0.05 to 1. Therefore, given the value of N^- , a 95% lower confidence bound for GM can be obtained when the following equation is satisfied.

$$p(non-\ finite\ at\ N^{-}\ cycles) = 0.05 \tag{4}$$

As the sample size is equal to one, this confidence bound leads to the same formula as would apply if the result were finite. For a sample of size 'n' the confidence test generalises to

$$\prod_{J=1}^{n} p(non-\ finite\ at\ N_J^{-}\ cycles) = 0.05 \tag{5}$$

That is, the probability of obtaining all non-finite results is equal to the product of the probabilities of obtaining the individual non-finite results in the sample. As for a sample of size one, a confidence interval on the GM is chosen such that the probability of obtaining the sample is equal to a value lying between 0.05 and 1. Thus, a conservative 95% lower confidence bound for the GM is given as that value for which this product is equal to 0.05 (see equation 4). This equation embodies a 'next-cycle-failure' assumption for up to one of the non-finite results. It is still slightly overconservative since the probability of even one of the non-finite results reaching dysfunction on its next cycle is extremely low. However, in general, removal of this overconservatism is complex and (without further information) does not result in a significant life increase.

It remains to substitute an expression for the failure distribution into equation 4. Firstly a change of variable is applied such that the parameter N_J^- is the resulting non-finite value associated with test result J, expressed in its logarithmic form, divided by the geometric mean life, N_μ, and rescaled to units of standard deviation. (Division by the population geometric mean life, N_μ, has the effect of placing the origin at the population GM life which has an indeterminate fixed value.) The following solution is obtained:

$$\prod_{J=1}^{n}\left[1-normal\left(\frac{\log\left(\frac{N_J^{-}}{N_\mu^{95\%}}\right)}{\frac{1}{6}\log\left(\frac{N_{+3\sigma}}{N_{-3\sigma}}\right)}\right)\right] = 0.05 \tag{6}$$

where the $N_\mu^{95\%}$ value is a 95% confidence estimate of the GM of the population fatigue life distribution. This is the only unknown in equation 6, and hence can be solved by substitution of the known values and iteration until the equation is satisfied.

Once $N_\mu^{95\%}$ has been established, the standard safety factor can be applied (as illustrated in equation 2) to ensure that not more than 1/750 components reach the defined dysfunction point. That is,

$$Ar = N_{-3\sigma}^{95\%} = \frac{N_\mu^{95\%}}{\sqrt{\frac{N_{+3\sigma}}{N_{-3\sigma}}}} \tag{7}$$

By way of illustration, the application of the above expression to 5 non-finite results of 10,000 cycles gives $N_{\mu}^{95\%}$ =10,374 cycles and substitution of this value in equation 7 gives an Ar value of 4,235 cycles. Thus, in this example and without any compromise to safety, the improved statistical analysis supports a 29% increase in the service life relative to that declared via current regulations. In service, the life extensions given by the method have ranged between 5% and 50%.

At one extreme a sample may contain all finite results; at the other all non-finite results. Consistency between the solution for mixed results and these two extremes can be demonstrated. Firstly for mixed results the solution copes with cases where all the results in the sample are finite and hence the confidence test used for the analysis of all finite results gives exactly the same results as that developed for samples of totally finite results. There is a subtle difference in that this confidence test uses a change of variable to a logarithmic normal density functional form as the reference datum, whereas the earlier test uses the mean of the sample as the reference datum. However, as this difference does not change the statistical model, it makes no difference to the solution. For illustration, at the other extreme (mostly non-finite) when all the results in the sample have the same life and all but one is non-finite, then the mixed results confidence test yields the same equation as is used in the all non-finite case. It can therefore be verified that there is complete consistency between the solutions for samples of mixed results and for samples of all finite results. Also, there appears to be a consistent level of conservatism between the solution for samples of all non-finite results and the solution for samples of mixed results. The confidence test used in the well-established method for handling samples of all finite results is fully optimised. However, as discussed above, the confidence test for samples of mixed results is essentially a more generalised version of that used to handle samples of all finite results. Thus, this is also fully optimised.

Finally, for a sample of non-finite results, a different confidence test has to be used. The one currently demonstrated is marginally overconservative for two reasons. Firstly, the additional complexity required to exploit fully the non-finite results is probably not justified in terms of the small increase in life it would give. Secondly, it enables the approach to be related to the familiar 'minimum life formula' and the Weibayes formula. This is important for verification of the theory.

The solution for mixed results is well behaved between the extremes of all finite and all non-finite results. One way of demonstrating this is to compare the variation of the safe service life as a function of the proportion of the results in the sample which are non-finite. For example for the sample of 5 results all of 10,000 cycles that were considered above, as the sample moves from all finite to only one finite result, the available safe life increases as illustrated:

No. of Finite Results	**No. of Non-Finite Results**	**PSCL (cycles)**
4	1	4,235
3	2	3,857
2	3	3,611
1	4	3,427
0	5	3,278

Table 1 Influence of non-finite results on predicted safe cyclic life. (All tests discontinued at 10,000 cycles).

Although somewhat obvious, the lower the value of a non-finite result; the less its significance. However, this property allows a further verification of the mixed results methodology (see Table 1). For example, suppose there is a sample of 2 results one has a finite value of 10,000 cycles and the other a non-finite value of 10,000 cycles. Then a safe service life of 3,254 cycles is given by the methodology (see row 1 of the Table 2). Suppose instead the non-finite value of the second result is 8,000 cycles, then a safe life of 3,000 is given (see row 2 of Table 2). This calculation is repeated with successively smaller values for the second result. As expected, eventually the safe life converges on a value of 2,498 cycles (the same value as if the second result did not exist at all).

First Test Result (cycles)	Second Test Result (cycles)	Predicted Safe Cyclic Life (cycles)
10,000 f.	10,000 n.f.	3,254
10,000 f.	8,000 n.f.	3,000
10,000 f.	6,000 n.f.	2,752
10,000 f.	4,000 n.f.	2,556
10,000 f.	2,000 n.f.	2,499
10,000 f.	1,000 n.f.	2,498 (convergence)

Table 2. Influence of low non-finite results on predicted safe cyclic lives (samples of 2)

A further example of the capability of the methodology to handle low non-finite results as expected is given in Table 3, this time using a sample size of 3.

First Test Result (cycles)	Second Test Result (cycles)	Third Test Result (cycles)	Predicted Safe Cyclic Life (cycles)
10,000 f.	10,000 n.f.	10,000 n.f.	3,693
10,000 f.	10,000 n.f.	8,000 n.f.	3,505
10,000 f.	10,000 n.f.	6,000 n.f.	3,346
10,000 f.	10,000 n.f.	4,000 n.f.	3,264
10,000 f.	10,000 n.f.	2,000 n.f.	3,254
10,000 f.	10,000 n.f.	1,000 n.f.	3,254

Table 3. The influence of lower non-finite results on predicted safe cyclic lives (sample size 3) Further examples, comparing mixed samples of size 3.

As for any lifing calculation, safe application of the non-finite methodology depends on the validity of the assumptions made. Primarily this means that the assumed scatter factor (i.e. the factor in life between the plus and minus 3σ points on the population failure distribution) must be conservative. Wherever possible a Chi-squared test should be undertaken verify that the assumed population scatter is consistent with that associated with the sample. In dealing with non-finite results, via an inequality condition a lower bound is established for the finite value associated with the non-finite result.

4.2 2/3 Dysfunction failure criterion

Experience over many years has shown that the ltfc approach is very conservative and disc failures in service are extremely remote with rates of only one or two for every 100 million flying hours. The approach has an in-built factor of safety associated with the cycles required to grow a crack from 0.78mm to the dysfunction crack size. Since this value is both material and geometry dependent, the method does not provide an overall quantifiable level of safety. To address this, an extension of this approach, the crack initiation life has been replaced by a set fraction of the total cycles to dysfunction. A figure of 'two-thirds' has been established in the UK and increasingly throughout Europe.

RB199 engine for which it has been shown that crack propagation β-factors can be 2-3 times greater than crack initiation β-factors [3]. In this paper, we consider a typical case in which $\beta_p/\beta_i=2.5$.

4.3 Crack tolerant designs

In crack tolerant designs, components have 2/3 dysfunction lives that can be significantly greater than those established via ltfc. This difference represents an additional safe crack growth phase which can be exploited. Assessments of life-limiting areas such as drive arm vent holes and disc rim features are typical of design features that can exhibit significant crack tolerance.

For such crack tolerant components, service lives can be safely extended beyond 100% Ar, without exceeding the extremely remote risk levels associated with conventional failure locations. However, to ensure a consistent level of safety, the associated ‘never exceed’ 2/3 dysfunction margin of safety must be imposed on the useable crack growth life. Thus, relative to component lives based on ltfc, life extension is available when the crack growth life is greater than 50% of the ltfc value. To ‘cash in’ the benefits of such designs, it is necessary to be able to predict accurately this crack growth phase and hence it is necessary to apply fracture mechanics procedures based on establishing operating stress intensity levels.

Although service lives are expressed in terms of the major reference cycles experienced, the damage induced by the large number of minor cycles must also be accounted for. Minor cycles occur as a result of adjustments to the thrust requirements during various stages of the mission flown by the aircraft. These are accounted for as the additional number of reference cycles per hour of flight that would inflict equivalent damage to one hour of the minor cycle loading. Then the exchange rate β is defined as the total number of reference cycles (major and minor) consumed during one hour of flight.

Both theoretical analyses and experimental test programmes have shown that, under an identical mission loading sequence, minor cycles are relatively more damaging during crack propagation than during the ltfc initiation stage. This can be explained in terms of the difference in the stress exponents associated with the life-to-first-crack curve and with the stress intensity exponents associated with the crack growth curve. Below the fatigue endurance limit, minor sub-cycles have virtually no influence on crack initiation, however they can contribute during the propagation phase. For the crack initiation phase (ltfc), a weighted average consumption rate of reference cycles per engine flying hour, β_I must be established. Similarly, during crack propagation a is the weighted average consumption rate of reference cycles per engine flying hour, β_p, must also be established. A consequence of crack initiation and crack propagation models is that β_p is generally greater than β_i. And typically, β_p/β_I has a value equal to 2.5. These observations have been confirmed recently for the Tornado aircraft.

Although life consumption and remnant life calculations are evaluated in terms of reference cycle damage, the end user is interested in the service hours available for specific components. In general to get this information, it is simply a matter of dividing the release lives (in cycles) by the relevant β-factor. The value obtained identifies the authorised service life in engine flying hours. For the situation in which the 2/3 dysfunction life exceeds the ltfc, this difference in reference cycles needs to be divided by the propagation β-factor to establish the authorised additional life in engine flying hours. Extensions of about 40% have been demonstrated for RB199 IPC and IPT rotor components. Details of the procedures involved in determining safe crack growth lives are given elsewhere (AGARD) [4].

4.3.1 Fracture Mechanics Based Procedures

A consequence of the development of very high strength disc alloys is that the critical crack size for the onset of rapid fatigue crack growth can be smaller than the 0.78mm associated ltfc. Additionally, aeroengine discs can have several life limiting features dependent on the specific design and operational requirements. The net effect is that it is impossible to construct a common databank for all these features on an ltfc basis. However a rationalised design approach can be developed via a fracture mechanics approach. In such cases, it is assumed that failure occurs either from initiation and growth of cracks induced by the imposed service loading sequences or as the result of crack propagation from inherent defects. Currently there are three fracture mechanics based life assessment methodologies approved for the certification of aeroengine fracture critical parts. These are Databank Lifing [5], Damage Tolerance Lifing and Retirement-for-cause [6]. The first approach is approved for use in UK Civil and Military engines and the others apply to US Military engines.

4.3.2 Databank Life Assessment

This procedure assumes that all discs contain small “pseudo” defects that grow under fatigue loading in a predictable manner from the first cycle. For the various features for which failure lives and final crack lengths are known, the ‘Paris’ crack growth equation is used in a back calculation mode to determine a “pseudo” crack size present in the component/test piece at cycle one. Since the fracture mechanics analysis accounts for variations in component geometry, stress field and crack shape, the approach offers a means of combining the results of different disc designs and large specimens into a common data bank. Statistical analysis procedures define the maximum effective "pseudo" crack size likely to be present in the total population. This initial flaw is then used as the starting point in a standard Paris summation to calculate the maximum allowable service life. The effective initial flaw sizes have no physical meaning, they simply provide a suitable parameter to enable the lives of different component geometries to be combined into a common data set. The inverses of the pseudo crack sizes are plotted in terms of a three-parameter Weibull model. Accurate estimates of component life can be established at the design stage thereby allowing greater design optimisation and more effective use of materials. Both CAA and FAA Airworthiness Authorities have approved fracture mechanics databank methods in declaring lives of civil engine components.

4.4 Damage Tolerance Life Assessment and Retirement for Cause

In this philosophy, damage is again assumed to pre-exist in newly manufactured components, but the starter crack size for residual life calculations is based on proven NDE capability. The residual life of the component is determined by applying a fracture mechanics approach to calculate the number of cycles required to grow from the starter crack size to a critical length. By assuming that damage exists in components as manufactured, the damage tolerance method avoids problems associated with the statistical distribution of crack initiation. Achievements of an acceptable level of safety now depends both on the reliability of the NDE system and on how often cracks or defects of a given size occur in practice. The most widely applied damage tolerance methodology is that developed by the US Airforce under its Engine Structural Integrity Programme (ENSIP). Here, the declared service life is based on an NDE crack detection size set to achieve a detection level of 90% with 95% confidence. Mean crack growth data are used to determine the available growth life to dysfunction and the declared safe service life is normally set at half this value.

4.4.1 Retirement for Cause

This is an extension of the damage tolerance approach which has been applied to high performance military components in the U.S. In contrast to the ltfc method, retirement is not implemented until actual cracks have been identified in individual discs. The safety of the approach is strongly dependent on the effectiveness of the NDE technology.

5 EXTENDED COMPONENT SERVICE LIVES VIA RISK ASSESSMENT AND RISK MANAGEMENT

5.1 Risk analysis

In the safe life methodology, the predicted safe cyclic life of a fracture-critical component is set to ensure that to 95% confidence not more than 1/750 service components suffer an engineering crack (>0.38mm radius). However, in applying this definition to the lifing of service components, several limitations have been encountered. Hence, although, ultimately safety should be concerned with ensuring that the possibility of fracture critical parts failure (dysfunction) must be extremely low, the risks associated with this event are not addressed directly. Also, in general, the declared life-to-first-crack is not a fixed fraction of the dysfunction life and consequently an inconsistent level of safety results. Additionally, it is found that minor cycles are usually more damaging in crack propagation than they are during life-to-first-crack. All these issues have important consequences for the implementation of a safe life, ltfc lifing policy. They have even greater significance for damage tolerance approaches.

Another aspect that has to be addressed is that in current methodologies little consideration is given to rate of increase in risk with increased component service life and hence these traditional approaches do not always provide the required statistical information for the managed withdrawal of service components. Current methods do not allow exploitation of the full life capability of components with long crack propagation lives.

Although the pscl methodology can loosely be considered to be a form of risk model, it is not sufficiently flexible to handle the wide variety of issues that arise in service. Indeed, when a significant downward life reduction occurs, strict implementation of the safe life criterion could result resulted in the grounding of aircraft fleets. Hence, within the context of airworthiness, it is now proposed that safety should be related to the predicted dysfunction distribution and safety factors associated with ‘95% confidence, 1-in-750’ should be replaced by the determination of maximum risk levels (e.g. for components a reasonable level might be set at 10^{-8} per engine flying hour).

To address these diverse issues it is strongly advocated that there should be a move towards the development and adoption of a more robust risk based life assessment methodology. Such an approach should include the identification of the component life-to-first-crack and propagation life distributions and account for all factors that affect them (inspection, fabrication knowledge etc.). These distributions should then be combined to obtain the estimated dysfunction distribution, whilst taking full account of sampling error. This should allow the instantaneous risk/hour to be derived from the estimated dysfunction distribution and hence allow the safety criteria to be defined in terms of risk.

Useful management parameters available from such an approach include the identification of an individual aircraft peak risk/hour by which time components must be inspected or else retired and also identification of a maximum fleet cumulative risk per annum. It should then be possible to correlate the identified risk to other available management tools such as the RAF Hazard Risk Index.

5.2 Identification of the component distributions

For aeroengine discs, the lives-to-first-crack form an approximate lognormal distribution. The associated probability density function can be expressed as

$$\mathrm{PD}^{\mathrm{i}} = 0.5.Erf\left[\frac{Ln\left[\dfrac{N^i}{N^i_\mu}\right]}{\dfrac{\sqrt{2}}{6}Ln[sf^i]}\right] + 0.5 \qquad (8)$$

where ‘Erf’ is the mathematical error function, ‘N^i‘ is the life-to-first-crack (cycles) of a given component, ‘ N^i_μ ’ is the log-mean of the ltfc distribution and ‘sf^i‘ is the scatter factor of the ltfc distribution. By replacing the occurrences of the superscript ‘i’ with the superscript ‘p’, a propagation life distribution is described. Similarly by replacing the superscript ‘i’ by the superscript ‘d’, a dysfunction distribution is defined.

5.3 Estimation of the dysfunction distribution

Often the limited available data are such that it is preferable to derive the dysfunction distribution from the life-to-first-crack and propagation life distributions for the component. The dysfunction life ‘N^d‘ is equal to the sum of its life-to-first-crack ‘N^i’ and its propagation life ‘N^p’.

$$N^d = N^i + N^p \qquad (9)$$

From a purely statistical viewpoint the two extremes are that the two component distributions are totally dependent or they are independent (total dependence means that a component with a long ltfc will have a proportionately long propagation life). Factors that contribute to the net effect include chemical composition, manufacturing and machining, heat treatment and surface residual stresses. The life-to-first-crack distributions and the propagation lives obtained in the nickel-base superalloys used in aeroengine discs appear to exhibit an interdependence. That is they are partly dependent and partly independent. Current evidence indicates that assuming 50-50 dependence-independence is conservative. Although the degree of interdependence does not affect the geometric mean of the dysfunction distribution and it has a relatively minor effect on the distribution form, it has a significant effect on the dysfunction scatter factor as illustrated in Table 4.

	Individual SF		Combined Scatter Factors		
Nature of component	**ltfc**	**Prop.**	**Dependent Dysfunction**	**Independent Dysfunction**	**50% Dependent**
Typical (N^p=0.5N^i)	6	4	5.6	4.4	5.0
Surface sensitive (N^p=0.5N^i)	10	4	8.6	5.8	7.2
Crack Tolerant (N^p=2N^i)	6	4	5.0	2.9	3.9
Crack Tolerant (N^p=2N^i) but surface sensitive	10	4	6.6	3.2	4.9

Table 4. Influence of initiation to propagation life ratio, and initiation scatter on the combined scatter factors

5.4 Estimation of the dysfunction distribution subject to inspection (Effect of inspection on remnant dysfunction distribution)

If a component is subject to inspection, then a dysfunction distribution can to be calculated for each inspection interval. Until the first inspection occurs the dysfunction distribution is derived as described above. For the second inspection interval, suppose that the probability of detection is 'POD'. Then POD*100% of the life-to-first-crack distribution is truncated and 100(1-POD)% is not truncated. The effective life-to-first-crack distribution is thus given by the following expression.

$$PD^i = POD.PD^i[truncated] + (1 - POD).PD^i[not \quad truncated] \qquad (10)$$

The modified life-to-first-crack distribution can then be combined with the propagation life distribution to obtain the dysfunction distribution using the approach described in the previous section. For simplicity, in the results to be presented later, 100% independence has been assumed. Numerical methods can be used to evaluate the required convolution integral.

Assuming 100% detection of all cracks exceeding 0.38mm surface length, inspection at 'n' cycles can be represented by a sharp truncation of the life-to-first-crack distribution at that point. Figure 2 shows the effect of inspection on the ltfc density distribution. (For the case shown, the geometric mean initiation life is 10,000 cycles and scatter factor (defined as the ratio between the plus and minus 3σ quantiles) has a value of '6'.

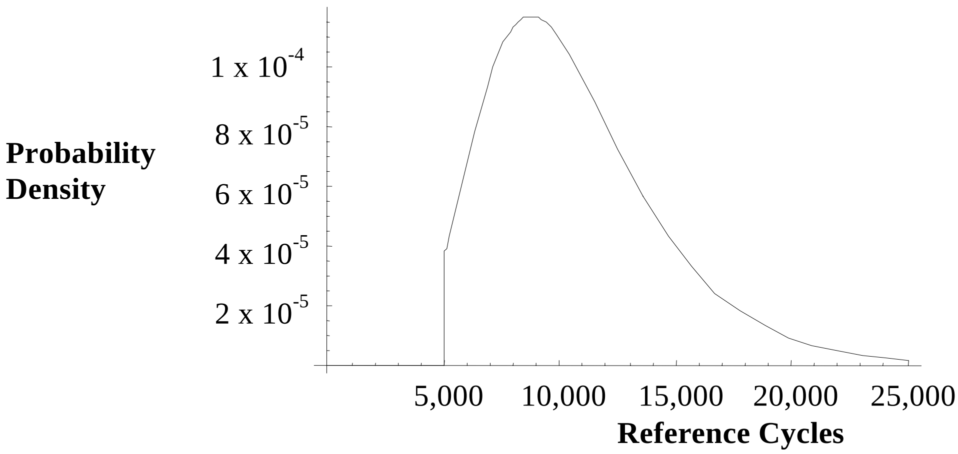


***Figure 2**. Truncated life-to-first-crack distribution (representative of perfect inspection.*

If crack detection is perfect prior inspections are irrelevant because the current inspection detects any crack missed by a previous inspection. If no cracks are detected, then the risk curve associated with the previous inspection applies. Thus the risk curve for a subsequent inspection ‘i’ can be expressed as

$$R_i[efh] = POD \times R_p[efh] + (1 - POD) \times R_{i-1}[efh] \tag{11}$$

An imperfect inspection can be represented as a mix of the truncated and non-truncated life-to-first-crack distributions.

5.5 Determination of instantaneous risk per flight hour

This section presents a model for determining individual aircraft dysfunction risk per aircraft flying hour and then indicates how the cumulative risk can be expressed via an integral form of this equation. Since in UK, most of the service components are lifed according to the PSCL approach described earlier, the following risk model is expressed in terms of the safe service life ‘Ar’ since originally, the risk equation was developed to allow management of components operating beyond their declared life, Ar.

The initial stage in the assessment therefore quantifies the safety factors associated with Ar. A procedure for determining the safety factor ‘y’, associated with a component that has reached its safe service life, Ar, was discussed. This factor is with respect to the crack initiation distribution and when related to burst a further factor has to be applied. In the absence of onboard usage recorders, service lives are expressed in terms of hours and mission exchange rates are used to convert these hours into reference cycles. For the typical case illustrated previously where the mean of the burst distribution is a factor of 1.5 times the mean ltfc, exchange rates for initiation and for propagation are based on different criteria and typically propagation rates are about a factor of 2.5 times initiation rates. This correlation allows propagation cycles to be converted to equivalent initiation cycles and hence allows the application of a single exchange rate to risk assessments. The propagation cycles can be converted into equivalent available initiation cycles by dividing them by the factor of ‘2.5’ thus, to relate the safety factor ‘y’ to the burst distribution it has to be modified by the ratio ‘1.2’ (=1+0.5/2.5). For any other service life in hours, the additional safety factor can be related to the safety factor at Ar through the multiplication factor Ar/(H.β^i). Hence

$$S = y.\frac{Ar}{h\beta} \quad = \quad 1.2 \times 2.449 \times 6^{\left(1.645/6\sqrt{n}\right)} \times \left(\frac{Ar}{H \times \beta_i}\right) \tag{12}$$

where H is the component life in hours.

The risk per engine flying hour is defined as the rate of increase of probability of failure with respect to increased engine flying hours.

$$risk/efh \quad = \quad \frac{\partial\{p(fail)\}}{\partial H} \quad . \tag{13}$$

To simplify the derivation, the assumed lognormal dysfunction distribution is transformed to a Gaussian distribution and to map the fatigue lives of the test results to the transformed distribution, the logarithm of their values is taken. Since it is easier to consider the risk in terms of the rate of increase of the transformed variable $\delta S'$ (i.e. log S), the following partial derivative is used:

$$risk/efh \quad = \quad \frac{\partial\{p(fail)\}}{\partial S'} \times \frac{\partial S'}{\partial H} = \quad -\frac{\partial\{p(fail)\}}{\partial S'}.\frac{1}{H} \tag{14}$$

For the artificial case of an infinite sample of component test results, there would be no sampling error. In this case, the location of the component service life relative to the GM of the burst distribution would be precisely determined as shown in Figure 3. Therefore, the risk per engine flying hour (risk/efh) is simply equal to the shaded area in the figure.

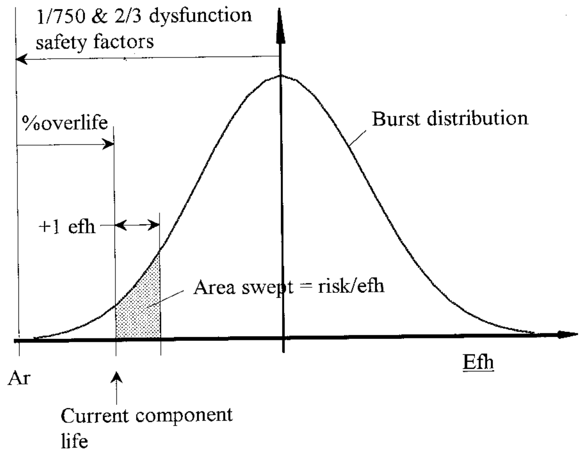


Figure 3. Illustration of the risk assessment model for the case where the geometric mean of the population burst distribution is known.

In current UK lifing procedures, for fracture critical components both the form and the scatter of the population burst distribution are assumed to be identified [4]. A consequence of these assumptions is that, given a (known) failure distribution of an infinite population, each result can be considered as the mean of a sample of one. If groups of samples of 2, 3, 4…n are randomly selected from the total population, the means of the individual samples of size 'n' will also have a known distribution. Hence, a single component test sample of size 'n' will have a known error band associated with its use in the estimation of the mean of the total population. This is particularly important in risk analysis since the location of the population mean solely from the mean of the sample and without consideration of sampling error can lead to an under-prediction of the 'best estimate' of the risks associated with individual service components. The sampling error x', is equal to the difference between the sample GM and the population GM. In other words, since the population mean life has been determined from the test sample mean life, the location of a component service life relative to the burst distribution is not precisely identified. Equally, the analysis can be considered from the viewpoint of an error in the service component life location relative to a fixed dysfunction distribution. This approach yields identical risk values as obtained with the assumption of dysfunction distribution error, but is simpler to illustrate, Figure 4.

It follows that for a specified service life and its associated sampling error, x', the risk/efh is equal to the integral of the probability of burst distribution over the interval (efh) to (efh+1). The total risk/efh is now the summation, or the integral of these risks multiplied by the probability that the sampling error is x'. Hence, to account for the sampling error x', the risk/efh is equal to the integral of the risk/efh given the location x' of the sample mean on the sampling distribution, times the probability of the sample mean being located at x' on the sampling distribution. Since the probability density function for the sampling error x' can be determined, then the following standard statistical equation can be used to solve the risk /efh.

$$P(y) = \int_{-\infty}^{\infty} P(y \mid x').PD(x').\partial\, x' \qquad (15)$$

where 'y' is the risk/efh. The solving of the above equations to give an expression in terms of risk levels associated with individual life expired discs is explained elsewhere [4]. Equation 16 is a basic form of this solution which has been used in some recent risk assessments.

$$\frac{\partial(risk)}{\partial(H)} = \frac{1.5}{H}\sqrt{\frac{n}{n+1.239}} \times Exp\left[-6.949\left(\frac{n}{n+1.239}\right)\left(\ln\left(\frac{H}{Ar \times \beta i}\right) - 1.0782 - \frac{0.49094}{\sqrt{n}}\right)^2\right] \qquad (16)$$

where 'H' is the life of the component service life in efh and 'n' is the component test sample size.

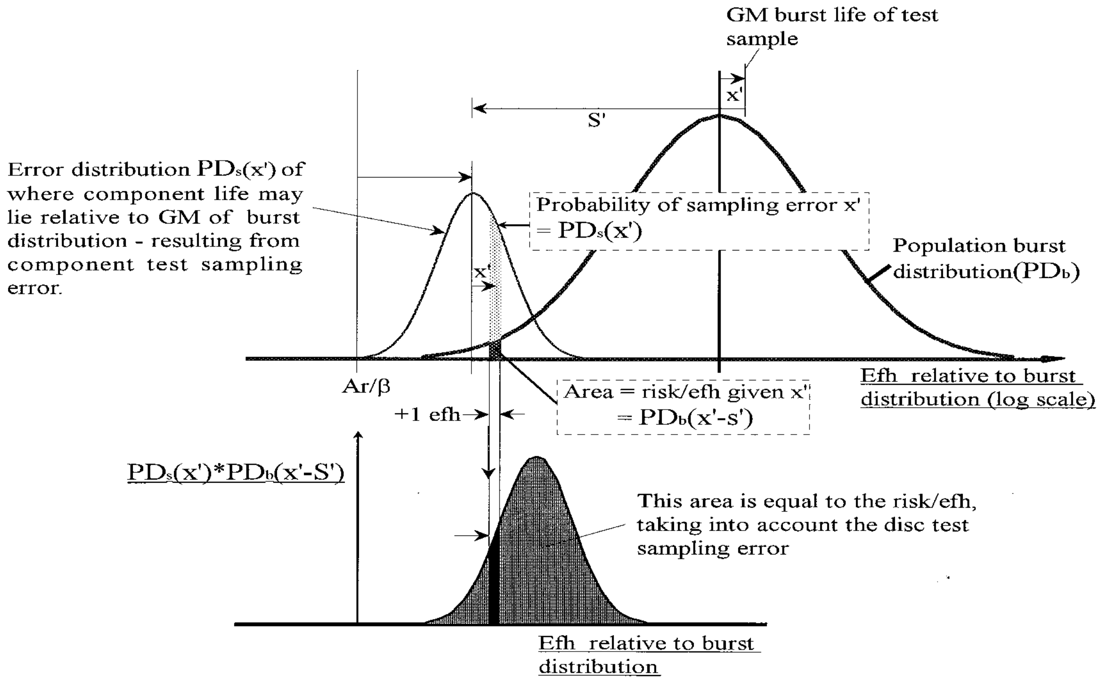


Figure 4. Incorporation of disc sampling error into risk assessment.

5.6 Risk based managed withdrawal of over-lifed parts

From an operational perspective, two of the most useful measures of the risk of fatigue failure of fracture-critical components are a) individual aircraft instantaneous risk per flight hour, and b) fleet cumulative risk per annum (or other suitable management period).

5.6.1 Individual aircraft instantaneous risk of fatigue per flight hour

As components are exposed to continued service usage, the instantaneous risk of failure increases at a highly non-linear rate. A consequence of this is that, in the event of an unforeseen component life reduction, service engines may suddenly have life-expired parts and the operator needs to know the risks that leading parts are incurring. Hence, for example, in a recent service case, the original declared safe service life of 10,000 cycles was calculated from a test sample of 3. With a cyclic exchange rate of 2.0, at the full declared ltfc, Ar, substitution into equation 16 gives an estimated risk/efh of 2.8×10^{-8}. A subsequent requirement reduced the declared safe service life of the component by 20%, to 8,000 cycles. However, at the original life of 10,000 cycles, the risk carried by such a component has now risen to 2.1×10^{-7}/efh. And hence for an individual disc close to the old service life, if this level of risk is deemed acceptable, then such a service life could be maintained for a limited period whilst replacement parts are procured.

An illustration of the form in of results obtained from a risk analysis of a typical aeroengine component, are illustrated in Table 5 (in this case the safe service life, Ar, equals 5,200 cycles). The table provides the calculated risk per efh at the service life for the component and provides estimates of the 'available' additional service hours until the operating risks reach the identified levels.

Total cycles	Cycles over PSCL	Additional Efh	Risk/efh/engine
5,201	0	0	7×10^{-8}
5,344	143	44	1×10^{-7}
5,828	627	194	3×10^{-7}
6,464	1,263	391	1×10^{-6}

Table 5. Estimated lives corresponding to specified risk level/efh/engine using a sample size of 5, and a mission exchange rate equal to 3. (Propagation rates have been assumed to be 2.5 times initiation).

5.6.2 Effect of Fixed Inspection Intervals

In both Damage Tolerance and Retirement for Cause life assessment methods, inspection is perceived to be the essential parameter that underpins safety. A detection limit is set such that to 95% confidence there is a 90% probability that cracks of the maximum specified size will be detected if present. Using this crack size as the input parameter for finite element crack growth analysis, the allowable inspection interval is then set at 50% of the calculated mean crack growth life. It is asserted that the procedure has a built-in safety in that should a crack be missed a further inspection will have occurred prior to dysfunction.

The above illustrations although strictly accurate do not reflect the true safety levels associated with damage tolerance. Indeed safety comes from the fact that extremely few parts are likely to be cracked at the specified inspection interval (although for high strength materials severe quenching, etc. can crack inherent defects at the microscopic level). Given either case, damage tolerance does not optimise the inspection interval. For conventional superalloys, the ltfc approach has demonstrated that typically the dysfunction life is 50% above ltfc. Assuming a crack initiation size of 0.38 mm as a realistic level at which cracks will not be missed, wide European experience indicates that exchange rates for growth beyond the 0.38mm size are not less than a factor of 2.5 greater the rates up to this crack size.

In a damage tolerance context, the inspection interval would normally be based half the crack growth life, that is 25% of the initiation life. However, since a crack growth mission exchange rate has to be assumed, this 25% in cycles translates to only 10% in equivalent service hours that can be flown. It follows that under damage tolerance a disc could be inspected up to 10 times prior to it reaching the European predicted safe cyclic life.

Specifically with regard to retirement for cause and set inspection intervals, Figure 5 illustrates the increased risk associated with running all components until cracks are identified. In this example, the inspection interval has been set at half the propagation life. Inspections 1-3 are unnecessary and inspections 5-7 are too infrequent to ensure acceptable risk management.

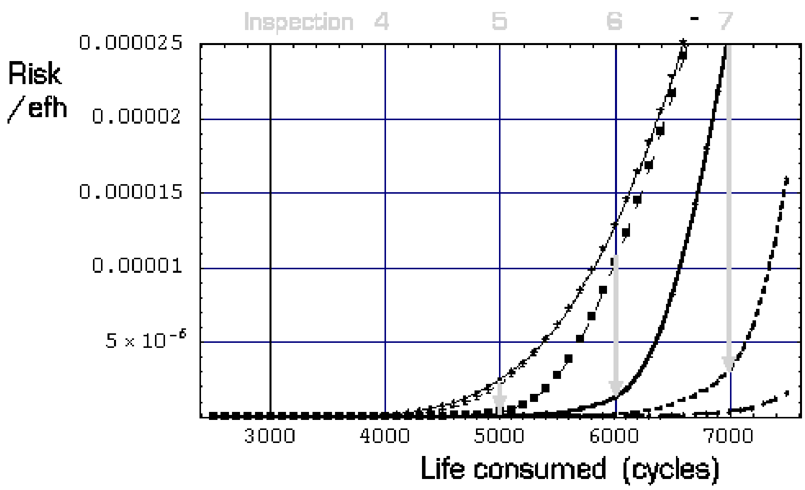


Figure 5. This is the effect of set inspection intervals on component service risk

5.6.3 Risk governed inspection intervals

An extension of the risk model allows new components to be left in service until the dysfunction risk/ reaches a specified limit. At this point component inspection is used to reduce the risk of failure. The model is again used to predict the dysfunction risk during subsequent service and hence the life extension prior to the set risk level being reached and the service intervals for all subsequent inspections. Figure 6 shows that the safe inspection intervals get progressively shorter until the life extension falls below a minimum economical viable service life. Although significant life extension beyond the pscl may be obtained using this approach while still ensuring that the risk of LCF failure does not exceed the specified level, the major benefit comes from identifying a safe service life beyond which retirement for cause would be unsafe. Hence the approach automatically sets an economic limit for retirement for cause but avoids the progressive risk levels associated with its current embodiment.

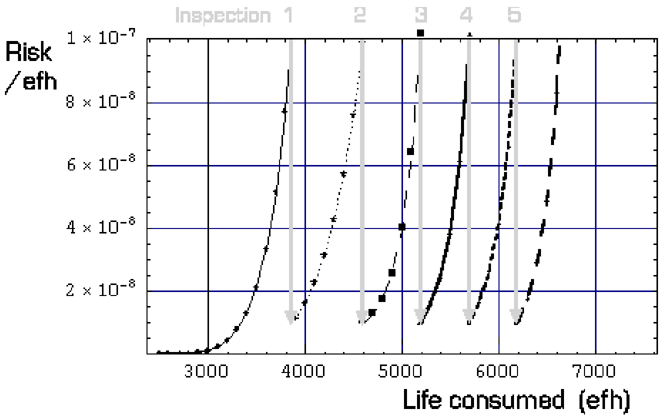


Figure 6. Prediction of the risk associated with application of Risk Based Damage Tolerance. Inspection intervals set to keep to remain within a set risk level of 10^{-7} per efh. (Note the long interval before inspection 1 and that the PSCL/β in this example is 2,880 efh).

6 FINAL COMMENTS

The paper has briefly summarised the common life assessment procedures for fracture critical parts. A procedure which allows the maximum benefit to be extracted from non-finite fatigue results whilst still maintaining safety. The application of risk based life assessment methods have been shown to provide more consistent levels of safety than standard procedures. The approach also enables the risks associated with the continued running of life-expired parts and enables informed management decisions to be taken. The model has significant potential for application within damage tolerance lifing procedures and should allow inspection intervals to be set on the basis of defined risk levels. In this context the approach automatically sets an economic limit for retirement for cause but avoids the progressive risk levels associated with its current embodiment.

7 REFERENCES

1. UK Military Defence Standards, Defence Stan 00-971, General Specification for Aircraft Gas Turbine Engines, 1986.
2. Joint (European) Airworthiness Requirements - Engines (JAR-E), Civil Aviation Authority, 1986.
3. ‘Critical parts life extension based on Fracture Mechanics’, P. Blüml and J. Broede, RTO Workshop ‘Qualification of Life Extension Schemes for Engine Components, Oct 1998.
4. ‘The Development of Life Extension Methods for Fracture Critical Aeroengine Components’, Proc. of NATO Conference on Qualification of Life Extension Schemes for Engine Components, A D Boyd-Lee and G F Harrison, 1999.
5. ‘The Role of Component Testing’, G Asquith, AGARD-R-770, 1989.
6. US DoD Engine structural integrity programme, MIL STD 1783, USAF, 1984.

This page has been deliberately left blank

Page intentionnellement blanche

Life Cycle Management Strategies for Aging Engines

J-P. Immarigeon, W. Beres, P. Au, A. Fahr and W. Wallace
NRC Institute for Aerospace Research
Montreal Road
Ottawa, Ontario K1A 0R6
Canada

A.K. Koul
LPT Incorporated
P.O. Box 46125, 2339 Ogilvie Road
Ottawa, Ontario K1J 8M0
Canada

P.C. Patnaik and R. Thamburaj
Orenda Aerospace Corporation
Gloucester, Ontario K1J 9L8
Canada

ABSTRACT
Diminishing budgets for new weapon systems are creating pressure within NATO nations to keep legacy aero engines in operation well beyond their service life expectancy. Techniques for safely extracting maximum usage out of aging components in these engines, to reduce maintenance costs, are discussed. The mechanisms responsible for the aging of components are described. The different strategies that fleet managers may adopt for extending component lives economically and safely are identified and discussed from an operator's perspective. The paper borrows from recent NATO activities in this area and shares related Canadian experience. It presents and discusses a qualification methodology for component life extension developed in Canada for the Canadian Forces. The methodology incorporates an Engine Repair Structural Integrity Program (ERSIP), which is used to identify structural performance requirements and the qualification tests required to ensure component airworthiness throughout the extended life. Examples of life extension technologies applied to gas path components and critical rotating parts are described, including the use of protective coatings and repairs to increase component durability. The application of damage tolerance concepts that allow safety-critical components to be used beyond their conventional safe-life limit is also addressed.

1. INTRODUCTION

This paper offers an operator's perspective of the problem of aging in gas turbine engines. It is aimed at NATO engine fleet managers and provides suggestions and advice for the cost-effective management of equipment through judicious use of component life extension technologies. The work presented is based on a fifteen-year Canadian team effort involving the development and implementation of a variety of component life extension technologies for Canadian Forces (CF) aero engines. This development program has involved extensive collaboration between CF engines life cycle managers (LCM), the NRC, the Chief Research and Development of the Canadian Department of National Defense (DND), Canadian industry (Orenda Aerospace Corporation and others) and universities. The paper raises a number of questions for the benefit of NATO engine LCMs. What does aging of engines mean and what does it entail? Why is aging of engines an on-going concern to NATO operators? What are the challenges faced by operators, overhaulers and the NATO research community to address aging engine problems? It also suggests what LCM can do to cost-effectively manage NATO aging engine fleets, while ensuring engine reliability and safety.

2. AGING OF ENGINES – CAUSES, EFFECTS AND MANAGEMENT

Besides gradually falling prey to obsolescence, engines age in service due to the gradual deterioration of many structural and functional components. This deterioration is due to damage incurred in service as a result of the highly demanding operating conditions. The aging of components is the cumulative effect of service time, quality of maintenance and the nature and conditions of operation. The rates of damage accumulation are difficult to predict due to uncertainties in operating conditions and, occasionally, changing requirements. Aging damage reduces component structural integrity and is therefore detrimental to engine reliability and safety. It may also reduce engine performance [1]. When damage exceeds allowable limits, or when lives dictated by design are reached, the components should be replaced with new ones. Replacing service-damaged or life-expired parts, is costly and is a significant contributing factor in life cycle costs. [2].

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

2.1. Aging Damage Modes

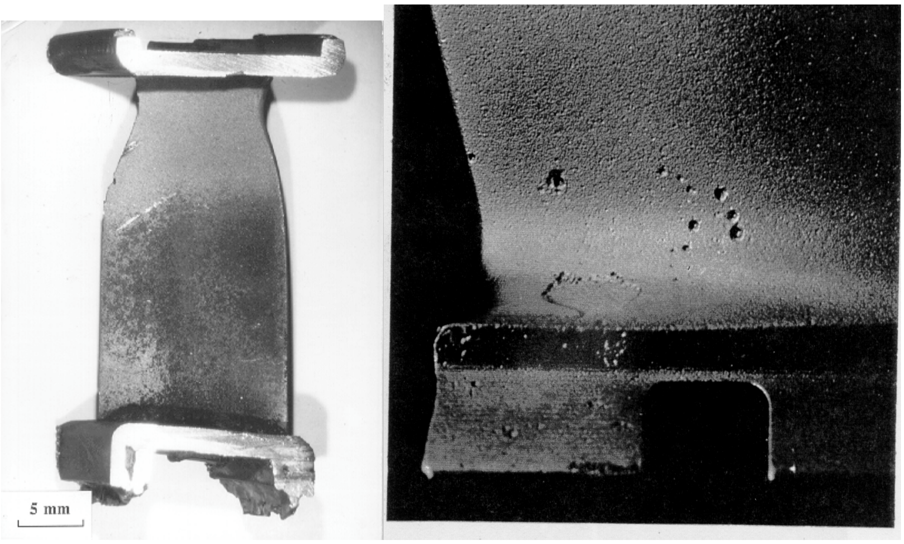
Aging of engine components may take many forms depending on component, engine type and operating conditions. The damage may be external affecting dimensions and surface finish, as a result, for instance, of erosion, wear, corrosion or oxidation. This form of damage affects the aerodynamic performance and load bearing capacity of gas path components. Surface cracks and notches induced by low cycle fatigue (LCF), fretting-wear or foreign object damage (FOD) may also lead to high cycle fatigue (HCF) failures. The damage may also be internal, affecting the microstructure of highly stressed and hot parts, as a result of metallurgical aging reactions, creep or fatigue. This form of damage may reduce component strength and lead to component distortion. Its accumulation may cause the initiation of flaws, which may lead to cracking and component failure [1,2]. Table 1 summarizes the generic forms of damage known to affect engine components by type.

Table 1. Life-limiting damage modes for turbine engine components

Section	Component	Failure Mode
Fan	Blades	FOD
Compressor	Blades	FOD, ER, COR, HCF
	Vanes	FOD, ER, COR, HCF
	Discs	LCF, C, HCF
	Spacer	LCF, C, HCF
Turbine	Blades	TMF, C, HC, LCF, HCF
	Vanes	TF, HC, C, HCF
	Discs	LCF, C, HCF
	Torque Ring	LCF
Combustor Case		LCF, TF, C, HC
Shaft		LCF, WR
Compressor discharge case		LCF, COR
Rotating seal		LCF, C, HCF

Abbreviations:

FOD: Foreign Object Damage
HCF: High Cycle Fatigue
LCF: Low Cycle Fatigue
TMF: Thermo-mechanical Fatigue
TF: Thermal Fatigue
ER: Erosion
COR: Corrosion
C: Creep
HC: Hot Corrosion
WR: Wear



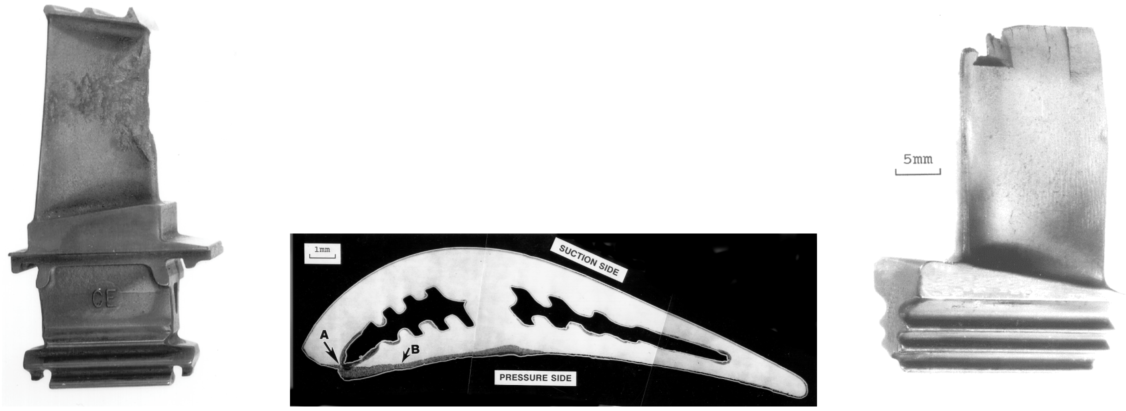
(a) (b)

Figure 1. (a) Erosion damaged compressor vane airfoil from a CF transport aircraft after approximately 5000 hrs of service (b) Corrosion pits close to the root of a compressor blade from a transport aircraft engine exceeding damage allowable limits.

2.1.1. Surface Damage

The eroded T56 vane segment shown in **Fig 1a** illustrates how erosion by ingested sand and other hard particulate matter can significantly alter compressor airfoils shape and surface finish. Such changes, in addition to reducing compressor efficiency, may lead to resonant excitation and HCF failures of airfoils [3]. Pitting corrosion may develop in marine environments. Both steels and titanium alloys are particularly susceptible to this form of damage. Corrosion pits provide sites for crack initiation and have been known to be responsible for HCF failures. The corrosion pits in the root section of a T56 compressor blade, **Fig 1b,** exceed allowable limits and would be cause for rejection at overhaul [4]. Corrosion pits in discs may also reduce LCF life.

Surface oxidation reduces the load bearing capacity of turbine blades and vanes. In marine environments, hot corrosion may rapidly destroy airfoils by surface melting, as evidenced by surface rippling, **Fig 2(a)** and metallography for a Mar-M246 T56 turbine blade, **Fig 2(b)** [5]. Turbine blades also often suffer tip oxidation, as protective coatings wear off rapidly at this location due to tip rub. In the case of directionally solidified (DS) blades, tip oxidation may lead to thermal fatigue cracking of the longitudinal grain boundaries that are embrittled as a result of boundary oxidation, **Fig 2(c)** [6].



(a) (b) (c)

Figure 2. (a) Evidence of hot corrosion damage on the pressure side of a Mar-M246 blade; (b) Metallographic section taken halfway across the airfoil of the blade showing evidence of hot corrosion damage penetrating the leading edge right through to the internal cooling passage; (c) Tip cracking of a DS blade caused by thermal fatigue of longitudinal grain boundaries embrittled as a result of grain boundary oxidation.

Fretting occurs where fan and compressor blades come in contact with discs. In CF F404 engines, the fretting occurs along the dovetails of fan and compressor blades as shown in **Fig 3**, which illustrates the various forms of damage incurred by this component in CF engines. Fretting scars may act as stress raisers, giving rise to fretting fatigue cracks. Fretting tends to reduce HCF life, but may also reduce LCF life. Another common form of damage affecting gas path components is foreign object damage (FOD), the result of impact by ingested foreign objects (e.g. rocks, ice pellets) with either static or rotating components. FOD is the predominant mode of damage for fan blades in CF F404 engines, and leads to their removal from service with considerable LCF life remaining. Most forms of external surface damage can be minimized through use of protective coatings or surface modification treatments [3,7-9], while FOD can be repaired within limits, as detailed below [10].

2.1.2. Internal (Microstructural) Damage

Internal microstructural damage is the result of metallurgical aging reactions and plastic strain accumulation. Time-dependent aging reactions occur primarily in hot parts, such as turbine blades and vanes. The reactions are varied and are invariably detrimental to mechanical properties, causing either loss of strength or embrittlement [11,12]. An example of coarsening of the strengthening precipitates in an alloy 713 blade arising from service exposure is shown in **Fig. 4**. Such coarsening causes loss of creep strength in nickel base superalloys. Loss of strength caused by service induced metallurgical aging reactions may lead to distortion of hot parts. Vane airfoils may bow while blades may lengthen or untwist. Extreme distortion may lead to HCF failures.

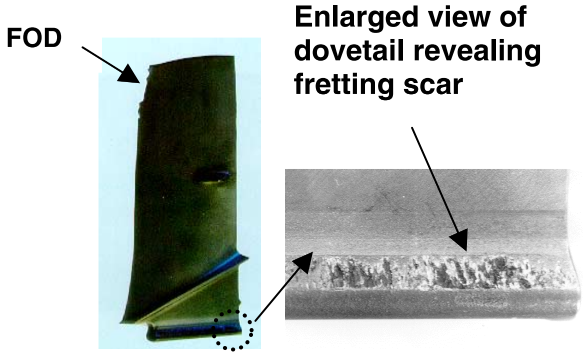
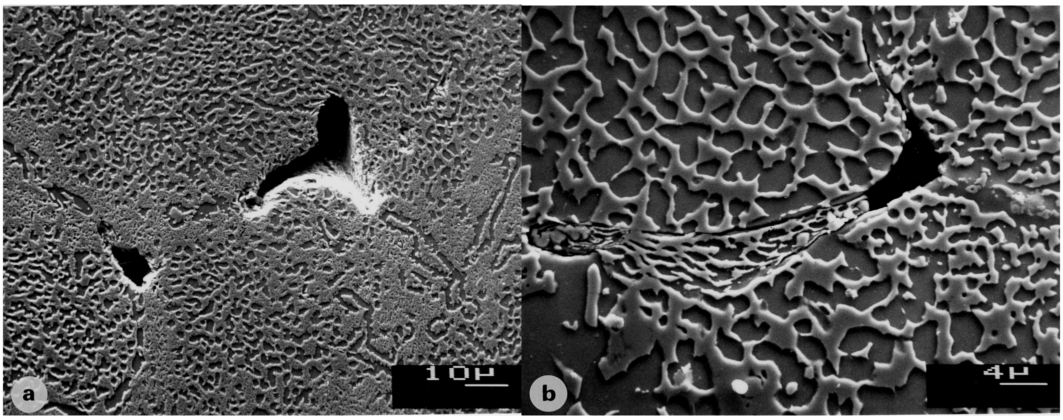


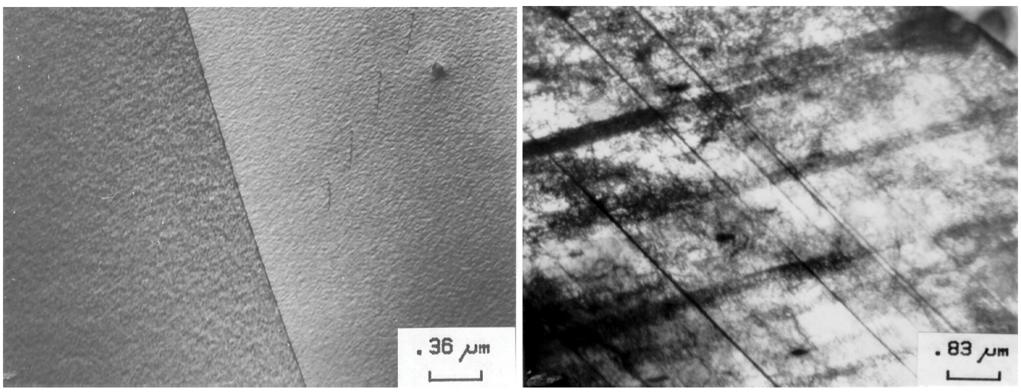
Figure 3. Foreign-Object-Damaged (FODed) fan blade from a CF engine also showing evidence of fretting fatigue damage along the blade dovetail. The surface fretting scar may lead to HCF or LCF failures.



(a) (b)

Figure 4. Microstructure of alloy 713C turbine blades halfway along the blade airfoil: (a) New blade microstructure (b) Microstructure after 5000 hrs of service showing evidence of coarsening of gamma prime precipitates and elimination of secondary gamma prime caused by service exposure. The internal porosity is characteristic of as-cast blades.

Plastic strain accumulation is the product of creep and/or fatigue. It manifests itself in the form of dislocation substructures [13]. Creep deformation leads to creep cavities and internal cracks. The presence of persistent slip bands in the bolt hole region of a disc is indicative of high temperature LCF damage, prior to crack initiation, **Fig. 5**.



(a) (b)

Figure 5. Effects of service exposure on microstructure of disc (a) Virgin disc; (b) Service exposed disc showing evidence of dislocation activity indicative of LCF damage accumulation [12].

Internal microstructural damage is an insidious form of damage because, in contrast with surface damage, it cannot be readily detected by NDI techniques. Its rate of accumulation is strongly influence by service stresses and temperatures, and since there are uncertainties in the temporal variations of these parameters, the extent of damage accumulation cannot be easily predicted. Consequently, the residual life of components is difficult to predict. Under LCF loading conditions, the build-up of microstructural damage in highly stressed components, such as discs and spacers, leads to crack initiation [14]. Cracks initiating at bolt holes or serrated blade slots may grow and lead to catastrophic failures.

2.2. Aging Damage Management

For the purpose of life cycle management, engine components can be classified as either durability-critical or safety-critical parts [2,15]. Durability-critical parts, are those for which aging deterioration affects mainly engine performance and fuel efficiency and may result in a significant maintenance burden, but will not normally impair flight safety. These parts include cold and hot gas path components such as vanes and blades. Safety-critical parts are those for which fracture may result in loss of the aircraft because of non-containment. These parts include most of the large rotating compressor and turbine components, such as wheels, discs, spacers and shafts. Quite different philosophies and techniques are used to manage the life of durability critical and safety critical parts [16].

2.2.1. Durability-Critical Parts

For Durability-critical parts, an "On-Condition" maintenance approach is normally practiced. Parts are removed from service when physical damage limits dictated by design and established through analysis are exceeded. No "hard time" life is set for such parts, but a minimum life expectation is usually guaranteed

2.2.2. Safety-Critical Parts

For Safety-critical parts, two life cycle management approaches are followed. They are (a) the "Safe-Life" approach, for which all components are retired before a first crack is detectable, and (b) the "Damage Tolerance" approach, for which all components are assumed to contain growing cracks, and individual components are retired when a crack is detected.

The Safe Life Approach

With the Safe-Life approach, it is assumed that, should a crack appear, the component has failed. The Safe-Life approach ensures that all components are retired before the first crack appears. This methodology follows a "cycles to crack initiation" criterion, with a minimum safe life (or hard life) capability established statistically through extensive mechanical testing of test coupons and components under simulated service conditions. The statistical minimum is based on the probability that only 1 in 1000 components (-3σ) will have developed a detectable crack, typically 0.8 mm long, at retirement, **Fig 6**.

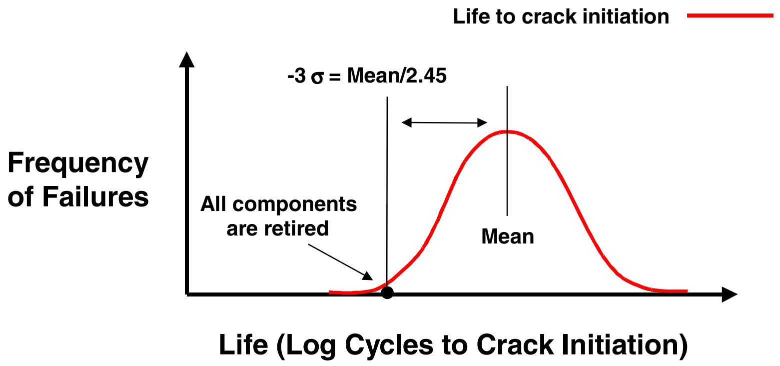


Figure 6. The Safe-Life approach

The advantages of the Safe-Life approach are that the maintenance requirements are kept to a minimum and time in service of components without inspection is maximized. Disadvantages are that the Safe-Life approach is overly conservative because components are retired with a significant amount of useful residual life (practically, 999 out of 1000 components are retired with no detectable damage). Furthermore, the approach is costly since all parts need replacing nominally at the same time. Under such conditions, the supply of spares may be a serious problem, which is often the case for older legacy engines.

The Damage Tolerance Approach

With the Damage Tolerance approach, it is assumed that fracture critical areas of components contain manufacturing or service-induced defects giving rise to cracks that may grow during service. It is also assumed that components are capable of continued safe operation as the cracks grow under service stresses, of both thermal and mechanical origins. It is further assumed that cracks grow in a manner that can be predicted from linear fracture mechanics, or other acceptable methods. Finally, it is assumed that cracks grow sufficiently slowly to allow their detection through regularly scheduled inspections. The approach follows an inspection schedule established by analysis that ensures cracks will not grow beyond a dysfunction limit. The interval between inspections, or safe inspection interval (SII), is based on the time it takes for a crack to grow from a size immediately below the detection limit of the method used to inspect the component, to a critical or dysfunction size, beyond which the risk of rapid or unstable crack growth becomes too high. The dysfunction crack size is obtained by analysis, for an assumed crack geometry, from the fracture toughness of the material and the stress intensity factor for the component of interest, using appropriate safety factors. The approach is described schematically in **Fig. 7** [2].

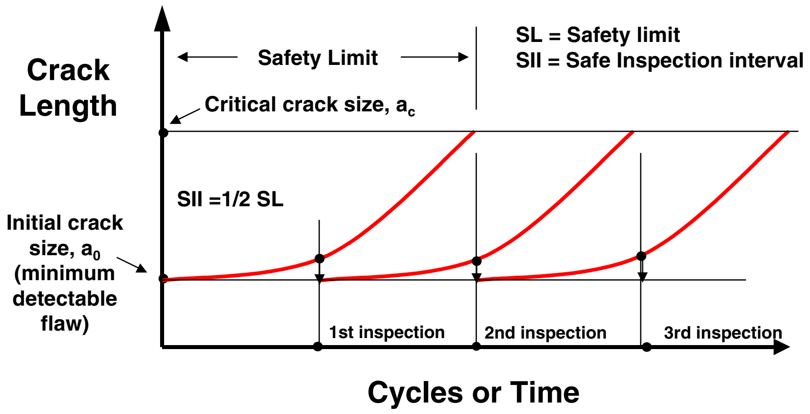


Figure 7. The Damage Tolerance approach

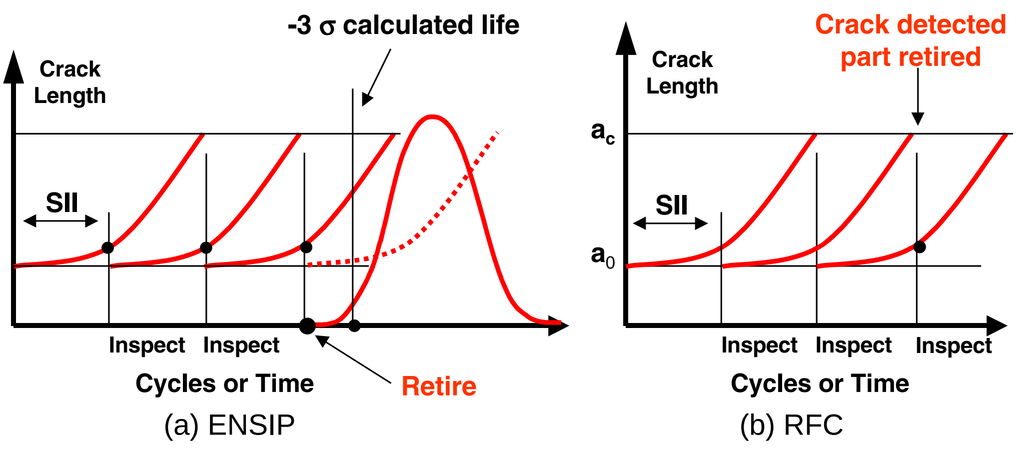


Figure 8. Schematic representations of the damage tolerance approaches based on (a) MIL-STD 1783 (ENSIP) and (b) Retirement for Cause (RFC), respectively.

Two methods may be used to implement a damage tolerance based life cycle management for safety critical parts. The first method, known as ENSIP (Engine Structural Integrity Program - MIL-STD-1783), was introduced in 1984 by the USAF for the structural design, analysis, development, production and life management of engines [16]. ENSIP embraces a fracture mechanics based damage tolerance approach to set safe inspection intervals for safety-critical parts. However, conventional structural design criteria are also used to minimize the risks of failure due to vibration, LCF, HCF and creep. With ENSIP, individual compressor and turbine discs are retired once their demonstrated Safe-Life (life to crack initiation) is reached.

The second method, known as Retirement for Cause (RFC) also relies on fracture mechanics to set safe inspection intervals for safety critical parts, as practiced with ENSIP. Retirement life is based on periodic inspections until a crack is detected, at which point the part is retired. With RFC, individual compressor and turbine discs are retired only once they are found to contain a crack. The ENSIP and RFC methods are described schematically in **Fig. 8** [2]. A third option, known as 2/3 Dysfunction, may also be practiced, as discussed elsewhere [17].

The advantages of the Damage Tolerance approach are that it ensures that cracks emanating from manufacturing defects (or service-induced cracks) in anyone component will not grow beyond allowable limits. Furthermore, the approach allows life extension beyond LCF based Safe-Life limits through use of the RFC method, if needed. Disadvantages are that the Damage Tolerance approach is more costly to implement than the safe life approach. It requires use of an elaborate NDI infrastructure to support increased inspection requirements. In addition, the handling of components is increased.

3. THE AGING OF ENGINES - WHY IS IT A CONCERN TO NATO OPERATORS

Because of diminishing budgets for new weapons systems, many NATO forces are faced with having to operate fleets of engines well beyond their anticipated service lives. These engines contain components often designed years ago made from materials that lack durability relative to new engine materials. This material obsolescence translates into short component lives and high maintenance costs. In addition, the residual lives of safety critical parts for many of these old engines are not accurately known. This is because the main design factors considered for compressor and turbine discs developed in the 50’s and 60’s were tensile properties in the bore and the rim of the discs and creep properties in the rim of turbine discs. This was done to provide an over-speed margin without disc burst. For engines of that generation, fatigue lives were not provided for safety-critical parts. Therefore, neither cyclic life consumption nor retirement criteria are available for these engines to dictate the retirement of safety-critical components. How long these engines can be kept in service safely without having to replace their critical components is a growing concern among NATO operators.

4. MANAGING NATO AGING ENGINE FLEETS – THE CHALLENGE

The need to balance risk and high maintenance costs is providing engine fleet managers with incentives to identify and implement strategies for extracting maximum life out of engine components, while ensuring the engines remain safe to operate and reliable in service. Several options to do this are available to LCMs.

4.1. Life Extension Strategies – The Options

One attractive strategy available to LCMs is to equip aging engines with modern health and usage monitoring (HUMS) systems to better predict parts life consumption. This allows component lives to be extended when the assumed mission severity is overly conservative. HUMS in combination with an engine parts life tracking system (EPLTS) allows an operator to optimize engine inspection schedules and the removal of service-exposed parts to achieve more cost-effective maintenance schedules. It also allows more optimal use of parts life potentials and minimizes risks of in-service premature failures. Other strategies for extracting maximum life out of durability-critical components include:

(1) returning service-damaged parts to functional serviceability through use of repairs, such as welding, brazing, rebuilding, re-contouring and rejuvenation heat treatments and

(2) delaying rates of damage accumulation through the addition of protective coatings or surface modifications treatments; a material change for the component is also possible.

For safety-critical parts, the life extension option is to implement a damage tolerance life cycle management approach. The challenge faced by LCMs is to decide upon what option, if any, to implement and to ensure operational safety of the vehicle.

4.2. Life Extension Strategies – Decision factors

The decision to replace or to extend the life of an engine component must consider:

(1) operational consequences of component failure,

(2) cost-effectiveness of the proposed life extension and

(3) qualification testing requirements. Tests are required to qualify as airworthy parts subjected to life extension. These tests must be selected to demonstrate that the parts once returned to service will remain safe and reliable through the life extension [2].

Safety considerations

These are best addressed through use of a Failure Mode and Effect Criticality Analysis (FMECA). A FMECA is a reliability analysis tool used to identify the possible modes of failure on a component-by-component basis, the probability of those failures occurring in service, and the potential consequences of failure [18].

Cost considerations

These are always important and can be addressed through use of a cost benefit analysis (CBA) [19, 20]. However, when there are no spare parts available, either due to a supply shortage, or because replacement components are not made anymore, and the engine must be kept in service, the cost considerations may be less significant or even irrelevant.

A recent trend has been the use of components manufactured by approved sources other than the OEM or OEM approved sources, known as Parts Manufacturing Authority (PMA) components. The PMA parts use is on the rise due to simple economics and parts availability for older generation engines. The users should however ensure that appropriate qualification and quality control procedures have been followed prior to using such parts, as described below.

Technical considerations.

These require that the life extension process be carried out to the same standard used to qualify the original product, or to an equivalent standard. The applicable standards for military aero engines are MIL-E-5007E, MIL-E-8593A, or MIL-STD 1783 (ENSIP) although other standards may apply depending on the engine type and country of origin [21]. MIL-STD-1529 (Vendor substantiation for aerospace products) describes procedures to qualify additional/alternate vendor and fabrication sources other than those qualified for the original product.

5. THE CANADIAN APPROACH TO AGING DAMAGE MANAGEMENT

In Canada, the Department of National Defence has recently adopted a Qualification Methodology, developed jointly by Orenda Aerospace Corporation and NRC, for engine component life extension [18]. This methodology is currently applied to repairs and life extension technologies for durability-critical and safety-critical parts, to reduce CF engine operating costs.

5.1. Canadian Qualification Methodology for Component Life Extension

Development of the Canadian methodology evolved from a careful review of civil and military regulatory requirements used in design and for life cycle management of aero-engines [21]. The methodology consists of (1) a FMECA to establish criticality of damage, (2) a CBA to establish whether it is more cost-effective to apply a life extension scheme than to replace a damaged part, and (3) an Engine Repair Structural Integrity Program (ERSIP), modeled after ENSIP, to ensure that parts will remain safe and reliable through the life extension [2, 18].

The Canadian ERSIP is conceived to establish structural performance requirements and to identify tests for the development and qualification of life extension technologies [22]. ERSIP modifies and extends the limits of ENSIP to satisfy needs for the management of components subjected to life extension. It incorporates the damage tolerance approach implied by ENSIP and is used by the CF to establish structural performance, process development and verification requirements that will ensure structural integrity of the components subjected to life extension. The ultimate goal of ERSIP is to ensure structural safety, durability, reduced life cycle cost and increased service readiness of engines. Depending on type and criticality of the targeted component, ERSIP calls for either all or part of the following qualification tests [2]:

(1) *Dimensional inspection* to ensure that the parts conform with drawing requirements;

(2) *Metallurgical verification* to ensure the material meets engineering specifications (depending on component, specification, this may cover grain size, precipitate sizes, degree of porosity, ductile-brittle-transition temperature, etc…);

(3) *Structural tests* to ensure that relevant mechanical properties are equivalent to or better than properties of original parts (depending on component, properties may include hardness, tensile strength and creep properties, HCF, LCF, TMF, etc…)

(4) *Functional tests* to ensure that part functionality is not impaired by the life extension process (e.g. cooling flow rates for internally cooled parts are identical to flow rates in original equipment);

(5) *Rig and engine tests* to verify that the parts after testing still meet the serviceable limits specified in the applicable R&O Manual and that their general condition is comparable to that of approved parts subjected to identical tests.

Different types of engine tests (accelerated endurance, stair-step, sand ingestion, simulated mission endurance, etc.) may be specified depending on the qualification objectives, as detailed elsewhere [2].

5.2. Life Extension Strategies Covered by ERSIP

Life extension schemes covered by ERSIP include:

(1) restoration of damaged components to serviceable conditions by repair or rework,

(2) modifications intended to improve the structural performance or damage tolerance of engine components (e.g. a material change, the addition of a coating or a surface treatment) and

(3) reuse of components under a damage tolerance-based life cycle management scheme to achieve life extension, all in conformity with the requirements of ERSIP.

5.2.1. Life Extension Through Component Restoration

Much work has been done in the commercial world to develop repairs and reworks for civilian aircraft engines. Repair vendors have been competing quite successfully with original engine manufacturers (OEM) in these developments. The delegation of authority by national aviation authorities to R&O organizations has encouraged such trends. Technologies developed for civilian products are for the most part applicable to military platforms and can be adopted by military organizations to achieve cost-effective management of aging engines [23]. Different types of component restoration schemes have been developed for aero-engines. Examples of schemes developed in Canada for the CF are provided below.

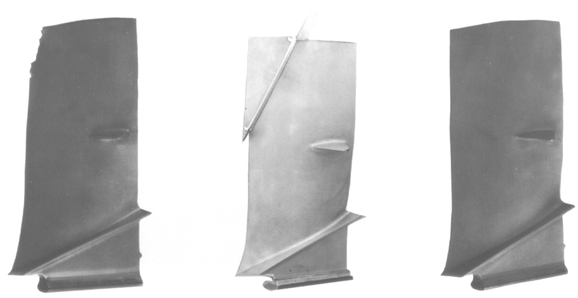


Figure 9. Repair of FODed F404 fan blade by electron beam welding a corner patch to replace damaged portion of blade.

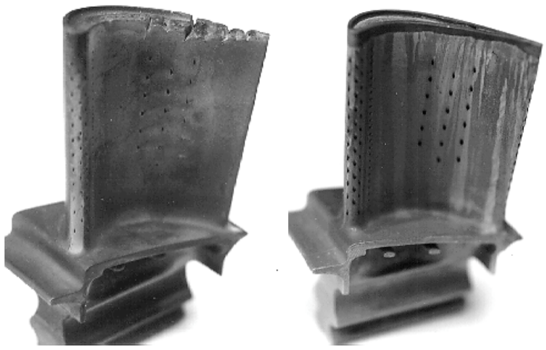


Figure 10. Weld tip repair of DS René 80 first stage turbine blade form CF F404 engine (Courtesy Liburdi Engineering Ltd., Hamilton, Ontario)

Electron Beam Weld (EBW) Repair of FODed F404 Fan Blades

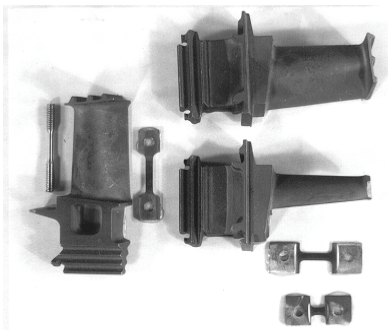
In this repair scheme developed by OAC to eliminate foreign object damage from fan and compressor blades, the damaged area is replaced with a patch of matching material (Ti64 in this case), which is joined to the airfoil by EBW, as shown in **Fig. 9**. Details of the technique and its qualification for the repair of F404 fan blades were presented at the AVTP Workshop on Cost Effective Applications of Titanium Alloys, in Loen, Norway, April 2001 [10]. Qualification testing for this repair included (1) an assessment of weld microstructure and residual stresses, (2) an evaluation of strength and fatigue properties of welded test coupons, (3) a comparison of the vibration characteristics, fatigue properties and ballistic impact resistance of new and weld repaired components and (4) engine block testing of repaired components, as part of an accelerated mission test (AMT) performed in one of IAR's test cells on a CF F404 engine.

Weld Tip Repair of High Pressure Turbine Blades

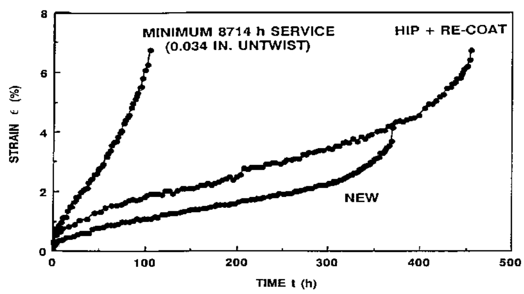
A weld tip repair was developed by OAC in collaboration with Liburdi Engineering (Hamilton, ON, Canada) for the DS Rene 80 HPT blade from the CF F404 engine, **Fig. 10.** For this repair, the tip damage is removed by grinding and the blade tip is rebuilt using an automated welding technique. The weld material is chosen to provide enhanced oxidation resistance at the blade tip to prevent oxidation of the underlying columnar grain boundaries. Details of this repair were presented at the RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998.[Liburdi Corfu, 10].

Advanced Braze Repairs for Nozzle Guide Vanes

Hydrogen fluoride (HF) cleaning, in combination with diffusion brazing, makes it possible to repair nickel base superalloys. The presence of thermodynamically stable oxides along crack faces makes these alloys difficult to braze. Cleaning with HF removes the oxides and promotes wetting and penetration of the crack by the braze alloy, thereby creating a structurally sound and durable joint. Special low temperature braze alloys have been developed for this type of repair. The braze alloys contain elements that lower their melting point. The molten braze alloys re-solidify at brazing temperatures once the melting point suppressant has diffused away from the joint into the bulk of the component. The AFOR-DBR process developed by Vac-Aero International in collaboration with NRC is an example of this type of repair [24]. The Liburdi Engineering LPM™ joining/cladding process [7,25] was developed as a hybrid wide gap brazing technique that has proven successful for the repair of both blades and vanes. The process enables a wide range of alloys to be used for crack repair and surface build-up. The damage is first removed by grinding and a powder metallurgy putty of matching or custom composition is applied and diffused into the surface to complete the repair and achieve the desired mechanical and metallurgical properties.



(a)



(b)

Figure 11 (a) Miniature creep specimens machined from the airfoils of different T56 turbine blades; (b) Creep curves for new, service-exposed and HIP rejuvenated 713LC blades [13,32].

Component Rejuvenation through HIPing or Heat Treatment

Combinations of heat treatments and hot isostatic pressing (HIPing) have been used to eliminate service induced microstructural damage in turbine blades and vanes to restore creep properties. HIPing eliminates creep voids that may have formed during service. It also eliminates casting porosity, which improves the component reliability by reducing scatter in material properties. The re-coating heat treatment completes the rejuvenation. HIP rejuvenation cycles have been developed at IAR for alloy 713C and IN 738, both of which are used for blades and vanes in T56 engines [26,27]. HIPing can also be used in conjunction with other forms of repairs, for instance to eliminate shrinkage porosity within braze joints [24]. Qualification of rejuvenation treatments for T56 blades relied in part on tests performed on miniature specimens machined from the blades. Results indicated that HIP cycles can be optimized to achieve rupture lives and creep elongation that are significantly higher that those for new blades with a nominally identical minimum creep rate, **Fig. 11.**

Rebuilding of Worn Seal Teeth
The repair of worn seal teeth from rotating air seal components is another economically desirable refurbishment procedure. The tips of the damaged teeth are first ground down and then rebuilt with over-lays of similar alloys, using welding techniques such as Dabber™ welding or pulsed laser or plasma torch welding [7]. Example of refurbished seal teeth are shown in **Fig 12**. The teeth are first rebuilt by laser cladding with a pre-alloyed powder and subsequently finish-machined to design specifications. Before implementing such types of repair, it is essential to assess the risk of cracks initiating at imperfections in the built-up weld or at the weld metal/parent metal interface. Simulated seal tooth specimens are desirable for this qualification work but the interpretation of results can be quite complex [28].

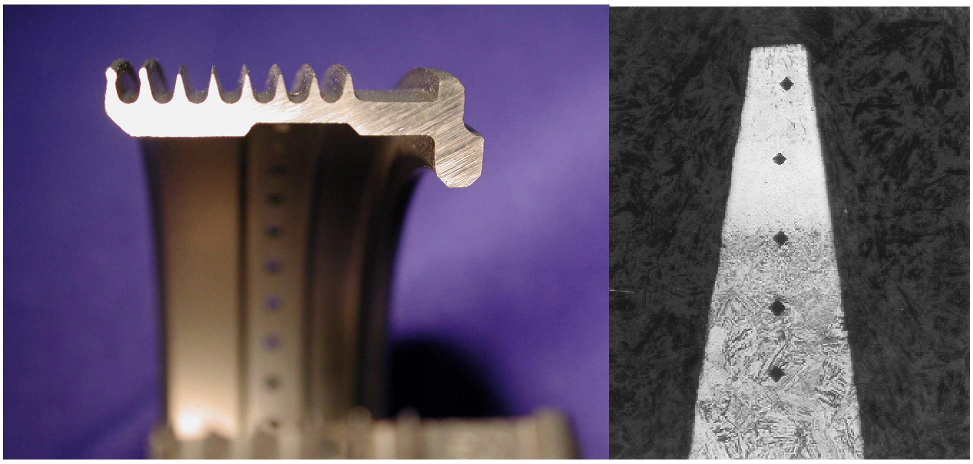


Figure 12. Example of seal teeth repair by laser welding (Courtesy of Standard Aero Ltd., Winnipeg, Manitoba).

<u>5.2.2. Life Extension Through Material Modifications</u>
The objective is to enhance component durability either by:

(1) substituting the component material for another to improve, for instance, the component creep properties, or to eliminate a problem,

(2) substituting or adding a protective coating to improve resistance to wear, fretting, erosion, oxidation or corrosion (e.g. adding a hard coatings to compressor airfoils or a TBCs to hot parts) or

(3) applying a surface modification treatment such as shot peening, ion implantation or laser surface processing to improve the resistance to various modes of surface degradation or fatigue.

Retrofitting with New Materials
This is occasionally implemented by an OEM, usually under the umbrella of a client-supported component improvement program (CIP). There are examples of material changes involving both blades and discs. For CF F404 engines, the first stage HPT blade material was changed from a conventionally cast René 125 to a directionally solidified (DS-columnar grained) René 80, to improve the creep strength and thermal fatigue resistance of the blades. A subsequent change to single crystal alloy N4 was approved to further improve properties of the blades. With changes of this type, the expected improvements are not always met, because a new and unexpected mode of damage may prove life limiting. This happened with the change to the DS blades in F404 engines, which suffered tip oxidation and thermal fatigue cracking along the columnar grain boundaries. Disc materials are more rarely changed. One of the CF J85 CAN40/15 engine compressor disc was changed from an AM355 martensitic stainless steel to a DA718 nickel base superalloy to eliminate premature bolthole cracking.

Substituting or Adding a Protective Coating
Titanium nitride (TiN) applied by physical vapor deposition (PVD) has been qualified as a coating for the protection of compressor airfoils against particulate erosion. The RIC™ PVD TiN coating from Liburdi Engineering is a bill-of-material option for T56/K501 RR engines. Testing to qualify TiN coated blades for CF engine use included fatigue testing in a specially designed test rig at resonant frequencies under the blade

1st bending mode [29]. Weibull analysis of the results for bare and coated blades indicated that coated blades have marginally higher fatigue strength and a lower probability of failure than bare blades at an equivalent blade root stress [3].

Turbine blades and vanes are routinely re-coated at overhaul. This provides an opportunity to change the coating for one that is better suited to a particular operating environment. For instance, Pt aluminides offer durability enhancement over conventional aluminides when hot corrosion is life-limiting. A coating can also be added to internal cooling passages [8] or to a normally uncoated component, as did OAC with the MA-754 high-pressure turbine nozzle in the F404 engine [30]. The addition of thermal barrier coatings (TBC) to airfoils, including the tip region of turbine blades, lowers metal temperature, thereby minimizing the rate of material consumption due to oxidation. When applied to the leading edge of an NGV, stress gradients may also be reduced, thereby reducing thermal fatigue cracking.

Qualification testing requirements for turbine coatings typically include:

(1) a metallographic evaluation of coating quality and its impact on microstructure and phase stability of the substrate,

(2) an assessment of the effects of coating on the mechanical properties of coated material test coupons (Tensile and creep strength, HCF LCF, TMF, DBTT)

(3) rig tests to ensure flow rates across cooling passages of internally cooled parts are identical for new and refurbished parts

(4) rig tests to compare durability and damage tolerance of new and repaired parts under simulated service conditions, for instance in a burner rig

(5) engine block testing as part of an AMT and

(6) field evaluation in a lead-the-fleet engine.

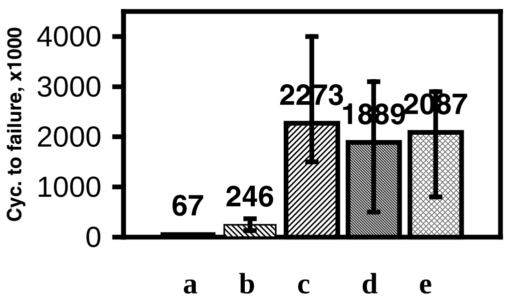


Figure 13. The fretting fatigue life of Ti-6Al-4V:
a = base metal; b = CuNiIn + MoS_2; c = shot peened; d = shot peened + CuNiIn + MoS_2; e = shot peened + MoS_2.

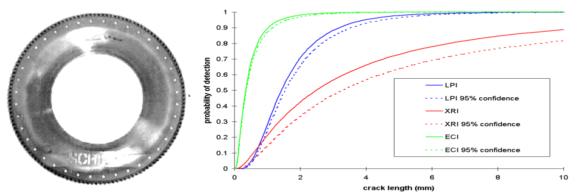


Figure 14. Probability of detection of natural cracks in Fe-Ni-Cr alloy turbine discs using liquid penetrant inspection (LPI), eddy current inspection (ECI) and x-ray inspection (XRI) [35]

Applying a surface modification treatment.

Surface treatments such as ion implantation and chemical surface treatments have been explored for use in conjunction with shot peening and soft coatings to minimize fretting fatigue damage along the dovetails of F404 fan blades. Shot peening in combination with chemical treatments is quite effective in laboratory tests but other surface treatments, including soft coatings and lubricants, appear to have the potential for improving the fretting fatigue resistance of titanium alloys. The effects of these treatments on fretting fatigue life of two titanium alloys (Ti64 and Ti17) have been evaluated using equipment developed at IAR. Some of the results, presented at an AGARD-SMP Workshop on Tribology for Aerospace Systems, Sesimbra, Portugal, April 1996 are shown in Fig.13 [9].

5.2.3. Life Extension Through Re-use of components under RFC based LCM

As noted previously, cyclic fatigue lives are rarely available for the fracture-critical parts of old engines. This is cause for concern to LCMs who have no safety criteria on which to base component retirement. Faced with this problem, it is not unusual for an OEM to establish LCF safe lives of engine rotors retroactively, at a late stage in the life of an engine. This is occasionally done for the benefit of users under a user supported component improvement program (CIP). However, experience shows that the service lives of a significant fraction of components from lead-the-fleet engines can be greater than the calculated LCF safe lives, sometimes by quite significant margins. This reinforces the often-expressed view that components retired at their design safe-life limits may have significant fractions of usable life remaining.

Implementation of a damage tolerance/RFC approach provides opportunities for safely managing parts from these old engines [31]. Components from three CF engines are being analyzed at IAR in collaboration with others for implementation of a life cycle management approach based on a damage tolerance/RFC concept. The engines include the Rolls Royce Nene X, the GE J85 and the Rolls Royce Allison T56. Implementation of a RFC based management philosophy for these legacy engine components requires that six basic steps be followed [2]. These are:

Step 1: Determination of stress and temperature data

This can be achieved by instrumenting targeted engines to determine temperature distribution and strain profiles across the components of interest. The results are used as boundary conditions in FEM models to establish the stress distributions within the components.

Step 2: Identification of the fracture critical location in the component of interest

This is normally achieved through finite element analysis. The lowest serration at the bottom of fir-tree slot is usually identified as the fracture critical location in discs.

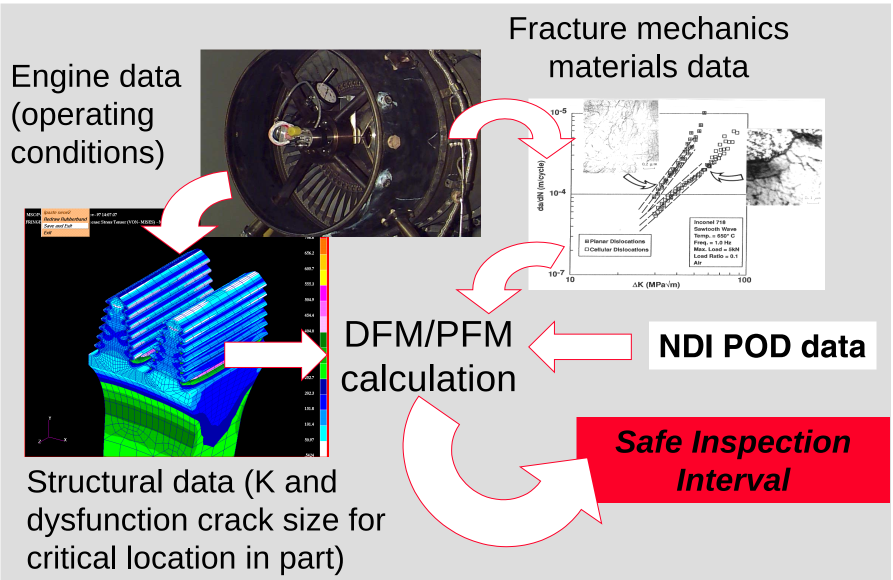


Figure 15. Steps associated with SII calculation for damage tolerance based life management of critical engine parts.

Step 3: Determination of the stress intensity factor (SIF) at the fracture critical location in the component

This is also obtained by finite element analysis at the tip of a crack of specified geometry, assumed present at the fracture critical location of the component. The FE analysis yields the variation of the stress intensity factor with crack depth for a single crack in the fracture critical region of the turbine disc [32]. The SIF values are obtained for a crack with a geometry preferably chosen to provide a worse case scenario.

Step 4: Generation of fracture mechanics data for safe inspection interval (SII) calculations

Relevant FCGR data should be generated at the operating temperature of the fracture critical location, using specimens machined from actual components. The reason for using specimens machined from components is to capture the effects of prior service history on the microstructure and therefore properties of the component [29]. Experience shows that high time IN 718 discs have higher fatigue crack growth rates than low time discs [33].

Step 5: Generation of NDI POD data

Probability of detection (POD) data for the NDI methods of interest are needed to establish the initial flaw size for the analysis. It is best that the POD data be generated from actual life-expired parts [34]. In a program sponsored in part by AGARD, a large number of retired J85 compressor discs were examined for fatigue cracks using a variety of NDI methods. The NDI results were verified by prying open each inspected bolthole and examining the pried-opened areas for evidence of service induced cracking. Actual crack sizes were established from microscopic examinations in a scanning electron microscope. The POD data were generated through standard POD analysis. The POD curves for three common NDI techniques, including Eddy Current Inspection (ECI), Liquid Penetrant Inspection (LPI) and x-ray inspection (XRI) obtained for the J85 Can 40 compressor disc are compared in Fig. 14 [35]. The NDI Detection Limit is defined as the crack size at 90% POD and 95% Confidence. For damage tolerance analysis, the initial flaw size is defined to be either the maximum crack length missed or the crack length value at 90% POD and 95% Confidence. The data indicate that the Eddy Current Approach is more sensitive than the other methods. The substantial amount of POD data derived from this work is available through the USAF supported Non Destructive Testing Information Analysis Centre (NTIAC) of Austin, Texas.

Step 6: Calculations of the safe inspection interval (SII)

The SII for the component of interest is obtained through the application of either probabilistic or deterministic algorithms [36,37], which require as input for analysis (a) POD data for the NDI used to inspect the component, (b) values of the stress intensity factor (SIF) and (c) crack growth rate data. The process is summarized schematically in **Fig. 15**.

Details of the damage tolerance analysis for the CF Nene X turbine disc are being reported at the AVT Symposium on Damage Mechanisms and Control – Part B on Monitoring and Management of Gas Turbine Fleets for Extended Life and Reduced Costs [38]

6. CONCLUDING REMARKS

The deterioration of engine components begins as soon an engine enters service. This deterioration or aging cannot be avoided and must be managed. Managing aging components to extract maximum life from expensive parts requires a good understanding of deterioration modes and their potential impact on engine performance, reliability and safety. From a fleet manager's perspective, a number of options are available to extend component lives beyond book limits. These options include repairing service-damaged parts, enhancing their durability through material modifications, including the substitution or addition of protective coatings, or applying a damage tolerance based life cycle management methodology for safety critical parts. Canadian related experience shows that significant savings in the operating costs of engines accrue from such initiatives [19].

ACKNOWLEDGEMENTS

The authors thank their industry colleagues from Liburdi Engineering, Vac-Aero International, Standard Aero Ltd. for supplying some of the illustrations. The financial support provided by Defence Research and Development Canada (formerly, Chief Research and Development of the Department of National Defence) through the Air Vehicle Research Section is also gratefully acknowledged.

REFERENCES

1. Ratwani M. M., Koul A.K., Immarigeon J-P. and Wallace W., "Aging Airframes and Engines", Future Aerospace Technologies in the Service of the Alliance, NATO-AGARD Conference Proceedings 600, AGARD-CP-600 Vol.1, December 1997, pp18-1/18-16.
2. Immarigeon, J-P., Koul A.K., Beres, W.,.Au, P, Fahr, A. Wallace, W., Patnaik, P.C. and Thamburaj, R, The Aging of Engines: An Operator`s perspective, NATO-RTO-AVT Lecture Series 218, October 2000.
3. Parameswaran, V.R., Nagy, D.R., Immarigeon, J-P., Chow, D.and. Morphy, D.D, "Erosion Resistant Coatings for Compressor Applications", in Advances in High Temperature Structural Materials and Protective Coatings", Eds, A.K. Koul et al, Pub. NRC, Canada, 1994, pp. 262-281.
4. Allison T56/501-D13 Overhaul Manual (Appendix 2-F).

5. Koul, A.K., Immarigeon, J.-P., Patnaik, P.C. and Dainty, R.V., “Degradation of Advanced Aero Engine Turbine Blades”, in Advanced Materials and Coatings for Combustion Turbines, Materials Congress 93, Eds.S. Cheruvu, V.P. Swaminathan, ASM International, Oct. 1993, p. 69.
6. Koul, A.K., “Hot Section Materials for Small Gas Turbines”, AGARD-PEP Specialists’ Meeting on Small Turbines, October 1993, Montreal, AGARD-CP-537, 1993, Paper No. 40.
7. Liburdi, J., “Enabling Technologies for Turbine Component Life Extension”, RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998, NATO Pub., RTO-MP-17, AC/323(AVT)TP/7, March 1999, pp. 11 –1/11-7.
8. Woloziuk, M., Dainty, R.V., Immarigeon, J-P. and Taylor, V., “Metallographic Evaluation of T56 Engine First Stage Turbine Blades Coated with C-30”, NRC LTR-ST-1811, February 1991.
9. Koul, A.K., Xue, L., Wallace, W., Bibby, M., Chakravarty, S., Andrews, R.G. and Patnaik, P.C., “An investigation on surface treatments for improving the fretting fatigue resistance of titanium alloys”, Tribology for Aerospace Systems, AGARD CP No. 589 (1996).
10. Azar, P, Ping Li, Patnaik, P.C., Thamburaj, R. and Immarigeon, J-P., “Electron Beam Weld Repair and Qualification of Titanium Fan Blades for Military Gas Turbine Engines”, NATO-RTO-AVT Workshop on Cost Effective Applications of Titanium Alloys in Military Platforms, Loen, Norway, 9-11 April 2001.
11. Koul, A.K., Castillo, R., “Assessment of Service Induced Microstructural Damage and Its Rejuvenation in Turbine Blades”, Met. Trans. Vol. 19A, August 1988, pp. 2049-2066.
12. Maccagno, T., Koul, A.K., Immarigeon, J-P., Cutler, L. and L'Esperance, G., “Rejuvenation of Alloy 713 C Turbine Blades”, Metall. Trans., Vol.21A, 1990, pp. 3115-3125.
13. Koul, A.K., Thamburaj, R., Raizenne, M.D. and Wallace, W., “Practical Experience with Damage Tolerance Based Life Extension of Turbine Engine Components”, AGARD Conference Proceedings, SMP Panel, San Antonio, Texas, U.S.A., April 1985, AGARD-CP-393, pp 23-1 to 23-22.
14. Koul, A.K. and Dainty, R.V., “Fatigue Fracture of Aeroengine Compressor Discs”, in Handbook of Case Studies in Failure Analysis, ASM International, Materials Park, Ohio, Eds. R.C. Uhi et al, 1992, p.p. 241-253.
15. Holmes, R., “Civil and Military Practices”, Chapter 2 of RTO Technical Report No. 28 on Recommended Practices for Monitoring Gas Turbine Engine Life Consumption, NATO Pub. RTO-TR-28 AC/323(AVT)TP/22, April 2000, pp. 2-1/2-15
16. Davenport, O., “Maintenance Policies and Procedures”, Chapter 3 of RTO Technical Report No. 28 on Recommended Practices for Monitoring Gas Turbine Engine Life Consumption, NATO Pub. RTO-TR-28 AC/323(AVT)TP/22, April 2000, pp. 3-1/3-15
17. Harrison, G.F., “Translation of Service Usage into Component Life Consumption”, Chapter 6 of RTO Technical Report No. 28 on Recommended Practices for Monitoring Gas Turbine Engine Life Consumption, NATO Pub. RTO-TR-28 AC/323(AVT)TP/22, April 2000, pp. 6-1/6-14
18. Patnaik, P.C. and Thamburaj, R., “Development of a Qualification Methodology for Advanced Gas Turbine Engine Repairs/Reworks”, RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998, NATO Pub., RTO-MP-17, AC/323(AVT)TP/7, March 1999, pp. 12 –1/12-11.
19. Hastings, R.R., “Aero-Engine Component Repair/Replacement Decision Factors”, RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998, NATO Pub., RTO-MP-17, AC/323(AVT)TP/7, March 1999, pp. 1-1 –1/1-6.
20. Tschirne, K.U. and Holzbecher, W., “Cost-effectiveness of Modern Lifing Concepts”, RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998, NATO Pub., RTO-MP-17, AC/323(AVT)TP/7, March 1999, pp. 2–1/2-9.
21. Au, P., Beres, W., Koul, A.K. and Immarigeon, J-P., “A Review of the Qualification Methodology for Turbofan, Turboshaft and Turbojet Engines for Military Applications,” NRC LTR-ST-2030, October 1995.
22. Xue, L., Wu, X.J., Koul, A.K., Au, P. and Immarigeon, J-P., “Engine Repair Structural Integrity Program”’ NRC Internal Report, 1995.
23. Thomas, W. and Junkin, B., “International Acceptance of Commercial Repair Approval”, RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998, NATO Pub., RTO-MP-17, AC/323(AVT)TP/7, March 1999, pp. 3–1/3-5.

24. Stoute, P., Manente, D. and Immarigeon, J-P., “Evaluation and Qualification of Diffusion Braze Repair Techniques for Superalloy Gas Turbine Components”, Proc. Symposium on Industrial Applications of Gas Turbines, Banff, AL, Canada, Sept. 1991; also NRC LTR-ST 1839, Sept. 1991.
25. Ellison K.A., Lowden P. and Liburdi J., “Powder Metallurgy Repair of Turbine Components”, ASME 92-GT-312, June 1992.
26. Maccagno T., Koul A.K., Immarigeon J-P., Cutler L. and L'Esperance G., “Rejuvenation of Alloy 713 C Turbine Blades”, Metall. Trans., Vol.21A, 1990, pp. 3115-3125.
27. Koul A.K., Immarigeon J-P., Castillo R., Lowden P. and J. Liburdi, “Rejuvenation of Service Exposed IN 738 Turbine Blades”, Superalloys 88 Book, Eds. Reichman S. et al., Met. Soc. of AIME, 1988, pp 755-764
28. Domas, P.A., Fracture Mechanics Evaluation of Weld Repaired Seal Teeth for Life Extension of Aircraft Gas Turbine Engine Components”, RTO-AVTP Workshop on Qualification of Life Extension Schemes for Engine Components, Corfu, Greece, October 1998, NATO Pub., RTO-MP-17, AC/323(AVT)TP/7, March 1999, pp. 16–1/16-8.
29. Au, P., Immarigeon, J-P. and Koul, A.K., “Test Methods for Evaluating the Mechanical Properties and Durability of Gas Turbine Materials”, Proc. Int. Conf. on Mechanical testing of Materials and Structures, The Hong Kong University of Technology, 24-26 Oct. 1998, pp. 66-85.
30. Elder J.E., Thamburaj R. and Patnaik P., “Braze Repair of MA754 Aero Gas Turbine Engine Nozzles”, ASME 89-GT-235, June 1989.
31. Koul A.K., Wallace W. and Thamburaj R., “Problems and Possibilities of Life Extension in Turbine Engine Components”, AGARD Conference Proceedings, PEP Panel, Liesse, Holland, May 1984, AGARD-CP-368, 10-1 to 10-32, 1984.
32. Koul, A.K., Beres, W., Cheng, S.M. He, R. and Wu, X.J., Predictions of Safe Inspection Intervals for the Nene X Turbine Disc”, NRC LTR-ST-2142, July 1998.
33. Pishva M.R., Koul A.K., Bellinger N.C. and Terada T., “Service Induced Damage in Turbine Discs and Its Influence on Damage Tolerance Based Life Prediction”, CASI Journal, Vol. 35, No. 1, March 1989, pp 4-11.
34. Koul, A.K., Fahr, A., Gould, G. and Bellinger, N.C., “Importance of Sensitivity and Reliability of NDI Techniques on Damage Tolerance Based Life Prdiction of Turbine Discs”, in NATO AGARD Report No. 768, 1988, pp. 4-1/4-22
35. Fahr, A., Forsyth, D., Bullock, M. and Wallace, W., POD Assessment of NDI Procedures Using a Round Robin Test”, NATO AGARD Report 809, AGARD-R-809, January 1995.
36. Koul, A.K., Bellinger N.C. and Fahr A., “'Damage Tolerance Based Life Prediction of Turbine Discs - A DFM Approach”, Int. J. of Fatigue, Vol.12, No.5, 1990, pp. 379-387.
37. Koul, A.K, Bellinger., N.C. and Gould G., “'Damage Tolerance Based Life Prediction of Turbine Discs - A PFM Approach”, Int. J. of Fatigue, Vol.12, No.5, 1990, pp. 388-396.
38. Beres, W., and Koul, A.K., “Damage Tolerance Assessment of Aging Nene X Turbine Disc”, NATO-RTO-AVT Symposium on Aging Mechanisms and Control (Part B – Monitoring and Management of Gas Turbine Fleets for Extended Life and Reduced Costs), AVT Meeting, Manchester, UK, 8-11 October 2001.

Assessment of the Efficacy of Polish Air Force Engines for Life Extension vis-à-vis Technology and Practices Prevalent in other NATO Countries

Col. Dr. Eng. Ryszard Szczepanik
Air Force Institute of Technology
Instytut Techniczny Wojsk Lotniczych
ul. Ksiecia Janusza 46
P.O. Box 60
00-961 Warsaw 42
Poland

Dr M. Witoś
Air Force Institute of Technology
str. Ksiecia Boleslawa 6
mail box 60
01-494 Warsaw
Poland

ABSTRACT

The goal of the paper is to share PAF former and present experiences about dealing with ageing engine structures and strategies for their repair versus technology and practices prevalent in other NATO countries. Some issues of safe operation of ageing aeronautical products have been presented on the grounds of the engine-operation model. Special emphasis has been put on some aspects of both: initial tolerances in components engineering and stochastic loads which occur in the course of engine performance affect the rate of consuming the operational "fit-for-use" reserve of (sub) assemblies. The above-mentioned issues have been approached using predefined functions which represent "life" curves for individual engine components and the engine as a whole.

1. Introduction

The end of "cold war" and shrinking military budgets have generated the need to use the engines far beyond the previously determined life limits. There are different approaches to ensure that as much life as possible is extracted from life limited engine parts. Considering those approaches we have to remember that we don't have to much knowledge about the design details of our engine which we have on our inventory list. During previous decades we used to get all needed spare parts for our engine from previous Soviet Union engine factories which helped us without difficulties to keep our engines fit for use. There was no room for any cost effective considerations. All our jet engines of Russian origin were design to fly and maintain basing on flying hours limit. Each type of engine had strictly limited service life. After reaching this number of flight hours the engine had to be sent to overhaul base usually located in Soviet Union. It was very simple algorithm to follow.

At present we started to organise the scientific and technical preparations for implementing at least some of the western countries approaches, lifing philosophies to our engine remembering all environmental constraints which we face. The goal of the whole work is their life extension.

Of course we can not directly implement the "Safe Life Design" approach nor "Damage Tolerance Design" approach.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

After careful studying some AGARD and the RTO/AGARD reports devoted to subject: Recommended Practices for Monitoring Gas Turbine Engine Life Consumption a well as Working Group AVT-046 Workshop papers held in November 2000 in Warsaw devoted to subject: Exchange Information About Experiences Relating to Ageing Military Aircraft Fleets with the Three New Member Nations of NATO we started to analyse the whole issue. Our efforts are directed to create step by step Engine Life Management Plan. We of course have limited time 2 years maximum to clarify the whole ELMP plan in regard to Engine Life Extension. Of course first of all we have to use the existing in PAF methodologies which are parallel or idententical with western countries approach. So first of all we have in our Air Force the data bank “San” system which enables from many years identification of causes of changes to reliability, safety, and quality of the processes of aircraft operation on the one hand, and on the other hand – determination of activities to improve the above-mentioned features.

From other point we can use the digital data recorders which reflects the Mission Analysis. Another parallel approach in regard to Engine Life Extension Program is to use fleet leaders to detect possible damage mechanism. Of course the conversion factor between EFH and LCF cycles is still unknown. The inspection frequency is based on an EFH limit. Life limited parts are scrapped once the life limit has been reached.

2. Brief Characterisation of the “San” System.

The San system has been intended to many-sided analyse and evaluate the aircraft’s operational-phase processes. All types and versions of aircraft operated by the air force can be covered with analyses. Both individual aircraft (assemblies, components thereof) and freely composed sets of aircraft (Fig. 1) can be given consideration. The system supports the management of the operational phases of various products of aeronautical engineering.

Plentiful needs proved decisive in shaping the system and defining the scope of information to be captured. They result from:

- the problems of current and long-time operation of aircraft (the aircraft operated by the air force were flown and maintained according to the service-life strategy, and to a high degree were deprived of any care of both the Polish and the Russian manufacturers; operators had to secure the highest level of reliability and safety of operated products in their own capacity; therefore, detailed information on each failure/damage to an aircraft had to be included in the system),
- the problems arising as the fleet of military aircraft was ageing,
- the need to utilise the service lives of individual aircraft still left (i.e. to extend the by-hours-defined and calendar-based service lives).

The assessment of the reliability and quality levels of the processes of operating aircraft is usually executed by means of analysing the assessment characteristics and rates necessary to rationalise and actually manage these processes.


DIVISION FOR SAFETY & RELIABILITY OF AERONAUTICAL SYSTEMS
THE SAN SYSTEM ...
... enables versatile analysis of the operational phase of aircraft of all types and versions operated in the air force, of both individual items and sets thereof.

Fig. 1.

The most fundamental characteristics comprise:

- the assessment of the system from the point of view of failure delectability and effectiveness to prevent them,
- the assessment of the aircraft technical availability,
- the assessment of flight safety from the technical point of view,
- substantial support of the Air Accident Investigation Board's activities,
- estimation of real service lives of aircraft and assistance to the service-life managing processes,
- forecasting some selected operational characteristics and rates, and those of safety.

Detailed objectives of the system include:

- to collect and store information in local and central data banks,
- to process the collected information according to the computational algorithms and procedures, as well as some selected probabilistic measures, estimators and statistical processing methods,
- to present – in some suitable form – a packet of characteristics and rates of the operational process to facilitate a set of rational principles of the process management to be laid, and right decisions to be made, concerning:
- the operational-phase strategies,
- frequency and scope of maintenance and the health/ maintenance status inspections,
- the fault-location processes,
- the routine maintenance and the pre-planned repairs,
- the emergency repair practice,
- weak points in aircraft structures to improve them,
- overhaul frequency and upgrade extent.

The SAN offer means of data collecting and direct loading into the computer (still at air bases which operate aircraft of interest) has to a high degree replaced techniques of `booking` the records and operating data (in use util now) and paper document circulation between individual units and command-and-control centres.

What has been provided within the system, includes:

- gathering detailed information on the course of operational phase of each aircraft,
- quick access to the resources of local and central data banks,
- good readability of the system-generated data; the effects of the processing procedures are displayed in the form of adequate reports which can be further on either printed or illustrated graphically, according to the needs,

- easiness and simplicity in using the system; there is no special training to operators required – the software has been developed in such a way as to lead the operator through (a system of directly displayed prompts, messages, descriptions has been used,
- wide spectrum of means of software protection against loading any incorrect data by the operator,
- system protection against unauthorised access,
- network communications, or that via magnetic carriers, between individual system's modules, according to hardware and financial capabilities of individual users,
- capability of selective acquisition of different, according to the module's assignment, information on individual aircraft or any set thereof.

The system has been developed with regard to the needs of, on the one hand, immediate aircraft users, decision-makers at every organisational level of the operational-phase-managing and maintenance systems, aircraft manufacturers, and on the other hand – to the needs resulting from research practice.

The SAN system has already been implemented into the Air Force of the Armed Forces of the Republic of Poland (LZS RP). It is founded on the relational database ORACLE 7/8 with more than 100 dictionary tables and tables with variable data. Applications have been developed with the ORACLE tools (i.e. OracleDeveloper/2000 and the Oracle Power Objects) and are operative in the MS Windows 98 or Windows environments.

3. Methodology and Experience of Operating the Ageing Aircraft Engines in PAF.

Having in view very little knowledge about the basic design criteria, as well as lack of flight test data on loads and stress spectra, lack of exact materials properties and data standards, lack of material crack-growth rate data, lack of knowledge of corrosion protection system, service experience with fatigue cracking and corrosion it makes us to create special methodology to operate safely and cost effectively our ageing turbojet engines without manufacturer support. The basic points of our approach to such task i.e. to keep our engine fleet fit to fly with high level of safety as well as cost effectively can be listed as follows:

3.1. Establishing good and reliable statistic data bank system to Engine Life Management Plan and Reliability Centred Maintenance which includes:

- engine description,
- engine life management concept
- design parameters,
- design life limits,
- inspections,
- depot maintenance,

- data tracking system,
- engine parameters monitoring,
- parts life tracking,
- documentation issues.

3.2. Conduct Mission Analysis based on installed digital recorders

The assumption is to conduct Accelerated Mission Testing on some aircraft which are called leaders. Usually such accelerated consumption of engine, aircraft life can be observed in special acrobatic teams.

3.3. Start with Analytical Condition Inspection, systematic disassembly and inspection of a representative engine sample to investigate such phenomena's like hidden defects, deteriorating condition, corrosion, fatigue, over stress, creep, to identify structurally significant, critical components whose failure could be expected to cause the engine shut down, in-flight break-up and or loss of the aircraft.

3.4. Put proper attention to engine trending and diagnostics.

We have to identify :

- mishap experiences and chronic problem areas.
- methodology to analyse service difficulty reports,
- special usage factors (for example: salt air environment),
- engine performance degradation.

3.5. Perform if needed so called "Reverse engineering" which can embrace also the Component Improvement Program.

This scope work usually embraces:

- some flight test data on loads and stress spectra,
- analysis/ test data on fatigue life,
- experience with fatigue cracking and corrosion.

4. Similarities and Discrepancies between Reliability so Called "Western" and "Eastern" Turbojet Engines and Their Components.

The best way to discuss these similarities and discrepancies is to use the engine diagnostic model.

The determined set of parameters which describe the engine life limit consumption together with their measuring technology can be expressed as pseudo-determined diagnostic model of the engine in the form of matrix equation

$$\mathrm{X}(t,\bar{r}) = \Phi[\mathrm{Y}(t)] + \eta_x(t,\bar{r}) \tag{1}$$

In this model, matrix operator Φ represents vector transformation of critical status of engine components into symptoms vector $\mathrm{X} = \mathrm{X}(t,\bar{r})$

Equation (1) is considered as a function of maintenance time t:

$$X = \Phi(Y) + \eta_x = \mu(t) + \eta(t) \tag{2}$$

If the symptoms X are properly chosen so equation (2) describe so called "life" curves for individual engine components and the engine as a whole. This "life" curves can represent the development of individual defects, damages. Such interpretation we can express by equation:

$$x_j\left(t, \bar{r}\right) = \sum_{i-1}^{n} \sum_{j=1}^{m} \left[\alpha_{ij} \mu_i(t) + \eta_j\left(t, \bar{r}\right)\right] \tag{3}$$

This model shows the representation of defect development about the **i**-th engine component $\{\mu_i(t), i = 1,2,...,n\}$ in "j" symptom values $\{j, j+1,2,...m\}$ measured in the point defined by measurement place vector $\bar{r}$ in period (0 ≤ t ≤ t $_{defect}$)

where: a_{ij} - importance coefficients about the **i**-th defect in **j**-th symptom.

$\mu_j\left(t, \bar{r}\right)$ - casual disturbance of **j**-th symptom

Having in view the stochastic loads of different structural engine components we can present $\mu_i(t)$ functions as standardised "life" curves $kz_i(t)$, defined as running capability of **i**-th engine component $N_i(t, n_{rot})$ related to entry, theoretical engine capability of component $N_t(t_0, n_{rot})$

$$kz_i(t) = \frac{N_i(t, n_{rot})}{N_t(t_0, n_{obr})}$$

Running capability of engine component is expressed by the set of these features which describe its ability to resist maintenance, flight loads spectrum.

The entry, theoretical capability of any engine individual component defines maintenance, flight loads which the component was calculated and design for with fatigue resistance preservation. $Kz_i(t)$ functions are monoton decreasing functions with "0" value when component fails.

Having in view:

– wide material components properties distribution resulted from technology processes
– loads differences during exploitation process

For the same type engine group we can present set of "life" curves of any particular engine component $\{kz_i(t), i = 1,2,..., N\}$ in so called "ribbonn" function with defined confidence (fig. 2) and normal distribution.

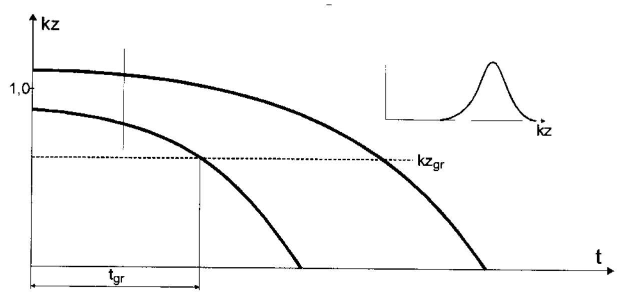


Fig. 2.

Summarising to extend the engine life we usually employ:

- the leaders accelerated usage concept,
- the reliable data coming from aircraft reliability data bank system called “SAN”,
- mission analysis,
- analytical condition inspection,
- engine trending and diagnostics,
- reverse engineering.

Those activities are similar to western proactive except reverse engineering which is we believe unknown area. Of course there are also some discrepancies. Among them we can list the inability to create and successfully perform Engine Structural Integrity Plan (Program) ENSIP and as consequence we can not implement. Damage Tolerance Philosophy which can not be put into our maintenance practise. All these methodologies are “passive” based on assumption that we are observing the whole population of the engines. So the rules refer to the whole population. In our every day practice we observe also very often the need to prolong the engine life of individual engine. In such cases there is the need to monitor the technical condition of strictly defined critical part or parts. Sometimes after full investigation the engine defect we implement on individual basis some so called active methodologies to prolong the real engine life of individual engine. Among such cases we can mention about creation of condition monitoring system based on non-interference discrete-phase compressor blade vibration measuring method. Design such reliable and useful monitoring system shows also how important is to make proper investigations of the engine failure and to take such proper countermeasures which can meet safe and cost effective flying requirement (Fig. 3 and 4).

5. Conclusions.

Ageing Engine Life Extension program in PAF has to be based on investment in such practical tools like:

- Diagnostics and monitoring systems,
- Non-destructive inspection,
- Reverse engineering,
- Statistical data bank,
- Mission analysis,
- Analytical condition inspection,
- Component improvement program.

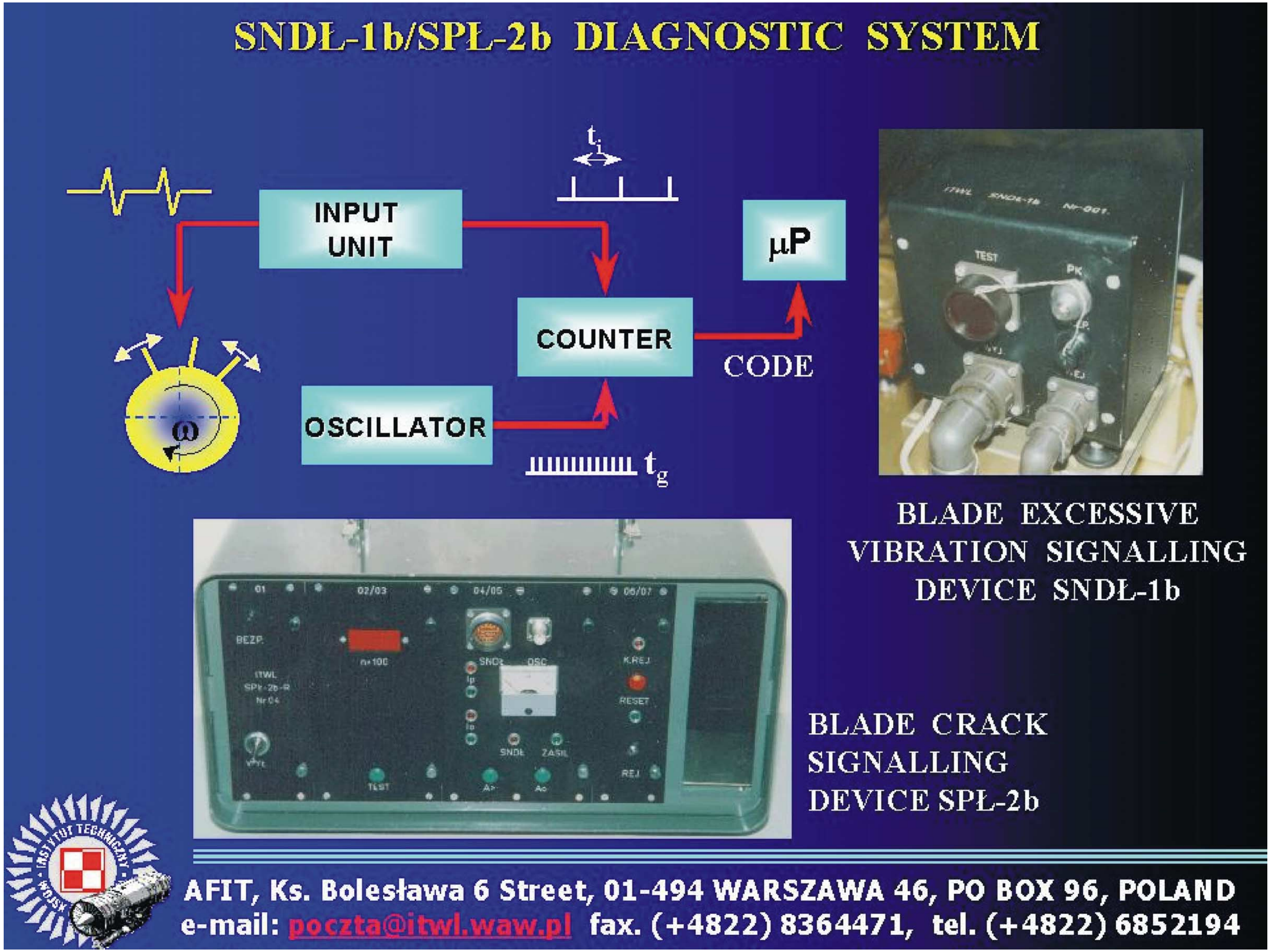

SNDŁ-1b/SPŁ-2b DIAGNOSTIC SYSTEM
INPUT UNIT
COUNTER
OSCILLATOR
μP
CODE
t_i
t_g
ω
BLADE EXCESSIVE VIBRATION SIGNALLING DEVICE SNDŁ-1b
BLADE CRACK SIGNALLING DEVICE SPŁ-2b
AFIT, Ks. Bolesława 6 Street, 01-494 WARSZAWA 46, PO BOX 96, POLAND
e-mail: poczta@itwl.waw.pl fax. (+4822) 8364471, tel. (+4822) 6852194

Fig. 3.

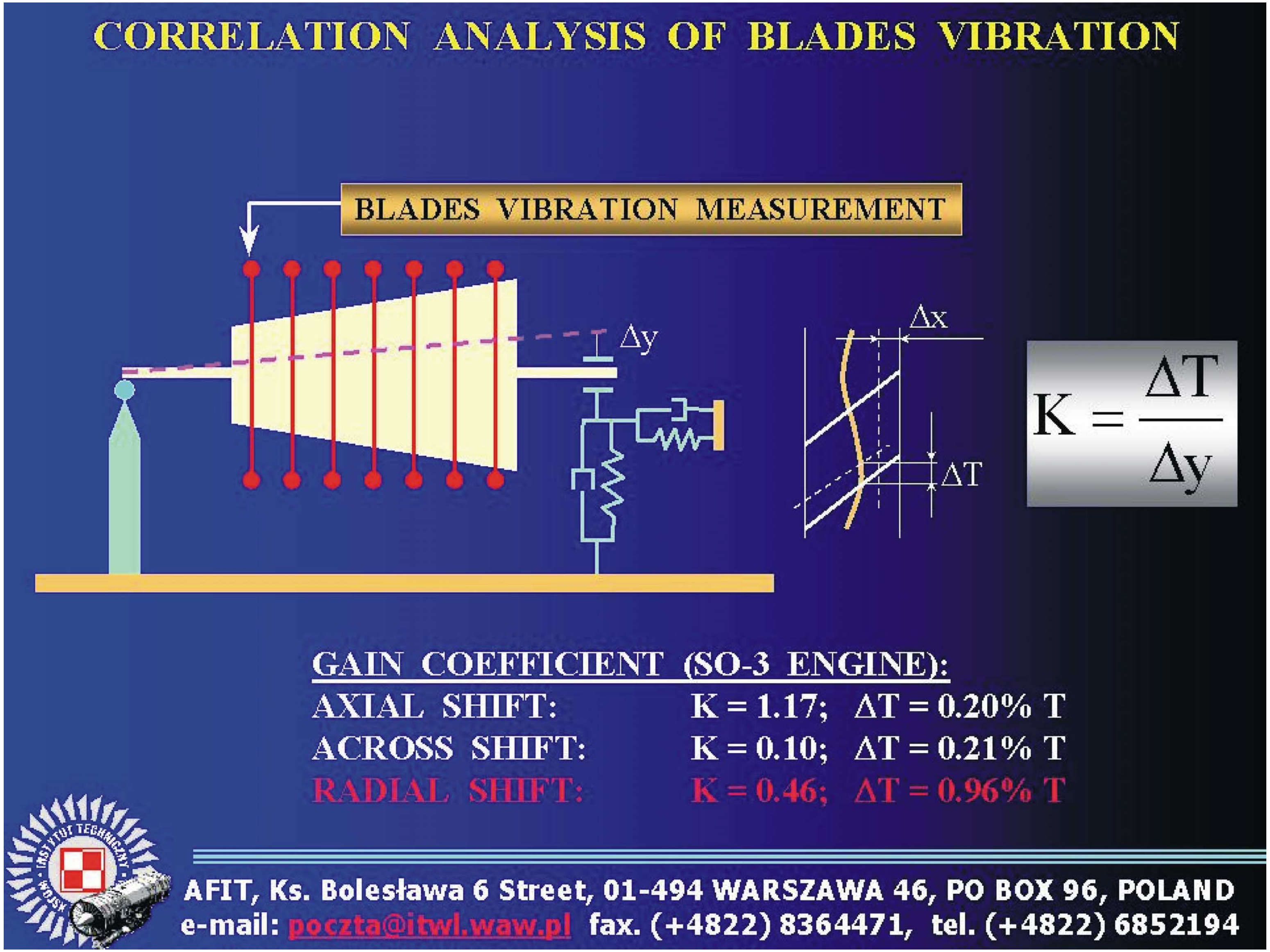

CORRELATION ANALYSIS OF BLADES VIBRATION
BLADES VIBRATION MEASUREMENT
Δy
Δx
ΔT
K = ΔT / Δy
GAIN COEFFICIENT (SO-3 ENGINE):
AXIAL SHIFT: K = 1.17; ΔT = 0.20% T
ACROSS SHIFT: K = 0.10; ΔT = 0.21% T
RADIAL SHIFT: K = 0.46; ΔT = 0.96% T
AFIT, Ks. Bolesława 6 Street, 01-494 WARSZAWA 46, PO BOX 96, POLAND
e-mail: poczta@itwl.waw.pl fax. (+4822) 8364471, tel. (+4822) 6852194

Fig. 4.

Engine Life Extension through the Use of Structural Assessment, Non-Destructive Inspection, and Material Characterization

Sharon Vukelich
Lead, Advanced Engine Projects IPT
USAF, ASC/LPJ
2145 Monahan Way
Wright-Patterson AFB, OH
45433-7017, USA

Abstract

For over twenty years, the USAF has had a life extension program for major rotating hardware, such as rotors, seals, and shafts. This program was called Retirement-For-Cause (RFC). Damage tolerance philosophies are essential to this lifing technique. Damage tolerance has been applied to all new engine programs as part of the Engine structural Integrity Program (ENSIP) within the USAF since 1984. An essential element of ENSIP and RFC is non-destructive evaluation (NDE). Not only is there a need for NDE equipment, but also the quantification of the NDE system/process. The RFC program provided state of the art NDE equipment and software to enable complex inspections. Due to these advancements low cycle fatigue no longer needs to be the governing factor in the retirement of engine hardware from service when you have a fracture control program. The process of fracture control or damage tolerance as it is also called governs the selection of material as well as the design and life management of the engine. Recent USAF experience clearly demonstrates that the damage tolerance philosophy has had a positive effect on safety within the design life and has shown with our limited experience in RFC that safety is maintained and it is cost effective when assessed on a life cycle cost basis. The initial RFC program had some problems and it is imperative for any future programs that we don't repeat those same mistakes. In order to make life extension of hardware more robust and more cost effective, a new program has been developed called the Engine Rotor Life Extension (ERLE) program. This paper will discuss the original RFC program and its shortcomings as well as our future needs.

Introduction

The problems with rising costs today are worse than they were 20 years ago. We were just anticipating having a few engines reaching their design life. Today the majority of the fleet are nearing retirement based on LCF, yet we still have a desire to lower our operational costs. The answer to this dilemma is to extend the life of our major rotating hardware. This is where the majority of the cost of the engine resides. This need is the same today as it was for the original RFC program. As always, in order to accomplish this task, it is necessary to use innovative lifing techniques, which can reduce the conservatism achieved through our analysis, while maintaining equivalent or even lower risk. In order to understand the magnitude of this problem we need to view Figure 1., which shows the breakdown by engine model of the replacement cost for rotors to be inducted over the next 10 years.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

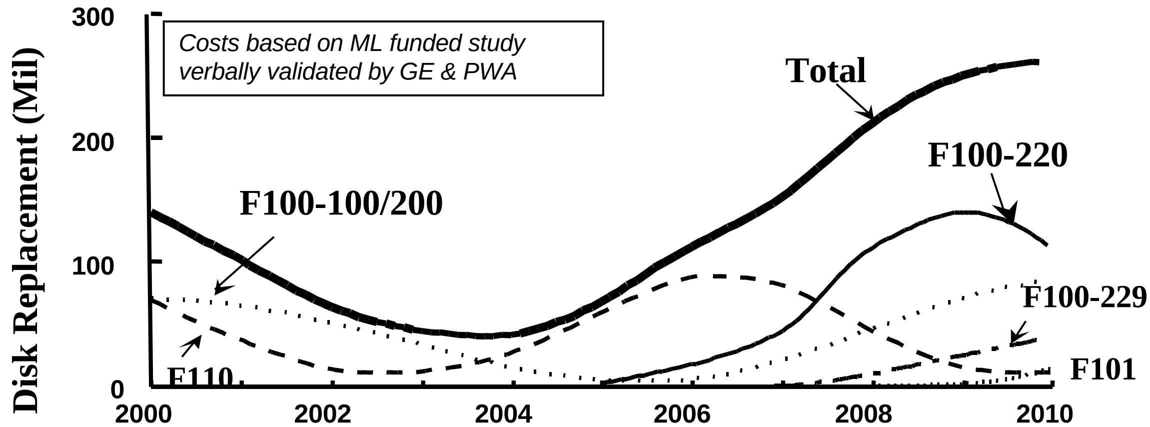


Figure 1. Future Disk Replacements

Figure 2. shows our anticipated savings based on implementing current technology and also if we have added technology investment. These replacement disk savings are based on the numbers of disk generations in Figure 1. As can be seen the savings are well worth the time and effort to achieve these goals. As also can be seen it is imperative to go after technology investment. We can more than double our savings by adding a small investment now.

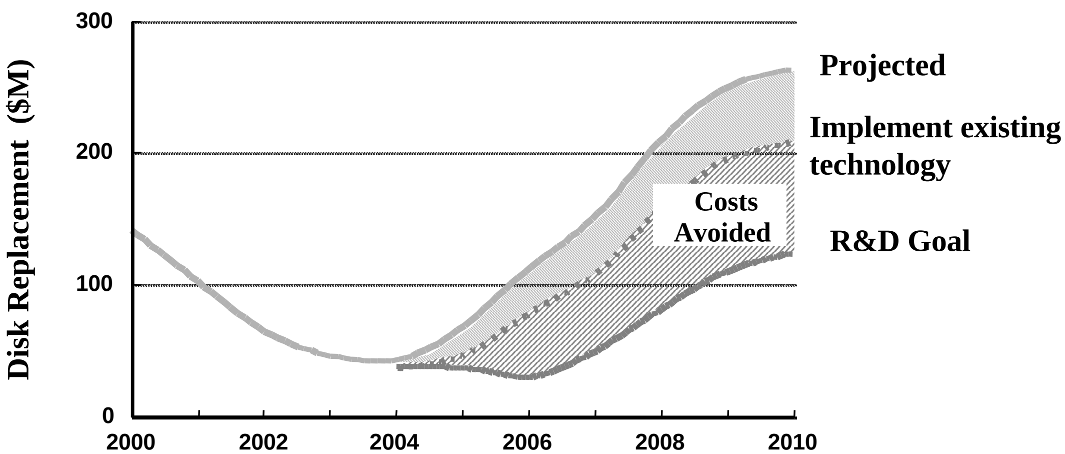


Figure 2. RFC Projected Savings

USAF Lifing Methodology

To help understand these savings, first I believe it is necessary to understand the standard USAF lifing methodology. The program is referred to as the Engine Structural Integrity Program (ENSIP). This program is more than just lifing, but for purposes of this paper we will only talk about the lifing and inspection areas of the program. The retirement life for hardware is the -3σ or equivalent life to crack initiation (defined as a 1/32 of an inch crack). Inspection intervals are based on ½ the average crack growth life from an inspectable flaw size to dysfunction. This inspection interval is termed the safety inspection. No credits are given for shot peening or other surface treatments, which can retard crack initiation, unless a full material characterization program shows a change to the -3σ material curves. This has seldom been done in the case of shot peening due to the variability in the processing. This results in a substantial increase in mean material properties, but an insignificant change in the minimum properties. As one can see, by using minimum material properties and allowing no credits this lifing method is conservative in nature. Unfortunately we can no longer afford to sustain this conservatism. Aircraft fleets are aging and yet, we need to keep that hardware usable and maintain safety of the fleet. The purchase of replacement engines has not been accounted for in our maintenance planning.

RFC was a program developed in the early 1980's. The theory behind this program was based on damage tolerance. We would continue to inspect hardware until we found a crack with our NDE techniques. A risk management program called the probabilistic life assessment program (PLAP) was developed (later changed to PLAT). This program would enable the assessment of the risk of achieving the next interval based on the FEM analysis, the inspection areas, and the inspection reliability. It took many years to develop the inspection equipment necessary to find the small flaws at the necessary reliabilities. The F100-PW-100/200/220 engines have used RFC, but not as it was fully intended and thus the predicted savings have not been achieved. The reasons for this are many. First and foremost was a change in flight missions for our aircraft. This change in mission reduced the LCF and fracture life of the hardware. These shortened lives prohibited the use of RFC for many of the anticipated hardware due to increased (unacceptable) risk. As with any lifing technique it is essential to have FEM analysis based on real updated mission usage. Thus, periodically there must be provisions for analysis updates due to mission usage changes. Second, it is imperative that maintainers understand that RFC inspections are different than normal safety inspections. Once you have reached the minimum LCF life, there will be more areas that need inspection to go on beyond this quoted life in order to maintain the same risk as in the first lifetime. This is not limited to surface inspections. When a rotor is first manufactured, it is sonic inspected for imbedded defects which would prohibit the rotor from making its' minimum life. This type of inspection (ultrasonic) would need to be done. However, we are no longer concerned with small void type defects, but with the cracks that would emanate from those defects. This requires sophisticated sonic equipment that can find the crack edges reliably. In the 1980s both Eddy Current (EC) and ultrasonic inspection systems were developed by Systems Research Laboratory for the USAF RFC program. These systems were put in place at the San Antonio - Air Logistics Center. The EC systems were used for safety limit inspections prior to true RFC use. Because of the misunderstanding about required inspections, the required analysis and inspection scan plan generation was not put in the funding cycle. Because of both the mission change and the delay in analysis required to extend the life with RFC, only limited hardware actually was extended, and then for only one interval. Thirdly, material characterization must be full and complete even on old/used material. This is necessary for crack growth data as well as stress rupture, and other strength characteristics. This also was not done and hence the lack of ability to extend the life intervals.

Revitalization of the RFC Methodology

In an effort to revitalize RFC. A new program called Engine Rotor Life Extension (ERLE) is being developed. It is intended to correct the shortcomings of the old RFC program. Enhanced inspection equipment are being developed. This includes a replacement system for the RFC ultrasonic system that was dismantled due to non-use. This system is anticipated to also be able to detect cracks emanating from acceptable defects in the normal forging process. Figure 3 below shows the ENSIP approach as well as the RFC process.

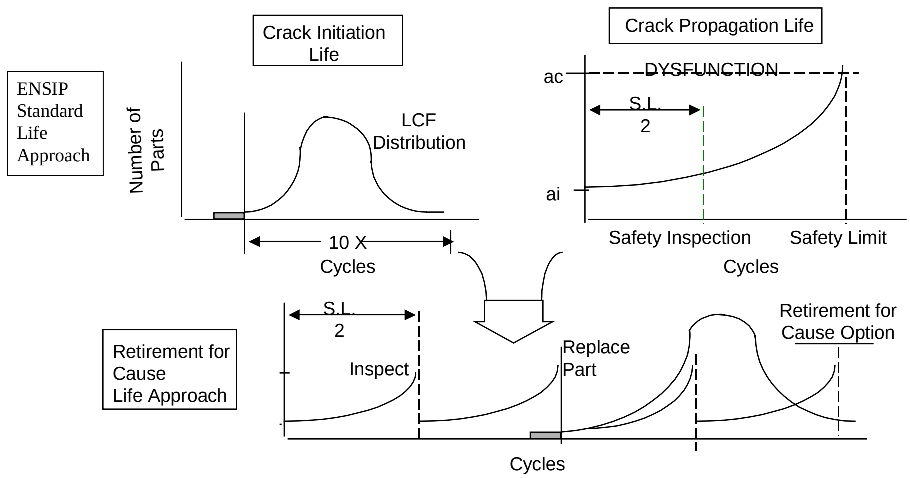


Figure 3. ENSIP Lifing and RFC

Both the RFC program and the ERLE program rely on the same theory. The USAF lifing methodology retires 999 good rotors when only one is bad. It is a simple matter to just keep the hardware in the fleet longer, but our customers and we, ourselves, have become accustomed to a certain level of risk. In order to keep that inherent risk level; we need to think about managing our inspection intervals differently. As more and more hardware is kept in the system the inherent risk goes up if nothing else is changed. However, it is possible to maintain the same risk by lowering the inspection interval. This requires a flexible tracking and Life Management system. Consider the following example: A high pressure turbine disk has a standard ENSIP LCF life of 9000 Total Accumulated Cycles (TACs) and an inspection interval based on fracture mechanics of 4500 TACs. The disk is inspected at half-life and returned to service. At 9000 TACs it would be retired under the standard system, but under the RFC program it would continue in service. However, to continue, one must look at all areas of the disk analytically. Suppose the bolt hole area had a fracture inspection interval of 10000 TACs. In its original life, it would not have needed to be inspected. Now for RFC we must develop an inspection technique and inspect that area. Suppose also that the imbedded defect life was exactly 9000 TACs. We now also need to re-ultrasonic inspect the disk, but looking for cracks not just the normal imbedded defects. As we keep doing this we find more and more areas that need to be inspected and the risk of having the original areas crack are becoming greater. Thus we need to alter the inspection interval to do this. The PLAT software program is a probabilistic life assessment tool designed to assess both risk and the areas needing inspection. The use of PLAT can help determine the useful life. It needs to be updated as well.

Even though we advocate RFC, the question becomes; “What is cause?” “Cause” can be based on inspection findings, unacceptable risk, or even economics. Consider the following example: A rotor stage has been extended 2 intervals. On inspection for a 3rd interval, 50 % of the rotors are rejected due to inspection findings. At this point, if the cost of the inspections is equal to or greater than the cost of rotor replacement, it would be best to retire all the disks after the 2nd interval for economic reasons.

Other ways to take conservatism out of analysis is to allow credit for beneficial stress that limits the ability of the part to initiate a crack. Compressive residual stress imparted by peening and cold working processes has not been allowed due to variability in the process and the inability to reliably use non-destructive inspection to verify the beneficial compressive residual stress. A new x-ray diffraction technique has now made the inspection possible. This will assist in management of fleet assets and add a measure of safety by finding below average disks ***before*** they crack, thus avoiding field problems. This new technique for tracking turbine engine component life may offer improvements to the engine life management tracking process, thus providing the customer with a more affordable and reliable system. The improvements will result from several factors, including full utilization of engine disk design life, life management on a module or component basis, improved system safety, and reduced spare parts requirements. The new life tracking technique employs technology that relates engine cycles to increments of disk life. This new approach uses residual stress to quantify engine disk life and compares each disk with other disks that make up that family database by alloy. At each subsequent inspection interval, new residual stress measurements will be made and this accumulated historical data will provide a realistic determination of which disks can safely reach full life and which disks should be retired at some earlier interval.

This new residual stress measurement technology is not intended to replace other NDT techniques such as EC crack detection inspections. It is, however, recommended that this technology be used as an inspection to complement and augment the existing NDE inspection and focus attention on those parts that display low compressive surface residual stress being the most likely candidates to have cracks. Those disks with tensile residual stress in critical locations become candidates for immediate removal from service. The use of residual stress measurement would thus help to negate subsequent NDE misses. The addition of residual stress measurement to any Retirement for Cause program such as ERLE will aid in differentiating disks with remaining life from those whose life is exhausted.

However, much work is yet to be completed on actually tracking residual stress with cycles and time at operating conditions. It is imperative to treat this data as one would any material characterization. As the database is collected for all materials, it is anticipated that we will be able to determine a minimum value of compressive residual stress, which is necessary to make the next inspection interval. It is also hoped that we would be able to determine a value associated with abusive peening (too high of a compressive residual stress). These measures only tell about the surface condition and would complement other techniques such as EC and ultrasonic surface/near surface inspections.

A further effort is being conducted to determine if there is a relationship or rather a correlation to the subsurface Stress State of the hardware. In other words, do the internal areas of the rotors degrade in stress levels/capability in a corollary fashion to the surface areas? This will take much longer to determine and may yield nothing. In the foreseeable future, ultrasonic inspections for internal cracks are the only reliable method of determining the internal health of the rotors.

Short Term Extension through Risk Management

Another method of life extension that has been used in several instances is called total life. Total life is defined as: (-3σ) LCF + (-3σ) crack propagation. This extension methodology is only used for small controlled populations when it is necessary to use a risk management philosophy due to extenuating circumstances (i.e. lack of parts). The USAF doesn't recommend or use this practice for long term life extension, such as Retirement-for-Cause.

Summary

In summary, many existing technologies can be used to extend the life of rotating hardware. These extensions will result in millions of dollars worth of savings while maintaining risk at current levels established during the normal LCF life. Many of theses can be done today. We have use of damage tolerant methodologies. We have state of the art inspection systems. However, there are still some holes, such as a new inspection system to look for cracks in the bores and webs of disks. Even with these, I believe we need to take conservatism out of our designs while maintaining the present risk level. This means we need to optimize our designs. It is imperative to pursue future technology advancements to provide longer extensions at reduced risk levels. The highest cost in these technologies is not the technology, but the true validation of the technologies based on our failure history and present inherent risk levels.

References

1. Engine Structural Integrity Program (ENSIP), Military Handbook 1783A (USAF), 19 December 1997.

2. Cowie, William D., and Tiffany, Charles, "Progress on the ENSIP Approach to Improved Structural Integrity in Gas Turbine Engines/ An Overview, ASME publication 78-WA/GT-13

3. Cassidy, Frances, Sammons, Jack, and Vukelich, Sharon, "Engine Life Maturation Process", AIAA 89-2464,

4. Rasmussen, Bruce, "Engine Rotor Life Extension (ERLE)", Aeromat Conference Briefing, 11-14 Jun 2001

Wire Integrity Field Survey of USAF Legacy Aircraft

Joseph S. Kuzniar, George A. Slenski
Air Force Research Laboratory
AFRL/MLSA
2179 Twelfth Street
Wright-Patterson AFB, OH
45433-7718, USA

Introduction

Wiring is so fundamental to current aerospace equipment that we often forget it is a system unto itself. The aging of a wire system can result in loss of critical functions in equipment powered by the system or in loss of critical information regarding the operation of certain parts of the equipment. Either result can lead to catastrophic equipment failure or to smoke and fire. Consequently, the safety of aerospace wire systems is an issue of major importance to us all.

Background

Aging of wire systems was first identified as an issue of national concern for the United States as a result of the White House Commission on Aviation Safety and Security. Efforts to address wiring issues in aviation revealed that aging wiring affects all electrical and information systems. With public attention being drawn to commercial aircraft wiring issues, a White House Interagency Working Group (IWG) was formed in June 2000 to examine policy, programs, investment priorities, and direction across the Executive Branch. Findings by the White House IWG were published in a report titled **Review of Federal Programs for Wire System Safety**, issued in November 2000 (See Figure 1). This report can be found at: http://scitech.dot.gov/polplan/wirerpt/index.html The report highlights the importance of wiring systems and concern for aging of these systems.

Also in 2000, the Air Force Research Laboratory (AFRL) conducted a field survey of the 3 United States Air Force (USAF) Air Logistics Centers (ALCs) and two F-16 field units. Results from the field survey were incorporated in part into the White House report.

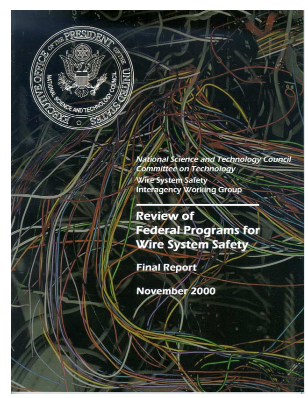


Figure 1: White House Report

Definitions

A wire/wiring system is defined as hardware which provides power, control, and information distribution. Wiring is just one part of the electrical interconnection system. Wiring, connectors, relays, switches, circuit breakers, power distribution panels, and generators are the components that makeup the wire/wiring system (See Figure 2).

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

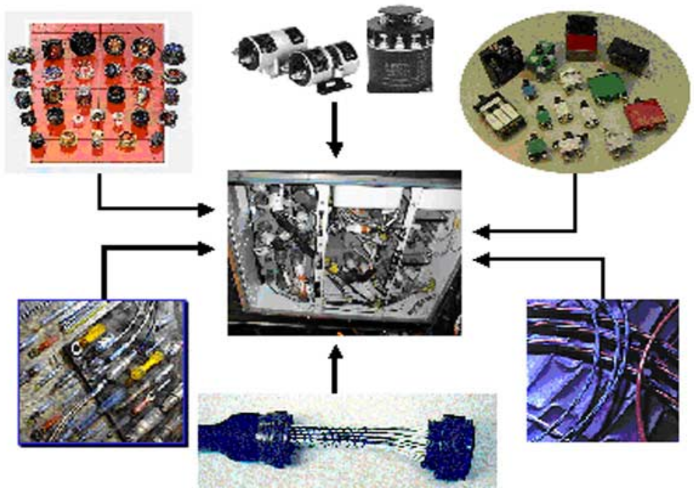


Figure 2: A wire/wiring system has multiple components.

The Issues

Wiring is the infrastructure linking electrical, electro-mechanical, electronic, and information systems in air vehicles. Wiring has emerged as vital in the control and safety of these systems, due to their ever increasing complexity:

- Increased emphasis on avionics: Fly-by-wire systems and glass cockpit
- Approximately 150 miles of wiring and over 500 connectors in a bomber or transport.
- Continuous exposure to harsh environments and mechanical stresses.
- Multiple wiring types used and use of materials with imperfect properties: Best commercial practices.
- Continuous wiring upgrades and modifications.

Aging Mechanisms

All wiring systems are subject to aging during their normal service life. Aging is the progressive deterioration of physical properties and performance of wiring systems with the passage of time, as well as with handling and usage. Stresses are often induced by the operational environment, as well as installation and maintenance practices. Wiring failures often appear as broken conductors and damaged insulation, which can disrupt electrical signals and/or lead to arcing. This aging is caused by the accumulated damage from exposure to the following stresses (See Figure 3):

- Chemical, including corrosion and moisture intrusion.
- Thermal, including fluctuations in temperature, which cause embrittlement… Radiological can also cause embrittlement.
- Electrical discharges, such as surges or arcs, and partial discharge transients.
- Mechanical, including vibration, chafing, overload, and fatigue.

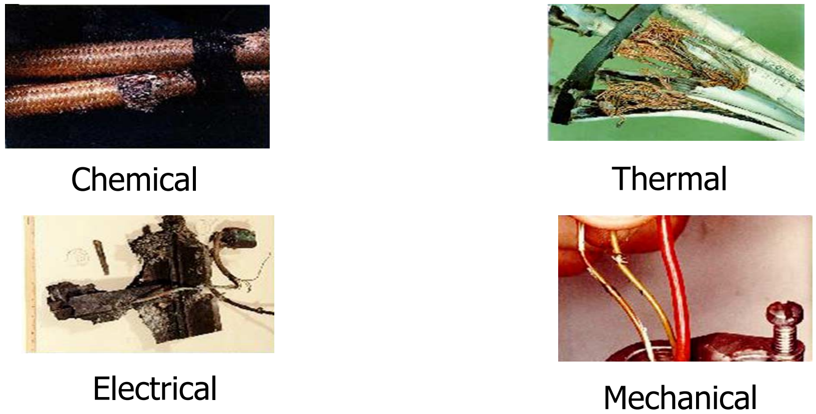


Figure 3: Wire failure mechanisms

Lack of Data

Military aircraft wiring integrity has gained increased visibility in the past two years as a significant aging aircraft issue. Problems in the wiring system can have a profound effect on flight control, avionics, and information systems critical to flight safety, as well as mission effectiveness. Similar to the commercial sector, the military does not typically report wiring maintenance and repair actions. The magnitude and pervasiveness of wire integrity issues are not well documented since the various parts of the wiring system are treated as low cost, throwaway, commodity items. The lack of an adequate data base poses a challenge in trying to determine just how extensive the aging wiring problem is in the current USAF fleet, as well as its actual impact on aircraft availability. Two available data sources are USAF and Navy mishap data (shown in Figures 4 and 5).

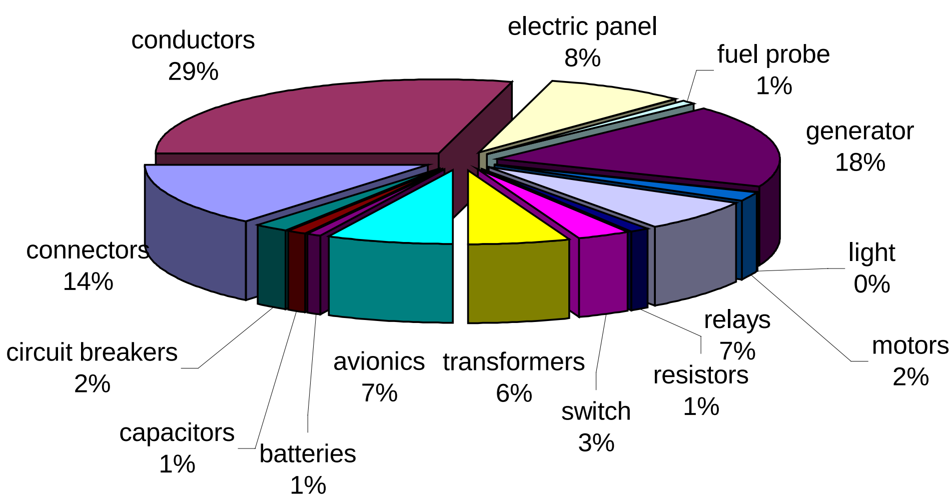


Figure 4: Electrical components contributing to USAF aircraft mishaps.
(Based on U.S. Air Force Safety Center Electronics Failure Data for 1989-1999.)

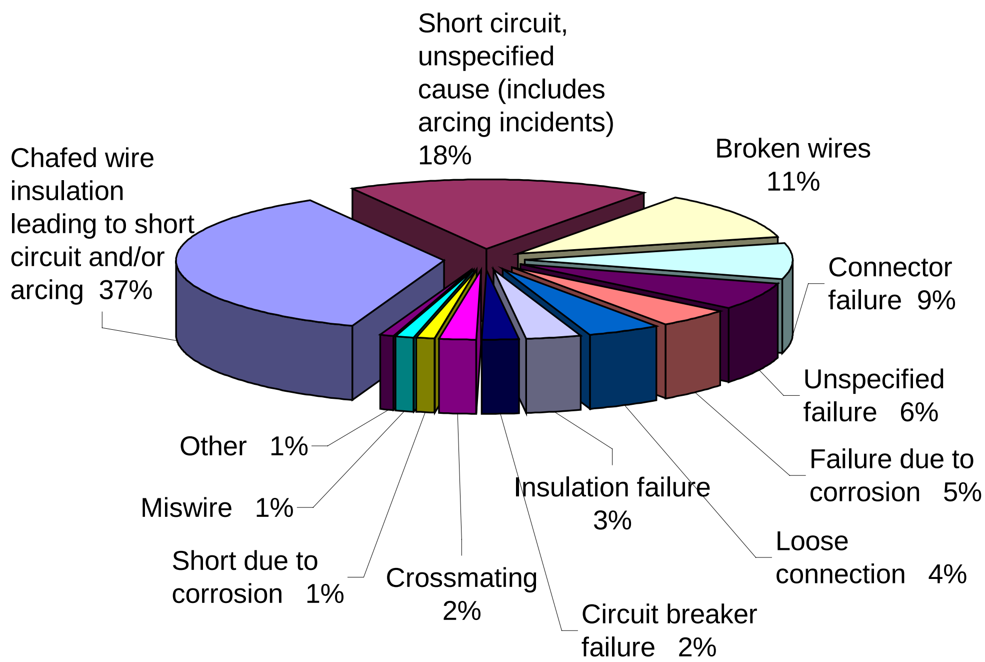


Figure 5: Typical wire system failure modes for aircraft.
(Based on U.S. Navy Safety Center Hazardous Incident Data for 1980-1999.)

USAF Legacy Aircraft

As the Air Force's aircraft fleet continues to age and accumulate increased operating hours, aging effects may result in the electrical systems. Lack of an adequate wiring system data base has to date precluded the determination of trends, aging characterization, and the degree various electrical systems are effected. Lacking a data base, AFRL conducted a seven month wire integrity survey (29 February through 30 September 2000) at all three ALCs and two USAF field units (Luke AFB, AZ and Springfield Air National Guard Base, OH). These field surveys included multi-site reviews of test methods, techniques, and types of wiring faults found at the ALCs and field units. The site surveys served to document wiring and maintenance issues at each location, as well as determine needs.

Survey Findings

On-site inspections of various USAF aircraft showed that some deterioration of wiring components (i.e., wire, wire bundles, connectors, grounds, clamps and shielding) is occurring in aircraft. During site surveys on fighter aircraft, most wiring problem areas identified were in the wing and flaps, fuel tanks, wheel wells and avionics bays. Further, it was found that almost two thirds of the failures occurred close to a connector or termination point. Many of the problem areas contained high wiring density and the limited space associated with these areas made them susceptible to chafing.

Most large transport type aircraft wiring faults occurred outside pressurized compartments where faults resulted from exposure to moisture, chemical contamination, heat, maintenance activities and indirect damage. These high wire system maintenance areas included landing gear and wheel well areas, leading and trailing edge flap areas, fuel cell wiring, flight control column wiring and other areas where wires were continually flexed. Wiring density was lower than in fighter aircraft and wire harness routing space generally was not an issue; however, long distances between termination points presented challenges in spatially locating faults. Maintenance personnel readily pointed out causes of wiring degradation include vibration, moisture, chemical contamination, heat, indirect damage and induced damage from maintenance actions (metal shavings, etc.). Wiring failures were most frequent in areas the wiring is directly exposed to the outside environment (wheel wells and leading edges), moving parts, and high maintenance areas (radar's, communication equipment and the cockpit) The majority of maintainers interviewed stressed the importance of checking wire clamping, connectors, terminations, conduits and grounding points. Failures tended to be from "wear and tear", chafing, environmental conditions, or from handling during modifications and upgrades. It was noted that most wire problems are not documented in recoverable database. The majority of wiring is also not tested during programmed depot maintenance periods. Only wiring that is modified or replaced is tested.

In discussions with maintenance technicians, it was found that most wiring problems are found through trouble-shooting not through visual inspection. Trouble-shooting is primarily accomplished by multi-meter. Maintainers expressed a high interest in the inspection, maintenance and repair of aircraft wiring. It was widely acknowledged that these areas must be given greater consideration as aircraft age. The routine inspection of wiring in a nonspecific nature was not found effective. Wiring problems often manifested themselves in the faulty operation of components that are controlled or powered by the wiring. Because of the high redundancy and conservative designs in these systems, for the most part a failure leads to a repair action and not a safety incident.

It was noted that wiring failures were often found only after an inability to resolve a system failure or during visual inspections. The field survey showed that wiring failures may have a significant impact on maintenance costs since:

- Maintenance tracking data bases do not show extent of problems.
- The evidence is primarily anecdotal.
- Damage to the wiring system is often due to maintenance actions, as well as exposure to moving parts and external environment.
- Material degradation was only apparent in older insulation materials (20+ years).

User Needs

After completion of the survey, a users' conference was held on 21 September 2000 at Wright-Patterson AFB, Ohio. The following near-term needs were identified and agreed to by the using community:

- Wiring codes and the establishment of a wiring system data base,
- Arc fault circuit protection,
- Additional wiring system inspection, troubleshooting, and test equipment,
- Additional repair tools, techniques, and materials.

With respect to inspection, troubleshooting, and test, as well as repair, the following equipment requirements were identified:

- The equipment must be simple, portable, compact, and easy to use with a minimum amount of setup time required.
- It must integrate with current test equipment and practices.
- It must be able to locate opens and shorts, as well as identify their location…with a strong desire to be able to identify and locate some intermittent and degraded conditions.

- The equipment should be capable of verifying the corrective action/repair.
- It should record and store the data/results.

Summary

AFRL is currently working with the Navy, Federal Aviation Administration, and the National Aeronautical and Space Administration to address the aforementioned needs, which are common to both commercial and military aviation. In summary, wiring systems can be managed with

- Emphasis on proper design, materials selection, installation, training, and maintenance practices,
- Collection and analysis of maintenance data,
- Understanding failure mechanisms,
- Use of surveillance programs,
- Treating wiring as an electrical interconnection system…not low cost, throw away, commodity,
- Use of proactive repair and replacement programs.

Bibliography

1. **Review of Federal Programs for Wire System Safety**, National Science and Technology Council, Committee on Technology, Wire System Safety Interagency Working Group, November 2000.
2. **Wiring Integrity Analysis of Air Force Weapon Systems**, GRC International, Inc., September 2000.

Wiring System Diagnostic Techniques for Legacy Aircraft

David H. Johnson
Materials Engineer for Electronic Materials Evaluation
Air Force Research Laboratory, AFRL/MLSA, 2179 12th St., WPAFB, OH 45433-7718, USA
Ph. 937.656.9163, Fax 937.656.4600, Email: David.Johnson2@wpafb.af.mil

Edward L. White
Materials Engineer for Electronic Materials Evaluation
Air Force Research Laboratory, AFRL/MLSA, 2179 12th St., WPAFB, OH 45433-7718, USA
Ph. 937.656.9162, Fax 937.656-9147, Email: Eddie.White@wpafb.af.mil

Joseph J. D'Angelo, Jr.
Deputy Director of Manufacturing Operations
GRC International Inc., 2940 Presidential Dr., Suite 390, Fairborn, OH 45324, USA
Ph. 937.429.7773, Fax 937.429.7769, Email: jdangelo@grci.com

Dwayne Dicks
Manager for Technical Services,
GRC International Inc., 2940 Presidential Dr., Suite 390, Fairborn, OH 45324, USA
Ph. 937.429.7773, Fax 937.429.7769, Email: ddicks@grci.com

Adam L. Decker
Senior Technical Services Specialist
GRC International Inc, 2940 Presidential Dr., Suite 390, Fairborn, OH 45324, USA
Ph. 937.429.7773, Fax 937.429.7769, Email: adecker@grci.com

Abstract

As aircraft continue to age, managing the overall wiring system is becoming an important issue. Over time, the accumulation of stresses from the operational environment, installation, and maintenance can induce wiring failures. In both design and maintenance, wiring is usually treated more as a commodity than a subsystem. A systematic process for managing wiring is only now just emerging. A major challenge is the development of wiring diagnostic equipment. The diagnosing and repairing of wiring failures can cause extensive downtimes for aircraft. Interconnection failures cannot be easily identified since most test equipment is designed to locate failures in avionics and not the connector or wiring. Additionally, wiring failures tend to be intermittent in nature and can take considerable time to isolate. Significant reduction in maintenance costs can occur by moving from unscheduled maintenance to a scheduled, preventative, or opportunistic maintenance philosophy based on the ability to isolate and repair degraded electrical systems. This paper will address the wiring system maintenance issues and concerns, field maintenance diagnostic requirements (needs), and compare available diagnostic tools in terms of utility, ease of use, and setup time. It will also address efforts currently being accomplished to modify/adapt commercial off the shelf testers to meet the requirements that will enable the United States Air Force to more effectively manage wiring as an aging subsystem.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Introduction

Aging wiring has become both a high interest and high cost item for the Air Force maintenance community. As the average age of the Air Force fleet continues to rise, wiring problems are becoming an increasing concern. Yet the term, aging wiring, is not well understood by the aerospace community. For the purposes of this paper, aged wiring is defined as wire exhibiting degraded performance due to accumulated damage from long-term exposure to chemical, thermal, electrical, and mechanical stresses. The operational environment, installation, and maintenance practices often induce these stresses. Diagnosing and repairing wiring faults can cause extensive downtimes for aircraft. In both design and maintenance, wiring is usually treated more as a commodity than a subsystem. Most troubleshooting systems, including aircraft built-in-test and automated testers for line replaceable units (LRUs), indicate failures occur in the black boxes. Interconnection failures cannot be identified. Additionally, wiring failures tend to be intermittent in nature and can take considerable time to isolate. Wire modifications and repairs in the field can also be difficult to verify since field evaluation systems are not effective. The time maintenance personnel spend diagnosing and repairing faulty wiring is becoming a leading cause of aircraft downtime. Current maintenance procedures do not adequately address wiring as a system. Visual inspection of individual wires in a bundle or connector is not practical because as wire ages it becomes stiff, and dismantling the bundle or connector may introduce collateral damage resulting in safety hazards. Without providing maintenance personnel improved diagnostic equipment, the costs associated with wiring failures will continue to increase. Air Force aircraft maintenance occurs in one of three levels: phase, field, and depot. Depot activities are focused towards maintenance modifications. Phase-level activities are focused towards scheduled maintenance inspections. Field-level maintenance activities are more reactive in nature with the objective to get the aircraft mission capable. There are no scheduled wiring inspections in any of these maintenance actions. Field-level maintenance is the area with the greatest need for an advanced testing system. Currently, a reactive approach is taken where the maintainers look at the wiring only after LRUs have been removed and replaced several times and the problem still exists. After performing a visual inspection with a flashlight and mirror, the most commonly used piece of equipment is the multimeter. This piece of equipment is preferred because it is easy to use, portable, and easy to interpret the results. Drawbacks to using the multimeter are that it is a time consuming process to try and isolate the fault pin-by-pin; its use requires two people to allow for connection at both ends; there is no data archiving or retrieving capability; and it is an extensive process to physically locate faults. During phase-level maintenance, the opportunity exists to perform a comprehensive inspection of wiring if needed. At this level, the aircraft is de-paneled at the beginning of the process which allows the maintainer greater access to the wiring. The aircraft is also parked inside a facility, allowing the use of any type of test equipment. Maintenance stands are available next to the aircraft, allowing the use of heavier equipment and decreasing the need to use ruggedized equipment. By collecting and archiving wiring data, incipient faults can be identified and corrected before they impact aircraft readiness in the field. Significant reduction in maintenance costs can occur only if the Air Force moves from unscheduled maintenance (fix it when it breaks) to a scheduled, preventative, or opportunistic maintenance philosophy based on the ability to isolate and repair serious degradation (integrity verification).

Background

The need for aircraft wire diagnostic equipment has become apparent through findings of recent mishap investigations, as well as, surveys of USAF maintenance personnel. The surveys identified a need to improve the efficiency of diagnosing, locating, and repairing damaged and degraded aircraft wiring, and a desire to track degradation trends. The investigation of the TWA Flight 800 fuel tank explosion, which caused the loss of the aircraft and 230 passengers and crew onboard, brought to light damaged and degraded wiring. The National Transportation Safety Board (NTSB) identified wire damage as part of the probable cause. Following this mishap, the Federal Aviation Administration (FAA) initiated a study of the condition of aged aircraft wiring under the guidance of the Air Transport Systems Rulemaking Advisory Committee (ATSRAC). The intrusive inspection phase of the ATSRAC study evaluated the condition of wiring on five different recently retired commercial aircraft models (A100, L1011, B747, DC-9, DC 10), each with over 20 years of service. The inspections involved three tasks: 1) detailed visual inspection with or without invasive follow-up, 2) nondestructive testing (NDT), and 3) laboratory analysis. Invasive follow-up meant spreading out the wire in wire bundles to allow evaluation of the condition of the interior wires and more complete evaluation of wires at the surface of bundles, sometimes with magnification.

In general, the visual inspections done during the intrusive phase of the ATSRAC study were more detailed than the general visual inspection procedures typically followed as a part of routine aircraft maintenance. Nondestructive testing to find insulation damage exposing the core conductor, high resistance, and conductor shorts and opens was done aboard the aircraft before wire bundle samples were removed for laboratory analysis. After the bundles arrived in the laboratory for detailed analysis of individual wires, they were again tested for insulation damage exposing the core conductor. It was expected that the very close, detailed inspections of the study would find most all defects in the sample bundles. The purpose of the NDT was to verify all defects had been identified during visual inspection, and to ensure the process of removing the bundle samples from the aircraft had not induced new damage. NDT done before the bundles were removed from the aircraft and follow-on laboratory procedures, however, found a significant number of defects that had gone undetected using these very rigorous visual inspection techniques. This is not an unexpected result given the findings of many studies done to evaluate the effectiveness of visual inspection in many different industries. Visual inspection has generally been found to be the least effective means of finding defects when compared with automated or instrumented inspection techniques.

Another concern about the use of general visual inspection as the primary wire management tool was the large amount of contamination on the wiring observed during the intrusive phase of the ATSRAC study. The accumulation had obviously taken place over a long period. The debris (e.g., lint, corrosion prevention compounds, drill chips from structural repairs, metallic shavings, coffee and other liquids, etc.), on/in many wire bundles in each of the study aircraft was so extensive that these bundles could not be seen; much less visually inspected adequately. There is apparently no requirement for aircraft wire to be cleaned prior to general visual inspection (GVI), or as a routine maintenance procedure.

As previously stated, general visual inspection is the only technique now used to monitor the condition of both commercial and military aircraft wiring on a continuing basis and to manage aging mechanisms and damage arising from normal operation and maintenance. Since substantially more damage sites were found using NDT and laboratory procedures during the ATSRAC study than had been identified using close visual inspection, it can be assumed that many wiring defects go undetected during normal maintenance operations. These are typically found only after system failures or very noticeable damage to a wire bundle, such as insulation charring, smoke, or electrical fire.

The results of the intrusive inspection are reported in the Transport Aircraft Intrusive Inspection Project Final Report (An Analysis Of The Wire Installations Of Six Decommissioned Aircraft) dated December 29, 2000 (Prepared by The Intrusive Inspection Working Group, Christopher Smith, Chairman). This report is available from the FAA web site. The data in the Intrusive Inspection Final Report (if not the conclusions and recommendations), as well as, data related to the effectiveness of visual inspection, and participation in the intrusive inspection process clearly indicates visual inspection is not able to identify many types of damage and degradation. Some examples of what may be visually undetectable are: damage or degradation hidden inside wire bundles; high resistance connections (connectors, splices, terminal blocks, nicked conductors/broken strands, etc.) and excessive current density in high energy circuit paths until insulation is visibly charred or a fire has ignited; damage and degradation hidden under accumulated lint and other contaminants commonly observed on all six aircraft studied; damage inside protective wrap materials, conduit, or in inaccessible zones; small cracks and other insulation breaches at arms length without magnification.

Analysis of maintenance data and surveys of maintenance personnel indicate that wire failures, especially intermittent failures, often cause good boxes to be repeatedly pulled and sent for testing and repair at great expense before wiring is considered as the possible cause. Once wiring is identified as the problem, finding the location of the open, short or high resistance is also very time consuming and difficult. The consequences are higher cost of ownership and reduced mission readiness.

For the reasons sighted above, the need for wire diagnostic equipment is clearly indicated. It is necessary for more effective management of aging mechanisms, to allow more efficient diagnostics and repair of system failures caused by wiring, and to reduce the number of events and amount of damage done when defects cause failures resulting in charring, smoke, and fire. If safety is to be improved, inspection techniques must be able to identify the precursors before defects become visually obvious by having already caused charring, smoke, and/or electrical fires.

The ideal set of diagnostic equipment would be capable of data storage and analysis capabilities sufficient to allow trend analysis and be able to identify the following:

- Open circuits and measure where along a signal path the break occurred
- High resistance interconnects (e.g., splices, connector pins, terminal blocks) as they develop and allow intervention before I^2R heating causes insulation to, or a fire ignited
- Non-linear propagation characteristics within connectors and coaxial cable, which cause sensitivity to RF interference and/or heating of transmission lines
- The distance to conductor nicks, cracks, and broken strands before current density causes heating sufficient to degrade insulation, or alter signal propagation characteristics
- Hard and intermittent shorts to other power feeders or signal paths, and to structure and measure the distance to the defect area.
- Small cracks through to the core conductor having the potential to allow stray voltage and current paths wire-to-wire and wire-to-structure, when moisture or contaminant intrudes into wire bundles
- The presence of chafed insulation before the insulation is completely worn away allowing a wire-to-wire or wire-to-structure short.
- RF signal paths with impedance mismatches, high VSWR, and other undesirable signal propagation characteristics
- Shielding and grounding faults and faults in as well as twisted pairs which make sensitive signal paths susceptible to EMI, or to radiate energy which causes EMI problems in adjacent signal paths
- Points in wiring where high voltage corona effects and arcing may occur at altitude

Some of these capabilities already exist and require only a simple instrument like an ohmmeter or a multimeter to identify the existence of the defect. These instruments however, sometimes do not identify the location of a defect along a circuit path. Other techniques such as a standing wave reflectometry (SWR) meter, or time domain reflectometry (TDR) meter must be added to ascertain the anomaly location. Some of the defects above, such as chafing and insulation cracking, which may or may not have penetrated to the core conductor, are much more difficult to identify. Significant development efforts will be required to arrive at test equipment that can be operated by a maintainer of limited experience, require little set up time, operate under adverse work conditions, and able to identify difficult to see degradation and defects, especially in inaccessible zones or within heavily contaminated bundles.

The ultimate goal is to detect high-risk conditions before charring, smoke, electrical fire, arc tracking, or the failure of attached avionics or electromechanical devices can occur. Using present visual inspection techniques often requires these high-risk events to occur before the underlying defect is detectable. Although aircrews are usually able to cope with various equipment/system failures, these types of events, however, do have the potential to cause substantial damage and can lead to an aircraft accident.

Wire Integrity Tester Evaluation Plan

The objective of the Air Force program with GRCI, entitled Improvement of Wire System Integrity for Legacy Aircraft, is to procure wire system testers in both bench top (wire analyzers) and handheld configurations, evaluate them in the laboratory on a test bed, and optimize them for evaluating wire system integrity. The foci of the program are on defining requirements for evaluating wire test systems; identifying actual aircraft wire bundles for evaluation; establishing a wire system integrity test set evaluation test bed; and analyzing, evaluating, developing, and procuring specific wire system integrity test systems for optimization and feasibility demonstrations. The ultimate goal of the program is to transfer and install the wire integrity test systems at a field location as a system that can be used to locate wiring system anomalies and maintain wiring system integrity.

Background Study

One of the objectives of the lead-in project (“Wiring Integrity Analysis of Air Force Weapon Systems”, AMTFD D064) was to evaluate aircraft wiring, document existing maintenance operations and conduct interviews with actual maintainers to determine field needs for wiring diagnostic tools. To achieve the overall objective, site visits were made to identify types of wire system faults that exist and to identify the types of tools and techniques needed to detect the faults. The field team was made up of AFRL and GRCI personnel who documented the current maintenance wire integrity testing practices. Areas of interest included techniques, equipment, documentation, data analysis, reporting, and corrective actions. Site surveys were used to gather information on maintenance operations and diagnostic test equipment requirements. The wire integrity team, GRCI and AFRL/MLSA, conducted site visits to Air Force Air Logistics Centers, and Air National Guard units. The aircraft reviewed included fighter and transport aircraft. The purpose of these visits was to identify potential wiring issues, and to engage in discussions with maintenance personnel. Topics included the usefulness of a wire integrity test system as a diagnostic tool and the benefit of test systems in the identification and resolution of wire anomalies. A major conclusion is that current visual inspection methods and hand-held tools are inadequate to identify most wiring problems. These findings and conclusions were consistent for all the types of aircraft evaluated. The input from the maintainers will be invaluable and directly used in providing a requirements list to the tester manufacturers. The following sections provide a summary of the findings that are applicable to this study for fighter and wide body aircraft:

Maintenance Issues Identified

Most of the maintenance personnel that were interviewed welcomed the idea of a new tester that would make their jobs easier. They stressed, however, that it must be easy to use. Many aircraft have endured a high rate of disturbance to wiring as a result of modifications and other maintenance actions. There was concern that any intrusive testing could result in a greater number of wiring problems. The following is a summary of comments made by maintenance personnel:

- Relays that are not fully seated cause intermittent problems and a tester is not available to evaluate installed circuit breakers. This can cause drastically reduced mission capability during the troubleshooting process. Over 90% of all respondents stated a need for circuit breaker and electrical panel tester. Circuit breaker panels currently have to be removed to the shop and each breaker tested individually. This process is very time consuming and the introduction of a tester that could check the breakers while installed on the aircraft would drastically reduce the time required during the overhaul/maintenance process.
- Nexus connection points are a source of many wiring problems. Removal of an LRU or troubleshooting one connector requires the disconnection of multiple connectors to access the area of interest. Troubleshooting the original problem can create new failures in surrounding systems.
- Intermittent connections typically could only be found by testing under a load. A continuity test is not sensitive enough to detect a broken wire if both ends of a fractured wire are in contact.
- Wiring faults can take 10 to 15 times longer to troubleshoot than LRU related problems. One reason for this is that the Technical Order usually has the maintainer replace the most probable LRU in order of probability of failure. Wiring is only identified as an area to check as a third or fourth item in the troubleshooting ladder. The result is that wiring may only be examined after the third LRU replacement.
- Approximately 80% of wiring jobs on the aircraft result in total replacement of a harness rather than repairing of particular wiring. This is due to restrictions of Technical Orders on the capabilities of the technician to repair wiring at the field level. Many harnesses have wiring from multiple systems running through them.
- A standard multimeter is the most often used piece of test equipment for testing wiring. It is a very time consuming activity to isolate failures.
- TDR is a technology that would be useful. Past versions, however, were difficult to use and interpret.
- The current method for testing wires for some aircraft involves multiple pieces of test equipment. One facility uses a programmable tester, a multimeter, and a TDR for detecting electrical anomalies in the wiring system. A single piece of test equipment to accomplish the same job would be highly desirable.
- For phase maintenance, the primary method for checking the condition of wires is through visual inspection. This process is very time consuming and inefficient due to the fact that most wires are difficult or impossible to see because of their location within the aircraft or position within a large bundle of wires.

Maintenance personnel of both types of aircraft are very limited in their resources to inspect for wiring problems. Flashlight and mirror inspections identify about one-fourth of all wiring problems discovered. This approach was not considered proactive. A targeted and specific inspection for particular problems was considered far more productive and beneficial.

It was widely acknowledged by all levels, that the wiring in these aging aircraft is becoming a greater concern each year. The wiring is being subjected to a greater number of intrusive maintenance actions (i.e. grabbing wiring bundles as a handle, rubbing LRUs against wiring during removal and replacement) simply as a result of the aging of other systems within the aircraft. As these systems age and fail in greater numbers, there is a need to disrupt wiring while fixing those systems. It was widely expressed by personnel that a better, more comprehensive approach to wiring inspection must be implemented in the near future.

The multimeter is the primary piece of test equipment that is used by maintainers. The multimeter is hand-held and very practical for using in a flight line environment. Size and ease of use were widely cited by personnel as favorable characteristics of test equipment. The multimeter is limited in its abilities to measure certain aspects of wiring anomalies. Specifically, with the aircraft on the ground, a wire could be hanging by a few strands and pass electrically with the multimeter. That same wire may be cause for failure when the aircraft is in the air or under “load” conditions.

Interviewees widely expressed the need for a wiring analyzer that could potentially narrow down the location of a wiring problem, within such long runs of wire. Typically locating a defect within two inches was desired. Maintainers routinely requested a wiring diagnostic tool that exceeds the capabilities of the multimeter. The ability of a piece of test equipment that could test under “load” conditions was highly desirable. Ease of use, portability and recurring training on the use of the equipment were major considerations for any type of test equipment that may be introduced.

The surveys indicated the majority of wiring problems are found during actual trouble-shooting of a system. A small number of wiring problems were discovered during the phase inspection process during visual inspections. These inspections are very limited and typically are performed with flashlights and mirrors. A vast majority of aircraft wiring is hidden or inaccessible so the rate of discovery using this process is very low.

Conclusions on Maintenance

Current maintenance practices are limited to repair actions and occasional inspection for a fleet wide wiring issue. Typically, wiring inspection is limited to an area under repair. Inspection is primarily visual, with limited use of diagnostic equipment or optical enhancements. Typically, a maintenance action may state “Perform a general visual inspection.” Inspection of individual wires, in a bundle or connector, is not a practical technique since handling may introduce new damage. In addition, wiring may also be difficult to inspect in various areas of an aircraft due to inaccessibility (i.e. wiring inside conduits and behind panels or equipment).

Organizations are “living” with the wiring issues. As aircraft age, wiring becomes more difficult to maintain with traditional methods. The current maintenance approach of flying aircraft until an electrical failure is encountered is becoming more difficult to continue. More proactive approaches are needed so that failures can be anticipated and wiring systems can be replaced during scheduled maintenance activities. This can be most efficiently achieved through a program that manages the aircraft wiring system as it ages.

During the aircraft field reviews, there were several common factors noted relating to the wiring system maintenance process. For each aircraft, “phase” inspections are primarily visual. Most wiring problems are actually found through trouble-shooting and not through visual inspection. When troubleshooting is required, the primary instrument is a multimeter. Typically when one wire fault is found, there are additional damaged wires present as well.

For troubleshooting equipment, maintainers wanted tools that are portable and have a setup time less than one hour. Additional capabilities include identifying where a fault is physically located, data recording that can be used for preventive maintenance, and sensitivity to contact resistance in connectors, relays, circuit breakers and splices. In general, field units preferred handheld tools, while depot units preferred comprehensive diagnostic tools.

Functional capabilities:
Hard shorts and opens
Physically locate damage sites
Locate intermittent and degraded interconnections
Store results
Use data collected to enhance wire integrity

In summary, the information collected from the site surveys provided valuable information to be considered in the overall findings of this project. The responses from the questionnaires will be used in helping determine which types of test equipment will best fit the needs of the Air Force aircraft maintenance community. Areas of the aircraft that need to be subjected to more stringent and periodic testing were identified in the survey. The survey indicated a need for a more comprehensive reporting process of wiring maintenance. Data collection is needed to support future decision-making activities concerning aging wiring in the Air Force's aging aircraft inventory.

Analysis of Current Wiring Types Used on Aircraft

What follows is a brief summary of the wire insulation types most commonly found on Air Force aircraft. The most commonly used type of wire appears to still be polyimide, commonly known by its trade name Kapton™ or MIL-W-81381 wire. This wire insulation is used on many commercial and military aircraft. This type of wire is used more often in fighter aircraft due to its lightweight small volume (6 mils) and high temperature capability (200°C). It typically consists of an aromatic polyimide over wrapped to form four layers with a fluoropolymer adhesive and coated with a polyimide lacquer for marking and identification. When new, polyimide exhibits excellent mechanical strength and good abrasion and cut-through resistance. It has a high dielectric strength and high temperature application. It is flame and environmentally resistant. While this insulation has exceptional mechanical properties, when nicked, flexing will propagate the crack to the core conductor. Once cracked, the wiring is vulnerable to arcing or disruption of electrical signals. It is also vulnerable to fluid penetration and exposure to the elements. It is also susceptible to arc propagation if a carbon char forms during an electrical arc event. The stiffness of the insulation can make it difficult to handle. MIL-W-81381 insulation is also susceptible to degradation from high pH (12 and above) cleaners and under certain conditions long-term exposure to moisture (hydrolysis) and ultra-violet radiation

Alternative wire insulation materials include cross-linked Tefzel™ (MIL-W-22759/33-44) which is a cross-linked fluoropolymer. It is higher weight and larger volume than polyimide constructions. It's mechanical properties begin to drop above 70°C specifically cut-through resistance. Additionally, Tefzel™ will generate considerably more smoke than other types of aerospace wiring when burned. There is also an issue with conductor corrosion in silver plated wire (red plague). Another widely used type of insulation in Air Force aircraft is Teflon (MIL-W-22759). Teflon is lightweight, arc tracking resistant, abrasion resistant, and heat resistant. Although, Teflon is known to flow under stress and high temperature, creating non-uniform thickness. For PVC, Kynar, and other materials such as polyalkene (MIL-W-81044), there are flammability, chafing/environmental resistance and thermal stability issues. One of the more recently available wire insulations is a composite construction containing primarily Teflon with a small percentage of a modified aromatic polyimide (MIL-W-22759/88-). Much of the original development and testing was part of an AFRL sponsored contract in the late 1980’s. The polyimide is sandwiched between two Teflon layers. This insulation has a good balance of properties, exhibiting excellent environmental performance while maintaining good mechanical properties over full temperature range, with improved flexibility and excellent arc propagation resistance compared to MIL-W-81381. This has become one of the recommended wire insulation alternatives for aircraft wired with MIL-W-81381.

The objective of the Improvement of Wire System Integrity for Legacy Aircraft Program is to select wire integrity test equipment and methods for use by Air Force maintenance organizations. The desired methods should be simple to use, easy to setup, adaptable to AF maintenance environments, require minimal training for technical school graduates to use and available for fielding during FY02. handheld meters and automated wire analyzers are the two categories of test equipment being considered.

Handheld

Handheld tools are categorized as battery operated, single or multi-function meters approximately the size of handheld multimeter. The readout format can be either digital or analog; however, digital displays are preferred. The desired functions are isolation of conductor shorts and opens, spatial indication of fault, indication of wire insulation degradation, and isolation of intermittent faults. The handheld meter should have a signal output for laboratory evaluation.

Wire Analyzers

These units are computerized circuit analyzer systems with the capability to store wiring architectures, generate and store test programs, conduct bulk and individual conductor testing, store test data, allow data manipulation, and provide analysis and report generation capabilities. The system should be able to test up to 5,000 points and be expandable to 100,000 points. It is desirable to have the handheld meters integrate with the portable systems. Options on the wire analyzers can include resistance, 4 wire Kelvin bridge, DC voltage, AC voltage, and capacitance measurements, voltage and current stimuli capability.

The two leading technologies used in test equipment currently being evaluated involve time domain reflectometry and standing wave reflectometry. Common to these reflectometry methods is the sending of a signal down the wire, which is treated as a transmission line, and sensing the reflected signal.

Time Domain Reflectometry

Time Domain Reflectometry is the analysis of a conductor (wire, cable, or fiber optic) by sending a pulsed signal into the conductor, and then examining the reflection of that pulse. Wiring and insulation anomalies may be precisely located by examining the polarity, amplitude, frequencies, and other electrical signatures of all reflections. Any device or wire attached will cause a detectable anomaly. TDR analysis will usually NOT detect capacitively isolated devices or inductive taps. In the case of capacitively isolated device or inductive tap, the TDR sweep is always supplemented by a detailed high frequency cross talk evaluation and a detailed physical inspection.

TDR use is based on the theory nearly all power and control circuits can be analyzed as radio frequency (RF) transmission line with a load on the end. This allows circuit components to be separated in time and analyzed individually while measurements are made from a remote location. Pulse width and rise time determine the length of line which can be tested using TDR. Wires are not perfect in construction and there will always be some sort of anomalies reflected in the waveform return and/or that the signals will experience loss as a function of time and distance. TDR sends out a square-wave pulse which contains frequencies from DC to 1 GHz that travels down a transmission line at a speed slightly less than the speed of light depending on the type of wire under test. During the travel down the transmission line the current and voltage wave can be measured as a function of time and distance. The resultant current and voltage wave is called the incident traveling wave. The characteristic impedance Zc is seen by a source connected to the transmission line. When a transmission line is terminated with a resistor equal to Zc then the incident traveling wave is absorbed and no waveform is reflected. But if the termination of the transmission line is lesser or greater than Zc then a reflection of the incident wave of the same or opposite polarity is seen. Therefore, when a pulse travels down a wire there will always be a return of X magnitude, given that the line does not terminate with a value equal to that of the original pulse.

Standing Wave Reflectometry

Basically, standing-wave reflectometry (SWR) involves sending a sinusoidal waveform down the wire. A reflected sinusoid is returned from the wire's end, and the two signals add to a standing wave on the line. The peaks and nulls of this standing wave give information on the length and terminating load of the cable. A healthy line's wave pattern will be distinct from that of a line with an open or short circuit.

A microprocessor-controlled oscillator injects a sweep of frequencies into the cable under test and determines whether the discontinuity is caused by a short or an open circuit. Voltage measurements are made before and after the reference resistor. Distance to cable defect is 1/4 of the wavelength injected at the time the impedance approaches zero for an open conductor. Distance to cable defect is 1/2 of the wavelength injected at the time the impedance approaches zero for an shorted conductor. A microprocessor is used to set the sweep frequencies using a numerically controlled oscillator (NCO). In a first pass, test signals are injected from 1 MHz to 50 MHz, in 50 kHz steps. In a second pass, test signals are injected in 4 kHz steps only in the frequency region where the discontinuity was detected.

Lab Evaluation

The objective of the laboratory evaluation is to validate equipment supplier specifications (precision accuracy, sensitivity, repeatability, functionality), determine the maturity of the equipment, and the applicability to the Air Force maintenance environment and finally to provide feedback to the equipment suppliers. Since some of the testers being evaluated are still being modified, the results will be fed back to the developer as suggestions to be incorporated in follow-on designs.

Material and equipment used in this study include a fraying machine which will duplicate common chafing scenarios, a dither (wire bender) used to break internal wires without breaking the insulation, and assorted standard electrical laboratory equipment such as oscilloscopes, multimeters, network analyzers, and an IR camera.

The testing not only verifies equipment specifications for finding the obvious shorts and opens but also tests the sensitivity of the wire analyzer, through varying degrees of insulation, conductor and shield damage. Special cases such as corroded connectors, splices and fraying will also be looked at. Wires will be aged in environmental chambers and the change in wire test results will be observed. Component testing such as relays and circuit breakers will also occur. The testing approach will be (1) Verify test bed configurations and establish a base line (2) Identification of induced defects and faults in the test bed (3) To determine ease of use using such factors as operator dependence or the amount of training required.

The test set up consists of three test beds. The first is designed to test handheld and wiring analyzers using representative F-16 wires. The second uses both the F-16 representative wires and selected F-16 harnesses. The third will use both new and used F-16 aircraft harnesses. Actual preliminary onboard aircraft testing evaluation and demonstration will be accomplished on an aircraft battle damage repair (ABDR) jet.

Test procedures consist of testing undisturbed and fault induced wire, then recording the results. Items to be considered while the laboratory evaluation is preformed

-Varying distance to fault (potential testing and accuracy/precision measurements)
-Varying degree of the induced faults (conductor,insulation, and shield)
-Format of reports (web-based, digital, and graphical)
-Test duration
-Preparation time, training required and ease of use
-User interface, ease of output interpretation
-Quality of artifacts (reports)
-Technical manuals (user instructions, updates from a remote site)

The testing not only verifies equipment specifications for finding the obvious shorts and opens, but special tests will be conducted to test the sensitivity of the device in identifying varying degrees of insulation, conductor, and shield damage. Special cases such as corroded connectors, splices, and fraying will be examined. Environmental testing to evaluate the effects on wire test results will further test the devices. In the future, component testing will also be accomplished.

Overall summary

The user/test equipment interface is the single biggest problem in automated electrical testing of aircraft. The test equipment needs to be able to display the results in a user-friendly manner. The primary method currently used by maintainers to identify problems is visual inspection. Most wiring problems, however, are found through trouble-shooting, not through visual inspection. Manual (multimeter) electrical testing is very time-consuming. Most troubleshooting is accomplished with a multimeter. Many problems erroneously check okay when tested with a multimeter. To find the problem the connector often has to be demated. Currently, there is no capability to test circuit breakers on the operational side. Maintainers would like a way to test circuit breakers and a method to tell the condition of outer insulation layer and the conductive properties of the wire. Wiring is part of the aircraft infrastructure and needs to be thought of in the same manner. Maintenance organizations are living with the wiring issues, conducting phase inspections visually. Currently, modifications/repairs use large, complex wire test systems and may take days to accomplish. Maintainers want something portable that has enhanced capabilities over current multimeter capabilities. They want it to be easy to use and portable and have a set-up time less than one hour. It should be able to record data that could be used for preventative maintenance. The tester should be able to identify where fault is physically located. It should be able to identify sticking relays and trends in contact resistance. Since many wires pass in a no-load condition yet fail under load, the ability to test a wire under load would be beneficial. Field units favor handheld tools, while depot units favor comprehensive tools. These are the goals this program is intended to address.

Wire integrity evaluation is a very dynamic area of research. New promising areas under development include frequency domain reflectometry (FDR) and impedance spectroscopy. FDR uses sine waves, but it directly measures the phase difference between the incident and reflected waves. Any faults in the wire will generated resonances between the two signals. Researchers at Utah State University with support from Management Sciences Inc, and the Naval Air Systems Command are developing this method. Impedance spectroscopy involves the extraction of electrical parameters from impedance measurements. Using the impedance spectra, you can determine the propagation function, which gives the location of the open/short, resistance function which correlates to conductor health, dielectric function that relates to insulation health. Rockwell Science Center in conjunction with Boeing Aircraft Company is accomplishing this work.

Diagnostic tools currently available are not comprehensive enough for maintaining an adequate wire integrity program. It is hoped the efforts of this program and current research will begin the transition from a reactive (fix when it breaks) to a proactive/prognostic health monitoring maintenance process resulting in extended and predictable failure free operating periods for the air fleet.

Bibliography:

1. Wiring Integrity Analysis of Air Force Weapon Systems, Final report on Delivery Order 64 of the Advanced Manufacturing Technology Feasibility Demonstrations (AMTFD) Program Contract Number F33615-96-D-5101, September 2000

2. “Down to the Wire”, Cynthia Furse, Randy Haupt, Utah State University IEEE Spectrum February 2001 Volume 38 Number 2

3. Review of Federal Programs for Wire System Safety, Final Report, November 2000, National Science and Technology Council Committee on Technology, Wire System Safety Interagency Working Group.

4. New Insulation Contructions for Aerospace Wiring Applications, Volume I: Testing and Evaluation, Final Report WL-TR-91-4066, performed by McDonnell Douglas Corporation under contract F33615-89-C-5605 to Wright Laboratories, Wright-Patterson AFB, OH.

Requirements for Risk Assessment Tools for Aircraft Electrical Interconnection Subsystems

Dr. Christopher Smith
FAA William J. Hughes Technical Center
AAR-430
Atlantic City International Airport
NJ 08405, United States

and

Mr. Robert Pappas
FAA William J. Hughes Technical Center
AAR-433
Atlantic City International Airport
NJ 08405, United States

Background

The continued safe operation of aircraft well into their expected service life depends on the safe and effective transfer of electrical power and signals between aircraft electrical devices. This in turn requires the enduring physical integrity of the electrical interconnect subsystem (EIS), which is comprised predominately by wire, connectors, switching devices (including relays and solid state switches) and protective devices such as circuit breakers. Recently there has been speculation that, under some conditions, the EIS on older aircraft may degrade to the point that it is no longer capable of ensuring the safe transfer of electrical current.

Though an EIS may be optimal with respect to aircraft design requirements, operational assumptions and the data existing at the time of certification, unanticipated demands on the EIS and changes to aircraft configuration can degrade EIS performance below acceptable limits. In addition, performance characteristics of the EIS over service lives of more than twenty years are difficult to predict. Inevitably, unanticipated failure modes will emerge requiring remedial action.

The EIS in modern transport aircraft provides the means of communication and/or power for nearly every subsystem aboard the aircraft. Failures in the EIS could result in the loss of critical functions as well as the potential for fire or other physical damage to the aircraft. Hence, the FAA is conducting a number of research projects addressing aging EIS concerns. The particular focus of one project is the development of advanced EIS risk assessment tools for design optimization and life-cycle management.

EIS Design Requirements and Risk Assessment

Certification requires that systems on airplanes be able to perform their intended function under all foreseeable conditions. The probabilistic safety analysis required under Part 25 (Para. 25.1309(b)) is just one of the requirements that all the systems on the airplane must meet. These analyses define the safety criticality of individual systems based on the functions they perform and requires a numerical assessment (based on a typical flight) of those systems that can participate in safety critical functions. The safety analysis done for certification is not a risk assessment in the truest sense of the word. For example, it does not take into account such things as fleet size when considering total exposure to an event. Typically, the various systems safety analysis done to satisfy 25.1309(b) have addressed the individual wires as parts of the various systems they serve, not as an individual subsystem that supports all the systems on the airplane. The EIS, in general, is designed and certified based on the individual systems needs, the fire marshal's requirement for the compartment the wire goes through, and good wiring practice (as defined in documents such as AC 43-13B). The fact the EIS has meet all requirements is not documented in one place.

Modern transport aircraft, with their digital systems and increased use of electrical and electronic command and control systems, are designed and certified to bound the risk associated with critical failure of one or more aircraft systems. The ever increasing complexity of these systems and the availability of more precise risk assessment methodologies (and the computational resources to implement them) have lead to a more sophisticated certification analysis than that applied to older generation aircraft. Federal Aviation Regulation

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

(FAR) 25.1309[1] identifies risk requirements and its companion Advisory Circular (AC 25.1309) interprets those requirements as the familiar 10^{-9} probability of catastrophic failure. The regulation also lays out other specific risk requirements and means to verify adherence to those requirements. AC 25.1309 and SAE Aerospace Recommend Practice (ARP) 4761 further elaborate on acceptable means of demonstrating compliance with FAR 25.1309.

First and second generation jet aircraft subsystems, on the other hand, were typically evaluated to specific, independent design or performance requirements – to the single-fault criterion or to a basic fail-safe design concept. The certification basis for older aircraft may not necessarily provide for an adequate assessment of risk associated with EIS failure, particularly when these airplanes are operated beyond their original design life or if they are updated with modern electronic/electrical systems. Unfortunately, qualitative analyses, which work well for simple subsystems, can breakdown when thousands of potential failures modes interact to produce consequences with unknown probability of occurrence.

For aircraft in the design stage, the means for demonstrating compliance to FAR 25.1309 or earlier requirements range from rather simple (qualitative arguments proposing an analogy to similar, already-certified subsystems) to complex (use of formal quantitative fault tree analysis and failure modes and effects analysis). Many of the more sophisticated analysis methodologies rely on multiple-cause failure condition evaluations using sophisticated probability tools. In all cases, the availability of precise design information is assumed.

FAR 25.1309

AC 25.1309 identifies a process that begins with a Functional Hazard Assessment (FHA) and then concludes with a safety assessment whose initial objectives are defined by the FHA. A FHA is a highly methodical and structured analysis of the functions performed by the aircraft and the aircraft systems that identifies all single failure conditions and combinations of failures and failure conditions that could hazard the aircraft. The FHA does not attempt to determine the cause of the functional failure. Instead, the purpose of the analysis is to identify the effects of the loss of each particular function and also the effects of the improper operation of the function (malfunction) and to classify the severity of the effects. The loss or inability to properly perform a function is considered to be a failure condition. The classification of the severity of each failure condition is based on the effect on the airplane and it's occupants. Failure conditions are classified as minor, major, hazardous, or catastrophic.

The FHA is a qualitative analysis, usually conducted using service experience, engineering and operational judgment. In some cases, the effects of the loss or malfunction cannot be accurately estimated. The FHA will then describe the type of tests that will be used to evaluate the severity of the failure condition. The FHA should consider all phases of flight, environmental conditions and abnormal/emergency operating conditions.

For systems where there is a clear correlation between functions and the system that performs the function, and where the systems (and hence functional) interrelationships are relatively simple, it may be feasible to conduct a separate relatively unstructured FHA for each system. The EIS does not, however, fall into this category. The EIS, like electrical, hydraulic, and pneumatic power systems do not directly perform aircraft functions. These systems provide services necessary for the operation of other systems; their loss or malfunction usually have widespread effects on many other aircraft systems. For the EIS a structured, top-down approach from an airplane perspective should be used. The FHA must also consider all factors that might alleviate or intensify the direct effects of the initial failure condition.

Figure 1 is a graphical representation of the development and manifestation of failures and failure conditions. Any FHA which fails address all of these factors is incomplete.

The system safety assessment uses various tools to assess the probability of occurrence associated with the failure conditions derived from the FHA. This analysis may be either qualitative or quantitative. Because the

[1] FAR 25.1309 was originally issued in 1965 with no reference to probability of failure, severity of failure, or aircraft safety risk. In 1970, FAR 25.1309 was amended to incorporate the concept of risk. The risk basis was further clarified in a 1977 amendment. FAR 25.1309 is currently being revised as part of the FAA/JAA Harmonization activities.

failure of an EIS can affect multiple flight critical systems, designers of new or highly modified EISs are usually compelled to apply sophisticated analysis techniques including FMEA, Fault Tree Analysis, Dependence Diagrams, Markov Analysis, and Common Cause Analysis.

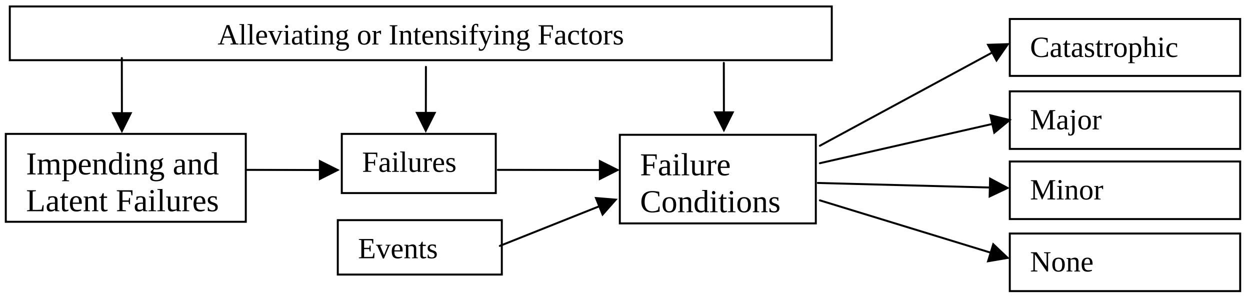


Figure 1: **elements of the Failure Hazard Analysis**

Figure 2 shows a flowgraph identifying the generic decision logic guiding the use of analysis techniques.

In-Service Risk Assessment

Though certification requirements mandate a thorough safety analysis prior to an aircraft's introduction to service, changing service profiles, subsystem modification, and unanticipated failure modes may effect the applicability of the original analysis. During evaluation of service problems, the focus of the investigation may narrow to parts, components, and subsystems. Under these conditions studying the original safety analyses may identify basic design weaknesses and help interpret the service event. A full re-analysis of the system is often not necessary and is impractical.

Where service history indicates a potential threat associated with degraded components in the EIS, remedial action needs to be taken and, if necessary, will be mandated by an Airworthiness Directive (AD). The remedial action may come in the form of operational restrictions, maintenance or inspection requirements, or aircraft modification. In any case, the remedy must be preceded by an analysis that conservatively estimates the probable frequency, severity, and exposure of the threat.[2] Wholesale re-assessment of the EIS to current standards of FAR 25.1309 may be a possibility, but is almost certainly not very practical and to a large degree redundant with prior analysis. Data collection would be extremely difficult (even for aircraft with few modifications) and the results of the analysis might not indicate practical means for upgrading the particular components of EIS.

These circumstances demand that the risk tools used by aircraft operators be sufficiently flexible to admit alternative data sometimes at a much more general level. One approach to risk abatement of an older EIS is to perform a risk analysis, utilizing service data, in addition to fault trees and other standard techniques, to narrow the focus on evidential safety threats. In January of this year, the Aging Transport Systems Rulemaking Advisory Committee's Intrusive Inspection Working Group (IIWG) published a report with such an analysis.

Note: The FAA is examining several concepts for facilitating compliance with existing and emerging rules regarding electrical systems safety assessment. The following approach represents neither a proposed change to the rules nor an approved means of compliance with those rules.

The IIWG Methodology

The IIWG was tasked with determining the state of wire on six decommissioned aircraft and assessing the risks associated with the wire flaws if those aircraft had been operating with those flaws. Because the IIWG's intention was to identify generic threats to the EIS (not specific conditions correctable by service bulletin or AD) and because the IIWG did not have complete design or configuration data on the EIS inspected, there was

[2] In a policy statement published in the Federal Register, July 2, 2001, the FAA stated that simple reliance on standard or best practice (including AC 43-13) was not sufficient for regulatory approval of a type design data package.

no effort to identify the specific threat associated with the conditions found. Instead, the IIWG identified generic conditions and postulated the risk associated with certain hypothetical (but realistic) situations surrounding those conditions.

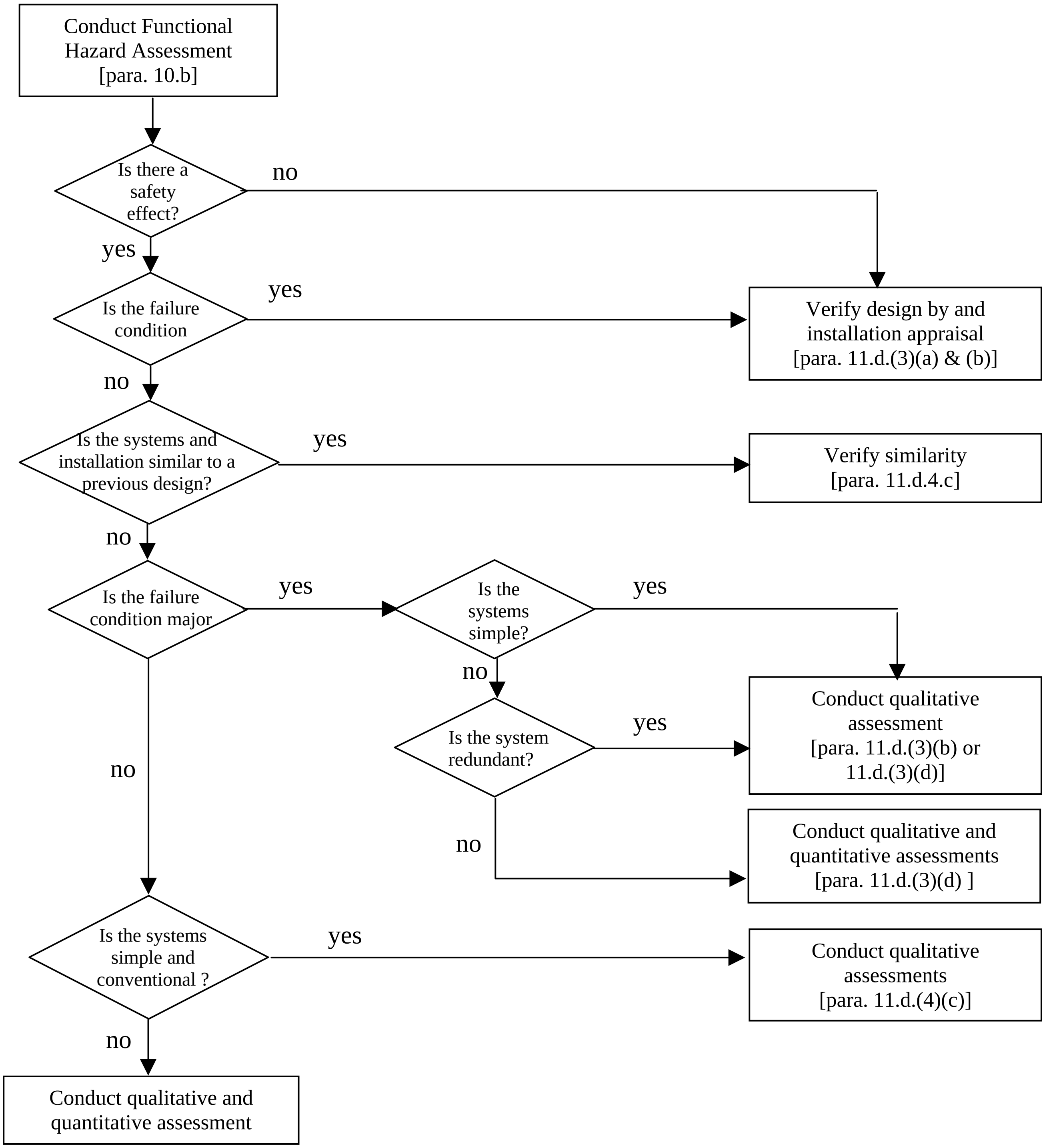


Figure 2: Depth of Analysis Flowchart

Despite its rather unique risk assessment objectives, the IIWG did base its methodology upon industry-accepted subsystems risk analysis. In order to conduct a System Safety Assessment, the analyst requires knowledge of at least the following four parameters:

- Failure Identification
- Failure Effect
- Probability of Occurrence
- Failure Condition Severity

In the aircraft design phase, the full availability of design specifications allows the failure condition severity and effects to be determined in an often lengthy but conceptually simple analysis (the FHA and derived safety requirements from the preliminary system safety assessment). On the other hand, the absence of relevant service data requires the use of sophisticated analysis techniques, testing and expert judgment to estimate the probability of occurrence (a systems safety assessment using FMEA, Fault Trees, etc).

Just the opposite was true for the IIWG assessment of the decommissioned aircraft. For the IIWG the probability of occurrence was relatively easy to assess from the frequency of the findings. On the other hand, because the conclusions the IIWG would generate were to pertain to generic subsystems and because the findings were often latent flaws or flaw precursors, the failure condition and effects were much harder to assess. As such, the IIWG developed a modified FHA and systems safety assessment referred to as the General Threat Analysis (GTA).

In the GTA, conditions (flaws and flaw precursors) were assessed for severity given plausible, hypothetical situations[3]. Hypothetical situations involve subsystems characterized, not by function and design, but by a set of factors, which would aggravate the degeneration of conditions into hazardous failures. The GTA begins with the development of two lists:

- A listing of the significant wire condition. In the case of the intrusive inspections, these observations were the direct result of the inspections. For revenue service aircraft, this data can be generated by inspection or by service difficult review.
- A listing of all generic conditions, which may aggravate – in any plausible situation – a failure associated with the terminal condition of any observed degenerative condition. Note that this list does not necessarily include factors that may have led to or may yet advance the condition; only those factors that could make some presumed subsequent failure more or less severe.

The two lists produced by the IIWG are presented in Tables 1 and 2. Table 1 contains only age-related wire conditions. For more comprehensive analyses, a larger and more detailed list would be created. Similarly, Table 2 contains only approximate aggravating conditions. A more precise analysis would require a more detailed list.

The two lists and the expertise and experience of the IIWG members were used to develop generic fault graphs. These fault graphs indicated the severity of the potential (worst case) consequence, if the fault were allowed to reach its fully degenerate state. Each branch of the fault graph terminated in one of three possible severities[4].

Undesirable – any condition that might – if left uncorrected – lead to a slight reduction in safety margins, slight increase in crew workload, or inconvenience to the occupants.

Severe – any condition that might – if left uncorrected – lead to significantly reduced safety margins or functional capabilities, a significant increase in crew workload impairing crew efficiency, or substantial discomfort to occupants.

---

[3] Plausible hypothetical situations will be those situations supported by the existence of data for that or a similar situation and/or the expert opinion that such situations could reasonably be expected to occur in the life of an aircraft.

[4] The fault graphs did not need to terminate in only three possible severities, but any such analysis must not have more terminal conditions than the resolution of the analysis permits. In other words, if there were 100 possible terminations, the analysis must be sufficiently sensitive to clearly and consistently distinguish between the 87th and 88th terminal condition.

Critical – any condition that might – if left uncorrected – lead to a large reduction in safety margins or functional capabilities, higher workload or physical distress such that the crew could not be relied upon to perform tasks accurately or completely, or adverse affects upon occupants.

Wire Condition	**Definition**
Deteriorated Repair	A faulty wire splice assumed to have met requirements when established (e.g., a splice originally established to be environmentally sealed but no longer so).
Heat Damage or burnt wire	Thermal damage to insulation resulting from the presence of elevated temperature due to internal or external heating.
Vibration Damage/Chafing	Insulation wear (material loss) resulting from the repeated application of a force which, if applied only once, would not result in noticeable damage.
Cracked Insulation	A breach in the wire insulation that does not include breaches resulting from the direct physical contact or traumatic force (e.g., knife cut or tears).
Arcing	One or more instantaneous electrical discharges evidenced by burnt spot on one or more wires and melted conductors.
Delamination	The unraveling of a tape-wrapped insulation. The separation of layers of insulation in a multilayered construction.

Table 1: age-related wire conditions

Aggravating Condition	**Definition**
Explosive Environment	An environment where there is a reasonable expectation of the presence of an explosive combination of gases during some phase of operation.
Flammable Materials	Surrounding materials that can sustain combustion. Includes the wire insulation itself (e.g., PVC but not polyimide).
Other Critical Systems	The wire in question is bundled with other wires, at least one of which supplies current or signals to systems required for safe flight.
Moisture	Normal relative humidity in excess of 90% during some phase of flight (landing, takeoff, climb, cruise, decent, approach, landing), resulting in enhanced likelihood of shorting.
Vibration	Sufficient relative motion between wires or between wires and structure to cause or accentuate intermittent shorting.
Contamination	Contamination as the result of normal operation or maintenance resulting in either enhanced flammability or likelihood of shorting.
Cockpit or Electronics Compartment	High consequence failure locations within the aircraft.
Arc Tracking Potential	The presence arc-track-susceptible materials in the bundle in conjunction with those conditions which could precipitate sustained arcing.
Potential for Excessive Resistance Heating	Wires with high current loads may fail as the result of excessive resistive heating at repair or splice locations. This failure can evolve into severely burnt, cracked, or melted insulation on the offending wire and its neighbors. With excessive heat and bare wire at these locations, the potential for fire is high.

Table 2: aggravating conditions

The fault graph was assembled by using the IIWG's expert judgement to identify the most significant aggravating condition[5]. The presence or absence of this condition leads to the first branch. The presence of successively important aggravating conditions was assessed until either the combined conditions necessarily resulted in an extreme outcome (critical or undesirable) or until all aggravating conditions were exhausted.[6] The fault graphs were further simplified by eliminating branches that necessarily lead to the same outcome. The fault graphs and flaw frequency data were then used to develop the IIWG's recommendations for EIS risk elimination or mitigation.

Having established the presence of an unacceptable condition, the aircraft designer may work backward or forward through the fault graph to identify interventions. For example, if the path through the fault graph in Figure 3 terminates immediately after the block that assesses the presence of vibration, the designer may force the terminal condition from "severe" to "undesirable" (less severe) by modifying the wire harness support hardware and ensuring protection from contamination

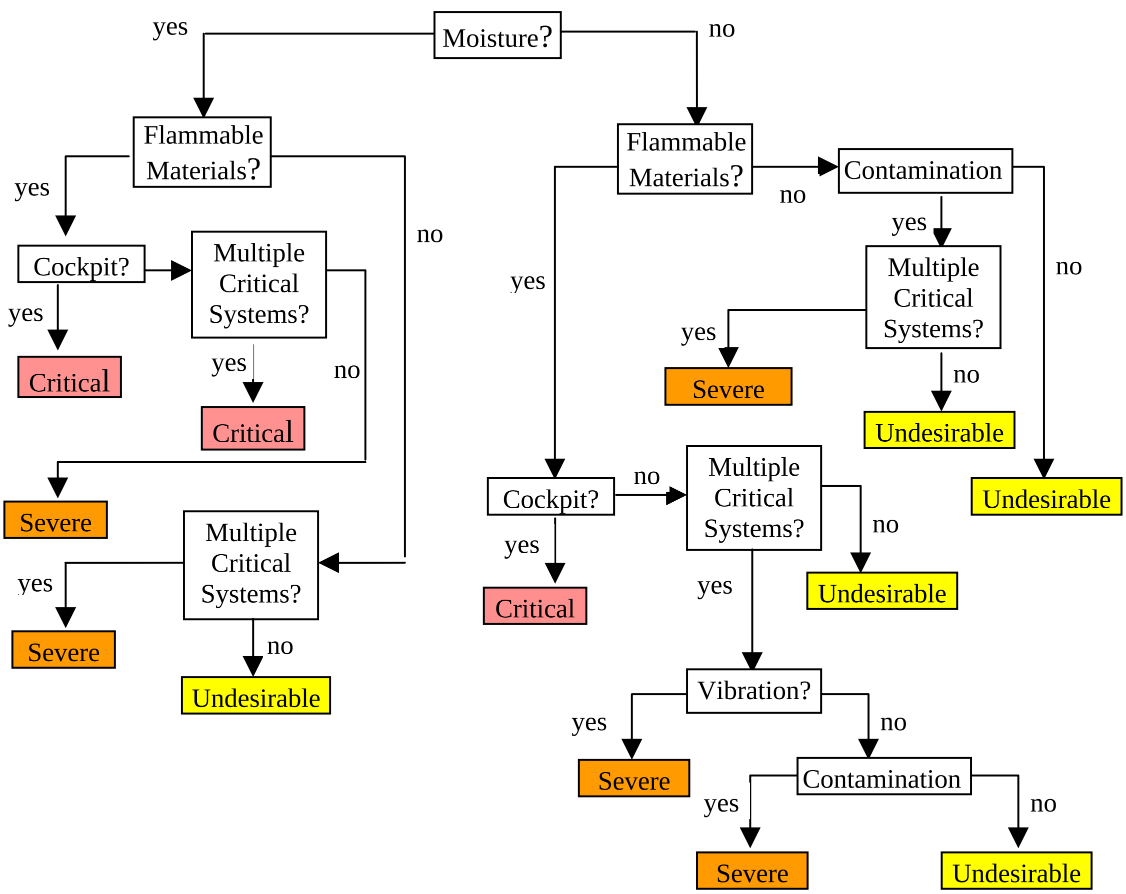


Figure 3: Example Fault graph for Wire Cracking

---

[5] The most significant aggravating condition depended somewhat on the wire fault under consideration. In general, it turned out to be the presence of moisture, but not always.

[6] Consideration of the aggravating conditions in order of importance is only necessary to minimize the complexity of the resulting fault graph. An arbitrary order should (if the same decision criteria are applied) lead to the same outcomes, though with more branches (i.e., redundancy).

Enhanced Risk Assessment Tools

Drawing from both standard design practice and the IIWG's risk assessment methodology, the FAA has initiated a project to develop risk assessment tools for application to existing EIS. The end product is expected to be an analysis methodology implemented in computer software that is:

Relevant: Model assumptions cannot be brushed off as simple parameters that can be changed as necessary. A risk model has no virtue if its assumptions are unjustifiable or parameters unknown or unavailable.

Practical: The end users of the risk assessment tools developed under this program will most likely be airline engineering and maintenance organizations. The risk assessment tools will have no virtue if the end users cannot embrace the tools because they are too complicated or because the tools seriously violate constraints of their operations (e.g., require unavailable data).

Useful: The tools should do more than confirm the obvious.

The IIWG fault-tree methodology was developed for the specific purpose of analyzing the intrusive inspection data. Because it was created in the course of the intrusive inspection project for the sole puropse of analyzing data from those inspections, it could be (and had to be) simple and reliable. In developing a more general methodology for application to revenue service aircraft, the FAA has the same requirements for high reliability but a greater need for broader applicibility.

Furthermore, the availability of design data for revenue service aircraft should be better, and the availability of configuration and modification data should be complete. Hence, while the risk analysis may not be able to quantify the risk of one specific wire in a bundle arcing to another specific wire, it should be able to quantify the risk of a newly installed in-flight entertainment wire acring to a wire in an adjacent bundle that is known to support flight critical subsystems.

Concluding Remarks

The FAA is committed to reducing the risks associated with electrical interconnect system failures. In doing so, the FAA is pursuing a program to develop risk assessment tools suitable for application to aging revenue service aircraft. The tools will support routine modification and configuration control as well as remedial action in response to safety threats.

The Pivotal Role and Current Status of Nondestructive Inspection Systems in the Maintenance of Aging Aircraft

J.P. Gallagher, W.C. Hoppe, Alan Berens, David Stubbs
University of Dayton Research Institute
University of Dayton, Dayton, OH 45469, USA

Charles Buynak
Nondestructive Evaluation Branch
Materials and Manufacturing Technology Directorate
Wright Patterson AFB, OH 45433, USA

Abstract

This paper discusses the pivotal role that Nondestructive Inspection (NDI) plays not only for maintaining safety through early crack detection in airframes and engines, but also for minimizing corrosion maintenance costs. The paper is based on multiple projects that have supported developing and validating NDI systems for crack detection in airframe and engine components and for corrosion detection in airframe structures. These projects have led to a new understanding of how to develop advanced (automated) NDI systems and how to quantify the capability of an inspection system for accurately detecting crack damage or corrosion damage in a maintenance environment. The paper also addresses the issues associated with how reliable, accurate NDI can also be used to detect (and quantify) the early stages of corrosion damage, so that corrosion control strategies can be implemented.

1. Introduction

Safety and economic issues drive the creation of maintenance plans for aging aircraft [1]. Over the last 30 years considerable development has taken place to ensure the effective management of potential crack damage on the structural integrity of aircraft and engine structures. The importance of nondestructive inspections (NDI), as part of the early crack detection maintenance approach to safety for these flight critical structures, is well recognized and practiced [2]. In contrast to cracks, the economic impact of corrosion on the maintenance of an aging aircraft is probably more significant than the safety impact, especially when one considers the costs for sustaining military transport aircraft. Thus, as the fleet of aircraft age, corrosion becomes a significant driver in airframe maintenance planning [3].

Over the years, statistically valid approaches have been developed to quantify the probability of detection (POD) for cracks using any given inspection system and these POD assessments have been employed in a growing number of applications [4,5]. Only now are similar POD assessments being attempted in order to quantify the capability of an NDI system for detecting corrosion; and this is possible, only because the current NDI metrics can be used to quantify the impact of corrosion damage on the structure [6].

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

As aircraft age, structural maintenance actions are required to ensure the continuing safe and economical operation of both the airframe and engine. These maintenance actions are required to contain the level of in-service created fatigue and corrosion damage below that which could (1) compromise aircraft structural integrity or (2) cause premature replacement of significant structural components. The USAF uses the Force Structural Maintenance Plan (FSMP) and the Engine Structural Maintenance Plan (ESMP) to summarize the anticipated actions and the times when known or suspected problems will be addressed in the airframe and engine, respectively. Initially, these documents are based on information generated during the design phase and then updated periodically as field experience is gained.

1.1 Principal Reasons Why NDI is Pivotal

The FSMP and ESMP define safety related surveillance programs that identify the presence of crack damage at some fraction of the expected crack growth life. NDI provides an essential tool for ensuring that any crack damage is found well before it could reach a size that would impact safety. Inspecting the structure for crack damage provides an alternative to replacing expensive structural elements on a strict time schedule such as dictated by a safe life approach. When NDI indications identify the occurrence of cracks, prescribed repair actions are triggered. In essence, NDI systems provide the basis for condition-based structural maintenance.

One important aspect of this surveillance is that the damage is sometimes found in known hot-spot locations well before it was anticipated. A basic feedback loop is required to provide the engineers responsible for keeping the FSMP and ESMP current with early observations of anticipated damage in aging aircraft. An effective feedback loop will allow the timely rescheduling of planned maintenance and repair actions or the development of new actions if hardware must be replaced.

To support aging aircraft, the maintenance plan must evolve, since unanticipated damage can be created during in-service operations; and, this damage is only found by experience. In-service, age-related damage (fatigue, corrosion, environmental degradation and wear) frequently occurs at unanticipated locations, and many times the damage is found by happenstance. It is important that this unanticipated in-service damage information be rapidly transmitted from maintenance operations to the FSMP/ESMP engineers.

Timely feedback on unanticipated damage will allow the maintenance and structural engineers to devise strategies for (1) establishing the breath of the potential problem and (2) developing and implementing cost-effective maintenance or repair actions. Having an effective inspection procedure that identifies the damage (before it reaches a structural limit) and a repair procedure that arrests (or slows) the growth of damage and reestablishes the integrity of the structure is the preferred approach for addressing unanticipated damage, and much more economically attractive than replacing hardware.

Thus, NDI is essential for routine surveillance to locate anticipated damage, for routine surveillance to detect unanticipated damage and for scoping the extent of any newly identified in-service damage problems. The NDI surveillance and problem solving activities that determine the presence of aging damage provide the maintenance manager with the ability to anticipate when major fleet-wide maintenance actions are required.

1.2 Crack Detection NDI Reliability – Driven by Safety

The reliability of crack detection NDI systems is driven by safety. Damage tolerant design assumptions require that crack size assumptions following an inspection are equal to the demonstrated probability of detection (POD) capability of the system. The POD is thus an important quantitative measure of the NDI system reliability. The POD established damage-tolerant crack size values are typically given as the 90/95 (probability of detection/ confidence level) crack sizes. For automated inspections of engine structure, the 90/50 crack size is used. The purpose of having a 90 percent probability of detection is to ensure that the inspection system will detect a critical flaw size with a high probability (only 1 in 10 cracks of this size will be missed). So the reliability for crack detection is driven by a need to qualify the inspection performance for safety reasons.

Guidelines for assessing NDI system reliability for crack detection are available in Military Handbook 1823 [7, 8]. This handbook represents an advanced development of NDI reliability assessment for detecting cracks in critical structural components. This handbook was developed to support the retirement for cause (RFC) program and provided the basis for establishing the reliability of the RFC NDI systems used to detect cracks in USAF F100 engine disks. Figure 1 provides a photograph of one of the RFC – Eddy Current Inspection Systems used to inspect F100 engine disks. An example describing the development of the 90/50 crack-size to balances the need for crack detection reliability (limited number of misses) with an assurance that cracks will be found if there is an indication (limited number of false calls) is presented in Figure 2.

1.3 Corrosion Detection NDI Reliability - Driven by Economics

Corrosion detection NDI reliability is driven primarily by the need to find hidden corrosion economically. There are two aspects to the corrosion NDI system requirement: (1) the system must detect corrosion reliably so that significant corrosion is found, and (2) the system must not indicate the presence of corrosion when no corrosion exists (false call) to avoid unnecessary disassembly.

The state-of-art for quantified corrosion detection is about 15 years behind that of cracks. Only recently have methodologies started to evolve that allow structural engineers and NDI engineers to communicate. The basic need for quantification of the NDI system was realized about five to seven years ago, and NDI researchers have been diligently working with structural and corrosion engineers to establish metrics that measure the impact of corrosion damage on the structural performance. Today, there is still no agreement on a standard approach for quantifying the level of corrosion damage, but there are approaches, some of which will be explored further in this paper.

The following sections of the paper discuss how the experience gained in developing and applying reliability assessments to the NDI techniques used to detect cracks in engine components have been applied to NDI techniques use for other aircraft structures.

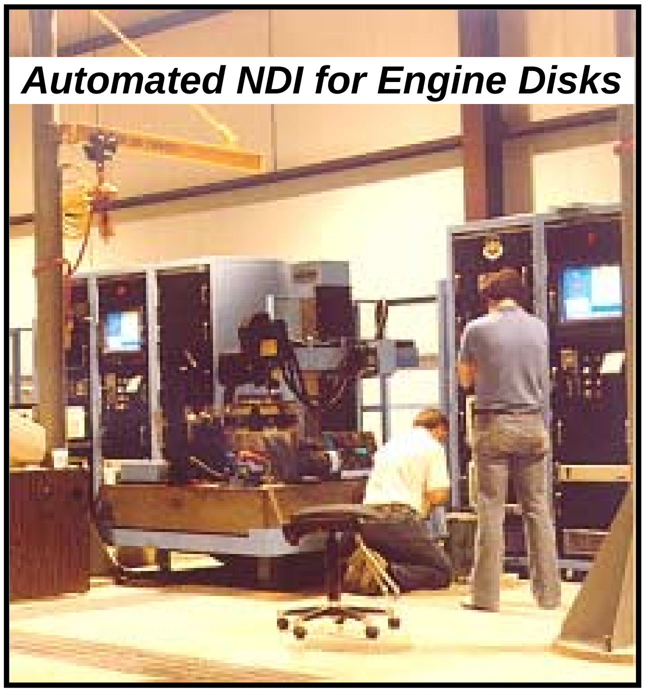


Figure 1 The modular eddy current inspection system (ECIS) is used to conduct surface inspections of USAF engine components.

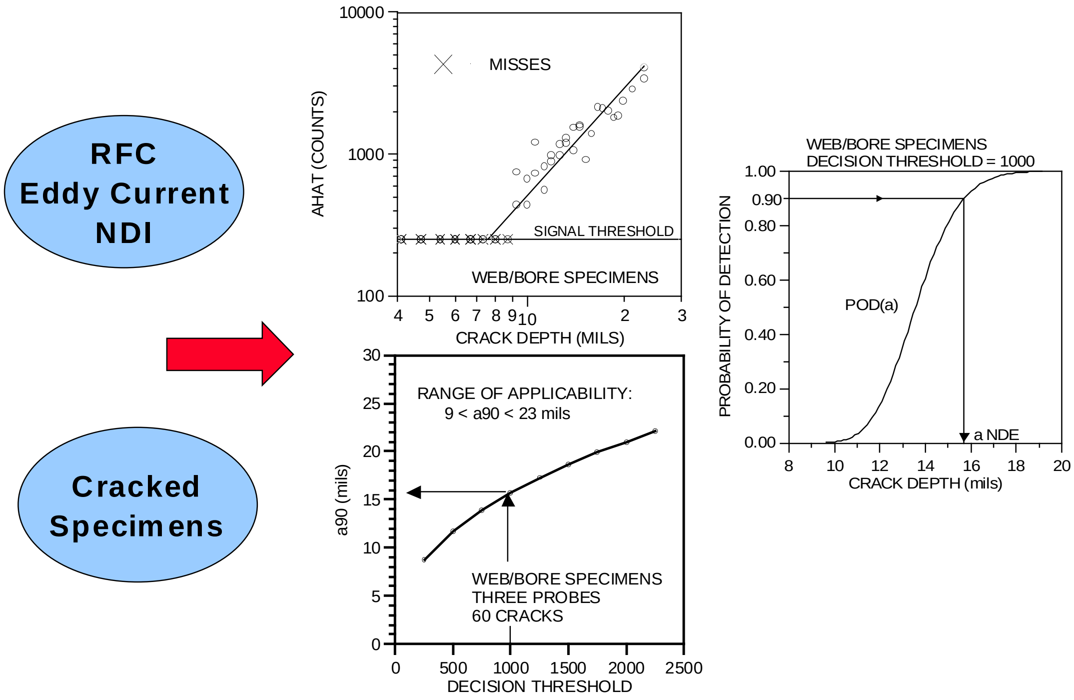


Figure 2 NDE capability improvements – detectability vs. throughput

It is important that NDI system reliability for corrosion detection be quantified. The concept of probability of detection is still an important measure of an inspection capability and the system's reliability. However, multiple opinions exist as to what must be detected. Therefore at this time, a fundamental need exists for a standard methodology that can be used to assess the capability and reliability of an NDI system for detecting corrosion damage. There is a basic and serious need for a standard that is accepted by the structures, corrosion and NDI communities, so that NDI capabilities can be evaluated for their usefulness.

1.3.1 The Metrics used for Corrosion NDI

Some of the controversy concerning standardization of NDI system for corrosion detection results from lack of agreement as to what is corrosion damage and exactly what impact does this corrosion damage have on the integrity of the structure. Fundamentally, any metric that is used to measure the capability and reliability of a NDI system for corrosion detection should have the following characteristics; the metric should: (1) measure the severity of damage, (2) have a structural impact ("effect of defect") and (3) consider the NDI system sensitivity. Because corrosion damage can take many forms (pitting, exfoliation, uniform/generalized, crevice, etc.), the metric has to be defined for the particular type of corrosion that is being experienced. For example, to characterize the effect of crevice corrosion in lap joints and doublers, one might utilize: (a) thickness loss, (b) joint pillowing, (c) surface roughness, or (d) pit depth. As another example, to characterize the effect of intergranular and exfoliation corrosion that occurs around steel fasteners in aluminum structure, one might utilize: (a) radial extent of the damage from the fastener hole or (b) the radial area of the damage from the fastener hole. Figure 3 provides two examples of metrics that may be used to quantify the structural impact of corrosion damage for the cases of a lap joint (crevice corrosion) and a fastener (exfoliation). [8, 9]

1.3.2 Concept of NDI Detection Reliability for Corrosion

The UDRI [8, 10-13] has taken the approach of developing a probability of detection (POD) for corrosion damage analogous to that for cracks. If one is using the thickness loss metric to characterize the level of corrosion damage in a lap joint structure, then the POD curve might look like that shown in Figure 4. As Figure 4 indicates, there are three zones in which structural and corrosion engineers have interest. The most important zone is on the right hand side of the chart where the level of corrosion damage has reached some critical structural or maintenance limit. If the corrosion detection capability is not high (i.e., POD<90%) for this limit, there is a strong possibility that the damage will be missed and serious economical consequences might result. Consider for example, the case where the limit is established for a blending operation – if the inspection capability is not capable of finding this type of damage, and if corrosion is present, the next time that an inspection is conducted, the level of damage may be such that structural elements will need to be replaced.

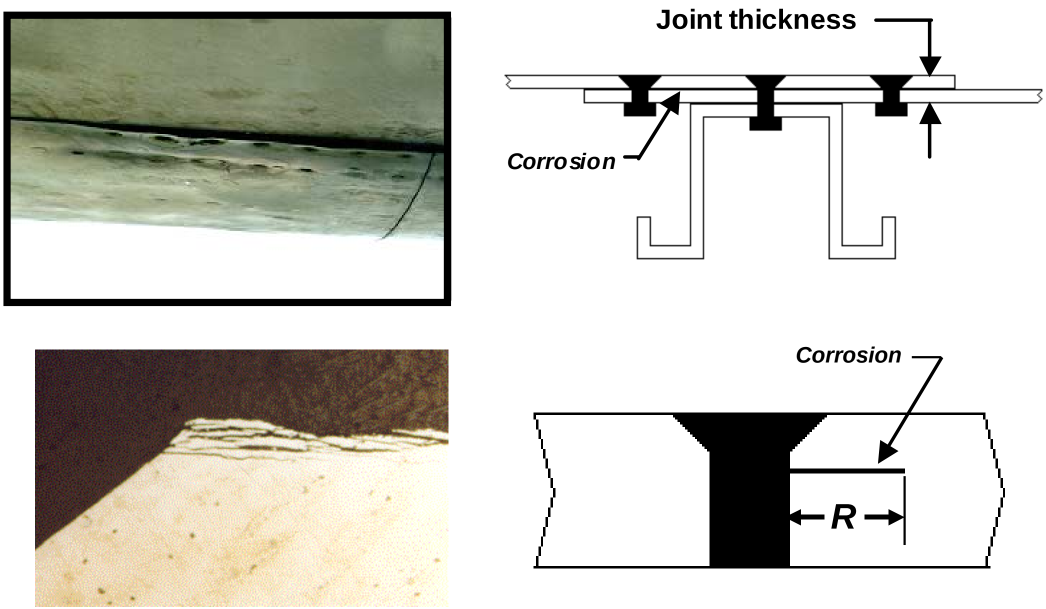


Figure 3 Lap joint and exfoliation Corrosion damage metrics [8, 9]

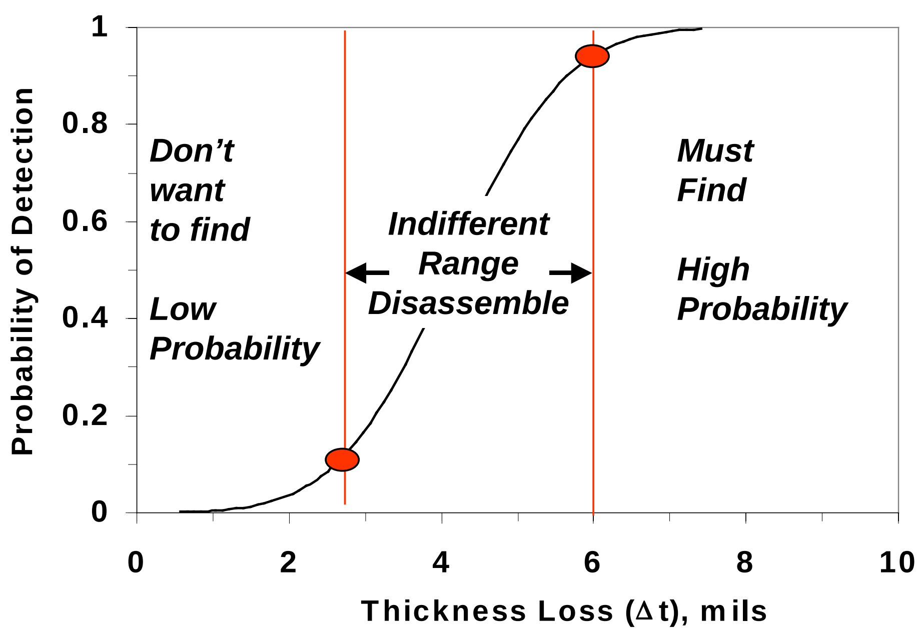


Figure 4 Corrosion POD – for corrosion control of hidden damage

The other two zones on the POD curve illustrate that the inspector also has a chance to find corrosion below the target structural or maintenance limit, although as one can note the probability of detecting corrosion damage below the target limit decreases with thickness loss. The importance of the slope of the POD curve becomes obvious. When the POD curve is not steep, not only does the chance to detect the presence of any corrosion damage decreases, but the chance of disassembling structure without finding corrosion damage also increases.

1.3.3 Developing the Assessment Methodology for Corrosion POD

The UDRI has been working both as a prime and subcontractor supporting the development and demonstration of methodology that can be used to assess the NDI system's capability for detecting corrosion damage [8, 9]. The suggested approach is based on the crack detection approach with the appropriate selection of the metric, and is patterned after the crack detection assessment methodology outlined in Military Handbook 1823. A major requirement of this methodology is that one must standardize the measurement of corrosion by type. Figure 5 diagrams the evaluation process patterned after Military Handbook 1823. The differences between the crack detection methodology and that associated with corrosion detection are shown in bold italic in the figure.

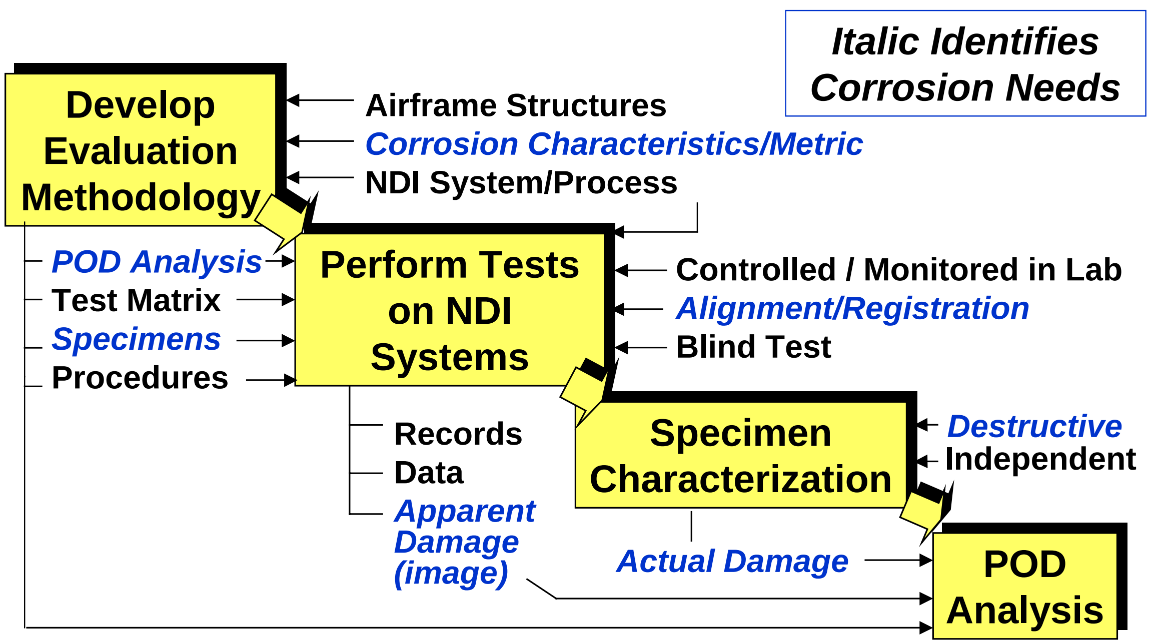


Figure 5 Corrosion NDE/I evaluation process patterned after Military Handbook 1823 for cracks

2. Addressing NDI Needs

2.1 Reducing the Inspection Burden

Over the last 20 years, the developers of NDI systems have taken a systems engineering approach to their designs. Several of these modern systems are modular, in that they: (1) process eddy current and ultrasonic signals depending on the sensor, (2) display the results in the same format, and (3) capture the data digitally so that the inspection results can be archived or reviewed off-line. See Table 1 for examples and Figure 1 for an example engine inspection station. These modern NDI systems are focused on automating the inspection processes as much as possible, since it has been demonstrated that the probability of detection (POD) for cracks is substantially enhanced when automated NDI systems replace manual inspection systems. The advantage of having an NDI system with multiple sensing and data processing capability is that the maintenance organization would eliminate the large number of NDI specialty systems purchased and could standardize their inspector training using the modular system.

Table 1
Examples of Modern NDI systems

System	Manufacturer	Advanced Capability
Retirement for Cause (RFC) for engine disks a.k.a. Eddy Current Inspection System (See Fig. 1)	Veridian Engineer	Highly automated; Digital output Historical database
Mobile Automated Scanner (MAUS) System	Boeing – Phantom Works	Eddy Current and Ultrasonic Sensors, Digital output
Ultra Image System	SAIC - Groton	Eddy Current and Ultrasonic Sensors, Digital output
High Resolution Real-time Digital X-ray System with Amorphous Detectors	GE	Digital Output Digital X-ray mode

2.2 NDE/I Specific Corrosion Needs for Airframe Structures

As discussed above, tremendous costs are incurred when corrosion damage needs to be detected in hidden locations. Limited NDI capability currently exists to detect the level of corrosion damage beyond the first layer of structure and, therefore, the only recourse for maintenance organizations is to disassemble the structure to determine if any damage is present. Where this limited capability exists, we must expand our research efforts to attack second and third layer corrosion to minimize the disassembly of non-corroded structure. Alternately, since there is some probability that corrosion damage will not be present at targeted locations, it behooves the structural community to better anticipate those locations that are actually experiencing corrosion attack, so that those which are not expected to be experiencing corrosion not be disassembled. This approach certainly helps to reduce the inspection burden (and therefore cost) of disassembly and reassembly without finding corrosion damage.

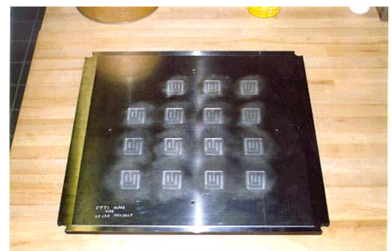
2.3 The Automated Corrosion Detection Program (ACDP)

In 1997, UDRI began a program with the USAF to develop and implement automated corrosion detection, principally in support of the KC-135 aircraft. A primary goal of this program was to develop a standard method of evaluating NDI corrosion detection capability for ultrasonics, eddy current, radiography and thermography. The focused approach was to determine if the assessment methodology of Military Handbook 1823 could be applied to lap joint corrosion with a target of detecting corrosion damage that resulted in less than a 10% thickness loss. A combination of engineered specimens and KC-135 fuselage structural joints were used to assess and demonstrate the assessment methodology and the corrosion capability. These demonstration specimens are illustrated in Figure 6.

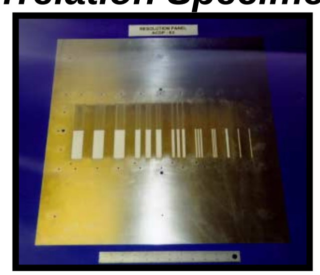
Actual Aircraft Specimens

Fuselage Specimens cut from decommissioned KC-135

Engineered Specimens



Correlation Specimen



Resolution Specimen

Figure 6 Actual and engineered specimens used in automated corrosion detection program

Thickness loss was selected as the metric for crevice corrosion in lap joints and doublers because thickness loss has a direct structural impact on the stresses in the joint. The evaluation followed the Military Handbook 1823 approach adapted to using the corrosion metric (see Figure 5 for an overview of this adaptation). Controlled tests were conducted on ten inspection systems representing four different inspection technologies. Table 2 summarizes the systems evaluated and the technologies on which they were based. Each of these technologies is sensitive to thickness loss either directly or indirectly.

Table 2
List of Corrosion Detection Systems, Techniques and Developers/Participants Evaluated During the Automated Corrosion Detection System

Corrosion Detection System	Technique	Developers / Participants
MOI II	Eddy Current	Physical Research, Inc.
MAUS IV	Eddy Current	Boeing/AFRL
Ultra Image IV	Eddy Current	SAIC
COREX I	Radiography	ARACOR
Reverse Geometry X-Ray ®	Radiography	Digiray/NASA Langley
Thermal Imaging	Thermography	Wayne State University
Line Scan Thermography	Thermography	NASA Langley
PULSE	Ultrasonics	AS&M, Inc.
Ultraspec	Ultrasonics	Southern Research Institute (SRI)
Ultra Image IV	Ultrasonics	SAIC

Except for the radiography techniques, the NDI system sensitivity was to the thickness loss in the top layer of the four-layer lap joint.

Figure 7 summarizes the method used to compare the t-hat (NDI response) with t (the actual thickness). In Figure 7a, each cell (denoted by "C"), defined by the system's special resolution, in the corroded joint represents an opportunity to correlate the corrosion damage of that cell (the average change in thickness) with a measurement of the NDI response (a single t-hat value) in the location of the image (denoted by the point "P"). Because NDI systems summarize their corrosion findings with images that can cover a wide area, it was possible to develop a scheme whereby the detection of corrosion in each cell could be considered independent of detecting corrosion occurring elsewhere. The collection of independent t-hat vs. t responses are collected and summarized in Figure 7b. The POD shown in Figure 7c is derived from regression analysis. The scatter about the mean t-hat response is used to calculate the probability of t-hat exceeding the detection threshold as illustrated in Figure 7b.

2.4. Corrosion Assessment Results

In the Automated Corrosion Detection Program (ACDP), the procedures described in Figure 7 were applied to all the techniques and NDI systems listed in Table 2. Figure 8 presents the results from one evaluated NDI system based on eddy current technology. Hoppe et al. [8] recently summarized the POD curves and t-hat vs. t behaviors for all the systems and techniques. Table 3 summarizes the results obtained from some of the NDI systems that looked the most promising. Both the 90 and 50 percent probability of

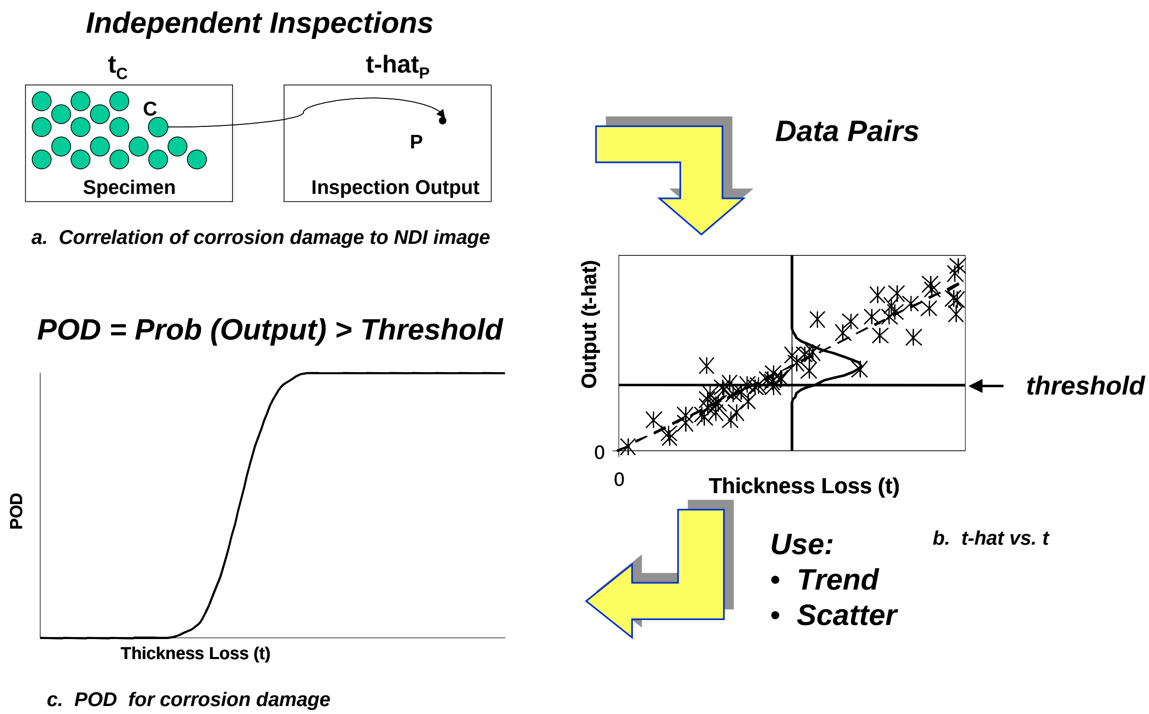


Figure 7 Formal evaluation process and methods used in automated corrosion detection program.

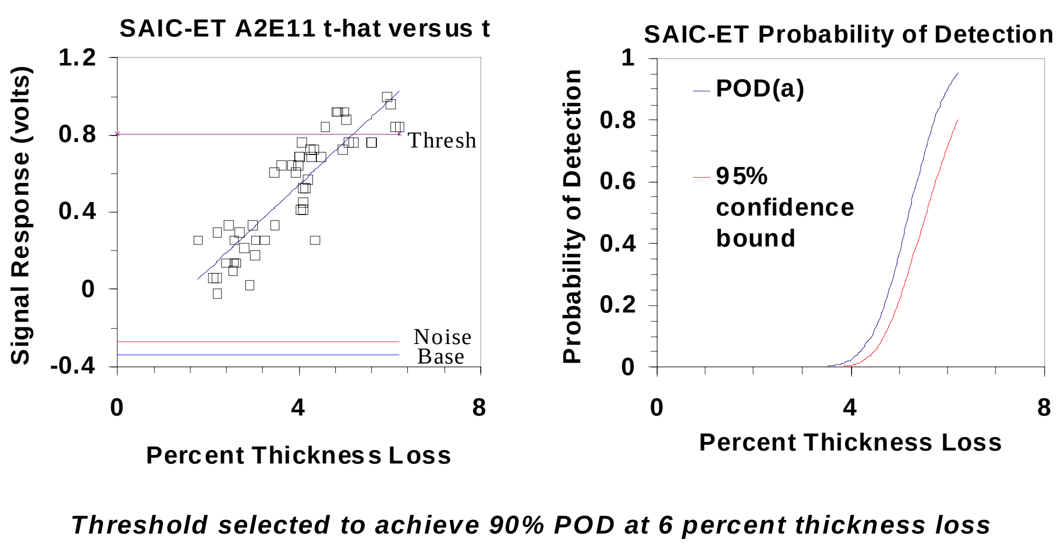


Figure 8 SAIC Ultra Image IV (eddy current) evaluation the using the ACDP method [8].

detection numbers are provided in the table along with the normal distribution parameter sigma and the signal-to-noise ratio. Sigma is related to the steepness of the POD curve, smaller values of sigma imply that the curve is steeper. Noise was estimated from zero thickness loss regions on a specimen without corrosion. Signal-to-noise measures the degree to which the threshold exceeds the noise level. One might note that the signal-to-noise ratios for the eddy current systems are, in some cases, about an order of magnitude larger than that of the ultrasonics systems. For Table 3, comparisons are for only the top thickness (0.063 inch, 1.5 mm) in the four- layer lap joint specimen. One observation made subsequent to the study was that POD comparisons between different systems are difficult due to the differences in cell size required in order to match the resolution of the NDI system.

Table 3
Summary of percent thickness loss parameters and signal to noise ratio

Manufacturer/Technology	Δt_{90} (%)	Δt_{50} (%)	$\Delta t_{90}/\Delta t_{50}$	Sigma	Signal /Noise
SAIC Ultra Image - Eddy Current	6.0	5.2	1.15	0.62	17
Boeing MAUS - Eddy Current	6.0	5.0	1.20	0.77	25
AS&M - Ultrasonics	5.6	4.0	1.40	1.11	2.2
SAIC Ultra Image - Ultrasonic	5.6	4.1	1.37	1.19	3.7
Southern Research Inst. - Ultrasonic	6.0	4.6	1.30	1.27	9.2

2.5 Near-Term Objectives for Evaluating NDI Capability

Because data were collected for both engineered specimens and real aircraft lap joints, one near-term assessment measure being pursued is the construction of diagram such as shown in Figure 9 which allows for a direct comparison between the POD values obtained from a typical laboratory type of experiment (Best) and that obtained from the an aircraft component (Standard). It is suggested that comparisons between the 90/95 POD values for these two types of experiments will lead to better expectations for adapting NDI systems from the laboratory to the field for the same detection problem.

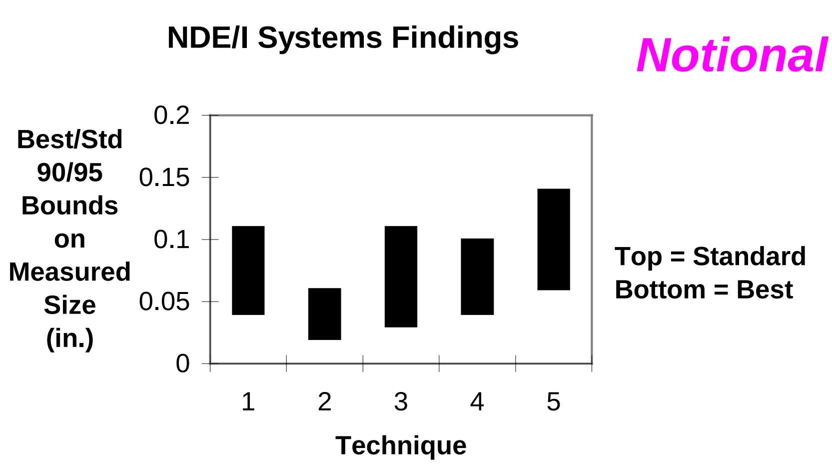


Figure 9 Corrosion detection technology assessment – comparison of NDE/I techniques

From a structural maintenance prospective, it is important to detect the presence of corrosion early enough through fleet surveillance techniques so that cost-effective decisions can be made about controlling the potential for damage. Corrosion detection NDI systems used in surveillance programs must have a high degree of reliability and a low occurrence of false calls to be valuable to the maintenance planner. Schemes in addition to that than shown in Figure 9 have to be devised to provide structural and maintenance engineers with a clear picture of the relationships between detection reliability and false call rates.

3. Concluding Remarks

The uses for NDI systems are widespread and pivotal in the development of cost effective structural maintenance programs. NDI systems provide the basis for surveillance programs that seek to detect anticipated in-service damage before it reaches critical levels in structural components. These systems also provide surveillance for detecting unanticipated in-service damage induced by fatigue loading and/or environmental attack, so that sufficient time is available for developing timely and cost-effective strategies for minimizing fleet wide costs. If hidden damage can be detected by NDI systems, significant cost savings also result since major maintenance costs are involved in disassembling and reassembling airframes, especially when the probability of crack or corrosion damage occurring is low.

To reduce the overall inspection burden, maintenance managers must strive to automate inspection systems and processes. The outcome of automation is to increase the POD, while enhancing the chance that the inspection will be conducted properly. The NDI research community needs to concentrate on increasing the inspection capability to eliminate disassembly of multi-layer structures.

Based on the results of the Automated Corrosion Detection Program [References], a demonstrated method now exists for assessing the reliability of detecting hidden corrosion. This method is based on the method that has been successfully used to assess the capability for a NDI system to detect cracks in either airframe or engine structures. This demonstrated method provides independent quantitative measures of NDI system performance/capability (POD, false calls) in terms of the selected corrosion metric. The Military Handbook 1823 based method assures that a given level/type of corrosion damage is below a target limit, while reducing the number of false calls, thus reducing the cost of disassembly when no corrosion is present. What is now needed to complement the assessment method is a clear definition of economical maintenance or structural limits for the allowable levels of corrosion damage in order to set the POD conditions to meet the target limit.

4. References

1. Lincoln, J. W., “The Findings of the Aging Aircraft Technologies Team”, presented at the 1998 USAF Aircraft Structural Integrity Program (ASIP) Conference, Dec. 1998, San Antonio, TX
2. Grimsley, F.M., Lincoln, J.W., and Zeigler, M.L., “USAF Strategy for Aging Aircraft Structures Research and Development”, presented at the NATO Applied Vehicle Technology (AVT) Panel Specialist’s meeting on Life Management Techniques for Aging Aircraft, 8-11 October, Manchester, UK

3. Cooke, G., Cooke Jr., G., Kawanishi George, Higginbottom, T., Miller, P., and Wacksman, S., "A Study to Determine the Annual Direct Cost of Corrosion Maintenance for Weapon Systems and Equipment in the United States Air Force", Final Report on Contract No. F09603-95-D-0053, Delivery Order 0006, CDRL No A001**,** NCI Information Systems, Inc, 6 February 1998
4. Berens, A.P., "What Metric Should be Used to Measure Intrinsic Capability? Is it Only POD and FCP – or is Theoretical?", *Materials Evaluation*, pp. 42, January,
5. Rummel, W.D., and Matzkanin, G.A., *Nondestructive Evaluation (NDE) Capabilities Data Book*, Nondestructive Testing Information Analysis Center (NTIAC), Texas Research Institute Austin, Inc., November, 1997. " His handbook developed for the NTIAC …
6. Hoppe, W.C., Schehl, N., Buynak, C., Gallagher, J.P., Berens, A.P., **"**From Cracks to Corrosion: The Evolution of a Probability of Detection Approach that Quantifies Corrosion Damage in Aging Aircraft", Proceedings of the 2000 USAF Aircraft Structural Integrity Program (ASIP) Conference, Dec. 2000, San Antonio, TX
7. Military Handbook 1823, Nondestructive Evaluation System Reliability Assessment (1994).
8. Hoppe, W.C., Pierce, J. L., Schehl, N., Ko, R., and Buchanan, D., **"**Results from the Hidden Corrosion Detection Evaluation on the Automated Corrosion Detection Program – Probability of Detection", Proceedings of 5th NASA/FAA/DoD Aging Aircraft Conference, 10-13 September, 2001, Orlando, FL
9. Schehl, N., Hoppe, W.C., Berens, A.P., and Ko, R., "Evaluation of an Intergranular Corrosion Detection Method on the Structural Repair of Aging Aircraft Program" Proceedings of 5th NASA/FAA/DoD Aging Aircraft Conference, 10-13 September, 2001, Orlando, FL
10. Hoppe, W. C., "Hidden Corrosion Detection Technology Assessment," Second Joint NASA/ FAA/DoD Conference on Aging Aircraft, NASA/CP-1999-208982, (1998), pp 349 – 358.
11. Hoppe, W. C., Schehl, N., Finch, J., "Specimen Development and Selection for Hidden Corrosion Detection Technology Assessment", Evolving and Revolutionary Technologies for the New Millennium, 44th International SAMPE Symposium/Exhibition, May 23-27, 1999 pp. 820 – 833.
12. Hoppe, W. C. and Pierce, J.L., "The Evaluation of Hidden Corrosion Detection Technologies on the Automated Corrosion Detection Program", Fourth Joint DoD/FAA/NASA Conference on Aging Aircraft – Proceedings, 15-18 May 2000.
13. Hoppe, W.C., Schehl, N., Buynak, C., "An Evaluation Methodology for the Assessment of Corrosion Detection", summary to be published in ASNT Fall 2000 Conference Proceedings

Advances in Structural Integrity Analysis Methods for Aging Metallic Airframe Structures with Local Damage

James H. Starnes*, Jr., James C. Newman, Jr., Charles E. Harris,
Robert S. Piascik, Richard D. Young and Cheryl A. Rose
*NASA Langley Research Center
Mail Stop 190
Hampton, VA 23681-2199, USA

ABSTRACT

Analysis methodologies for predicting fatigue-crack growth from rivet holes in panels subjected to cyclic loads and for predicting the residual strength of aluminum fuselage structures with cracks and subjected to combined internal pressure and mechanical loads are described. The fatigue-crack growth analysis methodology is based on small-crack theory and a plasticity induced crack-closure model, and the effect of a corrosive environment on crack-growth rate is included. The residual strength analysis methodology is based on the critical crack-tip-opening-angle fracture criterion that characterizes the fracture behavior of a material of interest, and a geometric and material nonlinear finite element shell analysis code that performs the structural analysis of the fuselage structure of interest. The methodologies have been verified experimentally for structures ranging from laboratory coupons to full-scale structural components. Analytical and experimental results based on these methodologies are described and compared for laboratory coupons and flat panels, small-scale pressurized shells, and full-scale curved stiffened panels. The residual strength analysis methodology is sufficiently general to include the effects of multiple-site damage on structural behavior.

KEYWORDS: fatigue-crack growth, critical crack-tip-opening-angle fracture criterion, nonlinear structural analysis, residual strength analysis, test-analysis correlation.

INTRODUCTION

Modern design philosophies for transport aircraft fuselage structures require that these structures retain adequate structural integrity when discrete source damage or fatigue cracks are present. As economic factors encourage the use of commercial and military transport aircraft beyond their original design requirements, it is important to develop methods that accurately predict the fatigue life and the residual strength of fuselage structures with cracks. During the past decade, research conducted at NASA Langley Research Center has resulted in a fatigue-crack growth analysis methodology for aircraft structures subjected to cyclic loads, and a residual strength analysis methodology for aluminum fuselage structures with cracks and subjected to combined internal pressure and mechanical loads. The fatigue-crack growth analysis methodology is based on small-crack theory and a plasticity induced crack-closure model (Ref. 1). The residual strength analysis methodology is based on elastic-plastic fracture mechanics and nonlinear structural analyses (Ref. 2), and is general enough to include the effects of multiple-site damage. This methodology includes a critical crack-tip-opening-angle (CTOA) fracture criterion (e.g., Refs. 3 and 4), and the STAGS (STructural Analysis of General Shells) nonlinear finite element shell analysis code (Ref. 5). The critical CTOA criterion assumes that stable crack growth will occur when the local crack opening angle reaches a critical value, and STAGS is used with the critical CTOA criterion to perform residual strength analyses for structures with geometric and material nonlinear characteristics.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

The present paper describes the fatigue-crack growth analysis and the residual strength analysis methodologies developed at NASA Langley Research Center, and presents results from several studies (Refs. 6-11) that have applied these methodologies to test specimens ranging in complexity from small laboratory coupon specimens to full-scale 2024-T3 stiffened fuselage panels. The fracture parameters used to predict the residual strength behavior of the more complex test specimens were obtained from the small laboratory coupon specimens. Results are presented for unstiffened and stiffened flat panels, small-scale unstiffened shells, and full-scale curved stiffened fuselage panels.

FATIGUE CRACK GROWTH ANALYSIS METHODOLOGY

The fatigue life prediction methodology developed at NASA Langley Research Center is based on ‘small-crack theory’ and a plasticity induced crack-closure model. 'Small-crack theory' is the treatment of fatigue as a crack propagation process from a micro-defect (or crack) to failure. The propagation of small fatigue cracks from surface defects (5 μm to 10 μm) constitutes a large percentage (50% to 90%) of the total fatigue life of structural components. Thus, accurate prediction of small-crack growth rate is required for damage-tolerance-based life predictions. The fatigue life prediction methodology is described in Ref. 6 and summarized in this section. First, large-crack fatigue-crack growth rate data from testing small laboratory coupon specimens are used to develop the relationship between the effective stress-intensity factor range (ΔK_{eff}) and crack-growth rate for a constant-amplitude loading condition. The effective stress-intensity factor range (ΔK_{eff}) accounts for plasticity-induced crack closure, and is used to define the closure-free ΔK-rate relation. A constraint factor, α, which accounts for three-dimensional state-of-stress effects, is used as a fitting parameter to correlate crack-growth rate data with ΔK_{eff} for constant-amplitude loading conditions with different stress ratios. Then, the ΔK_{eff}-rate relationship or curve in the near-threshold regime is modified to fit measured small-crack growth-rate behavior and fatigue endurance limits. The resulting ΔK_{eff}-rate relationship is used as input to the life-prediction code FASTRAN-II (Ref. 1) to predict the total fatigue life of a structural component based on crack propagation from micro-structural features. A crack is assumed to initiate and grow from a micro-structural feature (e.g., inclusion particle, void, corrosion pit) on the first cycle (e.g., Ref. 12). The crack-closure model and ΔK_{eff}-rate curve are used to predict crack growth from the initial crack size to failure.

Prediction of Crack Growth and Fatigue Life of 4340 Steel

Comparisons between small- and large-crack results have been made for 4340 steel (Refs. 13 and 14). A baseline effective stress-intensity factor range versus crack-growth rate curve for the material was developed in Ref. 13, and this curve was used to predict small-crack growth rate behavior from extremely small initial crack sizes on the notched specimens in Ref. 14. In Ref. 14, large-crack results were obtained from middle-crack tension specimens, and small-crack data were obtained from single-edge-notch tension specimens. The plastic-replica method was used to measure the growth of small cracks. Examination of the initiation sites for 35 fatigue cracks gave information on the distribution of crack-initiation site dimensions. The most dominant crack-initiation site particle was a spherical (calcium-aluminate) particle. The mean defect was about 13-μm in radius. Over 80% of all defects were represented by upper and lower bounds for the defect sizes of 8- and 30-μm in radius.

A comparison of small- and large-crack data for 4340 steel is shown in Figure 1(a). The symbols represent small surface-crack data from the single-edge-notch tension specimens. The dashed-dot curve represents the large-crack data obtained from middle-crack tension specimens. The small cracks were measured in the thickness or a-direction and large cracks were measured in the width or c-direction. The small- and large-crack data agree quite well. The dashed curve is the ΔK_{eff}-rate curve from Ref. 13, determined from middle-crack tension specimen data. The constraint factor α is 2.5 for rates less than 5E-4 mm/cycle. The solid curves are the predicted results from FASTRAN (Ref. 1) with either an initial semi-circular surface crack of 8- or 30-μm. All predictions start on the ΔK_{eff}-rate curve because the initial crack is assumed to be fully open on the first cycle. Because the effective stress-intensity factor curve is near to the large-crack curve, small-crack effects are weak. The predicted results for the largest defect size rapidly approach the large-crack behavior. The predicted results for the smallest defect size decrease very rapidly and then increase very rapidly to the large-crack curve. This behavior is due to the crack-closure transient and the shape of the ΔK_{eff}-rate curve at the lower rates.

The results from Ref. 14, shown in Fig. 1(a), were applied by Everett (Ref. 15) to predict the response of fatigue tests on 4340 steel (thickness B = 3.2 mm) using a specimen of width w = 12.7 mm and a single open hole with radius r = 3.2 mm. The material used in Ref. 15 had the same strength level as the material tested in Ref. 14, but the specimens were thinner and were taken from a different heat of material. However, it was assumed that the large-crack data and inclusion-particle sizes would be the same. A small-crack effective threshold, $(\Delta K_{eff})_{th}$, of 3.2 MPa√m was used to predict the endurance limits or the applied stress level where the initial defect would not grow. Results of constant-amplitude fatigue tests with a stress ratio R = 0 are shown by the symbols in Figure 1(b) for open-hole specimens. The maximum stress in the spectrum is plotted versus the number of cycles to failure. Predictions of total fatigue life were made using the FASTRAN code (Ref. 1) by calculating the number of cycles necessary to grow a crack from 8- and 30-μm initial semi-circular surface cracks located at the center of the hole. Near the endurance limit, the analysis results bound the test data quite well, but predict slightly longer lives at the highest stress levels for the tests. The defect size had more influence on life in the endurance limit regime than for the higher stress levels.

The Effect of Corrosion on Fatigue Life

Constant-amplitude fatigue-crack growth experiments were conducted (Ref. 7) in laboratory air and deaerated 1% NaCl environments to determine the effects of a corrosive environment on the ΔK_{eff}-rate relationship for 2024-T3 aluminum alloy. Extended compact-tension or eccentrically loaded single-edge cracked-tension specimens were tested for different R ratios ranging from 0.05 to 0.80. Small surface and corner crack growth rates and stress intensity factors were calculated assuming uniform semicircular crack geometry. The results of these tests are shown in Fig. 2(a) and indicate that the NaCl environment can accelerate the crack-growth rate. The fatigue cracks initiated at a corrosion pit approximately 10 μm in depth located at the root of the blunt notch in the specimens. Following initiation, the crack propagated along a transgranular semicircular shaped crack path.

A series of tests were conducted by Furuta, et al. (Ref. 16) to study the fatigue behavior of 2024-T3 (Alclad) countersink-riveted lap-joint panels exposed to a room-temperature laboratory air environment and a 3.5% NaCl corrosive salt-water environment. A typical test of a panel with two rows of rivets was conducted at a constant-amplitude loading condition with R = 0.125 and maximum stress S_{max} = 96 MPa to simulate fuselage skin stress conditions. A fastener interference level was not used in any calculations, and fastener bending effects were not included. The two-rivet row had a 50% rivet and by-pass stress. The results shown in Fig. 2(b) indicate that the fatigue life of the panels exposed to salt water (square symbol) is reduced by a factor of about 1/2 or 1/3 compared to the fatigue life in ambient laboratory air (circle symbol). The FASTRAN (Ref. 1) predictions for salt water and laboratory air environments are in excellent agreement with the experimental results. The fracture mechanics based calculations assumed a corner crack in a neat-fit riveted-loaded straight-shank hole (rivet fit-up and interference fit stresses are assumed small). The 6 μm radius equivalent initial flaw size used for each FASTRAN prediction is consistent with laboratory observations; 6 μm radius constituent particles and corrosion pits are observed at small crack initiation sites in fatigue test coupons exposed to laboratory air and salt water, respectively (Ref. 7). The predicted fatigue lives shown in Fig. 2(b) agree well with the test results.

RESIDUAL STRENGTH ANALYSIS METHODOLOGY

The residual strength analysis methodology developed at NASA Langley Research Center is based on the critical crack-tip-opening-angle (CTOA) fracture criterion (e.g., Refs. 3 and 4) and the STAGS nonlinear shell analysis code (Ref. 5). This analysis methodology accounts for both material and geometric nonlinear behavioral characteristics of the materials and structures of interest. The following sections describe the CTOA fracture criterion, and the geometric and material nonlinear finite element shell analysis code STAGS used in the residual strength analysis methodology.

CTOA Fracture Criterion and Plane-Strain-Core Height

The critical CTOA fracture criterion is supported by experimental measurements of the critical angle during stable growth and has been shown to be well suited for modeling stable crack growth in ductile materials and for predicting the onset of unstable crack growth in fracture analyses conducted using elastic-plastic finite element methods (e.g., Refs. 3 and 4). The CTOA is defined as the angle made by the upper crack surface, the crack tip, and the lower crack surface, evaluated at a fixed distance from the moving crack tip, as illustrated in Fig. 3. A fixed distance of 1 mm is used in the present paper to evaluate the critical CTOA value (e.g., Ref. 10). The CTOA criterion assumes that crack extension will occur when the CTOA reaches a critical value, $CTOA_{cr}$, and that the $CTOA_{cr}$ will remain constant as the crack extends. The critical CTOA value can be obtained experimentally using a photographic technique (Ref. 4), but significant scatter is usually present in the measurements. A better method of determining the critical CTOA value is to simulate the fracture behavior of a laboratory specimen with a three-dimensional elastic-plastic finite element analysis and determine the angle that best describes the experimentally observed fracture behavior.

An example demonstrating the use of a three-dimensional, elastic-plastic finite element analysis to determine the critical CTOA for three different thicknesses of 2024-T3 aluminum alloy, is shown in Fig. 4, where the critical value of CTOA for each thickness is represented by the symbol Ψ_c. Results of compact tension (C(T)) laboratory tests are shown in Fig. 4 for 2024-T3 aluminum-alloy sheets with the cracks parallel to the sheet rolling direction. The compact-tension test specimens are 152-mm wide with an initial crack length a = 61 mm. Data for three difference sheet thicknesses are shown on the figure. Analytical results from the geometrically linear elastic-plastic three-dimensional finite element code ZIP3D (Ref. 17) are also shown on the figures, and the critical CTOA values represent the best fit with the test data.

A three-dimensional finite element analysis code, such as ZIP3D, requires only the critical CTOA to predict the fracture behavior of thin ductile materials, since three-dimensional constraint effects that develop at the local crack tip (Ref. 18) are explicitly accounted for in the model. In a finite element shell analysis code, which typically uses two-dimensional plane-stress elements, a modeling approximation is required to simulate the actual state of stress near the crack tip. The modeling approximation used in the present methodology is to introduce a thin strip of plane-strain elements in a region on each side of the crack line. The width of the plane-strain region on each side of the crack line is commonly referred to as the plane-strain-core height, h_c, and is approximately equal to the thickness of the specimen. This strip of plane-strain elements has plane-strain conditions, while the remainder of the model has plane-stress conditions, as illustrated in Figure 5. The plane-strain-core height h_c is determined from analyses using the two-dimensional ZIP2D code (Ref.19), and the critical angle determined from the ZIP3D analysis is used to determine the value of h_c that makes the ZIP2D analysis results consistent with the ZIP3D results, as shown in Fig. 5.

To confirm that fracture parameters determined in the manner described above could be applied in a STAGS analysis, geometrically nonlinear elastic-plastic analyses were conducted to predict the fracture response in the T-L orientation of 1.6-mm-thick, 2024-T3 compact-tension (C(T)), and middle-crack tension (M(T)) panels, with and without buckling constraints. The critical CTOA used in the analyses was equal to 5.0º, and the plane-strain-core height, h_c = 1 mm. The experimental and predicted crack extension results for the C(T) and M(T) panels are shown in Fig. 6 as a function of the applied load. These results verify the selection of $CTOA_{cr} = 5.0^o$ and h_c = 1mm for the material and indicate that the analyses with STAGS accurately predict the reduction in strength of the panels caused by the geometrically nonlinear effect of panel buckling.

Nonlinear Structural Analysis Code

The STAGS (STructural Analysis of General Shells) nonlinear shell analysis code (Ref. 5) is used in the residual strength analysis methodology to predict the response and residual strength of unstiffened aluminum shells and stiffened aluminum fuselage panels with longitudinal cracks. STAGS is a finite element code for analyzing general shells and includes the effects of geometric and material nonlinearities in the analysis. STAGS can perform crack-propagation analyses, and can represent the effects of crack growth on nonlinear shell response. A nodal release method and a load relaxation technique are used to extend a crack while the shell is in a nonlinear equilibrium state. The changes in the stiffness matrix and the internal load distribution that occur during crack growth are accounted for in the analysis, and the nonlinear coupling between internal forces and in- and out-of-plane displacement gradients that occurs in a shell are properly represented.

Finite element models are constructed using a collection of two-node beam elements, two-node fastener elements, and four-node plate elements. Each node of the models has six degrees of freedom. Structural components including skins, stringers, frames, tear straps, and stringer clips are modeled by plate elements to represent accurately the cross sectional shapes of all components. Riveted connections between structural components are modeled using beam elements, or fastener elements in the region close to a crack, where fastener flexibility is thought to affect load transfer. The fastener elements represent the offsets of the joined components with rigid links that are connected by spring elements with six degrees of freedom. The spring elements can model elastic-plastic behavior, and fastener breakage if a prescribed fastener strength is exceeded. An example of fastener modeling details is given in Ref. 20. For conditions where deformation of the model would cause interpenetration of elements, the general contact capability in STAGS is invoked to prevent such element interpenetration from occurring. To simulate the experimental conditions for the specimens considered in the present paper, the finite element models include the load introduction hardware and replicate the loading conditions as applied in the experiments.

PANEL AND SHELL TEST AND ANALYSIS RESULTS

The residual strength analysis methodology described in the present paper has been experimentally verified for structures ranging in size from laboratory coupons to full-scale structural components. Results for small-scale pressurized shells, flat stiffened panels, and curved stiffened panels are presented in this section. Analysis results were obtained using values of $CTOA_{cr}$ and h_c that were determined for each material, sheet thickness, and crack orientation by correlating elastic-plastic finite element analyses and experimental results for small laboratory specimens.

Pressurized Cylindrical Shell Tests

Cylindrical shells were fabricated from 1-mm-thick 2024-T3 aluminum-alloy sheet, with the rolling direction orientated circumferentially. The shells were 99-cm long, 45.7 cm in diameter, and had a 3.8-cm-wide double lap splice with 1-mm-thick splice plates and a single row of rivets on each side of the splice. Each shell had a longitudinal crack that was simulated by a 0.025-mm-wide saw cut at the specimen mid-length, diametrically opposite to the lap-splice. Specimens with initial crack lengths of 50.8, 76.2, and 101.6 mm were loaded by internal pressure until failure occurred (Ref. 8). The crack length extension was recorded using crack wire gages.

The experimental measurements and the STAGS finite element predictions for the pressurized cylindrical shells are shown in Figure 7. Analysis predictions were made using $CTOA_{cr}$ = 5.6º, and h_c = 1 mm. For all crack lengths, the analyses predicted the maximum pressure to within 4% of the measured values, but tended to overpredict the pressure required to initiate crack growth. The use of saw cuts would generally cause the analysis to underpredict the pressure required to initiate the crack growth, since a saw cut would require higher loads to initiate crack growth than a sharp fatigue crack. One possible explanation for the overprediction of the crack growth initiation pressure could be that the intense crack-tip deformations might have caused the crack wire gages to register crack growth before the growth actually occurred.

Flat Stiffened Panel Tests

Fracture tests were conducted on 1016-mm-wide, 1.6-mm-thick, 2024-T3 aluminum alloy, flat, stiffened panels (Ref. 21). The stiffeners were made from 7075-T3 aluminum alloy and riveted to the specimens. The stiffeners were 40.6-mm wide and placed on both sides of the specimen, as shown in Figure 8. The crack configuration consisted of a single 203-mm-long center crack with an array of twelve 4.7-mm-diameter holes on either side of the of the center crack. Specimens with and without MSD were tested. The MSD crack length was 1.27 mm from the edge of the hole. The specimens were tested without guide plates to allow out-of-plane displacements.

The experimental and analytical results for the stiffened panels with a single center crack and without and with MSD are shown in Figures 9(a) and 9(b), respectively. Predictions of the fracture behavior were conducted with STAGS using a critical CTOA value of 5.4^{o}, and a plane-strain-core height of 2 mm. The results indicate that the analysis methodology represents the behavior of these specimens very well.

Curved Stiffened Panel Tests

Three stringer- and frame-stiffened aluminum fuselage panels with longitudinal cracks were tested and analyzed. These curved stiffened panels are referred to as Panels ASIP1, ASIP2, and ASIP3 in the present paper and are shown in Figure 10 prior to testing. The panels all had 3.09-m radii and initial crack lengths of 254 mm. Panels ASIP1 (Fig. 10(a)) and ASIP3 (Fig. 10(b)) had the initial crack located at the panel centerline, and panel ASIP2 (Fig. 10(c)) had the initial crack along a row of fasteners in a lap splice at the second stiffener. Panel ASIP2 also had MSD cracks along the fastener holes near the lead crack as shown in Figure 10(d). Panels ASIP1 and ASIP2 were 1.83-m long and panel ASIP3 was 3.5-m long. Additional details of the panels are given in Refs. 9 and 11. Panels ASIP1 and ASIP2 were tested in a pressure-box test machine and were subjected to combined internal pressure and mechanical hoop and axial tension loads. Panel ASIP3 was tested in the COLTS combined loads test facility located at NASA Langley Research Center. The panel was attached to a D-box test fixture, and subjected to internal pressure, axial compression and torsion loads. Details of the COLTS test facility and D-box test fixture for ASIP3, and the test and analysis results for ASIP3 are given in Ref. 11.

The test results for panel ASIP1 indicate that the panel arrested the propagating crack at the tear straps. As the internal pressure was increased, each end of the skin crack extended in the longitudinal direction until it intercepted an adjacent tear strap. The crack growth behavior was symmetric with respect to the central frame. The experimental and predicted crack extension results are compared in Fig. 11 as a function of pressure. Predictions of the fracture behavior were conducted with the STAGS analysis code using $CTOA_{cr}$ = 5.0º, and h_c = 1 mm. These results indicate good agreement in the pressure corresponding to crack extension values up to 25.4 mm, but a discrepancy in the predicted and observed responses occurs for crack extension greater than 25.4 mm. In the experiment, after 25.4 mm of crack extension, very small increases in pressure cause significant amounts of crack extension, while the analysis indicates that larger increases in pressure are required for additional crack extension. The values of the pressure for the test and the analysis differ by only 1% for 25.4 mm. of crack extension, but differ by 10% for 50.8 mm of crack extension.

The test results for panel ASIP2 indicate that the panel failed as a result of MSD crack link up. At a certain pressure magnitude, the lead crack suddenly extended on each end of the crack, and linked up with the series of MSD cracks ahead of the lead crack. The crack extended in the longitudinal direction in a fast fracture mode, and extended over the entire panel length in an instant. The crack growth behavior was symmetric with respect to the central frame. A typical solution with crack growth in the lead crack and the MSD cracks is shown in Fig. 12. The contour plot of the hoop stress in the region around the crack tip region, shown in Fig. 12(a), indicates the high stress regions near the crack tips of the lead crack and the MSD cracks. A contour plot of the plastic strains in the hoop direction is shown in Fig. 12(b) which indicates that there are regions of plastic deformation emanating from the lead crack and from the MSD crack tips, and that for the solution shown, the plastic zones from the lead crack and the first MSD crack have coalesced. The deformed shape shown in these plots indicates that the deformation on the side of the crack attached to the stiffener is much smaller than the deformation on the other side of the crack, demonstrating that the crack is not tearing due to a symmetric loading condition. The asymmetric loading could promote curvilinear crack growth, but it is assumed in the analysis that interaction between the lead crack and the MSD cracks will cause self-similar crack growth. The opening of the MSD cracks is also evident in the deformed shapes. The crack extension response from the analysis and the experiment are compared in Fig. 13 as a function of pressure. Predictions of the fracture behavior were obtained using $CTOA_{cr} = 5.0^o$, and $h_c = 1$ mm. The breaks in the solid curve indicate locations where the lead crack linked up with the MSD cracks to create a discontinuity in the length of the lead crack. Thus, the analysis predicts fast fracture and link-up at a pressure that is 11% greater than what was observed in the experiment. For comparison purposes, the predicted response of panel ASIP1 is also included in Fig. 13. The difference in the predicted stability of the tearing response of these two panels is caused by the interaction of the lead crack and the MSD cracks in panel ASIP2.

CONCLUDING REMARKS

A plasticity induced crack-closure model has been used to correlate large-crack growth-rate data for a constant-amplitude loading condition in ambient and corrosive environments. Comparisons made between measured and predicted small-crack growth rates indicate that the closure model predicts the trends of the test results. Using the closure model and some microstructural features, such as inclusion-particle sizes, a fatigue-life prediction method has been demonstrated for materials of interest. Predicted fatigue lives for notched specimens compare well with test data under constant-amplitude and spectrum loading. It is likely that a panel with a large number of fastener holes and other areas of stress concentration may have a critical size inclusion particle located at one of these sites. Thus, using the largest material defect for a material of interest, such as the 30-μm defect, would produce a somewhat conservative but reliable life prediction. If there are manufacturing defects larger than the material defects, they would control the fatigue lives of components subjected to cyclic loading conditions.

A residual strength analysis methodology for aircraft aluminum fuselage structures with cracks and subjected to combined internal pressure and mechanical loads has been used to predict the crack propagation characteristics of structures ranging from laboratory coupons to full-scale structural components. The methodology is based on the critical crack-tip-opening-angle fracture criterion that characterizes the fracture behavior of a material of interest, and a geometric and material nonlinear finite element shell analysis code that performs the structural analysis of the fuselage structure of interest. The methodology is sufficiently general to include the effects of multiple-site damage on structural behavior. Analytical results based on this methodology are compared with experimental results for aluminum-alloy laboratory coupons and flat panels, small-scale pressurized shells, and full-scale curved stiffened panels. The analytical and experimental results compare well.

The results of residual strength analyses indicate that elastic-plastic effects in a thin sheet can be effectively represented by a critical crack-tip-opening-angle fracture criterion. The results also indicate that geometric and material nonlinear structural analyses can accurately represent the changes in internal load distributions, local stress and displacement gradients, and crack growth behavior in stiffened fuselage shells with long cracks and subjected to combined internal pressure and mechanical loads. In addition, nonlinear fracture analysis and structural analysis methods provide higher fidelity results than traditional linear-elastic engineering analysis approximations for structures with significant plastic yielding and nonlinear out-of-plane deformations associated with internal pressure loads.

REFERENCES

1. Newman, J. C., Jr., “FASTRAN-II - A Fatigue Crack Growth Structural Analysis Program,” NASA TM 104159, 1992.
2. Harris, C. E., Newman, J. C., Jr., Piascik, R. S., and Starnes, J. H., Jr., "Analytical Methodology for predicting the Onset of Widespread Fatigue Damage in Fuselage Structure," *Journal of Aircraft,* Vol. 35, No. 2, 1998, pp. 307-317.
3. Newman, J.C., Jr., “An Elastic-Plastic Finite Element Analysis of Crack Initiation, Stable Crack Growth, and Instability,” ASTM STP 833, 1984, pp. 93-117.
4. Dawicke, D. S., Sutton, M. A., Newman, J. C., Jr., and Bigelow, C. A., “Measurement and Analysis of Critical CTOA for an Aluminum Alloy Sheet,” NASA TM-109024, September 1993.
5. Rankin, C. C., Brogan, F. A., Loden, W. A., and Cabiness, H. D., "STAGS User Manual, Version 3.0," Lockheed Martin Missiles and Space Co., Inc., Advanced Technology Center, Report LMMS P032594, June 1998.
6. Newman, J. C., Jr., "Advances in Fatigue and Fracture Mechanics Analyses for Metallic Aircraft Structures,"*Structural Integrity for the Next Millennium, Vol. 1*, Proceedings of the 20th Symposium of the International Committee on Aeronautical Fatigue, J. L. Rudd and R. M. Bader, eds., Bellingham, WA, July 14-16, 1999, pp. 3-40.
7. Piascik, R. S., and Willard, S. A., “The Growth of Small Corrosion Fatigue Cracks in Alloy 2024,” *Fatigue and Fracture of Engineering Materials and Structures*, Vol. 17, No. 11, 1994, pp. 1247-1259.
8. Starnes, J. H., Jr., and Rose, C. A., "Stable Tearing and Buckling Response of Unstiffened Aluminum Shells with Long Cracks," Proceedings of the Second Joint NASA/FAA/DOD Conference on Aging Aircraft, Williamsburg, VA, August 31-September 3, 1998. NASA/CP-1999-208982/Part 1, January 1999, pp. 610-626.
9. Young, R. D., Rouse, M., Ambur, D. R., and Starnes, J. H., Jr., "Residual Strength Pressure Tests and Nonlinear Analyses of Stringer- and Frame-Stiffened Aluminum Fuselage Panels with Longitudinal Cracks," Proceedings of the Second Joint NASA/FAA/DOD Conference on Aging Aircraft, Williamsburg, VA, August 31-September 3, 1998. NASA/CP-1999-208982/Part 1, January 1999, pp. 408-426.

10. Dawicke, D. S., Newman, J. C., Jr., Starnes, J. H., Jr., Rose, C. A.; Young, R. D., and Seshadri, B. R., "Residual Strength Analysis Methodology: Laboratory Coupons to Structural Components," Proceedings of the Third Joint FAA/DOD/NASA Conference on Aging Aircraft, Albuquerque, NM, September 20-23, 1999.

11. Ambur, D. R., Rouse, M., Young, R. D., and Perez-Ramos, C., "Evaluation of an Aluminum Panel with Discrete-Source Damage and Subjected to Combined Loading Conditions," Proceedings or the 40th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, St. Louis, MO, April 12-15, 1999. AIAA Paper AIAA-99-1439, 1999.

12. Newman, J. C., Jr., and Edwards, P. R., “Short-Crack Growth Behaviour in an Aluminum Alloy - an AGARD Cooperative Test Programme,” AGARD R-732, 1988.

13. Newman, J. C., Jr., Swain, M. H., and Phillips, E. P., “An Assessment of the Small Crack Effect for 2024-T3 Aluminum Alloy,” *Small Fatigue Cracks*, R. O. Ritchie and J. Lankford, eds., 1986, pp. 427-452.

14. Swain, M. H., Everett, R. A., Newman, J. C., Jr., and Phillips, E. P., “The Growth of Short Cracks in 4340 Steel and Aluminum-Lithium 2090,” AGARD R-767, P. R. Edwards and J. C. Newman, Jr., eds., 1990, pp. 7.1-7.30.

15. Everett, R. A., Jr., “A Comparison of Fatigue Life Prediction Methodologies for Rotorcraft,” NASA TM 102759, 1990.

16. Furuta, S., Terada, H., and Sashikuma, H., “Fatigue Strength of Fuselage Joint Structures under Ambient and Corrosive Environment,” *Fatigue in New and Ageing Aircraft*, R. Cook and P. Poole, eds., EMAS, Ltd., 1997, pp. 231-249.

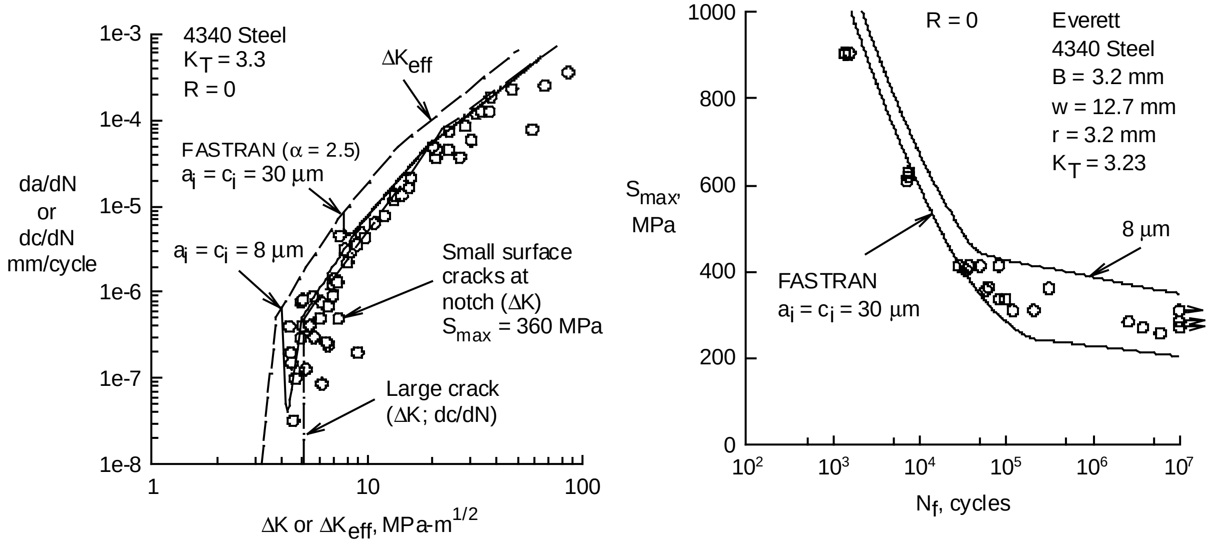
17. Shivakumar, K. N., and Newman, J. C., Jr., "ZIP3D - An Elastic-Plastic Finite-Element Analysis Program for Cracked Bodies," NASA TM-102753, 1990.

18. Dawicke, D. S., Newman, J.C., Jr., and Bigelow, C. A., “Three-Dimensional CTOA and Constraint Effects During Stable Tearing in a Thin-Sheet Material,” In *Fracture Mechanics: 26th Volume*, ASTM STP 1256, 1995, pp. 223-242.

19. Newman, J. C., Jr., "Finite Element Analyses of Fatigue Crack Propagation -- Including the Effects of Crack Closure," Ph.D. Thesis, Virginia Polytechnic institute and State University, Blacksburg, VA, May 1974.

20. Young, R. D., Rose, C. A., Dávila, C. G., Starnes, J. H., Jr., and Rankin, C. C., “Crack Growth and Residual Strength Characteristics of Selected Flat Stiffened Aluminum Panels,” Proceedings of the First Joint DOD/FAA/NASA Conference on Aging Aircraft, Ogden, UT, July, 1997.

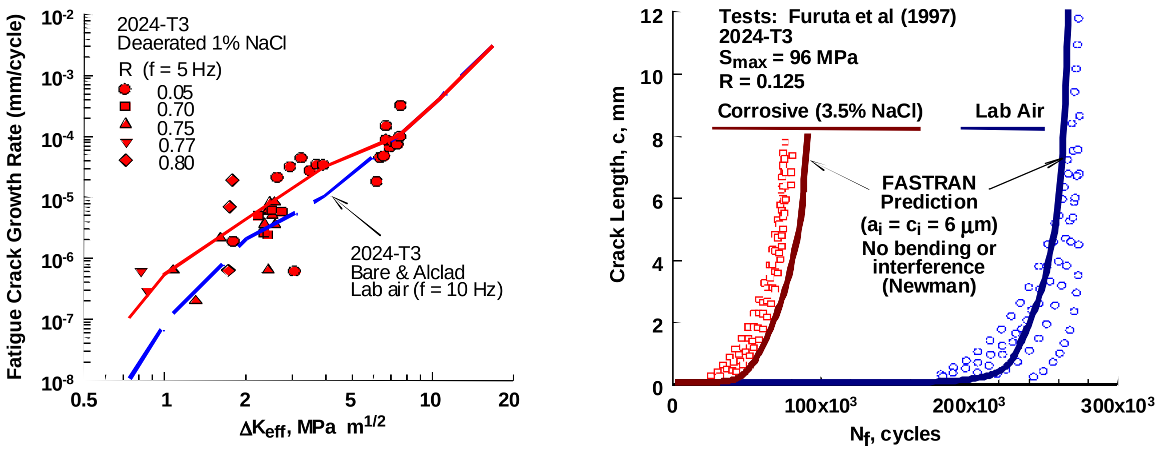
21. Seshadri, B. R., Newman, J. C., Jr., Dawicke, D. S., and Young, R. D., "Fracture Analysis of the FAA/NASA Wide Stiffened Panels," Proceedings of the FAA-NASA Symposium on the Continued Airworthiness of Aircraft Structures," DOT/FAA/AR-92/2, 1997, pp. 513-524.



(a) Small corner crack growth rate at a notch.

(b) Fatigue lives.

Figure 1. Measured and predicted short crack growth rate and fatigue lives for 4340 steel.



(a) Fatigue crack growth rate in air and salt solution.

(b) Lap joint fatigue crack growth.

Figure 2. The effect of corrosion on fatigue life.

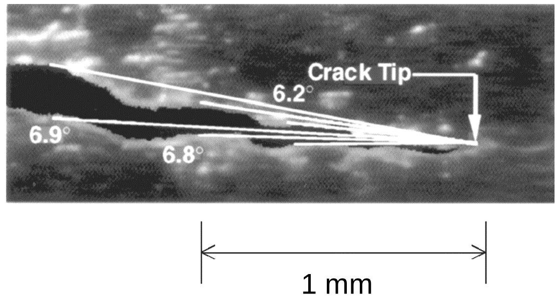


Figure 3. Critical crack-tip-opening-angle criterion.

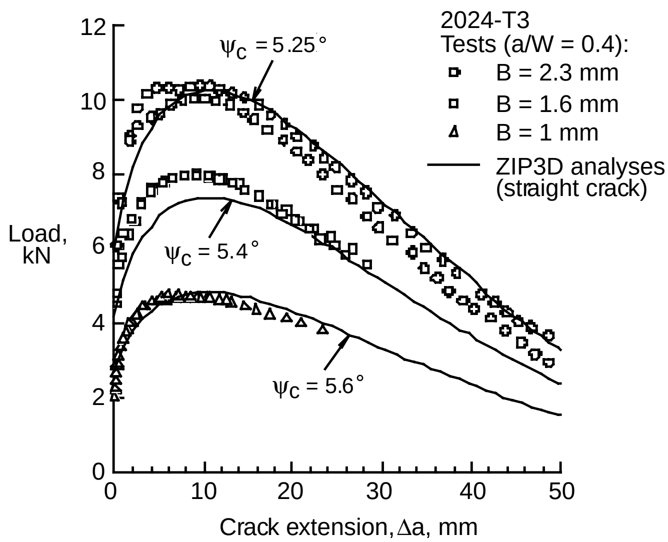


Figure 4. Experimental fracture measurements and ZIP3D finite element predictions for 152-mm-wide C(T) specimens of 2024-T3 aluminum alloy with an initial crack length of a/W = 0.4 and three specimen thicknesses.

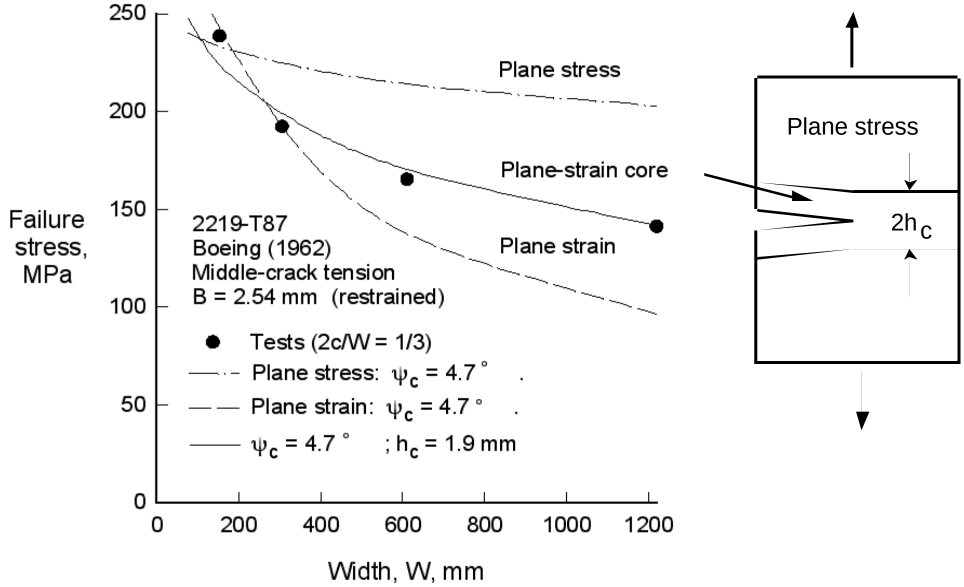


Figure 5. ZIP2D finite element predictions for C(T) specimens to determine plane-strain core height, h_c.

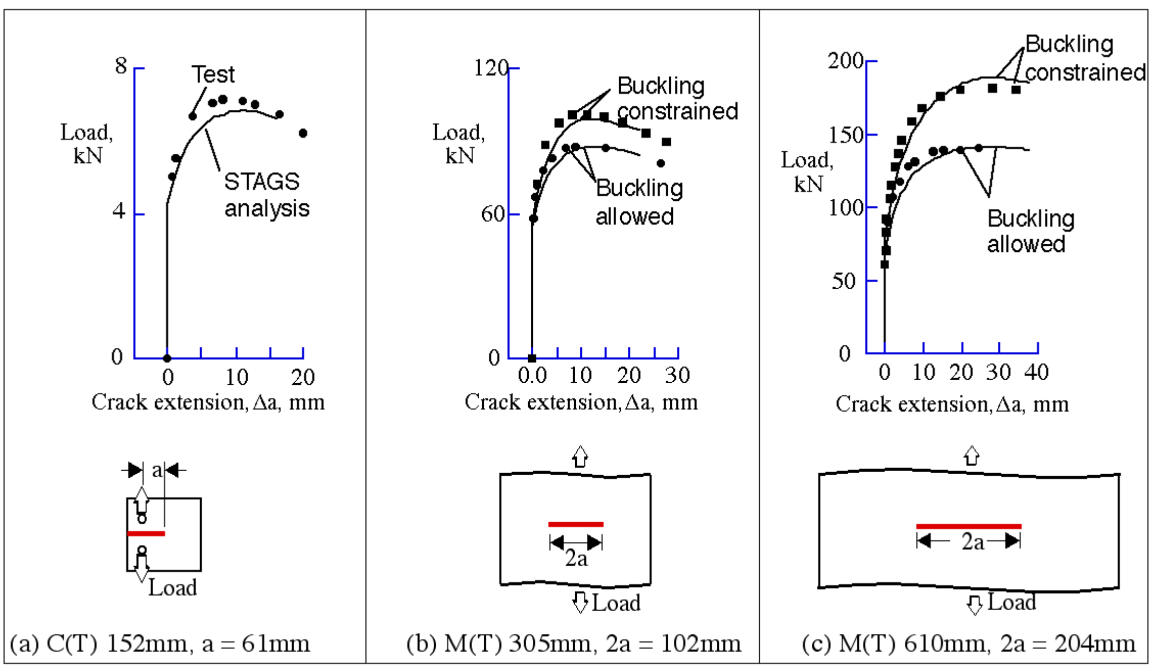


Figure 6. Load versus crack extension results from C(T) and M(T) tests, and nonlinear STAGS analyses with $CTOA_{cr} = 5.0^{\circ}$ and $h_c = 1$ mm. Specimen widths w = 152, 305, and 610mm, respectively.

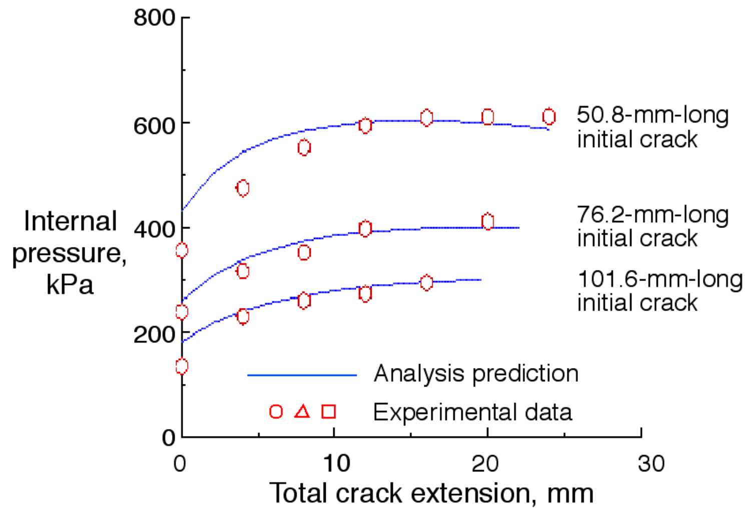


Figure 7. Comparison of analytical and experimental total crack extension results for 1-mm-thick internally pressurized unstiffened aluminum shells.

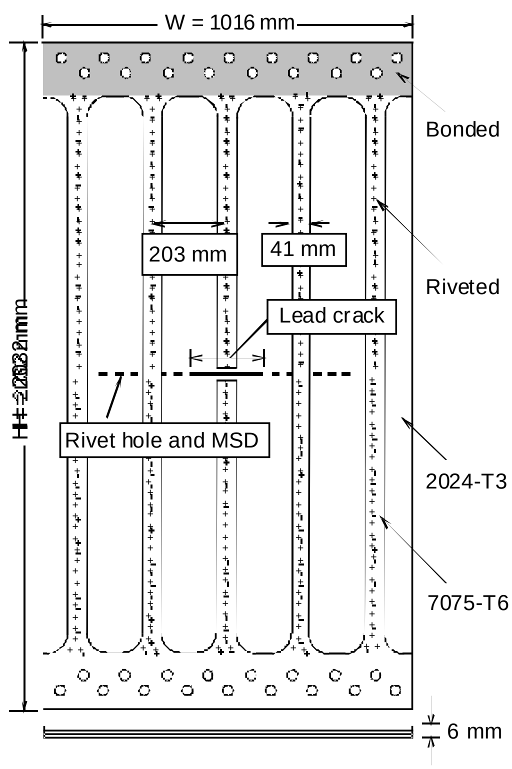
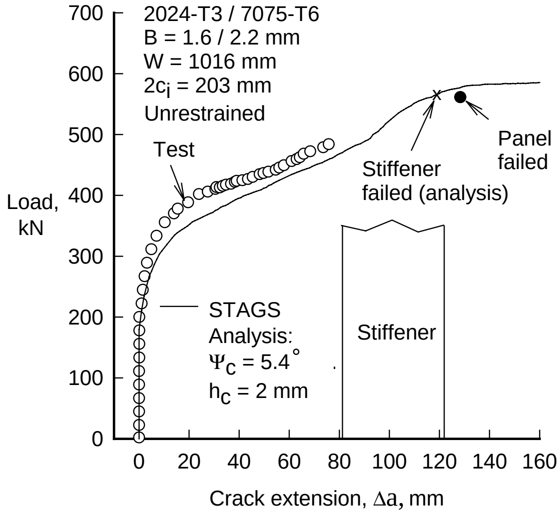
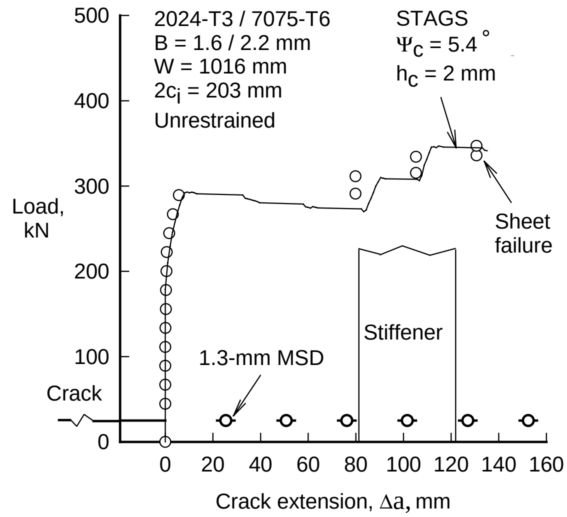


Figure 8. Wide stiffened flat aluminum panel and MSD configuration.

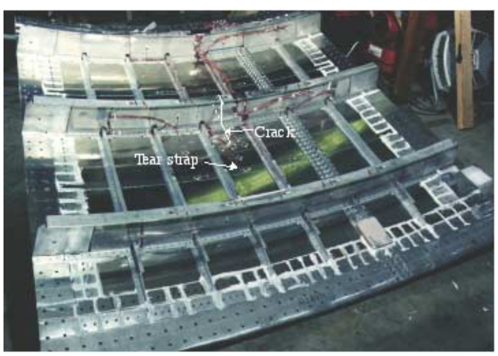


(a) Panel with a single crack.

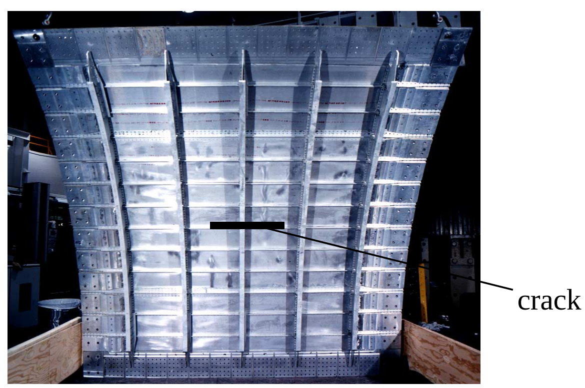


(b) Panel with MSD.

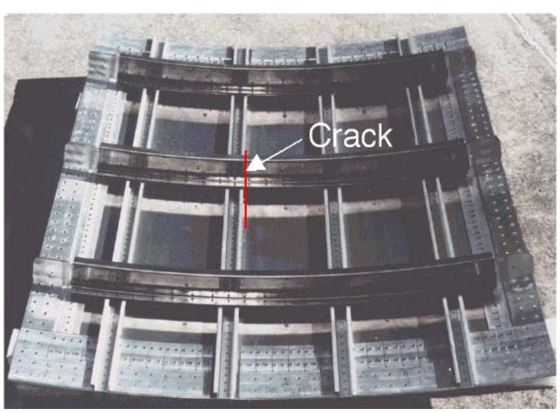
Figure 9. Experimental fracture measurements and STAGS finite element predictions for a 1016-mm-wide, 2024-T3 aluminum alloy unrestrained stiffened panel with a single crack without and with MSD cracks.



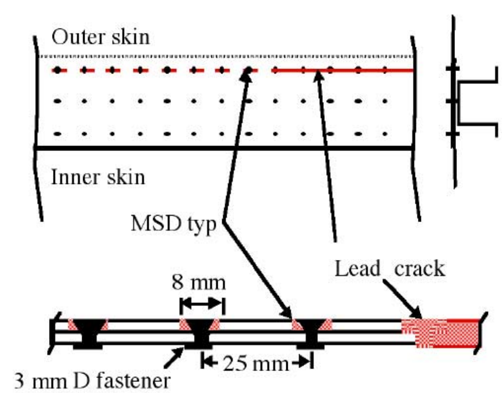
(a) Panel ASIP1 prior to testing.



(b) Panel ASIP3 prior to testing.



(c) Panel ASIP 2 prior to testing.



(d) Lap joint detail with lead crack and MSD cracks for panel ASIP2.

Figure 10. ASIP1, ASIP2, ASIP3 test panels.

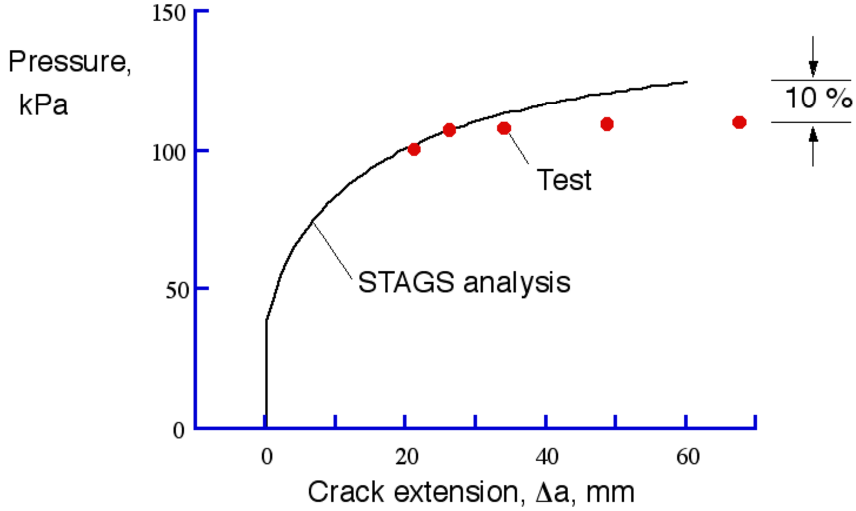
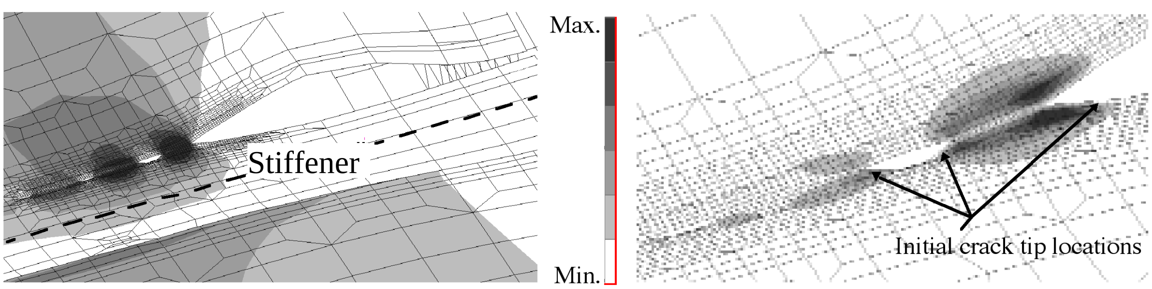


Figure 11. Panel ASIP1 test-analysis correlation of crack extension results as a function of pressure.



(a) Hoop stress, σ_y (b) Plastic hoop strain, $(\varepsilon_y)_p$

Figure 12. Typical analysis results for panel ASIP2 showing crack growth in the lead crack and MSD cracks.

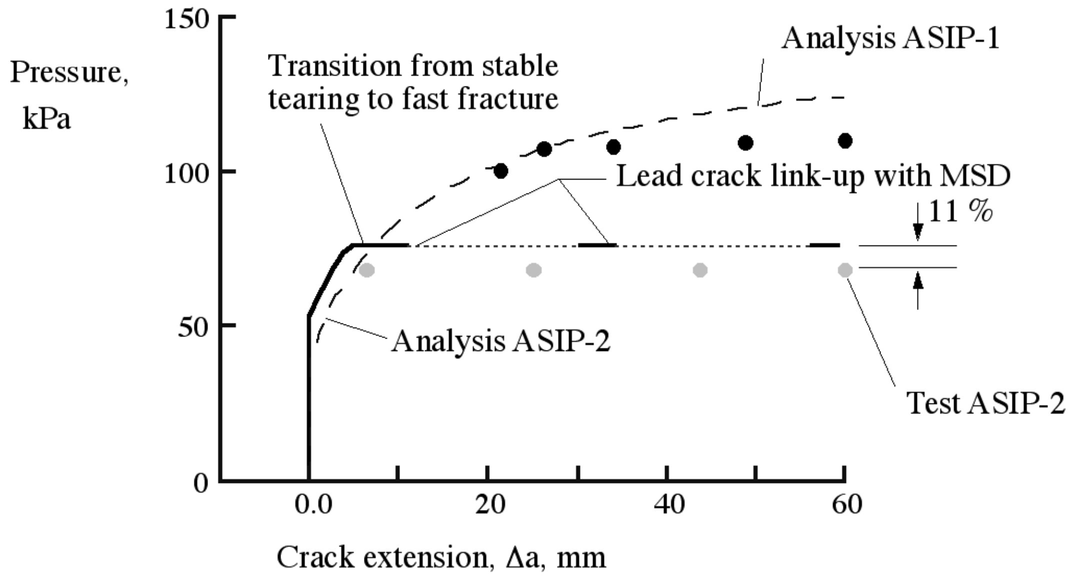


Figure 13. Panel ASIP2 test-analysis correlation of crack extension results as a function of pressure.

Widespread Fatigue Damage Assessment Approach

Paul W. Tan, Catherine A. Bigelow and John G. Bakuckas, Jr.
FAA William J. Hughes Technical Center
AAR-430
Airworthiness Assurance Research and Development Branch
Atlantic City International Airport, NJ 08405, USA

Abstract

A methodology to assess the development of widespread fatigue damage (WFD) and its effect on the residual strength of aircraft structure has been developed. The three major components of the methodology are crack initiation, crack growth and linkup, and residual strength. The crack initiation methodology uses experimentally generated equivalent initial flaw size (EIFS) data and an analytical closure model to determine initial flaw sizes and distribution for multiple-site cracking. The crack-tip opening angle (CTOA) and the T* integral, and plastic zone touch (PZT) criteria were used to predict crack growth and linkup. Elastic-plastic finite element analyses were used with the CTOA or T* integral to determine the residual strength in the presence of multiple-site damage (MSD). The methodologies were verified through a comprehensive test program.

1. Introduction

In response to public concerns after the Aloha Accident, Congress passed legislation known as the Aviation Safety Research Act of 1988. The Act directs the FAA to develop technologies and conduct data analyses for predicting the effects of aircraft design, maintenance, testing, wear, and fatigue on the life of aircraft and on air safety and to develop methods of analyzing and improving aircraft maintenance technology and practices, including nondestructive inspection (NDI) of aircraft structures. The Act also includes a requirement to develop a better understanding of the relationship between human factors and aviation safety and to identify innovative and effective corrective measures for human errors that could adversely affect air safety.

As a result of the Aviation Safety Research Act and concerns relating to the increasing age of the air carrier fleet, the Federal Aviation Administration (FAA) developed the National Aging Aircraft Research Program (NAARP) to ensure the structural integrity of high-time, high-cycle aircraft.

Within the NAARP, the FAA is actively pursuing research to address the problems associated with ensuring the continued structural integrity of the aging commercial transport fleet. The NAARP structural integrity research and development program area includes three major elements: methodologies to assess widespread fatigue damage, airframe repair assessment, and supplemental structural inspections for commuter aircraft. This paper discusses the first element, the assessment of widespread fatigue damage.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

2. Widespread Fatigue Damage

Widespread fatigue damage (WFD) in a structure is characterized by the simultaneous presence of cracks at multiple structural components where the cracks are of sufficient size and density that the structure will no longer meet its damage tolerance requirement. The two sources of WFD are multiple-site damage (MSD), characterized by the simultaneous presence of fatigue cracks in the same structural element; and multiple-element damage (MED), characterized by the simultaneous presence of fatigue cracks in similar adjacent structural elements. An industry committee on WFD identified 16 generic types of structure susceptible to WFD. A few examples are shown in Figure 1.

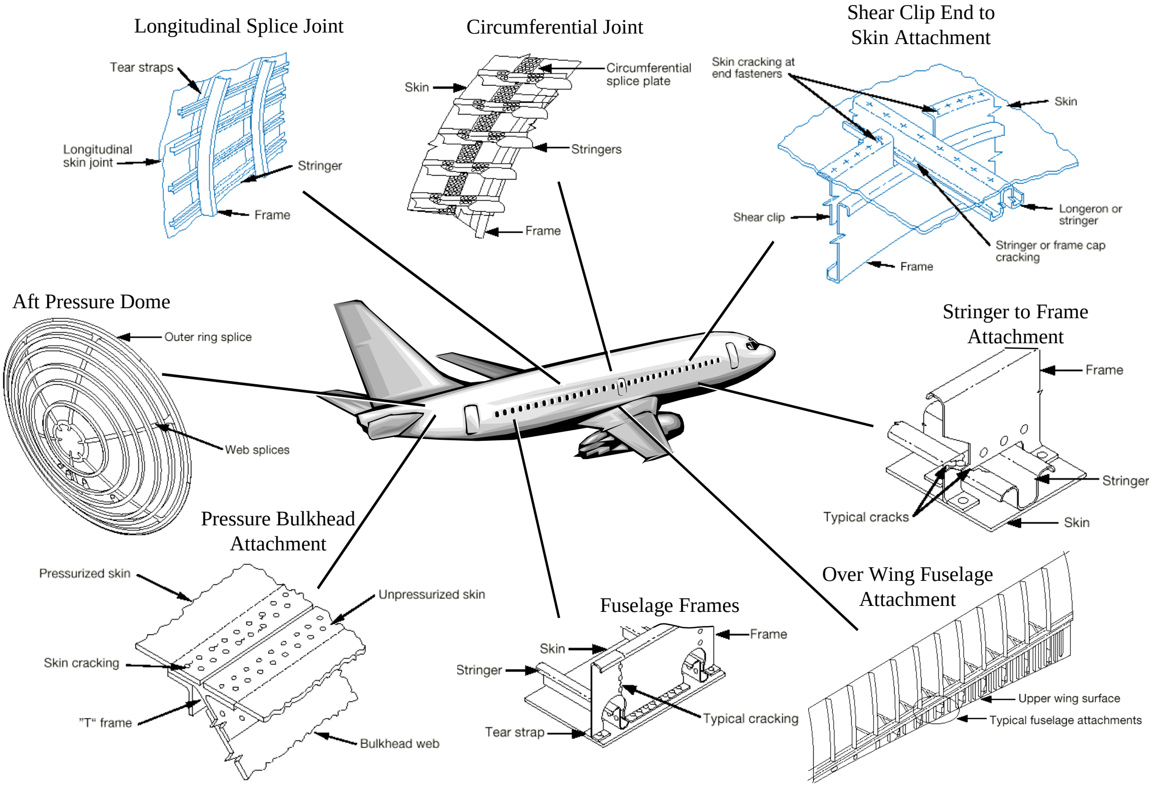


Figure 1. Susceptible widespread fatigue damage locations

WFD is a complex phenomena that is extremely difficult to analyze with standard methods developed from first principles of linear elastic fracture mechanics (LEFM). With limits on the applications of LEFM, more advanced methods have been explored and developed over the past decade with the support and sponsorship of the FAA and the National Aeronautics and Space Administration (NASA). This includes analytical tools to determine parameters governing the onset and growth of cracks and elastic-plastic fracture criterion for residual strength determinations. The tools include the finite element alternating method (FEAM) [1-3] a computationally efficient yet rigorous approach to calculate two- and three-dimensional stress-intensity factor (SIF) solutions governing crack formation and growth, FASTRAN [4], a fatigue crack growth analysis program using a crack closure model, and STAGS [5,6], an advanced finite element program implemented with fracture mechanics and stable tearing analysis capabilities for generalized shell structures. The elastic-plastic failure criterion include the plastic zone touch (PZT), originally developed by Swift [7], crack-tip opening angle (CTOA), originally developed by Wells [8] and later implemented and used extensively by Newman et al. [6,9], and the T* integral, developed by Atluri et al. [10,11].

These tools and criterion are capable of analyzing portions of the multiple-site crack initiation, growth, linkup, and catastrophic fracture process, and they also provide a framework for WFD assessment. They can also be used for future aircraft designs to prevent the occurrence of multiple-site cracking within the design life goal. A critical aspect is experimental validation of the tools and criterion in developing a WFD assessment approach. The methodology developed must be verified and validated using experimental data to ensure successful transfer of useable and accurate technology to industry.

The goal of this research is to ensure that the residual strength of an aging aircraft is not degraded below limit levels due to the occurrence of WFD. To realize this goal, a WDF assessment methodology has been developed to conduct a thorough residual strength evaluation of aircraft structure containing MSD. Throughout the development of this methodology, experimental tests were conducted to validate various components of the methodology including crack initiation, crack growth, and residual strength. Table 1 shows the variety of tests that were conducted by various government and industry agencies worldwide, including NASA Langley Research Center, Air Force Research Laboratory (AFRL), the National Institute of Standards and Testing (NIST), and the Dutch Nationaal Lucht- En Ruimtevaartlaboratorium (NLR), and the General Administration of Civil Aviation of China (CAAC). More recently, a collaborative test program was completed involving Boeing Aircraft Company, AFRL, CAAC, and the FAA William J. Hughes Technical Center as listed in the last two columns in Table 1.

Table 1. Test matrix for WFD assessments

	Foster-Miller (1991-1994)	Author D. Little (1990-1991)	Boeing, Seattle (1995-1996)	NLR (1992-1995)	NIST (1994-1995)	NASA Langley (1991-1999)	AFRL (1991-1994)	Boeing/AFRL/CAAC (1996-2001)	FAA-TC (1999-2001)
Coupons: Fatigue						93		36	
Coupons: Fracture						124			
Flat Panels: Fatigue		102		23					
Flat Panels: Residual Strength	12			23		5	18	12	
Stiffened Flat Panels: Res. Str.					12	5	2	12	
Subscale Cylinders: Fracture						5			
Unstiffened Curved Panels: Frac.	10					1			
Stiffened Curved Panels: Fatigue	8					2			7
Stiffened Curved Panels: Res. Str	19					4			4
Aft Pressure Bulkhead: Res. Str.								1	

3. Approach

The approach established to conduct WFD assessments, as shown schematically in Figure 2, is a product of the advanced tools and criterion described previously for the methodology development and test data for experimental validation. In this study, the approach is used to address WFD on two fronts: (1) characterizing MSD by studying the initiation and growth of cracks in the evolution of multiple-site cracks, and (2) determining the effects of MSD on the residual strength.

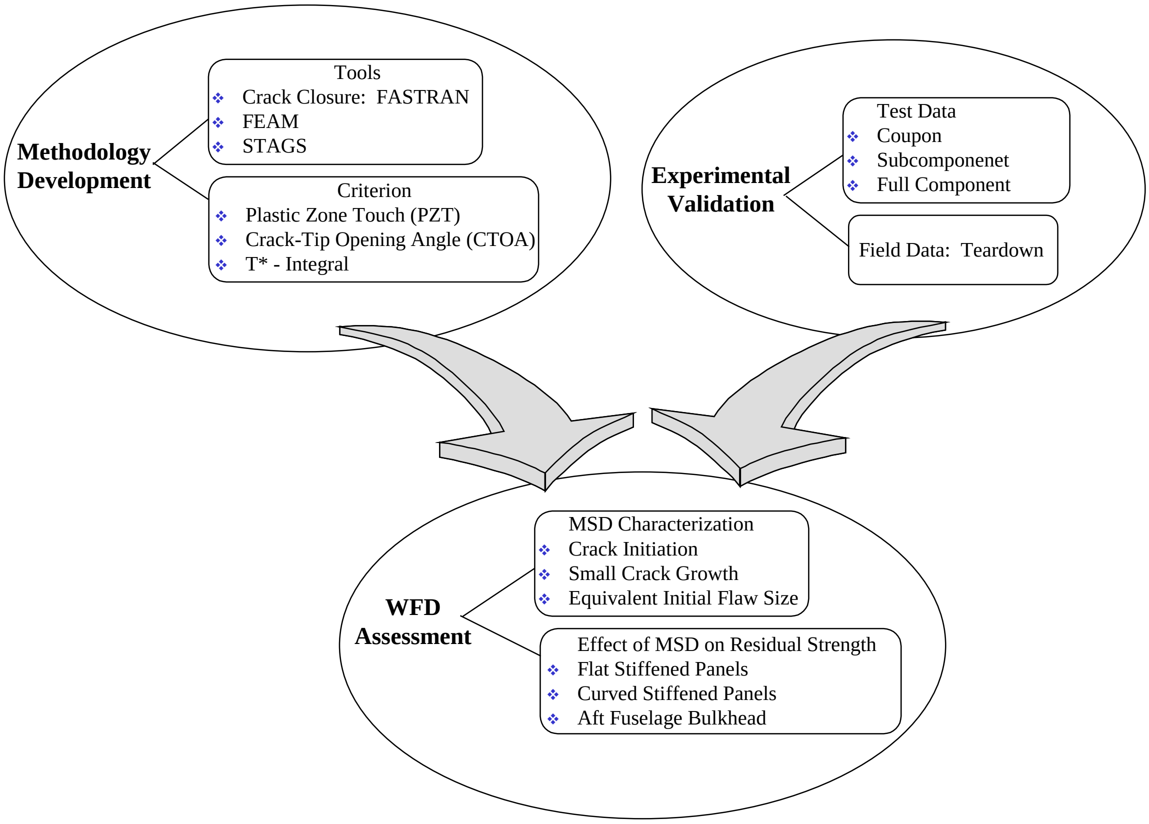


Figure 2. WFD methodology development, validation, and assessment

<u>Characterization of MSD Evolution</u>

One possible source of WFD is the occurrence of MSD along a structural component with similar details, for example, a lap splice with rows of riveted joints. Due to fatigue, small cracks of different sizes can emanate from each rivet hole. To accurately predict the initiation of these small cracks, the effects of rivet hole quality, interference fit, rivet load transfer, and rivet clamping forces need to be accounted for. However, these parameters are not easily determined individually. Therefore, an approximate approach, the equivalent initial flaw size (EIFS) approach was taken. In this approach, flat panels with structural splices were tested under fatigue loading. Crack growth was monitored using nondestructive inspection (NDI) techniques. The failed specimens were examined using a scanning electron microscope (SEM) to generate crack length versus cycles curves. The curves were then extrapolated to the "zero cycle" axis using the closure-based crack growth prediction code, FASTRAN, developed by NASA and that crack length was used as the EIFS.

Four common types of splice joints were considered as shown in Figure 3. A total of 16 flat-panel specimens were fabricated, four for each splice type. The splices represent three types of fuselage longitudinal splices and one type of fuselage circumferential splice.

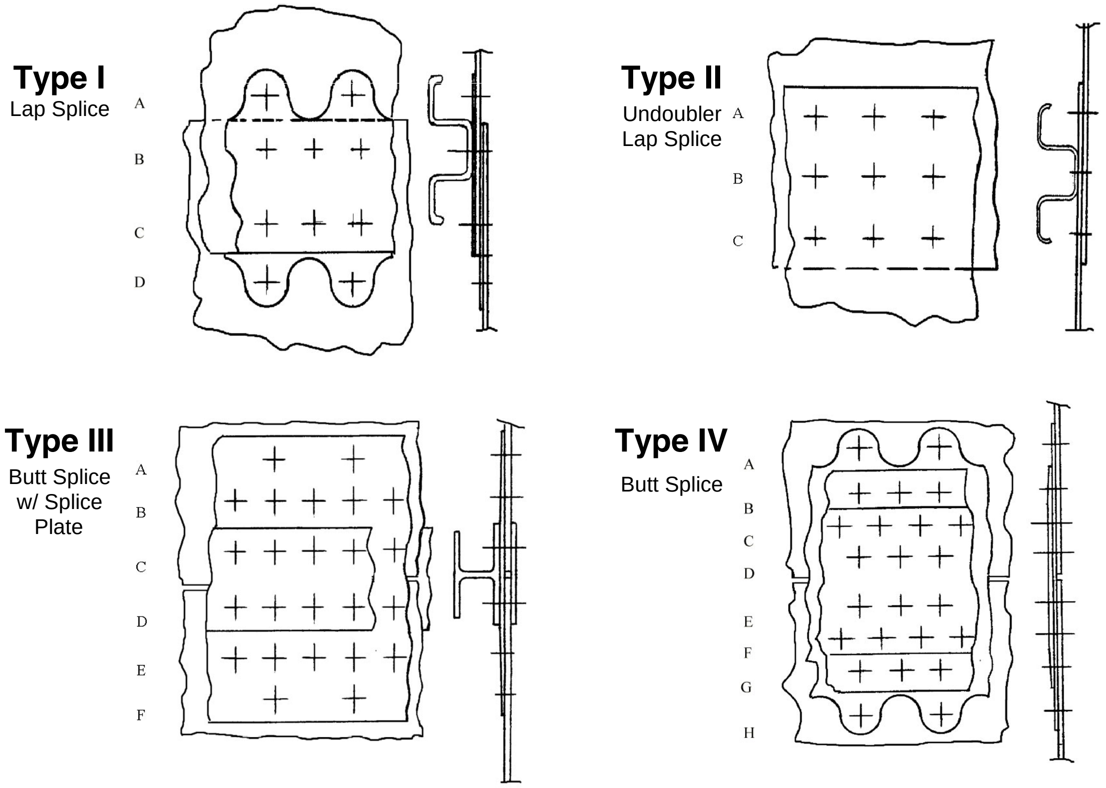


Figure 3. Three longitudinal splice type and one circumferential splice type

The three longitudinal splice types were (a) a lap joint with two finger doublers and a longeron; (b) a longitudinal lap joint without doublers but with a longeron; and (c) a longitudinal butt joint with a splice plate, a doubler, and a longeron. The circumferential splice type is a circumferential butt joint with a butt splice plate and a finger doubler. The skins were made of 2024-T3 aluminum alloy, the longerons were made of 7075-T6 aluminum alloy, and the doublers and splice plates were made of either 2024 or 7075 materials.

From the crack growth curves generated from the EIFS tests, as described above, the test and analysis were correlated using the NDI data and the SEM data. The closure-based crack growth prediction code, FASTRAN, and small crack growth rate data were used for the analyses. The parameters considered included the test specimen geometry, the test load history, magnitude of applied far-field stress, bending stress factor, bypass stress factor, neat-fit pin factor, and the effects of adjacent hole with cracks. A correlation factor as a function of crack length was then determined to obtain good agreement between analytical prediction and the experimental results for one specimen. This factor was subsequently used as the correlation factor for all other specimens. The EIFS distribution from the back tracking analysis using FASTRAN and an example of the correlation obtained between experiments and analysis using FASTRAN are shown in Figure 4.

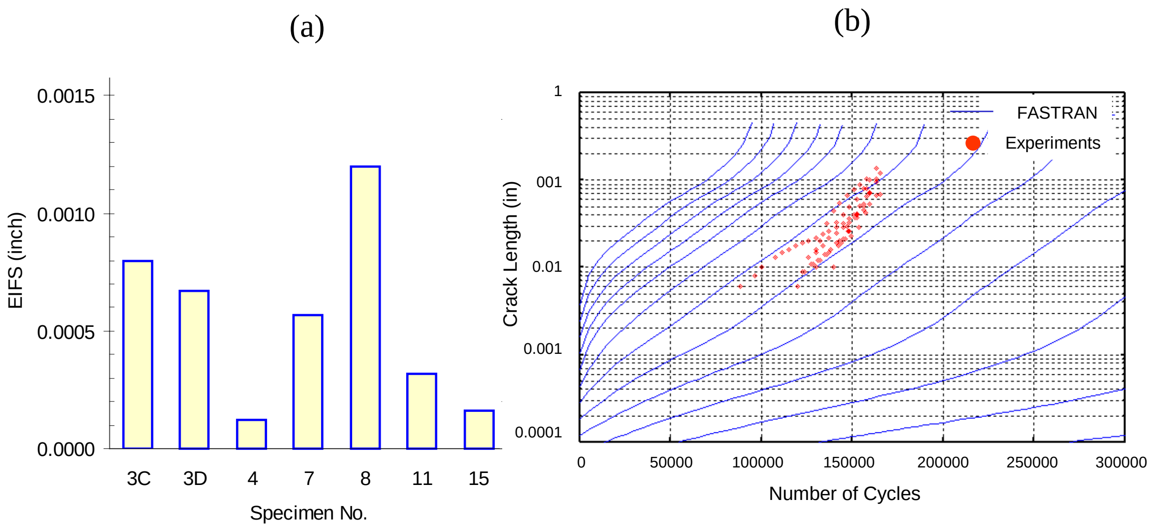


Figure 4. (a) EIFS distribution and (b) correlation between experiments and analysis using FASTRAN with different values of EIFS.

<u>Effects of MSD on Residual Strength</u>

A thorough residual strength assessment approach has been developed for aircraft structure containing MSD. The approach, based on nonlinear finite element analysis and the PZT, CTOA, and T* integral criterion, was applied and verified for three cases of problems outlined below.

MSD in Flat Panels

The effect of small MSD on residual strength was determined and the elastic-plastic criterion of T* integral, CTOA, and PZT fracture criteria were evaluated. These

Table 2. Residual strength prediction of flat panels.

Specimen Number	Joint Type	MSD Size (inch)	Criterion, Absolute Percent Difference Between Analysis and Experiments		
			PZT	T* integral	CTOA
1	1	0.00	-8	-	3
2		0.05	7	13	0
3		0.10	5	7	1
4	2	0.00	-13	-	-1
5		0.05	-10	-2	2
6		0.10	-1	-4	-6
7	3	0.00	-23	-	4
8		0.05	-3	-4	-2
9		0.10	0	-5	-5
10	4	0.00	-17	-	-5
11		0.05	-4	-3	-3
12		0.10	-4	-4	0
Average Percent Difference			7.9	5.2	2.7

criteria correlated well with the experimental results as shown in Table 2 with an average absolute percent difference of 2.7%, 5.2%, and 7.9% for the CTOA, T* integral and PZT criteria, respectively.

MSD in Curved Panels

A unique state-of-the-art facility to assess the structural integrity of aircraft fuselage structure was established at the Federal Aviation Administration (FAA) William J. Hughes Technical Center. The Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility is capable of testing full-scale fuselage panel specimens under conditions representative of those seen by an aircraft in actual operation. The test fixture features a novel adaptation of mechanical, fluid, and electronic components and is capable of applying pressurization, longitudinal, hoop, frame, and shear loads to a fuselage panel. A high-precision Remote Controlled Crack Monitoring (RCCM) system was developed to inspect and record crack initiation and progression over the entire fuselage panel test surface.

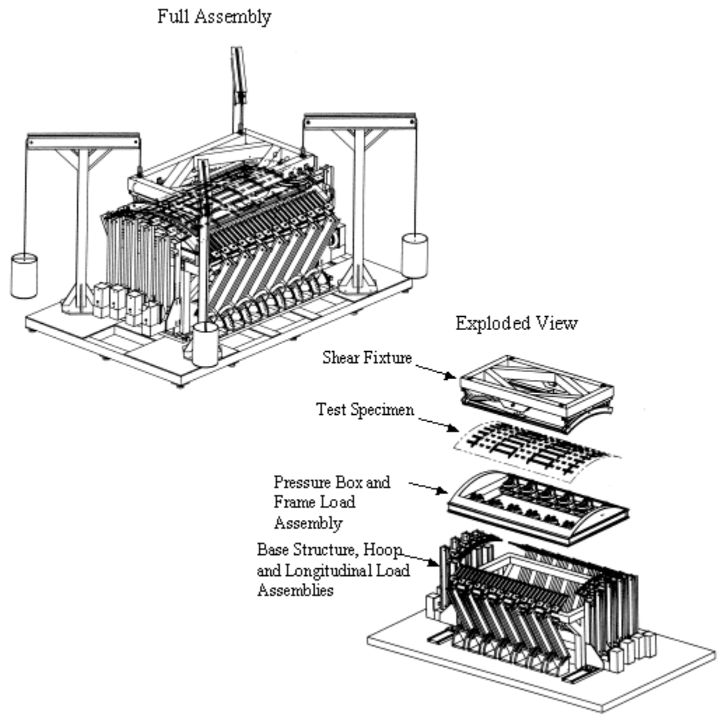


Figure 5. Full-Scale Aircraft Structural Test Evaluation and Research fixture

The effects of multiple cracks on the fatigue crack growth and residual strength of curved fuselage panels was studied using the FASTER facility. A total of four panels were tested, two panels with a longitudinal lap splice and two with a circumferential butt joint. For each joint configuration, one panel contained only a lead crack and the other contained a lead crack with multiple cracks located along the outer critical rivet row of the joints. Geometric nonlinear finite element analyses conducted using STAGS and the CTOA criteria were used to predict the residual strength. The strain distributions and fracture parameters governing crack formation and growth were determined. Comparisons with strain gage data verified the finite element models. Results include comparisons of strain distributions, fatigue crack growth characteristics, and the damage growth process during residual strength test for the two joint configurations. In general, the small multiple cracks did not have an effect on the overall global strain response. However, the small multiple cracks reduced the number of cycles to grow a fatigue crack to a predetermined length by 37% and 27%

for the longitudinal lap joint and circumferential butt joint panels, respectively. In addition, the presence of multiple cracks reduced the residual strength of the panels with a longitudinal lap joint by approximately 20%. The measured and predicted residual strength were in good agreement as shown in Figure 6.

	Residual Strength (psi)	
Panel	Measured	Predicted
CVP1	11.14	10.65
CVP2	9.16	9.58

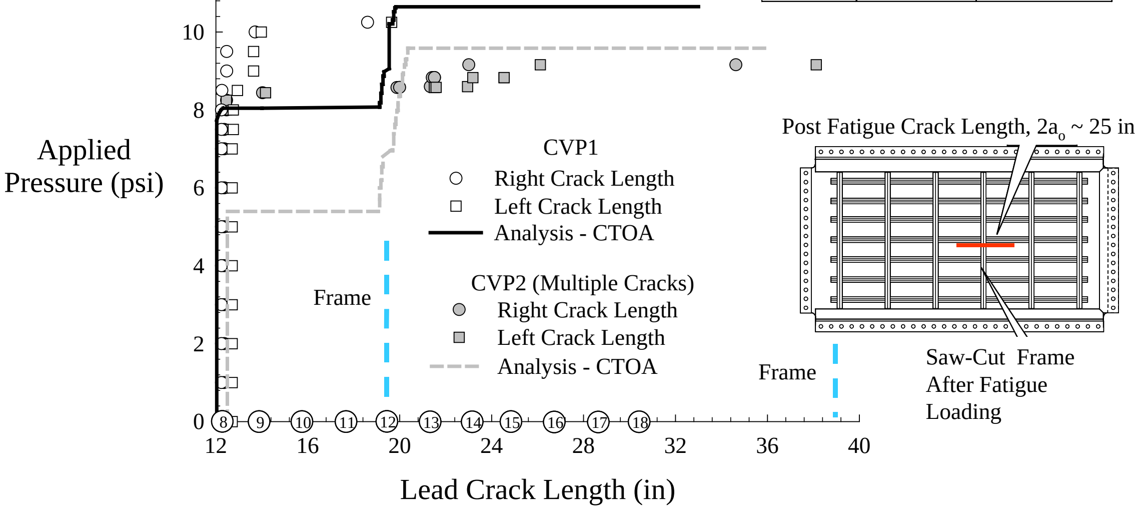


Figure 6. Measured and predicted residual strength

MSD in Aft Pressure Bulkhead

The purpose of this test was to verify the generality of the approach validated for the flat- and curved-panel cases by applying it to a different large-scale airframe structure, that is, to an aft pressure bulkhead. Geometric nonlinear finite element analyses using STAGS and the CTOA criteria predicted the residual strength. The measured and predicted residual strength were in good agreement as indicated in Figure 7.

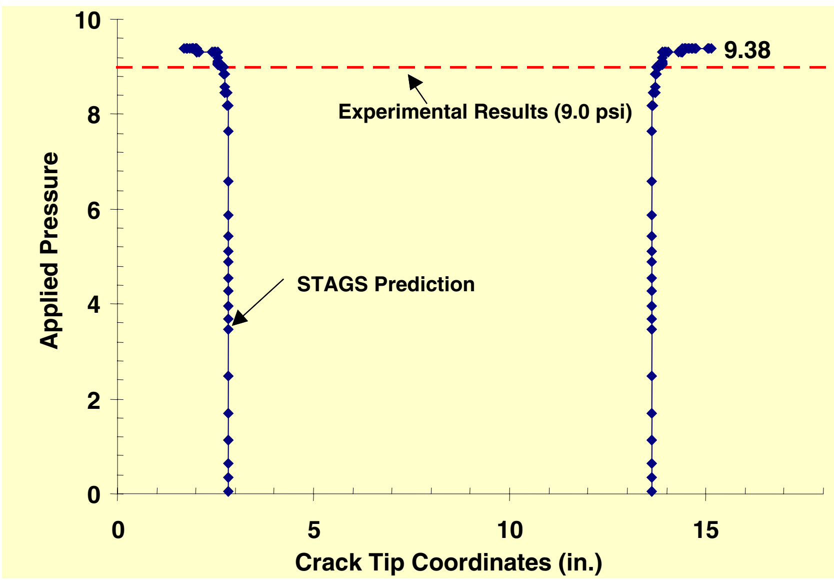


Figure 7. Measured and predicted residual strength

CONCLUDING REMARKS

An approach has been developed to assess the effect of widespread fatigue damage (WFD) on the residual strength of aircraft structure. The three major components of the approach are crack initiation, crack growth and linkup, and residual strength. The crack initiation methodology uses experimentally generated equivalent initial flaw size (EIFS) data and an analytical closure model to determine initial flaw sizes and distribution for multiple-site cracking. The crack tip opening angle (CTOA), T* integral, and plastic zone touch (PZT) criteria were used to determine crack growth and linkup and the residual strength in the presence of multiple-site damage (MSD). Good correlation was obtained between laboratory coupons and large-panel test data and the analytical predictive methodologies. The methodologies were verified for representative commercial aircraft panels under simulated flight conditions.

REFERENCES

1. Nishioka, T. and Atluri, S. N., "Analytical Solution for Embedded Elliptical Cracks, and Finite Element-Alternating Method for Elliptical Surface Cracks, Subjected to Arbitrary Loadings," *Engineering Fracture Mechanics*, Vol. 17, 1983, pp. 247-268.

2. Nishioka, T. and Atluri, S. N., "An Alternating Method for Analysis of Surface Flawed Aircraft Structural Components," *AIAA Journal*, Vol. 21, 1983, pp. 749-757.

3. Atluri, S. N. and Nishioka, T., "Computational Methods for Three-Dimensional Problems of Fracture," *Computational Methods in Mechanics of Fracture*, S. N. Atluri (ed.), North Holland, Chapter 7, 1986, pp. 230-287.

4. Newman, J. C., Jr., "A Crack-Closure Model for Predicting Fatigue Crack Growth Under Aircraft Spectrum Loading," American Society for Testing and Materials, USA, Special Technical Publication 748, 1981, pp. 53-84.

5. Rankin, C. C., Brogan, F. A., Loden, W. A., and Cabiness, H. D., "STAGS Users Manual," Lockheed Martin Missiles and Space Co. Inc., LMSC Report P032594, Version 3, March 1999.

6. Harris, C. E., Newman, J. C., Piascik, R. S., and Starnes, J. H., "Analytical Methodology for Predicting the Onset of Widespread Fatigue Damage in Fuselage Structure," Proceedings of *FAA-NASA Symposium on Continued Airworthiness of Aircraft Structures*, DOT/FAA/AR-97/2, July 1997.

7. Swift, T., "Widespread Fatigue Damage Monitoring–Issues and Concerns," Proceedings of *5th International Conference on Structural Airworthiness of New and Aging Aircraft*, Federal Republic of Germany, DGLR-Bericht 93-02, 1993, pp. 133-150.

8. Wells, A. A., "Crack Propagation Symposium Proceedings," Cranfield College of Aeronautics, 1961, pp. 210-230.

9. Newman, J. C., Jr., Bigelow, C. A., and Dawicke, D. S., "Finite Element Analysis and Fracture Simulation in Thin-Sheet Aluminum Alloy," Proceedings of the International Workshop on Structural Integrity of Aging Airplanes, eds. Atluri, S. N., Harris, C. E., Hoggard, A., Miller, N., and Sampath, S. G., 1992, pp. 167-186.

10. Atluri, S. N. and Nishioka, T., *Engineering Fracture Mechanics,* Vol. 20, 1984, pp. 209-244.

11. Wang, L., Brust, F. W., and Atluri, S. N., "Predictions of Stable Growth of a Lead Crack and Multiple-Site Damage Using Elastic-Plastic Finite Element Alternating Method (EPFEAM)," Proceedings of *FAA-NASA Symposium on Continued Airworthiness of Aircraft Structures*, DOT/FAA/AR-97/2, July 1997.

Programme for Life Extension and Widespread Fatigue Damage Evaluation to Ensure Continued Structural Integrity of Airbus Large Transport Category Airplanes

Hans Trey
Structure Analysis, Damage Tolerance Department (ESADG)
and Hans-Juergen Schmidt
Metal Design Principles (ESO) and Fatigue and
Damage Tolerance Department (ESG-O)
EADS Airbus GmbH
Fatigue and Fracture Mechanics Dept (EMF)
Kreestslag 10
D-21129 Hamburg
Germany

Keywords : ageing aircraft, life extension, widespread fatigue damage, Monte Carlo Simulation

Abstract

The Airworthiness Assurance Working Group (AAWG) has been charted by FAA to enhance and develop rules for continued structural integrity of large transport category airplanes. The subgroup AAWG – RWG (Rule Writing Group) has completed a draft proposal for enhancement of AC91-56 and the introduction of operational rules of aircraft operated under 14 CFR Parts 91, 121, 125, 129 and 135.
The Airbus activities to meet the new regulations are in progress for the Airbus types A300B2, A300B4-100, A300B4-200 and A300B4-600. This paper describes details of tests, analysis and procedures to meet the new requirements and recommendations. The presentation discusses especially the activities for the pressurized fuselage, which is mainly under the responsibility of Airbus Deutschland GmbH.
The Airbus Life Extension activities include a general review of the fatigue and damage tolerance analysis prepared for type certification, interpretation of full scale fatigue test findings, tear down results and SB review with respect to extended service goals.
Special emphasis will be given to the investigation and analysis of Widespread Fatigue Damage (WFD). Compliance must be ensured with the new requirements, which concern the need to limit the validity of the current structural maintenance programme and the need to impose operational requirements that mandate a structural maintenance programme to prevent WFD in the fleet. The WFD evaluation of the fuselage will be performed using a new analysis tool developed in a European research programme and extended and validated by Airbus Deutschland GmbH within the last years. The new analysis tool is verified by extensive coupon and panel testing comprising fatigue, crack growth and residual strength tests for the major areas potentially susceptible to WFD.
Another important topic of the life extension activities is the prevention of corrosion in the ageing fleet. Therefore a Corrosion Prevention and Control Programme (CPCP) was established which is reviewed periodically.
In addition the structure of a retired high-time A300 is investigated in detail to support the life extension activities. Tear down investigation of critical areas and areas potentially susceptible to WFD are performed. Several structural parts, e.g. large panels are cut out of the airframe and tested for fatigue, crack growth and residual strength capabilities. In addition material properties, i.e. crack growth data and fracture toughness data will be determined for the aged structure and compared to more recent values.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Status of Airbus Fleet

The A300 aircraft was the first of the Airbus types and approximately 500 aircraft are currently in service. This number includes an increasing number of older aircraft converted from passenger to freighter configuration. The A300 entered airline operation in May 1974 and production of the A300B4-600 variant continues today, however, some early models of the A300 are now approaching their Design Service Goal (DSG), i.e. *the number of flight cycles or flight hours during which the principal structure is expected to be reasonably free from significant cracking*.
As can be seen from Figure 1 the Airbus fleets are still relatively young with the majority of the airplanes below 50 percent of their DSG [1]. No Airbus aircraft has reached its DSG up to now.
However, in May 1999 26 A300 as well as 15 A300-600 aircraft had exceeded 75 percent of their DSG. The high-time A300 B2/B4 and A300-600 aircraft will reach their DSGs within the next few years. In 1997 a forecast of the fleet status was made for planning of the Airbus life extension activities. Figure 2 shows the development of the A300 variants for the years 2001 to 2005. Especially a considerable number of A300B4-600 aircraft will reach the DSG very soon. Consequently, life extension activities including widespread fatigue damage evaluation are needed.

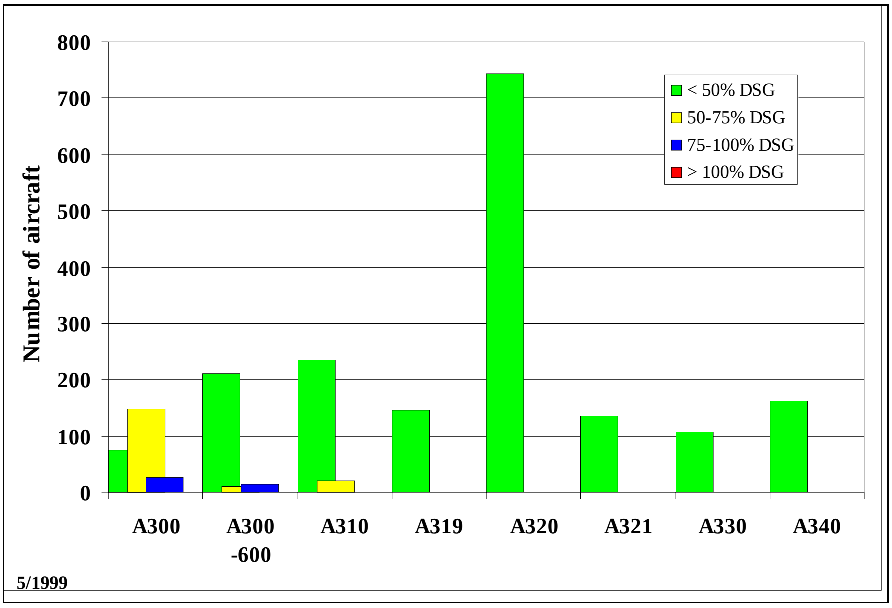


Figure 1 : Age of Airbus A300 Fleet

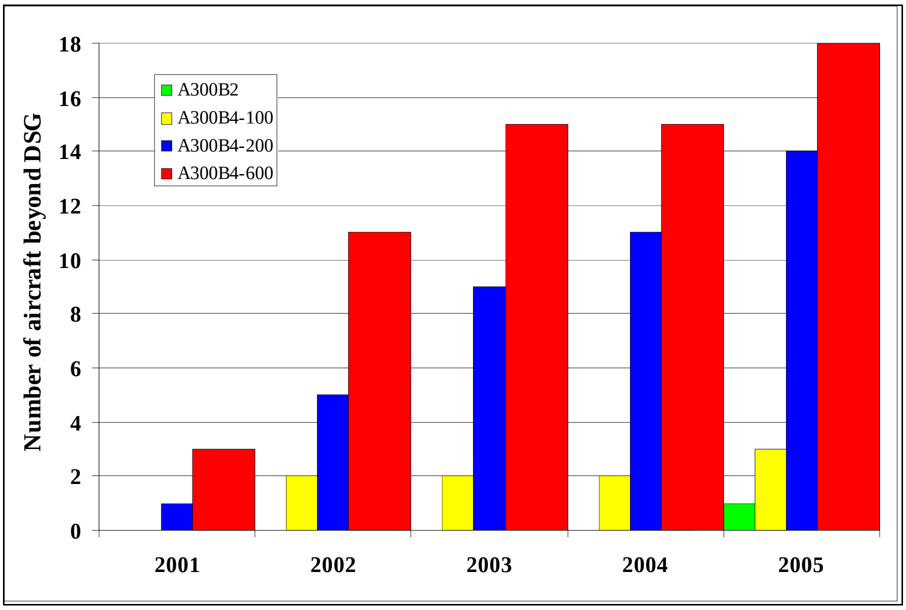


Figure 2 : Predicted Development of Airbus A300 Fleet

New Design Service Goals

The original DSG is generally established at the time of type certification and is not intended to limit the life of the structure, or to define the point at which the aircraft can not continue its operation. In the case of the A300 aircraft, operators have requested continued operation beyond the DSG, and the issue of an extension of the service life is now being addressed. In close cooperation with the operators Airbus has defined new, extended service goals (ESG) for the various A300 models. The table below summarizes the new ESGs :

A/C Type	Design Service Goal [FC]	Extended Service Goal [FC]	Increase [%]
A300 B2	48 000	60 000	25
A300 B4-100	40 000	57 000	42.5
A300 B4-200	34 000	42 500/57 000	25/67.6
A300 B4-600	30 000	42 500	42

General Approach

To reach the above new ESGs Airbus Industry has launched a Full Life Extension Programme for the A300 aircraft. In order to justify a further period of operation up to the new ESG, it is necessary to review service experience and re-assess the existing inspection programmes. This may lead to a modification of the maintenance strategy, including the inspection of additional items or an increased level of surveillance in some areas.

The following activities are performed under the A300 Full Life Extension.

- Fatigue and damage tolerance analysis of the original structure and modifications including :
    - Detailed identification of the concerned area
    - Review of Full Scale Fatigue Test (A300, A310) and in-service experiences
    - Loads comparison for all variants
    - Review of former fatigue justifications
    - New calculations
    - Review of Service Bulletins and current inspection programme
- Widespread Fatigue Damage Analysis
- Update of all inspection programmes incl. MRB, SSIP and the definition of new programmes
- Definitions of modifications or replacement of structure including embodiment threshold

In addition to these activities all repairs and in-service problems that were monitored by the operator must be considered in the ESG analysis. Therefore a complete review of all repairs, in-service problems and of the Structural Repair Manual (SRM) is required.

The activities are especially tuned to the pressurized fuselage, which is mainly under the responsibility of Airbus Deutschland GmbH. Additionally some areas of the Vertical and Horizontal Tailplane are investigated.

All activities are supported by comprehensive WFD and local damage testing and tear down investigation of a retired airframe.

Furthermore, the life extension activities including WFD evaluation are significantly supported by the results of the full scale fatigue tests. The A300 aircraft was tested in a multi-section test for at least two lifetimes. Figure 3 shows the four A300 FSFT specimens together with the number of simulated flights.

For the assessment of areas susceptible to local damage only a conventional fatigue and damage tolerance analysis may be used. However, for the analysis of WFD susceptible areas a new approach was developed.

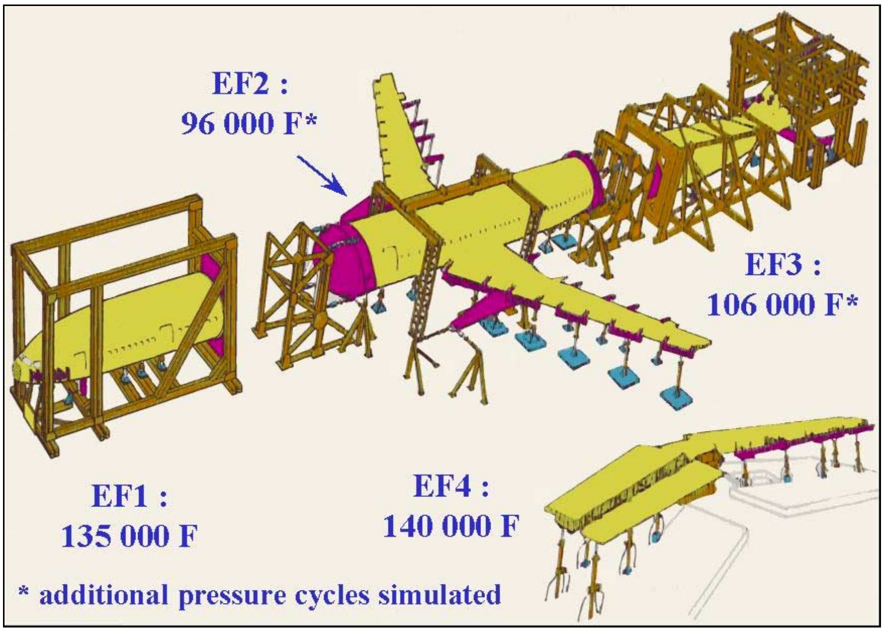


Figure 3 : A300 Full Scale Fatigue Test

Conventional Local Damage Evaluation

The evaluation of all structural items potentially susceptible to local damage will be performed according to the philosophy outlined Figure 4. The evaluation is based on analysis supported by comprehensive testing. This kind of structure is fully protected by the existing maintenance programme up to the DSG. Areas where fatigue damages occurred during the FSFT are covered by inspection and/or modification or by production improvements.

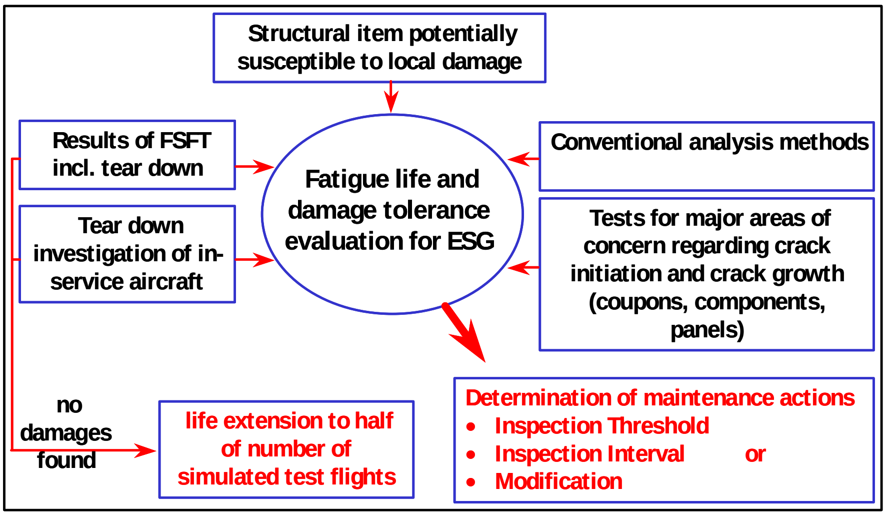


Figure 4 : Approach used for the evaluation of local damage

For life extension the results of the FSFTs are evaluated again in order to define additional areas which need to be assessed. This assessment is done for all areas by fatigue life and damage tolerance analysis using conventional state of the art analysis tools. For some specific areas this analysis is again supported by testing of coupons, components or panels.

A more simple justification is possible for all areas where no damages occurred neither during FSFT nor during tear down investigations. The life of these areas may be extended up to half of the number of flights simulated in test, adjusted by the relevant flight mission data.

Widespread Fatigue Damage Evaluation

Since fatigue crack initiation becomes more likely with extended service, the evaluation of structural items potentially susceptible to WFD requires a more complex analysis.

With extended operation there is an increased probability of Multiple Site Damage (MSD), *i.e. the simultaneous presence of multiple fatigue cracks in the same structural element*, or Multiple Element Damage (MED), *i.e. the simultaneous presence of multiple fatigue cracks in similar adjacent structural elements*. MSD and MED can seriously degrade the damage tolerance capability, i.e. the residual strength of the structure, and may develop into Widespread Fatigue Damage (WFD), which is defined as the point where *the structure is no longer able to meet the required level of residual strength*. Thus, the prevention of WFD is an important issue to the continued safe operation of ageing aircraft, and special WFD analysis methods have been developed.

Following discussions between the manufacturers, aircraft operators and regulatory authorities undertaken by the Ageing Aircraft Working Group (AAWG), a draft proposal for enhancement of AC91-56 and the introduction of operational rules for aircraft operated under 14 CFR Parts 91, 121, 125, 129 and 135 was issued [2]. Compliance must be ensured with the new requirements, which concern the need to limit the validity of the current structural maintenance programme and the need to impose operational requirements that mandate a structural maintenance programme to prevent WFD in the fleet.

The new rule requires the introduction of specific detailed inspections for MSD/MED, and the declaration of an '*Inspection Start Point (ISP)*', where special WFD inspections must be started, and an operational limit, known as the '*Structural Modification Point (SMP)*', beyond which a structural item may not be used without modification because of the increased risk of WFD. The SMP is derived from the average expected behavior. Beyond this point the airplane may not be operated without further evaluation and modification. The SMP is established so, that operation up to the point provides equivalent protection to that of a two lifetime fatigue test.

For structure where the MSD/MED situation is reliably detectable before it becomes critical a so-called monitoring period may be defined and applied before other means have to be taken. The monitoring period is the period of time between ISP and SMP. Repeat inspection intervals are established based on the length of time from detectable fatigue cracks to the average WFD divided by a factor.

Consequently, for each structural item potentially susceptible to WFD the analysis method must provide both the ISP and the SMP, as well as the interval of repeat inspections during the monitoring period. An example is shown in Figure 5, where MSD inspections start at 36000 FC with an repeat inspection interval of aprox. 7000 FC until the SMP is reached at 54000 FC. These values were derived according to the recommendations provided by the AAWG, which state that SMP and ISP are determined by applying a factor of 2 and 3 on the WFD Average Behaviour, respectively.

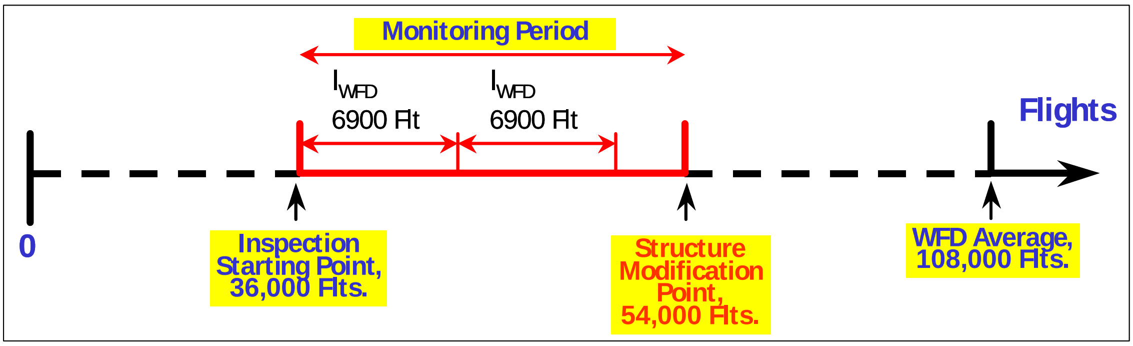


Figure 5 : Example of Determination of Inspection Programme

The WFD phenomenon is commonly associated with longitudinal or circumferential fuselage lap joints, such as in the *Aloha Airlines* Boeing 737 accident of April 1989, in which widespread fatigue and corrosion damage in a high-time aircraft led to an in-flight structural failure and explosive decompression [3]. Although initially independent, the development of these cracks may eventually lead to interaction and link-up of damage sites, and the possibility of structural failure.

The structural items within the fuselage structure of a commercial large transport category aircraft that are potentially susceptible to MSD/MED have been identified by the AAWG [2]. These locations are summarized as follows:

- Longitudinal Skin Joints, Frames and Tear Straps (MSD/MED)
- Circumferential Joints and Stringers (MSD/MED)
- Lap Joints with Milled, Chem-milled or Bonded Radius (MSD)
- Fuselage Frames (MED)
- Stringer to Frame Attachments (MED)
- Shear Clip End Fasteners on Shear Tied Fuselage Frames (MSD/MED)
- Aft Pressure Dome Outer Ring and Dome Web Splices (MSD/MED)
- Skin Splice at Aft Pressure Bulkhead (MSD)
- Abrupt Changes in Web or Skin Thickness Pressurized or Unpressurized Structure (MSD/MED)
- Window Surround Structure (MSD/MED)
- Latches and Hinges of Non-plug Doors (MSD/MED)
- Skin at Runout of Large Doubler (MSD) Fuselage, Wing or Empennage

In the case of the Airbus A300, a number of specific structural features may be identified that correspond to the generic items in the above list. These locations must be considered during the WFD assessment required to justify extended service of the aircraft.

The general approach for the WFD assessment used by Airbus Deutschland GmbH is presented in Figure 6. As for the local damages the results of the FSFTs including tear down are taken into account, but furthermore specific analysis methods are necessary to determine the WFD parameters.
Additionally, a large testing programme was launched to investigate the behaviour of structural items in the presence of MSD.
Both the WFD analysis tool and the WFD Testing Programme are described in the following chapters.

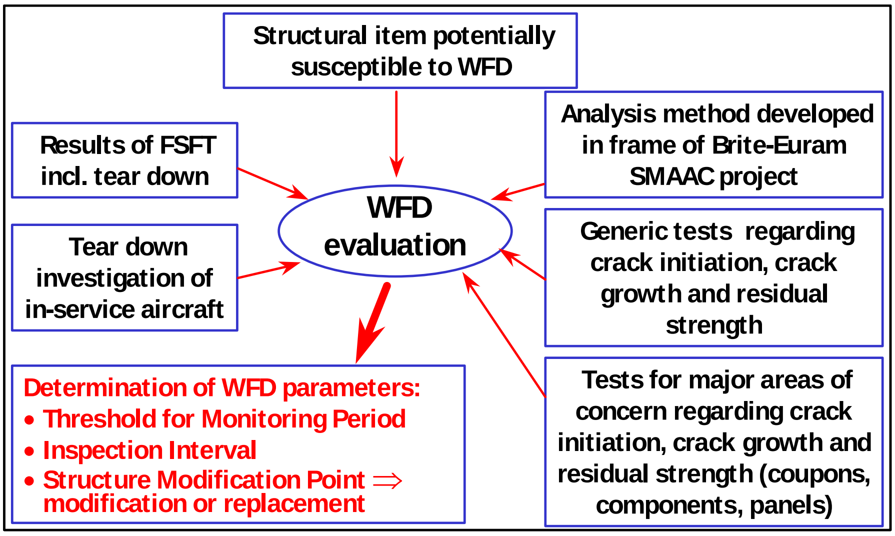


Figure 6 : Approach used for WFD Evaluation

Widespread Fatigue Damage Analysis

Effect of MSD

The cause of the *Aloha Airlines* accident was assigned to the presence of small cracks at adjacent fastener holes, MSD, in a skin lap splice leading to inflight structural failure of the upper part of the forward fuselage [3]. Therefore, the main issue of the aging aircraft fleet is the occurrence of multiple

damages at adjacent locations (MSD, MED) which influence each other. This so-called 'interaction of cracks' leads to higher stress intensity factors at the crack tips and consequently to higher crack propagation rates.
The effect of MSD is shown in **Figure 7**. In the presence of MSD adjacent to a lead crack the residual strength is reduced drastically. The drop of the residual strength from the capability of the intact structure to the capability required to withstand the design loads occurs in a much shorter time compared to the case of a local damage. Furthermore the crack growth for a MSD scenario is increased compared with the single crack. Together with the reduced critical crack length, this results in a significantly reduced crack growth period between the detectable and critical situation. The diagram also acknowledges the fact that while the MSD crack growth and residual strength degradation occurs in a more rapid sense, it is also expected to occur later in the life of a structural detail. This very rapid decrease of the residual strength in MSD situations due to crack interaction and accelerated crack growth must be taken into account during each analysis process.

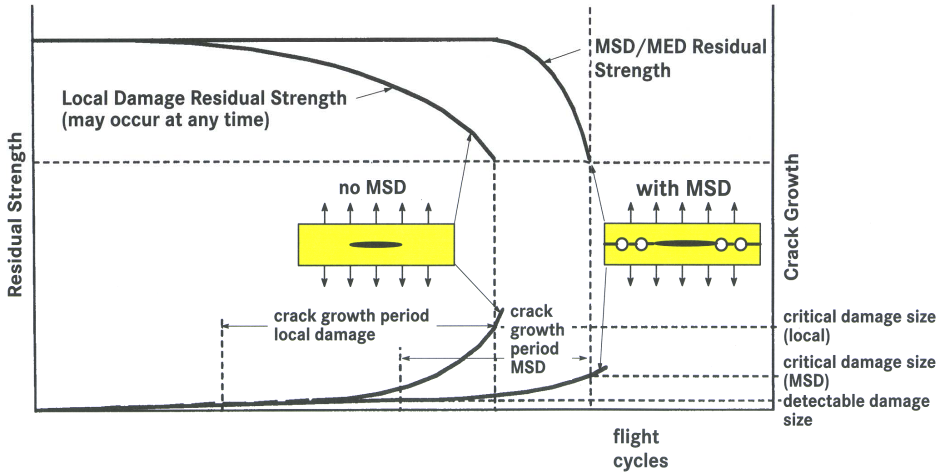


Figure 7 : Effect of MSD

Analysis Method

The analysis method for WFD evaluation was mainly developed by of the European research project 'Structural maintenance of Ageing Aircraft' (SMAAC) [4], partly founded by the European Commission. Significant additional effort was undertaken at Airbus to derive and validate an engineering tool for the Airbus structure to be evaluated. The development of this engineering tool was supported by extensive testing, which is described in the next chapter.
The main objective of the method is to estimate the development and evolution of MSD/MED in an aircraft structure. This information would permit the definition of an inspection programme for MSD/MED cracking. At a fundamental level, the methodology attempts to model both the initiation of multiple cracks at repetitive features within a structure susceptible to MSD/MED (e.g. a fastener hole in a lap joint), and the subsequent growth of those cracks after initiation. In the approach used by Airbus this is done by means of a Monte-Carlo simulation, where multiple initial crack scenarios are randomly generated and for each scenario subsequent crack propagation and failure is calculated (Figure 8). The method is summarized as follows [4]:

- The first step is the definition of the initial damage scenario. Each potential crack location (e.g. two sides per fastener hole in a lap splice) is allocated a crack initiation time. The initiation time is determined by randomly drawing a "life" from an overall log-normal distribution of fatigue lives for simple coupons.
- The propagation of each initiated crack is calculated through the techniques of linear elastic fracture mechanics. The major parameter within each crack propagation calculation is the stress intensity factor at the crack tip. Due to the nature of the MSD problem, within this model a number of solutions account for the interaction of cracks with other cracks. Plasticity effects are accounted for by considering Irwin's plastic zone in front of each crack tip. There are different ways of calculating the stress intensity factor, for example FEM, BEM, complex stress functions or compounding. Since a very important feature within a Monte-Carlo Simulation is the

computer time consumption, this model uses the compounding method, because it combines reasonable accuracy with very short calculation time compared to other methods.

- A damage accumulation procedure accounts for the effect of stress redistribution on the initiation of additional cracks. This stress redistribution is a consequence of the occurrence and growth of cracks Three main effects are considered here: the increase of net stress, the stress increase at the uncracked side of a cracked hole and the stress increase at an uncracked hole adjacent to a cracked hole. The increased accumulation of damage due to these effects leads to earlier crack initiation at the considered locations. This type of damage accumulation ensures a realistic simulation of the MSD behaviour in WFD susceptible components.
- Damage accumulation and crack propagation stop at failure. Three different failure criteria may be applied : reaching a critical stress intensity factor, net section yielding or exceeding a given lead crack size. A reliable estimate of the residual strength of a component in the presence of MSD is needed to complete the WFD analysis. Therefore, a R-curve analysis procedure has been implemented, which accounts for crack interaction. In this procedure each crack tip of a simulated crack scenario is investigated for unstable crack extension

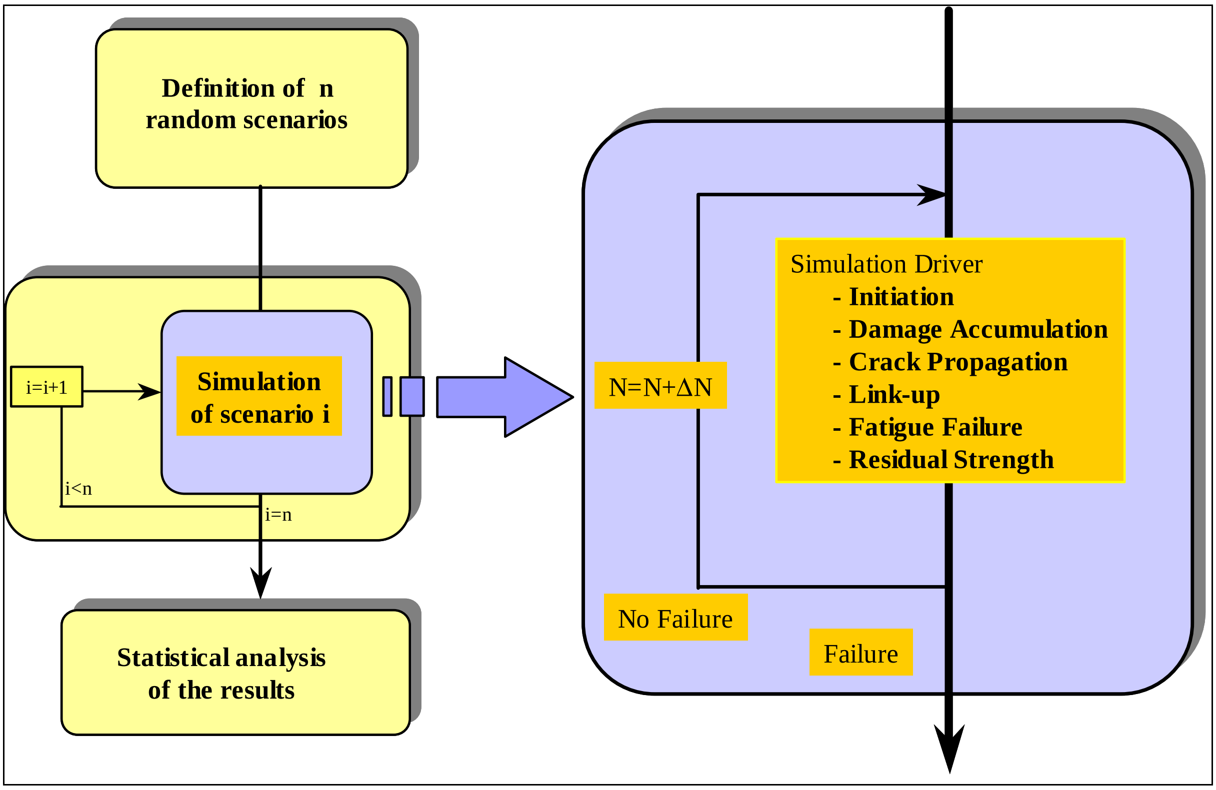


Figure 8 : Flow Chart of WFD Analysis Method

These stages form a single Monte-Carlo iteration (Figure 9) and are repeated many times to form a complete Monte-Carlo simulation of the structural item under investigation. The results of n iterations are evaluated statistically to obtain probability distributions, mean values and standard deviation for the Time to Initiation (left/blue curve in Figure 10), the Time to Detectable (mid/green curve in Figure 10)and the WFD Average Behaviour / mean of Time to Failure/Endurance (right/red curve in Figure 10).

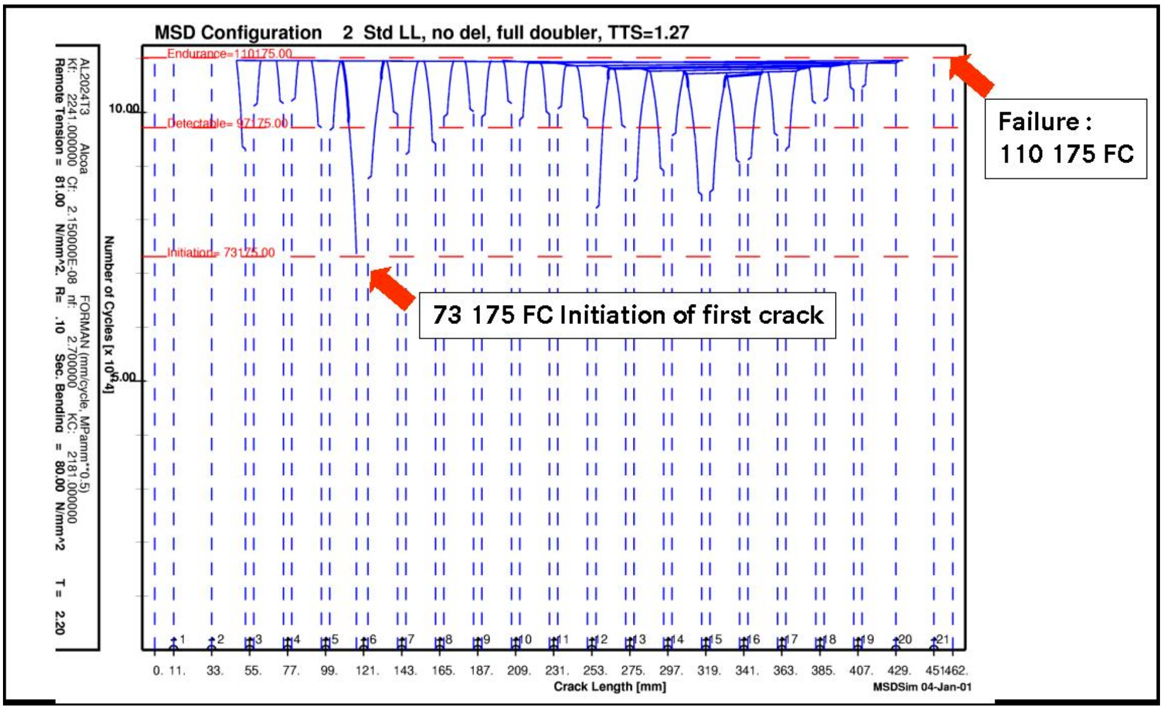


Figure 9 : Result of a single Monte Carlo Iteration : shown is the evolution of a scenario in a lap joint, i.e. the position of crack tips/holes against the number of cycles

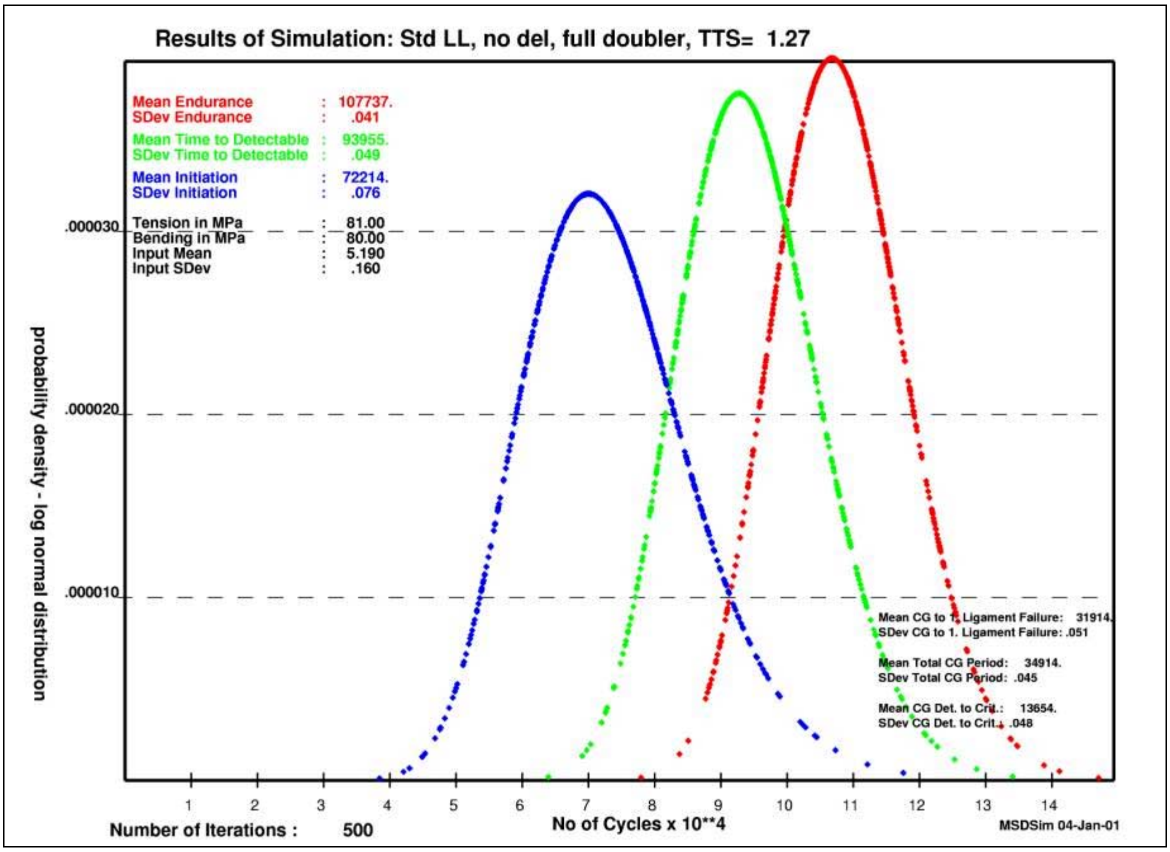


Figure 10 : Results of a complete Monte Carlo Simulation comprising 500 iterations

For relatively simple situations, such as constant amplitude cyclic loading, it is possible to evaluate rapidly many individual Monte Carlo scenarios with reasonable computational effort (of the order of 100 scenarios per minute on a UNIX workstation). To obtain stable results the number of Monte Carlo iterations performed should be in the order of 250 to 500 as illustrated in Figure 11. However, the incorporation of more complex features within the model, such as the calculation of crack growth under spectrum loading or the use of sophisticated techniques such as finite element modeling would require significantly more computation time and would therefore limit the number of scenarios. A careful balance between accuracy and speed needs to be achieved.

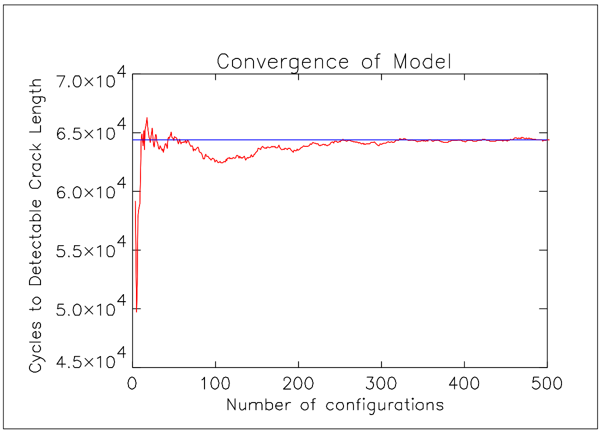


Figure 11 : Number of Monte Carlo iterations to obtain a stable mean value

The analysis tool (MSDSim) has been validated theoretically through comparison to existing fracture mechanics tools, such as NASGRO [5], FRAN2D [7][8][9], AFGROW [6], etc. An example is given in Figure 12 : Special emphasis has been given to the investigation of the crack interaction phenomenon. The stress intensity factors calculated for two approaching crack tips have been compared to the FRANC2D calculation to ensure accurate performance of the code.

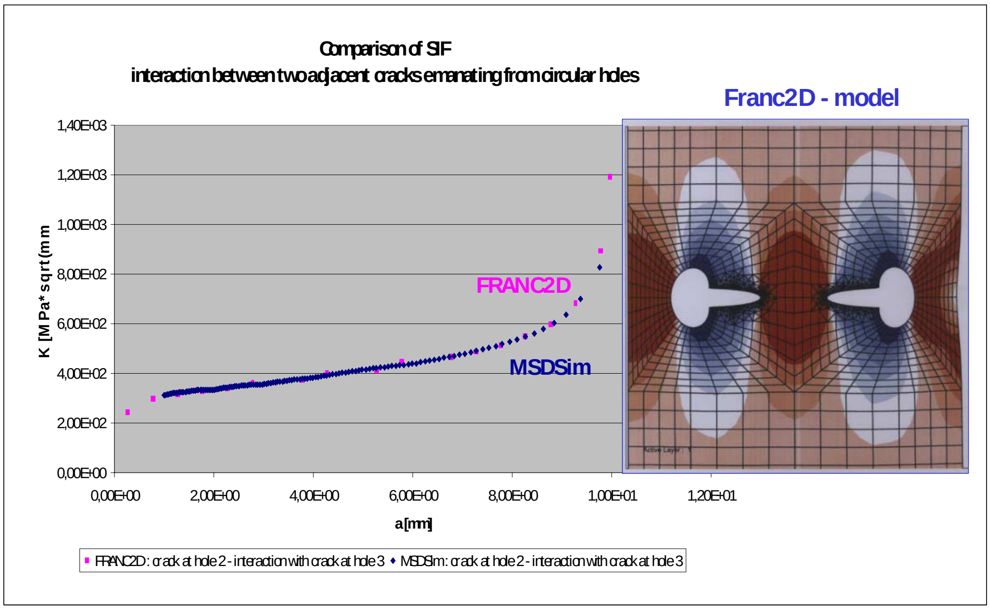


Figure 12 : Comparison of stress intensity factor calculation for two interacting cracks

Additionally, extensive comparisons to test results derived from the Airbus WFD Testing Programme as well as from Full Scale Fatigue tests have been done to validate the code on an experimental basis. The next chapter outlines details of this test programme and provides examples for the experimental validation.

WFD Testing Programme

Two general objectives are the background for the test programme performed by Airbus Deutschland GmbH:

1. provide a sound experimental basis for the WFD assessment and
2. to provide experimental data for the verification of the analysis tool described above.

The test programme is intended to cover the major structural areas of the A300 under responsibility of DA, which are potentially susceptible to Multiple Site Damage and subsequently to Widespread Fatigue Damage. Since longitudinal and circumferential joints have a high potential to develop MSD simply due to their large number of structural details operating at a similar stress levels, these areas form a large part of the WFD test programme. Another high priority MSD item is the rear pressure bulkhead skin.
Consequently, three major test batches covering different variations in geometry and conditions have been performed :

1. Longitudinal Lap Joints
   - Standard A300 lap joint
   - Standard A300 lap joint without doubler delamination
   - Lap joint variants
   - Panels from retired A/C
2. Circumferential Butt Joints
   - A300 circumferential joint
   - A300 Circumferential joint with doubler delaminated
   - New panels
3. Rear pressure bulkhead skin at splice of attachment angle

To fulfil the second general objective an additional batch comprising generic tests has been tested, tuned to investigate specific features of the analysis tool.

The detailed objectives of the test programme were :

- Evaluation of multiple crack propagation, i.e. MSD crack scenarios,
- Investigation of residual strength for different length of lead crack in combination with different MSD scenarios
- Investigation of the occurrence of WFD, i.e. the fatigue concept.

The crack propagation specimen contained artificial corner or through cracks of different lengths. The implementation of artificial cracks ensured that MSD scenarios were obtained, i.e. crack scenarios, where multiple cracks start growing, link up and finally cause failure, rather than local damage scenarios, where an isolated crack grows to failure.

In total 463 small (width 440mm) and large (width > 1000mm), unstiffened and stiffened coupons were tested and 4 curved panel tests were conducted.
The two longitudinal lap joint panels were cut from a retired aircraft, which had already accumulated 36 000 flight cycles. The panels were then tested for remaining fatigue life, crack growth and residual strength.
For the experimental validation of the analysis tool the majority of the coupon tests have been simulated analytically. Figure 13 provides an example of the comparison between calculation and test results. Shown is the crack propagation associated with each rivet hole in a joint containing a combination of 1.27 mm (0.05 inch) and 0.127 mm (0.005 inch) cracks.

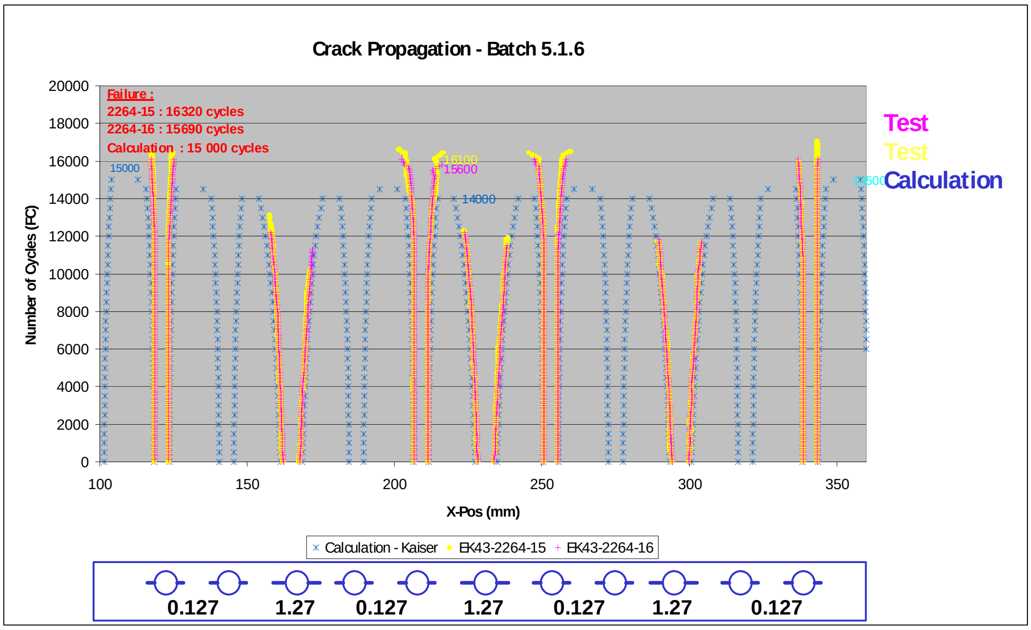


Figure 13 : Experimental validation of the analysis tool using data from the WFD Test Programme

The validation of the analysis tool included the growth and link-up of multiple cracks until specimen failure in complex structural items such as riveted joints. The comparison between test and analysis proved that the WFD analysis tool is able to simulate crack scenarios of varying complexity ranging from simple hole cracking to complex MSD crack scenarios at riveted joints. The analytical results were generally slightly conservative compared to the test results, as expected from any analysis tool. Furthermore, lap joint damages resulting from MSD crack initiation and propagation in A300 Full Scale Fatigue Test were successfully simulated.

Additional Programmes : Tear Down and Corrosion Prevention and Control Programme

A special activity under the Airbus A300 life extension programme is the tear down investigation of an old in-service aircraft. Airbus bought the airframe of the A300 MSN 008 which has accumulated approximately 75 percent of its DSG. The following investigations have been performed :

- Inspection and tear down investigation of areas potentially susceptible to WFD, especially longitudinal lap joints, circumferential joints, rear pressure bulkhead attachment to fuselage.
  - Inspection of 50 m longitudinal lap joint according to NTM
  - Inspection of special areas of long. lap joints with Rototest and Microfractographics
  - Inspection of rear pressure bulkhead according to NTM
  - Inspection of complete rivet row one of pressure bulkhead with Rototest
- Inspection and tear down investigation of areas where local damages (cracks) may occur.·
- Inspection and tear down investigation of areas susceptible to corrosion.
- Crack growth testing of skin material to investigate possible material degradation : several structural parts will be tested to derive material data, especially da/dN and fracture toughness data, for comparison with former and actual material properties

Another important topic under the life extension activities is the prevention of corrosion in the ageing fleet. Therefore a Corrosion Prevention and Control Programme (CPCP) was established which is reviewed periodically.

Conclusion

Since an increasing number of Airbus A300 aircraft are approaching their original DSG or will reach their DSG in the near future, Airbus Industry has defined new ESG and launched a Full Life Extension Programme for the A300 models, i.e. A300B2, B4-100, B4-200 and B4-600. In order to justify a further period of operation up to the new ESG, it is necessary to review in-service experience and re-assess the existing inspection programmes. For pressurized fuselage, which is mainly under responsibility of Airbus Deutschland GmbH, this includes a complete fatigue and damage tolerance analysis of the original structure, modifications and repairs, which may lead to a modification of the maintenance strategy, including the inspection of additional items or an increased level of surveillance in some areas.
In order to ensure compliance with the new requirements, which concern the need to limit the validity of the current structural maintenance programme and the need to impose operational requirements that mandate a structural maintenance programme to prevent Widespread Fatigue Damage in the fleet, special emphasis is given to the analysis of MSD and MED, which are both defined as sources of WFD.
A new analysis tool was developed to assess structural items potentially susceptible to WFD and a large Widespread Fatigue Damage Test Programme, including 463 small and large coupon tests and 4 large curved panel tests, was conducted. Special inspections tuned to MSD/MED will be established between the Inspection Start Point (ISP) and the Structural Modification Point (SMP), the point beyond which the aircraft may not be operated without further evaluation or modification. The calculation of these points and the inspection interval is done by means of a Monte-Carlo simulation to reflect the statistical/probabilistic nature of MSD, where multiple initial crack scenarios are randomly generated and for each scenario subsequent crack propagation and failure is calculated.
Significant effort was undertaken to develop and validate the computational tool for the Monte Carlo simulation with special consideration of balancing the needed accuracy against the speed of complex calculations.

The large number of activities undertaken to obtain the goal for the A300 Life Extension Programme may be summarized as follows :

- Evaluation of structure potentially susceptible to local damage
    - Evaluation of in-service and Full Scale Fatigue Test Experience
    - Tests for areas susceptible to local damage
    - Conventional fatigue and damage tolerance analysis
- Evaluation of structure potentially susceptible to MSD/MED
    - Evaluation of in-service and Full Scale Fatigue Test Experience
    - WFD testing
    - Development of MSD/MED analysis tool
    - Validation of analysis tool
    - Complex MSD/MED analysis
- Tear Down investigation of a retired aircraft
- Evaluation of repairs, in-service problems and Structural Repair Manual
- Definition of new inspection programmes, adjustment of existing inspection programmes
- Definition of modifications

Figure 14 presents the distribution of financial effort spent for the major Full Life Extension activities (evaluation of repairs and definition of modifications not included).
Approximately 30% of the total effort was required for the evaluation of local damage (including local damage testing), while tear down and definition of inspection programmes only make up a small part of the total amount. It becomes obvious that the largest portion, almost 70% of the total effort, is assign to the evaluation of WFD with approximately half of that amount being consumed by WFD testing. However, 15% of the total amount was required for the development and validation of the analysis tool – this amount would not be counted for any follow-up life extension programme, e.g. for A310, A320 models. Furthermore, a large portion of the WFD testing was conducted in order to obtain

data for the validation of the analysis tool, which would not be necessary for future projects, and large datasets are applicable to other Airbus aircraft.
Consequently, as shown in **Figure 15** for future life extension projects the effort assigned to WFD analysis would be significantly reduced (roughly in the order of 40%).

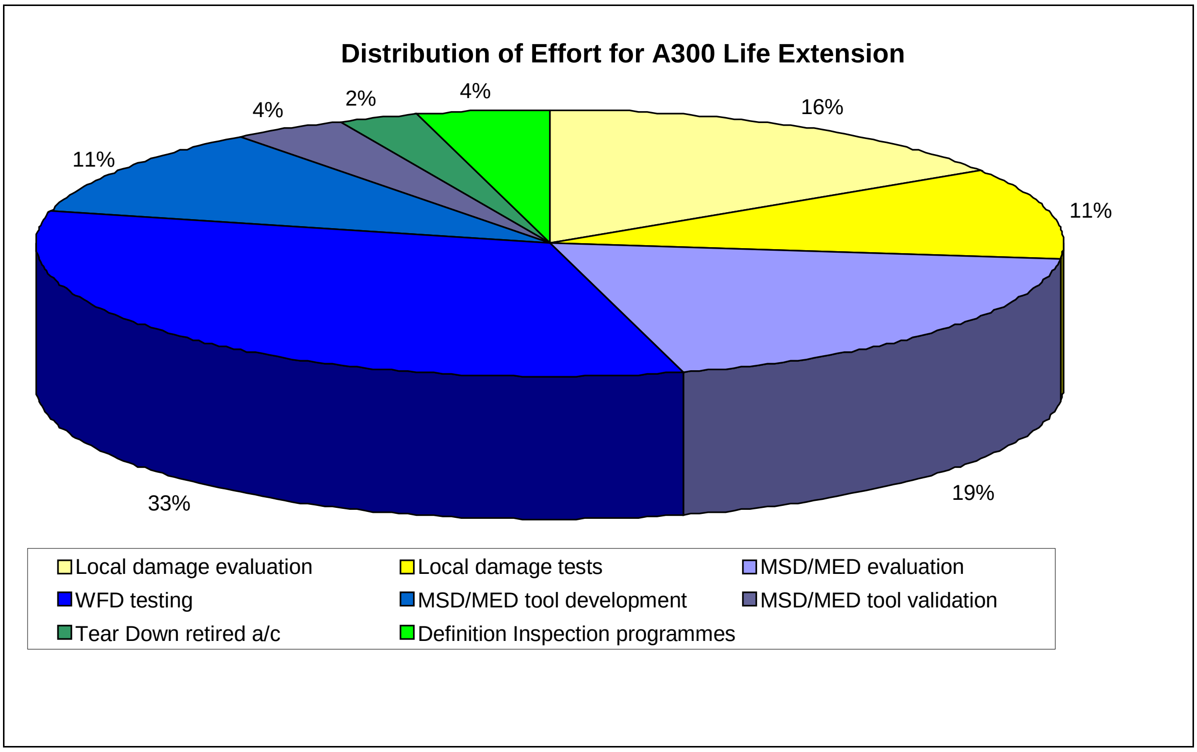


Figure 14 : Distribution of Financial Effort for A300 Life Extension (on man-months basis)

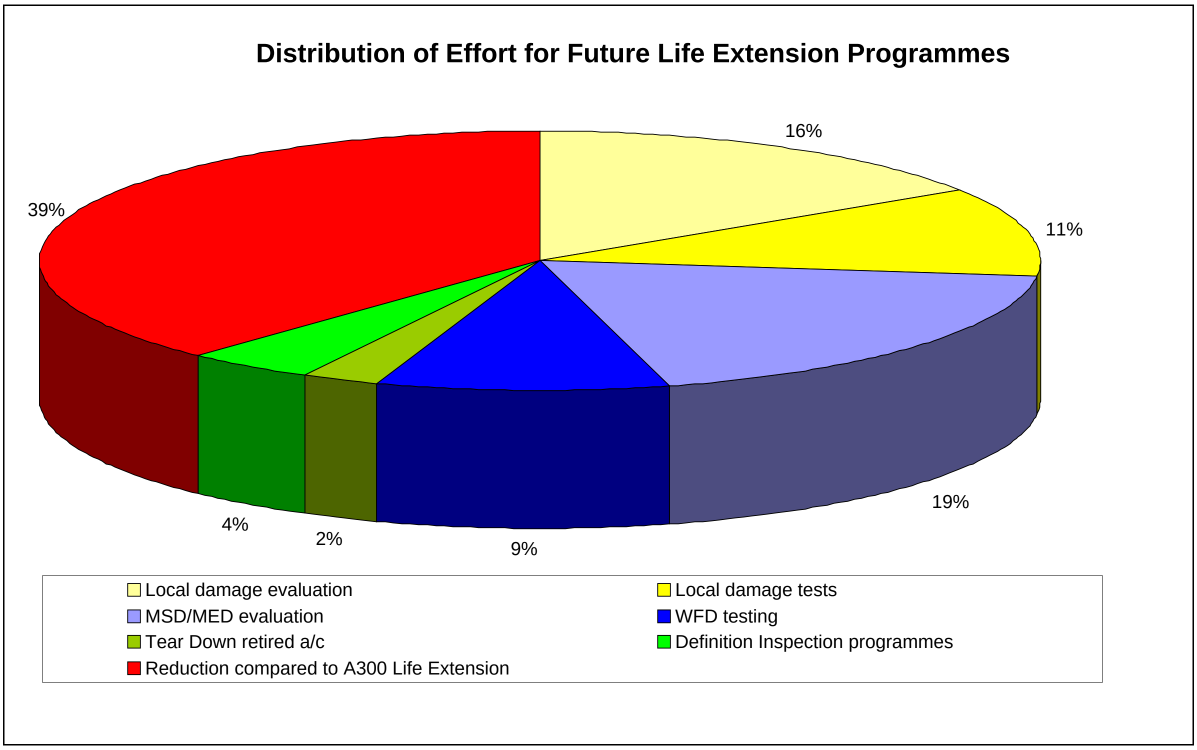


Figure 15 : Distribution of Financial Effort for Future Life Extension Programmes

References

[1] H.-J. SCHMIDT, B. SCHMIDT-BRANDECKER, H. TREY, ‘Airbus A300 Fuselage Program for Life Extension and Widespread Fatigue Damage Evaluation’, The Third Joint FAA/D0D/NASA Conference on Aging Aircraft, Sept. 1999, Albuquerque, New Mexico, USA.

[2] J.McGUIRE & J.FOUCAULT, ‘Recommendations for regulatory action to prevent widespread fatigue damage in the commercial airplane fleet’, Airworthiness Assurance Working Group, Transport Airplane and Engine Issues Group, Aviation Rulemaking Advisory Committee, Federal Aviation Administration, Washington, D.C., USA (March 1999)

[3] ANON, ‘Aircraft accident report, Aloha Airlines, flight 243,Boeing 737-200, N73711, near Maui, Hawaii, April 28, 1988’, Report No. NTSB/AAR-89/03, National Transportation Safety Board, Washington, D.C., USA (June 1989)

[4] P. HORST, H. TREY, ‘Structural Maintenance of Ageing Aircraft : SMAAC’, Air & Space Europe, Vol 1, No. 5, 1999.

[5] R.G. FORMAN, Shivakumar, V., Mettu, R.S., Newman, J.C., “Fatigue Crack Growth Computer Programme NASGRO, JSC-22267B, Apr. 1999.

[6] J.A. HARTER, “AFGROW Users Guide and Technical Manual“, AFRL-VA-WP-TR-1999-3016, Feb. 1999.

[7] J.C. NEWMAN, D.S. Darwicke, C.A. Bigelow, ‘Finite Element Analysis and Fracture Simulation in Thin-sheet Aluminium Alloy’, NASA TM 107662, August 1992, NASA Langley Research Center, Hampton, V.A.

[8] P. WAWRZYNEK, A.R. Ingraffea, ‘Interactive Finite Element Analysis of Fracture Processes : An Integrated Approach’, Theoretical and Applied Facture Mechanics, v.8, pp. 137-150

[9] D. SWENSON, M. James, ‘FRANC2D/L : A Crack Propagation Simulator for Layered Structures’ – User’s Guide.

This page has been deliberately left blank

Page intentionnellement blanche

Bonded Repair Technology for Aging Aircraft

Dr. Alan Baker and Dr. Richard Chester
Airframes and Engines Division
Defence Science and Technology Organisation
Aeronautical and Maritime Research Laboratory, Australia

Mr. James Mazza
United States Air Force Research Laboratory
Materials and Manufacturing Directorate (AFRL/MLSA)
Wright-Patterson AFB OH, USA

Abstract

NATO weapons systems are being extended well beyond their design lives due to decreases in defence budgets and the rising costs associated with procuring new hardware. This situation makes it increasingly important that methods for extending the lives of these weapon systems in a cost-effective manner be developed and implemented to the greatest extent possible. Adhesive bonding technology, particularly bonded composite repairs/enhancements, has been successfully applied by several nations to extend the lives of aircraft by bridging cracks in metal structure, reducing strain levels, and repairing areas thinned by corrosion. Bonded composite reinforcements are highly efficient and cost effective when compared to conventional mechanically fastened approaches. In some cases, bonded repair technology is the only alternative to retiring a component. This technology has already resulted in the documented savings of hundreds of millions of dollars in Australia and the United States.

This paper describes the advantages of bonded composite repairs over conventional repair methods. Bonded joint design/analysis, installation procedures, nondestructive inspection, certification issues, and other key aspects of the technology are generally addressed. Examples of applications to aircraft are used to illustrate these issues as well as demonstrate bonded repair advantages. The capabilities and resources required to successfully apply bonded repairs are discussed. Finally, several recent reviews of this technology area are summarised to indicate where the key scientific gaps remain and to suggest research that should be undertaken to further enhance the usefulness of the technology.

1. INTRODUCTION

Military aircraft are increasingly being required to remain in service for times that are longer than their original design lives. This is due to both decreasing defence budgets and high costs for replacement aircraft. As the aircraft age, managers will face ever-increasing amounts of damage that will require repair. Conventional repair methods typically involve either replacement of the damaged component or installation of a

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

mechanically fastened repair. Both approaches are well established but suffer from lengthy aircraft down times and high costs. In addition, mechanically fastened repairs may not be viable for certain applications. The approach of using adhesive bonding to repair or reinforce damaged aircraft structure has been shown to be a highly cost effective alternative to the conventional repair methods [1]. Thousands of adhesively bonded repairs have now been applied to hundreds of aircraft in service with the Royal Australian Air Force (RAAF) [2] and United States Air Force (USAF) [3] since the middle 1970s.

Mechanically fastened repairs are usually less expensive than component replacement, however, the need to drill new holes for the fasteners in the structure is a major limitation. These holes will act as stress concentrators and may result in the initiation of new fatigue cracks. As a mechanically fastened repair transfers load only through the fasteners, it is not particularly stiff and so the damage must usually be removed from the structure. Adhesively bonded repairs are much more efficient due to the uniform load transfer mechanism and repairs can therefore typically be made to untreated or "live" cracks. Validated, predictive methods now exist to calculate the crack growth rate after repair to assist with certification. By avoiding the need to remove the damage, adhesively bonded repairs are much less intrusive, are less likely to cause unexpected damage (to hidden wiring or hydraulics for example) and are faster to apply.

The technology can also be used very effectively to reinforce undamaged structure that is known to be under designed and in danger of developing fatigue cracks, for example, at some later stage. In this regard, the technology is highly effective in extending the life of an aging aircraft structure.

As the technology has matured and repairs have become more routine, the aircraft operators have required the development of rigorous engineering standards to underpin the application of repairs. Importantly, the technology has also been incorporated within aircraft structural integrity programs that provide the required level of engineering management. This includes issues such as training of staff in design and application procedures, quality control and configuration control. Successful use of the technology requires an overarching engineering management framework such as this.

Adhesively bonded repairs are routinely applied to tertiary and secondary structure, and there have been significant applications to safety-of-flight-critical (primary) structure. Research is now focusing at making these repairs even easier to design and apply. It is also addressing the certification concerns that currently prevent aircraft maintainers from taking full benefit of the technology on primary structure applications.

2. DESIGN

2.1 Design Approaches

The first requirement is to assess the defect, assumed here to be a crack, in terms of its length and depth, and to determine the thickness and geometry of the cracked region as well as the local loading conditions. Of particular importance in adhesively bonded repairs is the available overlap length on either side of the crack. The thicker the structure and consequently the higher the loads, the longer the overlap length needed to transfer the loads into the patch. Highly loaded repairs require appropriately thick patches

[4] a) to provide adequate reinforcement, b) to prevent failure of the patch and c) to prevent failure of the adhesive.

In the most usual case where the repair can be applied to only one side of the structure, the degree of support of the structure against secondary bending must also considered. Secondary bending which results from the displacement of the neutral axis of the parent structure by the patch can markedly change the loads experienced by the patch and structure [5].

The temperature and environment experienced by the region to be repaired must also be considered, since this will help determine the type of adhesive to be used. Adhesive choices are considered later.

It is unlikely, other than for relatively simple cases such as fuselage repairs, that information on local loading will be available, unless there is access to detailed design data. Failing this, a good approximate estimate of the stress at design limit load (DLL) is achieved by equating design ultimate load (DUL) with the material yield stress σ_y. The basis for this is that a far-field stress level equal to material yield stress failure could occur for aluminium alloys at regions where Kc > 1.2. Then DLL is the yield stress divided by 1.5. This has proven to be a conservative estimate in all cases examined by one of the authors (AAB) where the DLL was known. Table 1 is an example of a correlation. Clearly, assumption of the ultimate strength σ_u as equal to σ_{DLL} is over conservative.

DADTA Item No	σ_{DLL}	σ_y	σ_u	σ_u/σ_y	$\sigma_y/1.5$	σ_y/σ_{DLL}	$\sigma u/\sigma_{DLL}$
67	202.9	400.2	462.3	1.2	266.8	1.97	2.28
70	167.0	400.2	462.3	1.2	266.8	2.40	2.77
70a	204.2	400.2	462.3	1.2	266.8	1.96	2.26
78	149.7	400.2	462.3	1.2	266.8	2.67	3.09
154	171.8	400.2	462.3	1.2	266.8	2.33	2.69
194	165.6	400.2	462.3	1.2	266.8	2.42	2.79

Table 1: *Data on Design Limit Stress σ_{DLL} for F-111 for several (DADTA) data points in the lower wing made of aluminium alloy 2024-T581, compared with the yield stress σ_y.*

A knowledge of the loading spectrum for the region is also unlikely to be available, so the best approach, if such detail is required for design, is to assume one of the standard spectrums, FALLSTAF or TWISS, according to whether the aircraft is a fighter or transport.

Once sufficient information is available concerning the loading and other parameters and it is considered on the basis of the forgoing discussion that a bonded repair is feasible, a patch design can be undertaken. For patch design, there are two basic approaches: analytical usually based on the Rose model [6] for simple loading and geometries or, for more complex geometries or loading, finite element analysis. CalcuRep, developed by the USAF Academy, is an established software program for the analytical approach. Earlier, an iterative approach was developed by Baker [4] to determine optimum patch dimensions. In patch design, the aim is to determine the reduction of stress intensity experienced by the repaired crack as well as the stress levels in the patch and adhesive system. The goal is to avoid failure of the patch system, while providing a sufficient reduction in stress intensity to ensure the required service lifetime for the repair.

Fatigue studies [7] were undertaken to validate use of the Rose model for patch design. Generally, it was shown that satisfactory prediction of patching performance could be made over a range of variables, including stress patch stiffness and R ratio based on the usual semi-empirical crack growth equations and the predicted the stress intensity. It was also found possible to extend the model to include allowance for disbond growth. Figure 1 shows a standard log/log plot for crack growth versus predicted ΔK for a range of patch thicknesses.

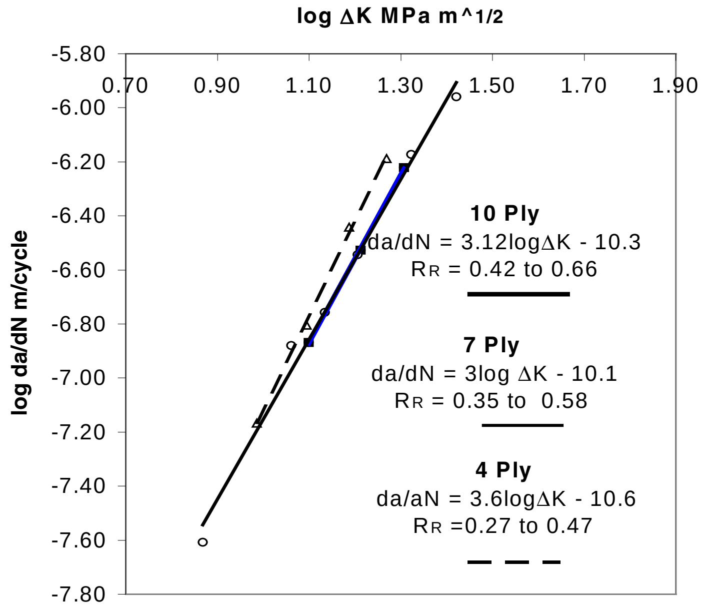


Figure 1: *Plots of crack growth rate da/dN versus log ΔK for various thickness boron/epoxy patches.*

2.2 Patch and Adhesive Choices

For the patch material, there are three main options generally considered: the fibre composites boron/epoxy and carbon/epoxy and the aluminium alloy-glass/epoxy laminate GLARE. The need is for high strength and stiffness, fatigue and environmental durability and formability. The composites satisfy most of the requirements; however, their main disadvantage is their low thermal expansion coefficient which gives rise to undesirable residual tensile stresses in the repaired component.

Most Australian and U.S. repairs to date have used boron/epoxy as the reinforcement rather than graphite/epoxy by virtue of its superior properties for this application:

- Better combination of strength and stiffness.
- Electrically nonconductive, avoiding galvanic corrosion problem with aluminium and facilitating eddy-current nondestructive inspection (NDI) of cracks under the repair.
- Higher coefficient of thermal expansion (CTE), minimising residual-stress problems.
- Better fibre alignment under cocure conditions as a result of much larger fibre diameter — 125μm compared with 8μm for graphite fibres.

However, compared to carbon/epoxy, boron/epoxy is much more costly, less readily available and because of the large fibre diameter less formable. Thus carbon/epoxy is used whenever it is more cost effective or where very high formability is required. Work by Poole [8] in the UK, has shown that carbon/epoxy provides very satisfactory properties as a patch material.

GLARE (aluminium/fibreglass laminate) is an alternative patch material that has high fatigue resistance and important benefits [9] where minimising residual stresses is important; however, it has limited formability and relatively low stiffness so is best suited to the repair of thin-skinned fuselage components.

The optimum choice for the adhesive is generally a structural epoxy film. The main adhesive used in Australian repairs is Cytec Fiberite FM 73, a nominally 120°C-curing epoxy-nitrile structural film. This adhesive, or a similar epoxy film, is also most often used in the U.S.. Reasons for this choice include the following:

- Excellent strength and toughness from low to moderate temperatures
- Resistance to aircraft fluids.
- Ability to form strong durable bonds using appropriate prebond treatments.
- Ability to cure (with some sacrifice in properties) at relatively low temperatures - as low as 80°C (with extended times) compared with the standard one hour at 120°C.

The first three advantages are typical of most moderate-temperature-curing structural epoxy-nitrile film adhesives. However, the ability of FM 73 to cure at temperatures as low as 80°C is both unusual and valuable for repairs where the higher temperatures cannot be achieved or where there is a need to minimise residual stresses. For higher-temperature applications (above 80° C) the adhesive FM 300-2, also by Cytec Fiberite, is most often selected. This adhesive also has a capacity to cure at a relatively low temperature (120°C) while providing properties typical of a 175°C-curing adhesive. Finally, modified acrylic adhesives have been found to be highly effective for less demanding applications (temperatures not exceeding 60°C or not below –10°C, if peel stresses are high) or where the use of elevated cure temperature is not feasible. Some two-part epoxy paste adhesives may also be used when elevated-temperature curing is impractical or undesirable, but these are typically confined to non-critical applications.

2.3 Processing Choices

The processes by which the adhesive and patch materials are installed on the aircraft have a direct influence on the final properties and long-term durability of the repair. The material properties considered for design should take into account the effects of these processes, such as the cure cycle (time/temperature) and pressure application method used for adhesives and cocured patches. Selection of repair area and patch surface preparation processes is also an important design consideration. This task can be somewhat difficult since no tests exist that can quantitatively correlate laboratory test performance with service life. Prior experience and screening using the wedge test [10] have been used to select metal surface preparations for most Australian and U.S. repairs. However, it is important to note that the test specimen but not necessarily the moisture conditioning and acceptance criteria are per the referenced wedge test standard.

3. INSTALLATION

Successful bonded repair installation is not necessarily difficult but requires proper execution of a number of steps. These include surface preparation both for the aircraft structure and patch material as well as heating and pressurisation. Other considerations include the nature of the repair installation environment, handling of repair materials, health and safety issues, training of repair installers and post-bond operations.

3.1 Surface Preparation

Preparation of adherend surfaces prior to bonding is the single most important application process step for ensuring a successful repair [11]. Proper surface preparation is necessary to achieve initial bond strength and long-term durability in the service environment. Although the environment includes temperature extremes and exposure to many aircraft fluids and maintenance chemicals, moisture tends to be the biggest impediment to long-term durability, particularly for bonded aluminium joints [12].

A metal surface preparation typically must remove contaminants and naturally occurring oxide from the metal surface. It must also chemically and/or physically modify the surface to promote adhesion with the adhesive (or primer) and enable it to resist moisture attack. Cleaning and deoxidising alone can sometimes provide adequate initial adhesion but rarely result in bonded joints with long-term service lives. For repair applications, only very simple and nonhazardous treatments that can be applied on aircraft under field conditions are considered viable for most applications. Ideally, such a surface preparation will yield highly durable bonded joints with a variety of metal substrates.

To satisfy these requirements, Australian work is focused on the use of silane coupling agents. The coupling agent found most suitable for epoxy adhesives is the epoxy-terminated silane, γ-GPS [13]. This coupling agent provides high-strength durable bonds to aluminium alloys, stainless steel, nickel and titanium alloys. It is applied from an aqueous solution to the metal surface following mechanical conditioning by alumina grit blasting. The silane treatment is safe since it does not rely on noxious chemicals or electrical power. The process does not use acids, so it eliminates the corrosion concerns they cause if not properly rinsed and their potential to embrittle high-strength steel fasteners.

The durability against moisture degradation provided by the silane can be further enhanced by use of a standard corrosion-inhibiting primer. Figure 2 shows typical results for crack growth in the wedge test for 2024-T3 aluminium bonded with FM 73 adhesive following a) grit-blasting, b) grit-blasting + silane and c) grit-blasting + silane + primer. Although this test is considered by the authors to be the best available method to screen surface preparations, it is not ideal since test criteria that directly relate to bonded joint service cannot be established. The test can compare different surface preparations, with all other factors being equal, and relate the results to service experience. Minimal crack extension following exposure to hot/wet conditions and a crack that remains predominantly in the adhesive (rather than at a metal interface) indicate a good surface preparation. For FM 73 adhesive, conditioning at 50°C and 100% relative humidity (RH), crack growths of about 5 mm after 7 days of exposure tend to indicate an adequate surface treatment. This behaviour is exhibited by system c) in the figure. However, despite the longer crack extension and interfacial failure modes, system b) has performed adequately in service for many applications.

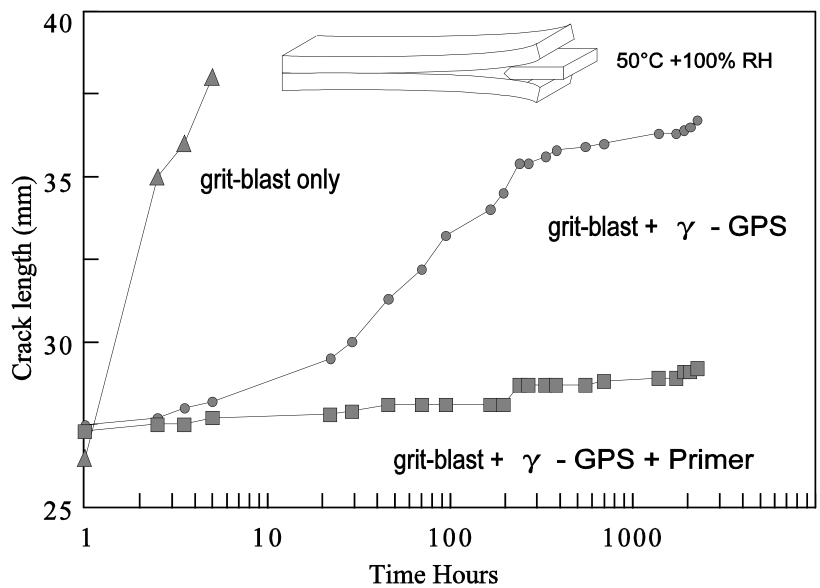


Figure 2: *Plots of crack growth versus time for wedge-test specimens (illustrated inset) made of 2024-T3 aluminium and subjected to the surface treatments indicated, prior to bonding with FM 73 adhesive.*

Most U.S. repairs have followed the Australian lead and employed a grit-blast/silane (GBS) surface preparation [14], including application of a corrosion-inhibiting adhesive bond primer, as a practical on-aircraft prebond treatment that yields good in-service environmental durability. Recently, a process based on sol-gel chemistry developed for the USAF by The Boeing Company has emerged for applications on the same metal alloys currently treated using GBS [15,16]. This approach is similar to GBS but employs a more reactive chemistry. The results are a simpler, quicker process that performs as well or better than GBS in laboratory tests, including the wedge test. A lesser-performing variant of the sol-gel approach that may be suitable for many noncritical applications eliminates the grit-blasting and priming steps [17].

For precured (thermosetting) fibre composite patches, surface removal by light grit blasting with alumina is a highly effective treatment that provides excellent bond strength and durability [18]. The standard peel-ply surface treatment procedure is not as effective unless followed by grit blasting or some other effective mechanical method of surface removal. Often, a cocured layer of adhesive is applied to the surface of the boron/epoxy patch to increase the toughness of the surface resin and to provide a layer more suitable for grit blasting. The surface of the as-received GLARE material is a cured corrosion-inhibiting adhesive bond primer. Solvent cleaning followed by lightly abrading constitutes an effective prebond treatment. Abrasion debris should be dry removed (without solvent), and care must be taken not too damage the thin primer layer.

3.2 Heating and Pressurisation

Heating and pressurisation are key installation issues since both can have a direct impact on the quality of the repair. Controlled heating is required to cure adhesives and cocure composite patches on the aircraft. Heating may be required with certain surface preparations and for adhesive primer cure. It may also be necessary for drying structure prior to repair installation. Pressure application is needed to mate the patch to the aircraft structure. Adequate pressure must be applied to ensure proper bondline thickness and minimise bondline voids and porosity. It also causes the adhesive to flow and properly "wet" the treated surfaces to achieve adequate adhesion. In the case of cocured composite patches, pressure may be required to consolidate the composite in order to obtain the desired mechanical properties.

Heating may be conducted by any of a number of methods provided they are able to safely control the temperature in the repair area within prescribed tolerances without contaminating the repair. Typical on-aircraft heating methods include electric-resistance heat blankets, infrared heat lamps and hot air devices. Application specifics determine the method best suited to a given repair. Heat blankets are typically used to cure adhesives, whereas heat lamps are usually the choice for silane drying and precuring primers. All heating devices must be controlled by some means so that heat can be applied when and to the levels required. This is particularly important for adhesive cure since controlled heat-up and cooling rates are usually prescribed. "Hot bonders" that automatically control heating based on temperature feedback from the repair area are normally used with heat blankets. These units or similar means can be employed to control heat lamps and hot air devices.

Attaining specified repair temperatures within desired tolerances can often be difficult on aircraft, since portions of the structure can act as "heat sinks." These regions conduct heat away from the repair site and become locally cooler, creating the possibility that the adhesive may not fully cure. Thermal surveys of the repair area are important to ensure proper heating will be attainable. The surveys should be conducted on the actual repair area using the equipment to be employed during the repair. They can determine the placement of insulation materials or the locations needed for supplemental heating, and they will reveal the required temperature readings for surrounding "monitoring" locations that can be used to determine the temperature in the repair area.

Pressure application on aircraft may also be achieved by a variety of means. These include vacuum bag, inflated bladder or various forms of mechanical pressure. The use of a vacuum bag is the most common since it is almost always the most convenient. Vacuum bags are light, conform to almost any surface, apply uniform pressure, can remove volatiles from the repair area and can hold a heat blanket in place. To apply pressure this way, a bag is built over the repair area and air is extracted, allowing atmospheric pressure to be applied. In most cases, it is not desirable to achieve a full vacuum throughout the cure cycle since the vacuum allows volatiles in the adhesive, such as moisture or solvents, to volatilise more readily, and it allows entrapped air to expand more easily. Often, high vacuum levels are applied initially to remove volatiles from the repair, then vacuum levels are reduced before the adhesive gels in order to minimise porosity in the bondline. Levels corresponding to 0.034-0.069 MPa pressure are common. Bladders inflated with air can be used to apply positive pressure (as opposed to vacuum) on a repair area. This may be desirable to minimise void formation due to the evolution of volatiles. However, this approach is not usually convenient since bladders must be held against the structure in some way and they require a frame or fixture to react against. If this fixture is fastened to the structure, pressure is limited to prevent damage. Mechanical pressure may also be applied by clamping or other means. Again, these forces must be reacted, and it may be difficult to apply uniform pressure over a large area.

Fibre composite patches should be precured under positive pressure per manufacturer's recommendations whenever possible. This is normally done in an autoclave to minimise porosity and achieve the per ply thickness value envisioned by the design. The cured patch can be nondestructively inspected prior to application on the aircraft. The knowledge that the patch has minimal porosity, typically a couple percent or less, eases the on-aircraft inspection burden for the adhesive bondline. At times, fibre composite patch materials are cocured on the aircraft with the adhesive. This may be done to allow them to conform to a complex geometry where the alternative of and secondarily bonding would require a tool to recreate the surface of the repair area. Prior to cocuring, fibre composite patches are often consolidated in an autoclave to minimise porosity and achieve the desired fibre volume.

3.3 Additional Installation Considerations

Although many successful repairs are accomplished on aircraft in the field environment, some care must be taken to control the repair site. Repairs should be conducted in aircraft hangers to provide protection from inclement weather. This environment allows for easier access to the repair area and to required equipment and facilities, such as power, air and vacuum source. Often, the repair must be shielded from air currents, especially when heating with infrared lamps. Temperature and humidity should also be controlled within reason. Temperatures should be in the 10°C to 32°C range with between 30% and 70% RH; a narrower range is desired. In all cases, the repair area must be protected from airborne and other contaminants. Cleanliness must be stressed for all adhesive-bonding operations. Prepared surfaces must not be touched, and handling of adhesives and patch materials should be minimised and conducted by personnel wearing appropriate gloves.

Adhesives, primers and surface preparation chemicals must be stored, handled and disposed of properly in order to achieve a successful repair and ensure worker safety. The material safety data sheets (MSDSs) for all materials must be read and understood. Local fire, safety and environmental regulations must be understood and followed, and appropriate personal protection equipment must be worn for some operations.

The typical repair film adhesives are stored frozen in sealed bags and are ready to use after warming to ambient temperature prior to bag opening to minimise the chances that moisture will condense on the material leading to porous bondlines during cure. Freezer storage allows for a shelf life typically between 6 months and one year. Time out of freezer storage must be monitored. This "out time" for repair adhesives depends on the temperature/humidity environment and is in the order of several days. Bond primers have similar reduced-temperature storage issues. Film adhesives generally contain a carrier cloth that helps control adhesive flow and final bondline thickness.

Many two-part paste adhesives have the advantage that they can be stored at ambient temperatures for extended periods and can cure, given enough time, at typical ambient temperatures. However, they have several limitations. In general, they do not possess the good overall mechanical properties obtained with epoxy-nitrile films. Also, the two parts must be mixed properly prior to use. Once mixed, the "pot life," generally less than 60 minutes, must not be exceeded prior to application. Bondline control is an issue that must be addressed. This and the mixing issue can be minimised by packaging schemes, including glass beads for bondline control, developed by the material supplier.

4. CERTIFICATION

Certification of any repair is an important process and must address a number of design, installation and in-service inspection issues. Unfortunately there is currently no widely accepted standard for the certification of bonded repairs. Repairs must be designed on the basis that the repair efficiency can be predicted and they should be designed conservatively with respect to the various failure modes to include the surrounding structure. In this regard, an approach similar to the one outlined by the USAF Structural Integrity Program identified in MIL-HDBK-1530 [19] can be used as the basis for certification.

For the purposes of this discussion, a distinction will be made between repairs to non-critical structure and those to safety-of-flight-critical structure. Additional certification requirements apply to the latter and these will be considered separately. For the former, the following list is indicative of the type of issues that should be examined in a certification program:

- The ability of the patch to operate under the environmental conditions that can be expected. This largely concerns issues such as the operating temperature and the choice of materials that were considered in the design. Some repairs that could be subjected to repeated impacts may need appropriate materials selection and possibly the design of a suitable protection system. Repairs that may be subjected to high-velocity airflow may also require appropriate sealing.

- The design of the repair with regards the expected operational loads. The design of the repair should show that the repair (patch and/or adhesive) will not fail under design ultimate load and the repair is not susceptible to any expected fatigue stresses. Note that a number of failure modes may be possible and these should be considered. If the repair is to be made to highly loaded structure, checks should be made on whether any growth of the defect may in turn cause damage (disbonding or delamination) within the patching system.

- The influence of the repair on the underlying structure. Checks need to be made to confirm for example, that the stress intensity at the defect, following repair, has been reduced to a level such that any crack growth rate is manageable. Repair installation must be performed using validated and appropriate materials and processes to ensure no additional damage is caused during the application.

- Setting of appropriate in-service inspection intervals. For noncritical repairs, even the complete failure of the repair will not compromise flight safety, however, inspections will be required to confirm the rate (if any) of subsequent growth of the underlying damage. NDI can also be used to confirm that the repair has been applied without any serious defects such as large bondline voids or disbonds.

- Quality control procedures have been met. These include the use of effective training methods for design and installation staff, the use of qualified materials that are in-life, the use of validated design and application procedures and the control of the repair environment to ensure full cure of the adhesive.

Where repairs are considered for safety-of-flight-critical structure, some additional considerations are likely to apply, as mentioned above. Chief amongst these is the issue of long-term bond durability in the service environment, particularly moisture, and the current inability to satisfactorily predict or measure this durability. NDI methods are able to measure the presence of defects within an adhesive bondline, however this is necessary but not sufficient for assuring bondline integrity. No NDI method is currently able to measure the strength of an adhesive bond and therefore (through repeated measurements over time) any gradual degradation in the strength. For this reason, a fail-safe approach is often currently adopted for flight critical structure. This does not allow full credit to be given to the repair for restoring residual strength and reducing the fatigue crack growth rate [20]. The assumption is therefore made that the repair could fail at any time by adhesive bond failure and certification of the repair is performed on this basis. Clearly this is very conservative as the requirement is then that the damaged, unrepaired structure has to be capable of withstanding design limit load in the absence of the repair. While this is a safe approach that minimizes certification requirements, it also prevents many cost effective bonded repairs from being considered.

At this point it is important to clearly state that properly designed and installed adhesively bonded repairs have a most impressive track record for outstanding long-term durability [2,3]. Unfortunately, even a well-designed repair can suffer from adhesive bond durability if is not applied in accordance with approved procedures. For this reason strict adherence with a comprehensive and validated quality plan is necessary to ensure the long-term durability of the bond.

There are at least three approaches that could be used in the future to overcome this problem, and one of these involves the use of a simple accelerated test method to predict the long-term behaviour of the repair. A simple adhesive “witness coupon” can be made alongside the actual repair using the same materials and processes. If it can be shown that an accelerated test on such a coupon accurately reflects the actual long-term behaviour, this would provide confidence in the durability. The other two approaches will require further research to be effective and are considered in the final section of this paper.

5. REPAIR EXAMPLES

5.1 F-16 Lower Wing Skin “Vent Hole” Repair [21]

Fatigue cracks, running forward and aft, initiated at fasteners around the vent holes in lower left wing skins in older F-16 aircraft, Figure 3. These holes vent fuel from the fuselage via a tube that is attached to the wing skin by two concentric rows of fasteners through a flange on the tube. Some cracks extended to the second fastener row. Cracks extending beyond the vent tube flange would allow a direct path to the wing fuel tank and result in a fuel leak. This was considered the end of the service life for the lower wing skin.

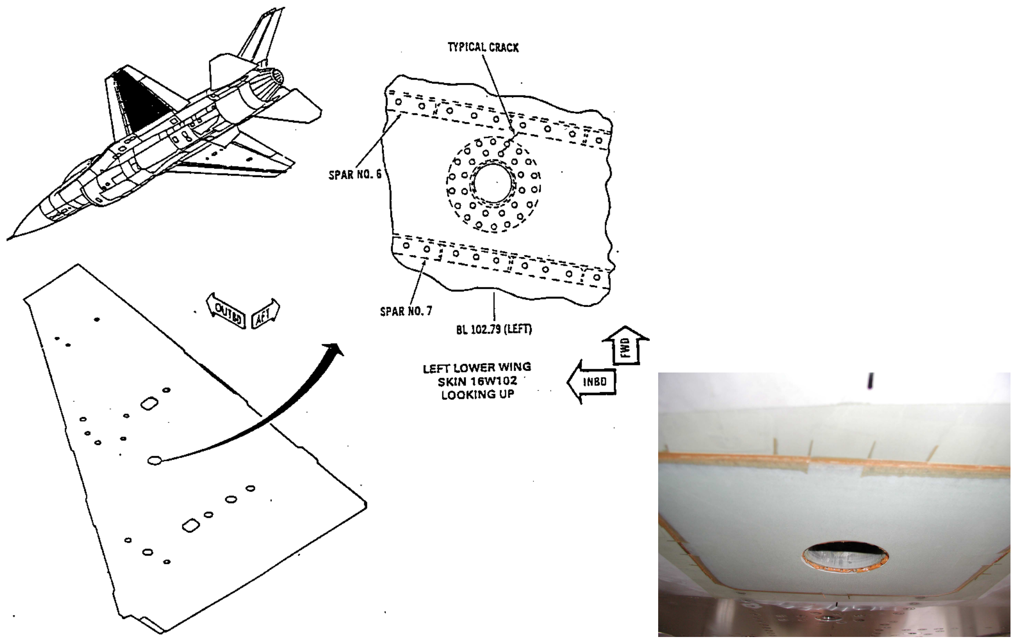


Figure 3: *Typical Crack Location in F-16 and, inset a photograph of an installed repair patch.*

A mechanically fastened aluminium patch repair was designed for the application. Analysis showed this repair would not stop fatigue crack growth but would extend service life from about 3300 hours to about 5700 hours, but short of the 8000-hour goal. Installation of the mechanically fastened repair would have been time consuming since the upper wing skin must be removed to permit access to the required fasteners in the lower wing skin. The need to drill new fastener holes in the skin was also a very significant disadvantage of this approach.

A bonded fibre composite patch repair was considered the best option since it could be performed without drilling additional holes or removing the upper wing skin. F-16 engineering at the USAF Ogden Air Logistics Center at Hill AFB teamed with engineers at AFRL to design and install bonded boron/epoxy patches to repair the vent hole cracks. F-16 engineering designed the repair and AFRL installed the patches.

The starting point for the composite patch design was to match the stiffness of the initial aluminium doubler design. Finite element analyses were conducted using the IDEAS code for both the wing and the patch. Before the patch design was completely optimised, an immediate opportunity arose to test the repair on two vent hole subcomponents. The results of the fatigue tests indicated the repair would meet the life extension goals for the repair, and further patch optimisation was discontinued. The precured patch consists of 14 unidirectional plies of boron/epoxy, to be aligned on the structure normal to the fatigue cracks, and ±45° plies on the top and bottom of the patch. After cure and prior to installation, a hole is established in the patch for the vent. The patch covers existing fasteners in the structure.

The important metal surface preparation step was initially grit-blast/silane (GBS) with application of Cytec Fiberite BR 127 primer which was cured prior to the application of the adhesive. The latest repairs were installed using grit-blast/sol-gel with Cytec Fiberite BR 6747-1 primer cocured with the adhesive, reducing repair time by over 3 hours compared to the GBS process. In order to keep cure temperatures reasonable for on-aircraft application while meeting the service temperature requirements, FM 87-1 and FM 300-2 adhesives from Cytec Fiberite were selected as repair materials with on-aircraft cures in the range of 104°C to 127°C. Extensive thermal surveys on an F-16 wing were used to determine the heating methods and necessary insulation. Electric-resistance heat blankets cure the adhesive while infrared heat lamps are used for silane drying and primer cure. Pressure is applied via vacuum bag. The fuel tanks are air purged during the repair operations and the lower explosion limit (LEL) is maintained well below that required for safety.

The first F-16 vent hole bonded repair installation was completed in early 1993. Only this initial repair utilized FM 87-1. That repair patch material was Textron 5505 boron/epoxy cured at 177°C. Most subsequent repairs were conducted using Textron 5521 cured at 121°C with FM 300-2 adhesive. Twenty aircraft from three countries have been successfully repaired. To date, there are no known problems with the repair installations, and nondestructive inspections of cracks beneath the patches reveal no concerns.

5.2 RAAF F-111 Lower Wing Skin Repair

F-111 aircraft in service with the Royal Australian Air Force (RAAF) have recently been found to suffer from fatigue cracking in the outboard section of the aluminium lower wing skin [22]. The cracking is caused by a stress concentration from a runout in the forward auxiliary spar to create a fuel flow passage. When the first crack was discovered, fracture mechanics calculations indicated that it was beyond critical length at design limit load. A conventional mechanically fastened metallic repair was considered, but this was unattractive from an aerodynamic standpoint (excessive thickness). New fastener holes would not have been acceptable in this highly stressed primary structure, and the crack would have been uninspectable beneath such a repair. A bonded composite repair was the only alternative to scrapping the wing. It must be emphasised that because of its criticality this repair is not a typical example, but rather represents the limit of what bonded repair technology can achieve. Because of certification concerns [20], repairs to critical defects in primary aircraft structure are unlikely to become commonplace in the near future, and in this case an extensive program was required to certify this repair.

Extensive and detailed 2-D and 3-D finite element analysis was conducted so that the stress distribution around the defect could be quantified. This revealed that the wing skin at this location was subject to secondary bending and this was the explanation for the observation that the crack had initiated on the inside surface of the wing skin. The model was validated with strain measurements from a full-scale wing test undertaken at the Aeronautical and Maritime Research Laboratory. In addition, three levels of specimen testing were undertaken:

- Small, inexpensive coupon-sized specimens were used to investigate the effects of impact damage, temperature and moisture and load spectrum truncation effects.
- Panel specimens with a full-scale representation of the local wing geometry were used as structural details in a fatigue and environmental study.
- Large box specimens were used to represent the wing as a quasi full-scale test article in testing static and fatigue strength and an examination of thermal residual stresses.

A repair was designed using boron/epoxy as the repair material as this provides lower levels of thermally induced residual stress compared with graphite/epoxy and enables the ready use of eddy current NDI methods to confirm the length of the crack that was left in the wing. Cytec Fiberite FM 73 epoxy adhesive was selected and cured at the comparatively low temperature of 80°C to minimise the thermally induced residual stresses [23]. This cure cycle had previously been carefully validated for another complex repair [24]. The surface treatment used was the GBS process described above. Advantage was taken of nearby hard-points on the wing to make use of positive pressure during the cure. An inflated bladder was used to apply pressure to the repair and the pressurisation loads were reacted out via a rigid plate to the hard-points. A similar system was used in the earlier application of doublers to the upper surface of F-111 wing pivot fittings [24].

This lower wing skin repair was predicted to, and subsequently proven in service for around 3 years to, reduce substantially the crack growth rate of the defect. This wing was later used as a fatigue test article and survived for over 5000 hours of F-111 spectrum loading with no further detectable crack growth. It was also noted that a test wing with a much smaller crack in this region failed from the crack in less than 1000 hours of simulated flights.

As most wings in the RAAF fleet have not yet developed cracks, these repairs are currently being applied to the fleet as preventative reinforcements to prevent the initiation of cracks in the future. This is an excellent example of how the technology can be used to extend the life of airframes. The repair to the wing is shown in Figure 4.

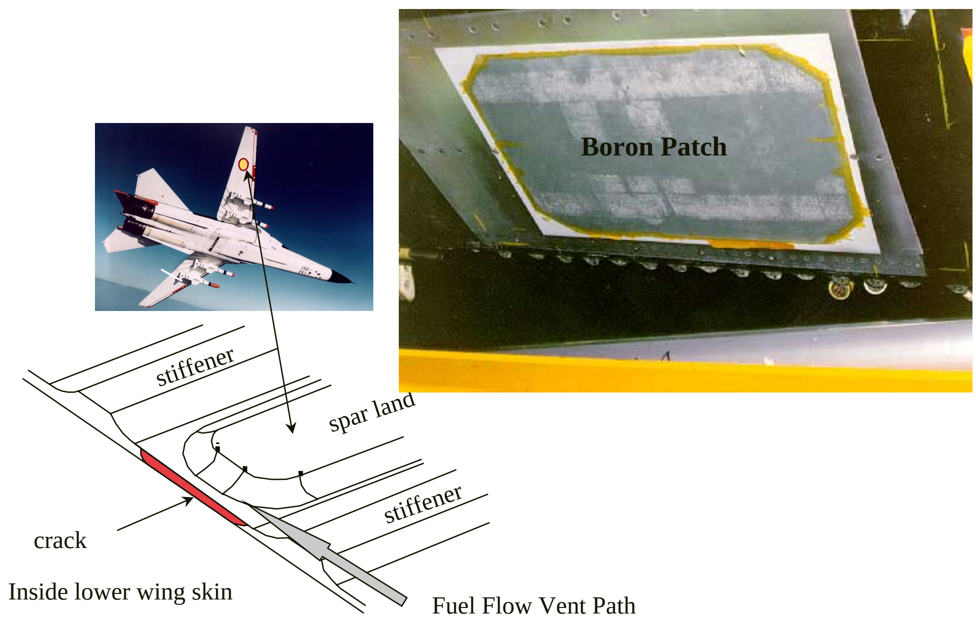


Figure 4: *Region of fatigue cracking in F-111 lower wing skin and, inset a photograph of the boron/epoxy repair.*

6. FUTURE TECHNOLOGY REQUIREMENTS

While this technology is now reasonably mature and is routinely used for aircraft repairs, further research is required to address certification concerns for complex repairs, to further simplify design and application procedures and to extend the capability to address damaged structure that cannot currently be repaired. Recently, three major reviews were undertaken independently to define the general R&D needs of bonded repair technology. These reviews were by the Committee on Aging of U.S. Air Force Aircraft in 1997, The Technical Cooperation Program (TTCP) Aeronautical Vehicles Action Group on Certification of Bonded Structure [25] in 1999 and an Australian Defence Science and Technology Organisation (DSTO) strategic review in 1998. This section aims to summarise some of the key findings from these reviews.

R&D proposals are grouped here into Design, Certification and Application areas. Although there is some overlap, these are useful groupings for this discussion. In the design area, there is scope to simplify routine design procedures through the use of validated software design packages for personal computers; the USAF has work in progress on a package called CalcuRep, as mentioned earlier. Expert systems could also be of use in the assessment of damaged structure and design analysis. There is scope to optimise the stress distributions within repairs using routines within Finite Element Packages. Work in Australia [26] has shown that it is theoretically possible to reduce some of the peak stresses in bonded joints through careful shaping of the adherends.

The scope of the technology can be increased if design procedures can be developed for repairs to more complex structure such as thick, highly stressed components or highly curved components. There is a limit to the amount of load that can be reliably transferred through an adhesive joint. For very thick structure with severe damage, bolted repairs may be the more suitable alternative. There are however, some limitations that may be overcome by further research. In a curved repair, stresses are developed normal to the plane of the repair (peel stresses) and current understanding of these and the associated design allowables for various repair materials is poor. Repair designs for corrosion and, more particularly, for acoustically generated fatigue stresses are other areas where there is scope for improvement. The nature of acoustic fatigue and the need for dampening of the structure means that an entirely different repair philosophy is usually required. Both Australia and the U.S. have active programs in this area [27,28]. Finally, the ability to calculate more readily the magnitude of any thermally generated residual stresses would be advantageous for some repairs. The problem is that it is difficult to estimate accurately the degree of thermal expansion of the repaired area due to the constraint that is provided by the surrounding cooler structure.

The certification issues requiring further research are dominated by the need to provide confidence in the long-term durability of the adhesive bond. While it is well known that well-designed and produced adhesive joints do have excellent durability, this is only the case if they are prepared using well-documented and validated procedures. There is a need for a simple accelerated test method that can be used to indicate the likely long-term performance of the joint. Such a method could either be used as a witness specimen during a repair (prepared at the same time and with the same materials/processes) or as a laboratory method to indicate the likely effect of changes to procedures. Any such test method will, of course, need to be validated against actual long-term performance and in this regard the wedge test is particularly attractive [20].

Models currently exist which can be used to predict the rate of continued damage growth after repair, and this will be required in some cases for certification of repairs to primary structure. Validation of these models is required and further development of them to ensure that they can accurately predict the effect of various parameters such as the nature of the loading (spectrum and R ratio effects), the influence of residual stresses and environmental effects.

Certification issues also arise when repairs are required to materials other than the standard aluminium alloys, for example titanium, nickel or stainless steel. Considerable work is required to ensure that acceptable levels of bond durability can be achieved and the availability of the validated accelerated test method mentioned above would be of considerable help. Related issues include bonding over fasteners where it may be necessary to prove that the repair will not be compromised by fastener movement or durability problems caused by the dissimilar materials and surface preparation methods. The development of either a Smart Patch or a post-bond NDI method would help certification concerns [20]. Unfortunately, the scientific problems with the NDI method are considerable, and it is unlikely that a one will be available in the near future that can measure the level of bond strength as described in the previous section. Of more immediate hope is the development of a Smart Patch that is able to sense its state of health by virtue of embedded microelectronic sensors. Such a patch has been developed and is currently being flight tested on a RAAF F/A-18 [29]. It is intended that these patches will be self-powered using piezoelectric elements to “harvest” power from the parent structure. Future developments will also include active patches. Using piezoelectric sensors and actuators, these patches will, for example, be able to provide active damping as well as reinforcement to counter acoustic fatigue.

Finally, in the area of repair application, research is required to develop new methods of preparing metallic surfaces for adhesive bonding. While current methods are extremely effective, improvements would lead to further reductions in repair time (and hence cost) as well enabling repairs to be applied with reduced levels of quality assurance. This would assist for field-level repairs or perhaps battle damage type repairs. In the area of materials, rapid screening methods would be helpful to quickly assess the likely potential of the material for use in repairs. Currently, large and expensive test programs are required to generate B-basis allowables for new materials. Methods that provide the required design data at reasonable cost are required. In the area of NDI, there is scope for the development in the short term of methods that can indicate if surface preparation methods have been correctly applied. While not in itself a guarantee of long-term bond durability, such NDI methods would be helpful in a risk management context.

References

1. A.A. Baker and R. Jones. Bonded Repair of Aircraft Structures, Martinus Nijhoff, Dordrecht, 1988.

2. A.A. Baker, Bonded Composite Repair of Metallic Aircraft Components – Overview of Australian Activities, Proceedings of the 79th Meeting of the AGARD Structures and Materials Panel on “Composite Repair of Military Aircraft Structures”, AGARD CP 550, Jan 1995, pp 1-1 to 1-14.

3. W. Schweinberg, et al, “Bonded Composite Doubler Repair of Severely Corroded C-130 Primary Wing Structure,” Proceedings of The Fourth Joint DoD/FAA/NASA Conference on Aging Aircraft, St. Louis MO, May 2000.

4. A.A. Baker, Crack Patching: "Experimental Studies, Practical Applications," Chapter 6 in Bonded Repair of Aircraft Structures, Editors A.A Baker and R Jones, Martinus Nijhoff, 107-173 1988.

5. C.H. Wang, and L.R.F. Rose, (1999) "A Crack Bridging Model for Bonded Plates Subjected to Tension and Bending," *International Journal of Solids and Structures*, Vol.**36**, 1985-2014.

6. L.R.F. Rose, "Theoretical Analysis of Crack Patching, " Chapter 6 in Bonded Repair of Aircraft Structures, Editors A.A Baker and R Jones, Martinus Nijhoff, pp. 107-173 1988.

7. A.A. Baker, "Fatigue Studies Related to Certification of Composite Crack Patching for Primary Metallic Structure," 1996, Proceedings of the FAA/NASA Symposium on Continued Airworthiness of Aircraft Structures, Atlanta GA, pp. 313-330.

8. P. Poole, A. Young and A. S. Ball, In: Composite Repair of Military Aircraft, AGARD-CP-550, 1994, 3.1-3.12.

9. R.Fredell, A. Vlot and G. Roebroeks, "Fiber Metal Laminates: New Frontiers in Damage Tolerance," Proceedings of the 15th International European Conference of the Society for the Advancement of Material and Process Engineering, Toulouse, France, June 1994, pp. 319-328.

10. ASTM D 3762, "Standard Test Method for Adhesive-Bonded Surface Durability of Aluminum (Wedge Test)." Annual Book of ASTM Standards, Volume 15.06: Adhesives, pp. 254-257, 1997.

11. R.E. Horton, J.E. McCarty, et al, "Adhesive Bonded Aerospace Structures Standardized Repair Handbook," Final Report for USAF Contract F33615-73-C-5171, AFML-TR-77-206, p. 5-1, 1977.

12. H.M. Clearfield, D.K. McNamara, and G.D. Davis, Engineered Materials Handbook, Vol. 3 Adhesives and Sealants, H.F. Brinson (technical chairman), ASM International, pp. 259, 261, (1990).

13. A.A. Baker and R.J. Chester, "Minimum Surface Treatments For Adhesively Bonded Repairs," International Journal of Adhesives and Adhesion 12, (1992), pp. 73-78.

14. R.J. Kuhbander and J.J. Mazza, "Understanding the Australian Silane Surface Preparation," Proceedings of the 38th International SAMPE Symposium and Exhibition, Anaheim CA, May 1993, pp. 1225-1234.

15. D.B. McCray and J.J. Mazza, "Optimization of Sol-Gel Surface Preparations for Repair Bonding of Aluminum Alloys," Proceedings of the 45th International SAMPE Symposium and Exhibition, Long Beach CA, May 2000, pp. 53-54.

16. J. Mazza, et al, "Faster Durable Bonded Repairs Using Sol-Gel Surface Treatments," Proceedings of The Fourth Joint DoD/FAA/NASA Conference on Aging Aircraft, St. Louis MO, May 2000.

17. D.B. McCray, et al, "An Ambient-Temperature Adhesive Bonded Repair Process for Aluminum Alloys," Proceedings of the 46th International SAMPE Symposium and Exhibition, Long Beach CA, May 2001, pp. 1135-1147.

18. L.J. Hart-Smith, R.W. Ochsner and R.L. Radecky, Engineered Materials Handbook, Vol. 3 Adhesives and Sealants, H.F. Brinson (technical chairman), ASM International, 841, (1990).

19. U.S. Air Force (ASC/ENFS), MIL-HDBK-1530, "Aircraft Structural Integrity Program, General Guidelines for," 4 Nov 1996.

20. A.A. Baker, "Bonded Composite Repair of Primary Aircraft Structure," Composite Structures **47** (1999), pp. 431-443.

21. J.J. Mazza, M.S. Forte and B.D. Cramer, "F-16 Fuel Vent Hole Bonded Repair Update," Proceedings of the Air Force 4th Aging Aircraft Conference, United States Air Force Academy Colorado, 9-11 July 1996, pp. 665-693.

22. A.A. Baker, L.R.F. Rose and K.F. Walker, "Repair Substantiation for a Bonded Composite Repair to F111 Lower Wing Skin," Applied Composite Materials 6, (1999) pp. 251-267.

23. M.J. Davis, K.J. Kearns and M.O. Wilkin, “Bonded Repair Cracking to Primary Structure: A Case Study,” Proceedings of the 6th Australian Aeronautical Conference, Melbourne, 20-23 March 1995, pp. 323-329.

24. A.A. Baker, R.J. Chester, M.J. Davis, J.A. Retchford and J.D. Roberts, “The Development of a Boron/Epoxy Doubler System For The F111 Wing Pivot Fitting - Materials Engineering Aspects,” *Composites* **24** (1993), pp. 511 - 521.

25. “Certification of Bonded Structures,” The Technical Cooperation Program (TTCP) Action Group 13 Report, J.W. Lincoln (chairman), February 2001.

26. R. Kaye and M. Heller, “Shape Optimisation of Bonded Lap Joints and Crack Repairs,” (accepted for publication in International Journal of Adhesion and Adhesives), 2001.

27. R.J. Callinan, W.K. Chiu and S.C. Galea, “Optimisation of a Composite Bonded Repair to Cracked Panels Subjected to Acoustic Excitation,” Proceedings of the 21st Congress of the Aeronautical Sciences (ICAS 98) Melbourne, Australia, 1-18 September 1998. (Paper A98-31631 ICAS Paper 98-5,3,2).

28. L. Rogers, et al, “Durability Patch: Repair and Life Extension of High Cycle Fatigue Damage on Secondary Structure of Aging Aircraft,” Proceedings of the First Joint DoD/FAA/NASA Conference on Aging Aircraft, Ogden UT, pp. 595-624, 1997.

29. A.A. Baker, S.C. Galea and I.G. Powlesland, “A Smart Patch Approach for Bonded Composite Repairs to Primary Airframe Structures,” Proceedings of the Second Joint FAA/DoD/NASA Conference on Aging Aircraft, Williamsburg VA, 1998.

Sustaining an Aging Aircraft Fleet with Practical Life Enhancement Methods

Len Reid
Vice President of Engineering, Fatigue Technology Inc.
401 Andover Park East; Seattle, Washington 98188, USA
E-Mail: lreid@fatiguetech.com
Telephone: (206) 246-2010, Fax: (206) 244-9886

Abstract
Extension of aircraft beyond their original design service life or operation in expanded or different roles pose challenges in continuing to operate these aircraft economically or safely. Management of the fleet generally entails increased structural inspection and maintenance, repair of inevitable fatigue damage or possible structural replacement. In a large number of cases structural elements become life limiting or require major rework due to fatigue cracks and damage originating at holes. Hole cold expansion to induce residual compressive stresses around the hole, and thereby minimize the stress concentration effect of the hole, is a proven method for retarding cracks at holes. Reworking existing structure using this technology can significantly extend the fatigue and damage tolerance life and ultimately reduce or eliminate the need for costly ongoing structural inspection while ensuring continued airworthiness without compromising performance or weapons systems capability. This paper discusses the hole cold expansion method as well as derivative technology used to repair fatigue cracked or corroded holes using high interference fit cold expanded bushings. Examples of where the technology is used to extend service lives and restore structural integrity on aging military and commercial aircraft will be presented.

Introduction
Management of an aging aircraft fleet encompasses, among other things, increased structural maintenance and repair of inevitable fatigue damage and normal wear of structural attachments. Refurbishment of the structure will typically entail repair to restore static strength with acceptable durability and damage tolerance requirements for primary structure. The challenge facing the industry and operators is how to economically achieve these objectives without compromising long-term structural integrity, imposing restrictive ongoing inspections and incorporating expensive major structural replacement and/or repair.

In a large number of cases structural elements become life limiting or require major rework due to fatigue cracks and damage originating at holes. Economical life extension has been attained without replacement of major structural elements or reduced risk to continued safe operation by use of technology that can pre-stress fatigue critical holes and damaged holes in structure and attachment lugs and fittings. Hole cold expansion is a proven method for retarding growth of cracks originating at holes. The induced residual compressive stresses minimize the stress concentration effect of the hole. In addition, these beneficial residual stresses are useful in meeting ongoing damage tolerance requirements by reducing the stress-intensity factor for residual cracks thereby permitting use of smaller initial flaw sizes in crack growth analysis. The resultant increase in fatigue and damage tolerance life can ultimately reduce or eliminate the need for costly ongoing structural inspection while ensuring continued airworthiness without compromising performance or weapons systems capability.

Repair of fatigue cracked, damaged or corroded holes can be accomplished using derivative technologies such as high interference fit cold expanded bushings and a new cold expanded rivetless nutplate system. These methods resize the damaged hole back to nominal and synergistically cold expands the surrounding material, resulting in a repair with generally a better fatigue and damage tolerance life than the "as-built" structure. Several military and commercial applications have successfully used these methods as terminating repair solutions as well as avoiding structural or component replacement. This is very beneficial when trying to manage or extend the life of aircraft as an interim measure while awaiting replacement aircraft. These proven life enhancement technologies are routinely

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

incorporated into both new an aging aircraft structures and systems. Examples of applications and supporting test data are presented.

The Problem with Holes

Holes in metal structures are unavoidable in most conventional aircraft designs and are inevitably the weakest part of the structure. They concentrate stresses, intensifying the magnitude of the applied load by factors of three or more. The additions of cyclic tensile loading frequently produce crack initiation, growth and eventually fatigue failure. Small defects in the hole from manufacturing defects, tooling marks, fastener installation and repairs are stress risers that can accelerate crack initiation. Over time and enough accumulated cycles, these flaws can lead to single or multiple fatigue cracks that can result in catastrophic structural failure. The advent of damage tolerance requirements attempts to account for these “rouge” flaws in analysis to ensure structural integrity through conservatism in determining ongoing structural inspections or structural lifing.

The fatigue life of holes can be enhanced in several ways. Good hole quality, minimizing flaws; use of interference fit fasteners in the hole to modify the fatigue stress amplitude; and increasing material thickness to reduce the net section stress although it carries a weight penalty. The other way is to introduce beneficial residual stresses around the hole, which like interference fit fasteners, primarily shield the hole from the applied stress to reduce crack growth rate. This method can compliment the other methods, and carries no weight penalty. Of the methods available to induce compressive residual stresses around holes the split sleeve cold expansion method developed by Fatigue Technology Inc (FTI) is the most commonly used by the aerospace and other industries. The technology was intended for new production aircraft structures, however it has been proven effective in aging aircraft by reducing the impact of increased localized stress levels following rework and increasing the fatigue and crack growth resistance of repairs.

Overview of Split Sleeve Cold Expansion Process

Split Sleeve Cold Expansion, or the Cx process, is accomplished by using an oversize tapered mandrel pre-fitted with an internally lubricated stainless steel sleeve. A nosecap assembly restrains the sleeve in the hole while the mandrel is pulled through the hole, as shown in Figure 1. The sleeve protects the hole from damage and allows the tapered mandrel to radially expand and yield the area surrounding the hole in a repeatable, controlled manner. The sleeve also allows the process to be a one-sided operation. After cold expansion of the hole is completed, the sleeve is discarded. Mandrel insertion and sleeve removal requires no access to the backside of a component, which is an important consideration for repairing existing structure and process automation. The process can also be applied through a stack-up of multiple materials or structural elements.

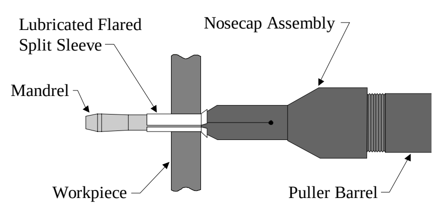


Figure 1
Split Sleeve Cold Expansion Process

Hole cold expansion improves the fatigue life of holes in metallic structure by generating a large, controllable zone of permanent residual compressive stress around the hole. A typical photoelastic pattern for a cold expanded hole (Figure 2) shows the residual stress field created by this process. These stresses are formed as a result of plastic yielding of the material caused by the mechanical expansion of the hole, and the subsequent elastic “springback” of the material lying beyond the plastically deformed hole.

Figure 2
Residual Strain Pattern Around Cold Expanded Hole As Viewed

Typical residual radial and circumferential stresses generated by cold expansion are illustrated in Figure 3. The annular zone of compressive stress extends radially up to one diameter from the edge of the hole for typical fastener diameters and has a peak magnitude roughly equal to the compressive yield strength of the material. A balancing tensile stress zone with a peak stress of 10 to 15 percent of the material tensile yield strength lies just beyond the compressive stress region. Since it is unlikely for the applied cyclic tensile stresses to overcome the residual compressive stress, the hole is effectively shielded from the tensile stresses that propagate flaws into fatigue cracks. Fatigue life improvement from the Cx process usually ranges from 3-to-1 to 10-to-1 in typical aircraft structures.

Optimal fatigue performance is achieved when the hole is expanded by at least 3 percent for aluminum, and at least 4.5 percent for titanium and high strength steels, in typical hole diameters (up to 25 mm) and plate thickness. Cold expanded hole fatigue lives generally range from 3 to 10 times the fatigue life of similarly non-cold expanded holes as shown in Figure 4 for aluminum alloy [1] and Figure 5 for titanium [1].

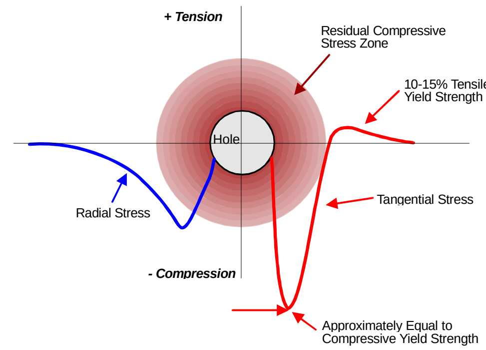


Figure 3
Typical Residual Radial and Circumferential Stress Distributions Around a Cold Expanded Hole

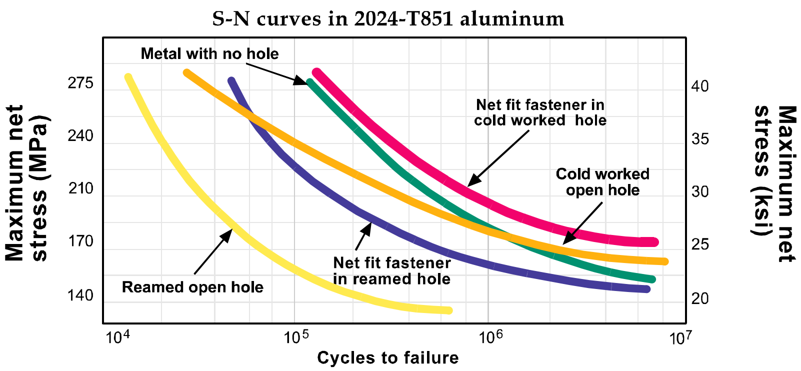


Figure 4
Fatigue Life Improvement - 2024-T851 Aluminum Alloy

Damage Tolerance

Crack growth characteristics serve as the basis for fleet structural maintenance planning for many commercial and military organizations. Durability and damage tolerance analysis (DADTA) revolutionized the USAF structural design and repair philosophy in the 1970s. Theoretical analysis of crack growth life from an assumed 0.050-inch (1.25-mm) initial flaw size and material fracture toughness characteristics, has proven a reliable and somewhat conservative technique to determine structural life and inspection cycles.

The primary effect of the cold expansion residual stresses is to reduce the crack growth rates by reducing the stress intensity factor range (ΔK) and the stress ratio (R, min. stress/max. stress). This effect is shown in Figure 6 [2]. The stress intensity factor is a measure of the stresses acting on the crack. Additionally, the presence of residual stresses may change the critical crack length for unstable fracture, because it reduces the static stress intensity factor. The reduction in crack growth rate and the increased critical crack length significantly improves the damage tolerance of the structure.

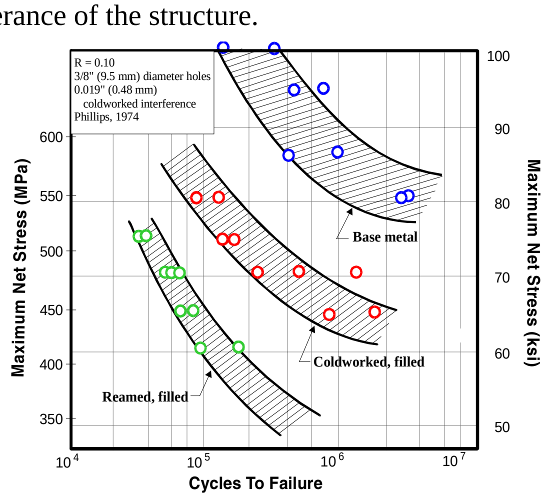


Figure 5
Fatigue Life Improvement –Titanium (6Al-4V)

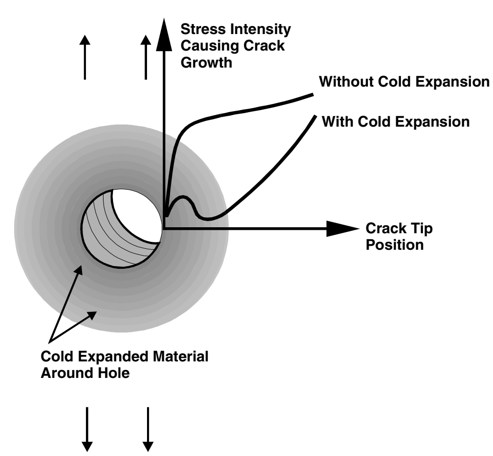


Figure 6
Reduction in Stress Intensity Factor Range Under Residual Compressive Stress

When reworking aged structure the probability of missing a crack in a hole during inspection and rework is high. The Cx process can be effective in preventing these cracks from growing. Figure 7 shows that cracks about 1mm in length growing from a 6mm (1/4") diameter hole in 7075-T6 aluminum alloy, under 248 MPa (35 ksi) net stress, are totally arrested when subjected to the same applied cyclic loads [3]. The residual compressive stress zone acts like a strong clamp on the material around the crack minimizing crack opening displacement, thereby preventing growth. The process is just as effective in high-strength steel and titanium [4]. Fatigue life improvement of 3:1 is typical.

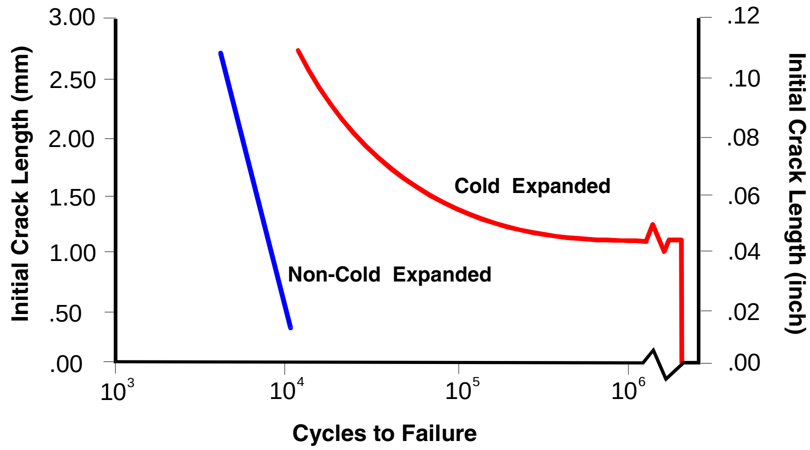


Figure 7
Effect of Cold Expansion on Stopping Crack Growth (7075-T6 Aluminum)

Results from crack growth tests incorporating cold expansion of holes prompted a revision of the DADTA initial flaw size philosophy. For many aircraft in the USAF, advantage is taken of the crack retardation benefits of cold expansion by reducing the initial flaw size to as small as 0.005 inch (0.125 mm) if cold expansion is incorporated. The same philosophy could be applied to repairs on other aging military aircraft and commercial aircraft for fatigue-related service bulletin repairs.

Rework of Previously Cold Expanded Holes

Although hole cold expansion has traditionally been used in new production and repair of fatigue damaged structure, the general perception is that once utilized, the beneficial residual compressive stresses induced could not be improved upon, thereby limiting future rework options. Extensive tests and investigations [5] revealed that additional split sleeve cold expansion of these holes can further enhance the damage tolerance and fatigue life, particularly after a period of cyclic strain aging from in-service structural loads.

The resultant increase in fatigue life after rework is comparable to, or exceeds, the life of initially cold expanded holes. The Cx process performs well even when cracks (up to l.25 mm for the conditions tested) are present. The evaluations and subsequent in-service evaluations showed conclusively that holes may be cold expanded multiple times with commensurate fatigue life improvement making it a very versatile process for rework of aging structures.

Life Cycle Cost Benefits

Split sleeve cold expansion is a measurable and quantifiable way of improving the fatigue life and quality of the hole without relying on the integrity of the fastener fit. The overall result is enhanced fatigue life, which can allow flying an aircraft that might otherwise have reached its fatigue limit.

Life cycle cost benefits derived from use of this process include: (1) Added safety and operational assurance through improved structural integrity, both in production and repair; (2) Reduced maintenance costs by virtual elimination of fatigue problems associated with fastened joints; and (3) Reduced inspection costs, by extending inspection intervals resulting from the enhanced fatigue/durability and damage tolerance of the structure.

The additional cost of incorporating cold expansion of holes, in either production or repair applications, is practically insignificant in comparison with the total cost of preparing the joint and installing the fastener. The overall airworthiness, structural integrity and operational safety benefits far outweigh any additional process costs.

Applications of Hole Cold Expansion on Aging Aircraft

The split sleeve cold expansion process has been used to repair and restore structural integrity and fatigue life to a large number of aged aircraft that were operating at or beyond their original design life. In many cases it has been used to enable aircraft to reach the original design life after premature or unpredicted structural fatigue problems were encountered due to design deficiencies, increased operational load spectrum, or changing roles of the aircraft.

Aircraft such as the F-4, T-38, F-16 (which includes cold expansion of a number of non-round hole configurations), F-111, B-52, KC-135, B-2, F-14, F-15, F-18, C-141, C-5, C-130, Mirage III, Tornado, EA-6B, JSTARS (Boeing 707)—virtually all Western World military and commercial aircraft—incorporate the cold expansion process to some extent or another. It has been effective in repair of existing structure without major structural replacement that otherwise would have been cost prohibitive or uneconomical to repair. The principles of hole cold expansion have been applied to derive other processes used to repair badly damaged holes, including expansion of repair bushings and will be discussed next.

Repairing Damaged Holes

Damaged or discrepant holes are common in aircraft production and structural repair or modification programs. Damage induced by drilling or mis-alignment requires oversize of fasteners or in extreme cases bushing or plugging of the hole to reposition it or remove all incipient damage. In repair of aging aircraft, removal of fatigue or corrosion damage in holes necessitates similar oversizing and frequently needs application of splice repairs or component replacement.

The use of cold expanded bushings in repairs result in a convenient high integrity repair with generally a better fatigue and damage tolerance life than the original structure. Several military and commercial aircraft applications have successfully used these methods as terminating repair solutions.

Derivative Cold Expansion Processes

The principles of hole cold expansion are used to provide the interference fit required for bushing installation in structural components. Bushing interference is defined as the degree to which the bushing outside diameter is greater than the inside diameter of the hole. Traditional techniques using cryogenic fluids or dry ice to shrink the bushings are limited to diametrical interference of 0.05 to 0.075 mm (0.002 to .003 inch). Fatigue Technology Inc. has derived a process and tooling under the trade name of ForceMate® (FmCx™), which has the ability to achieve interference of 0.1 to 0.225 mm (.004 to .009 inch) in a nominal 25-mm (1.0-inch) diameter bushing. A schematic of the ForceMate process is shown in Figure 8. Besides being a convenient way to install high interference fit bushings, cold expanded bushings are an effective way to repair damaged holes in structures.

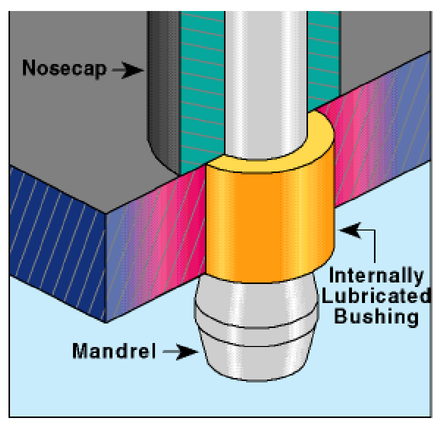


Figure 8
Typical ForceMate Process

There are three methods of installing cold expanded bushings that have been developed by FTI. These methods are:

1. ForceMate® which installs initially clearance fit bushings with high interference fit resulting in significant fatigue life improvement,
2. BushLoc® A convenient repair/resizing bushing with high interference fit using a variation of the split sleeve cold expansion process, and
3. ForceTec® A rivetless nut plate, which can be used for repairing fatigue damage associated with conventional riveted nut plate installations.

Each of these methods has its own merits and benefits.

Besides the higher retention forces and ease of installation, cold expanded bushings are superior to shrink and press fit bushings in many ways. The primary advantage is the fatigue and crack growth life improvement resulting from the unique state of residual stress around the hole. Increasing the fatigue life reduces the need for frequent inspections and increases the overall integrity of the repair. The typical life improvement, ranging from 3:1 to greater than 20:1, allows the cold expanded bushing to be used as an integral part of a terminating repair.

The action of cold expanding the bushing generally imparts compressive residual stresses around the hole, depending on the bushing/parent material combination, that reduce the mean stress at the hole thereby improving fatigue life. The high interference fit of the bushing acts to reduce the stress amplitude at the hole and works synergistically to significantly improve fatigue and crack growth lives of both new and repair bushing installations as shown in Figure 9.

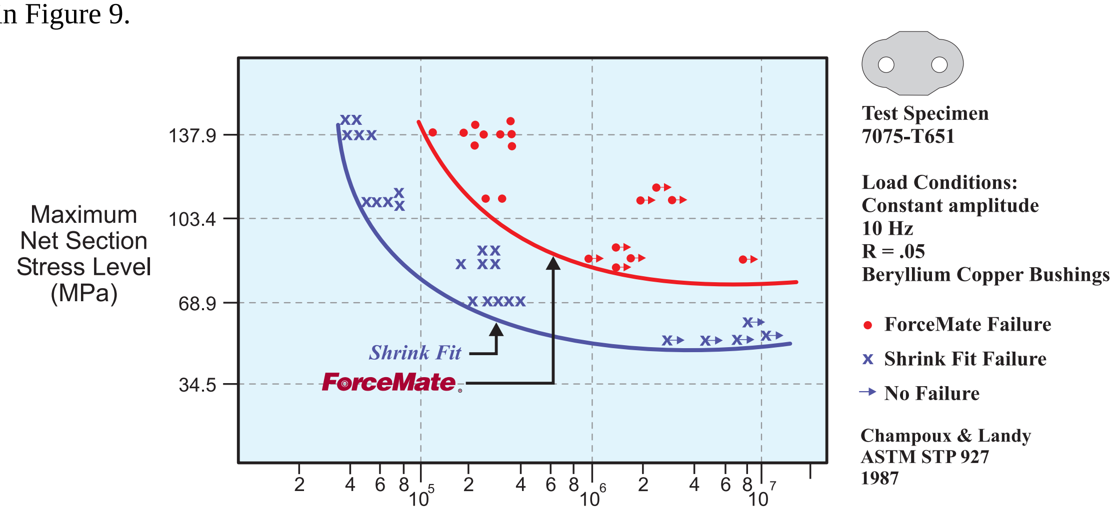


Figure 9
Fatigue Life Comparison of Shrink Fit and ForceMate Bushing Installations

Each of the cold expansion bushing process comes in a wide variety of diameters and lengths to meet just about any application. ForceMate bushings can be made to meet the most exacting specifications including dimension, tolerance, and material. The BushLoc process offers the greatest flexibility for cold expanded repair bushing installation. Just about any combination of bushing length, inside diameter and outside diameter can be installed with this process. The manufacturing/rework tolerances of holes and repair bushing are generally wider than standard bushings.

Figure 10 shows a multi-layered stack-up with individual segmented bushings. All three bushings are installed simultaneously. Different outside diameter bushings can be simultaneously installed allowing minimum material removal to correct hole discrepancies or to remove corrosion damage or fatigue cracks.

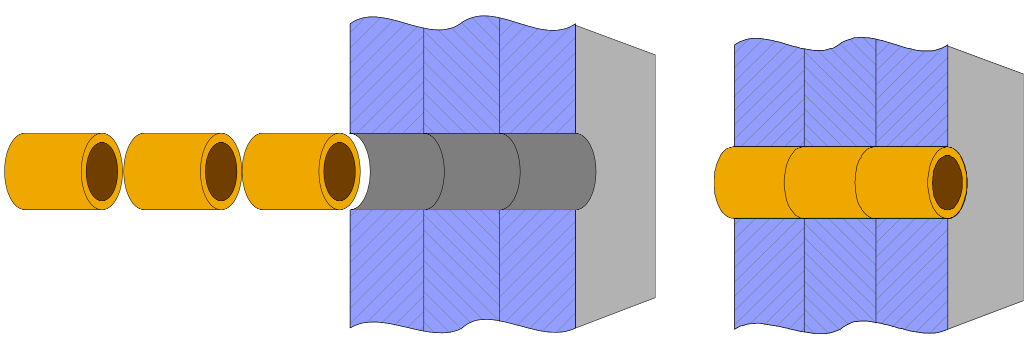


Figure 10
Schematic of Multiple Bushing Installation

BushLoc™ Bushing Installation Process

This process was primarily designed to repair or re-size damaged fastener holes and has been successfully used in these applications. The installation of a bushing using BushLoc is accomplished using specially designed tooling similar to that used to spilt sleeve cold expanded a hole.

The method is currently used on a large number of repairs to existing aircraft and has eliminated the need to replace major structural components such as the inner to outer wing (rainbow) fitting on the C-130 transport aircraft. Thousands of holes are repaired on the aging Boeing 707 aircraft being refitted for the JSTARS program. The superior benefits of cold expanded bushings in fatigue, damage tolerance, improved corrosion resistance, ease of installation and the flexibility of adapting to almost any repair configuration, gives a technically and economically advanced alternative to traditional repair methods. The cost of using cold expanded bushings is generally similar to the cost of conventional repairs with the added benefit of providing terminating repair action or at least reduced inspections, and reduced follow-on maintenance costs.

BushLoc Applications

In a spanwise splice repair evaluation for military transport aircraft, FTI completed a test program comparing the spectrum fatigue life of both FTI's BushLoc hole repair and resizing process and the ForceTec Rivetless nut plate system in a short edge distance application. The test was conducted on 7075-T6511 pre-cracked, short edge margin aluminum specimens taken from actual wing structure. The original 1/4 inch diameter [edge margin (e/D) =2.0] holes were pre-notched and the crack grown to a part-through surface length up to 0.070 inch. A portion if not all of the pre-crack was removed when preparing the bushing repair starting hole. Residual crack lengths for the repaired holes ranged from not visible (zero) to 0.025 inch. The holes were installed with BushLoc bushings and ForceTec retainers to determine the crack growth life of the repaired holes. All specimens, baseline and repaired, were tested at the same gross stress. This means that repaired specimens were tested at higher net stresses due to metal removal. The repairs reduced the edge margin from 2.0 to a range of 1.25 to 1.4 depending on the repair bushing diameter. Results of these tests are shown in Figure 11.

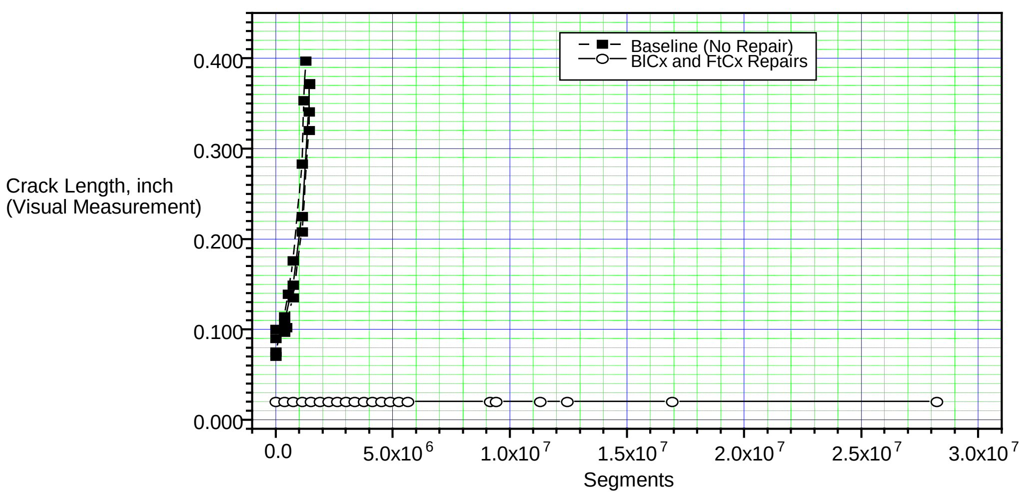


Figure 11
Crack Length Versus Segments

All repaired specimens ran to run-out life of 45,405 flight hours with very little or no additional crack growth when repaired with either BushLoc or ForceTec.

In another commercial aircraft BushLoc repair, extensive testing was performed to simulate the repair of a spar cap to wing skin fastener hole on a commercial aircraft. The repair was to remove a large crack from the spar cap without removing the skin or oversizing the original 3/8-inch fastener hole in the wing skin. Several bushing repairs were examined. In one case, a 1/8-inch wall thickness BushLoc bushing was installed. The load transfer test coupon configuration is shown in Figure 12.

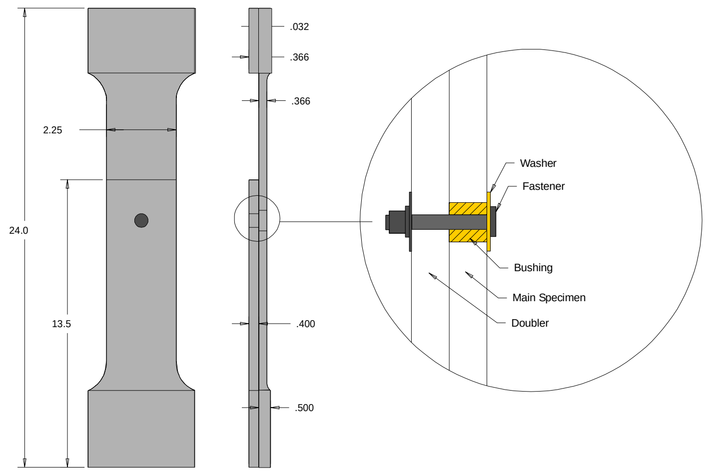


Figure 12
Load Transfer Test Specimen

A typical commercial aircraft wing spectrum load was applied to the specimen and results for various repair scenarios were compared to baseline (fastener only) configured specimens. Test results in Figure 13 show that specimens repaired using either aluminum or steel BushLoc bushings performed better than the baseline configuration and were substantially better than shrink-fit repaired specimens. Results of this BushLoc test were accepted by the FAA as terminating repair actions for this location, which further justified its use in a number of other military and commercial rework applications.

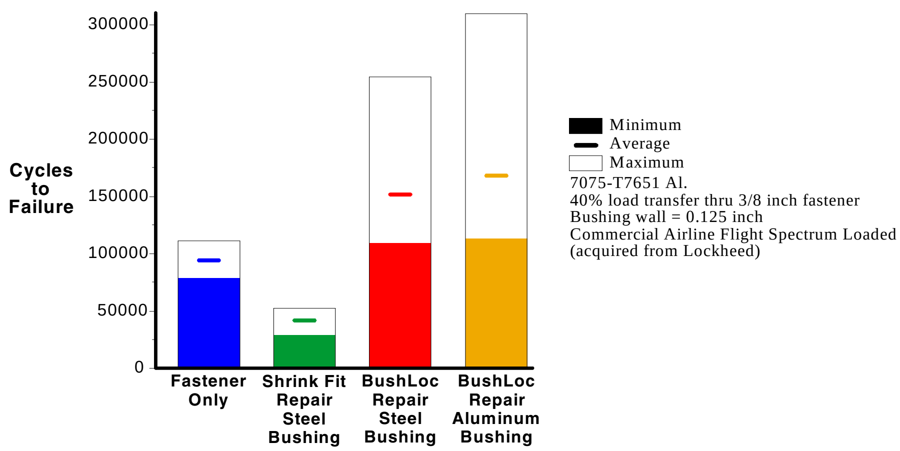


Figure 13
Comparison of Fatigue Life for BushLoc Repaired Specimens

ForceTec Rivetless Nut Plate Process

The last method to be reviewed is the ForceTec rivetless nut plate process shown schematically in Figure 14. While this system is not specifically a bushing installation method it does have a bushing like member (called a retainer) that is installed in the hole and offers the potential for terminating repairs of riveted nut plates. The installed retainer provides superior fatigue performance when compared to riveted nut plates by virtue of the beneficial residual stress imparted to the material surrounding the hole. The resultant high interference fit of the retainer provides resistance to torque and removal forces normally encountered during fastener installation and removal. ForceTec retainers meet and exceed the demanding torque and push-out requirements of MIL-N 25027 and can be used as a substitute for most riveted type nut plates.

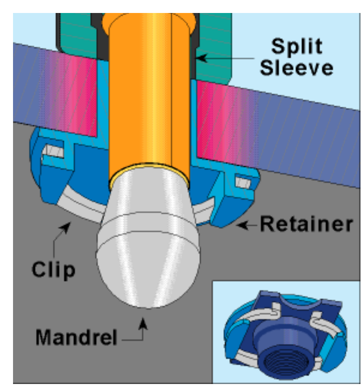


Figure 14
Schematic of ForceTec Rivetless Nut Plate Installation

By way of an example of the use of ForceTec in a rework situation, the upper fuselage skin of the F-16 has been identified as an area requiring fatigue life improvement for the aircraft to meet its service life objective. This area of the F-16 has a number of access panels secured with conventional riveted nut plates as shown in Figure 15. The highly loaded corners of the cutouts are developing fatigue cracks from either the fastener holes or the attaching rivet holes of the nut plate.

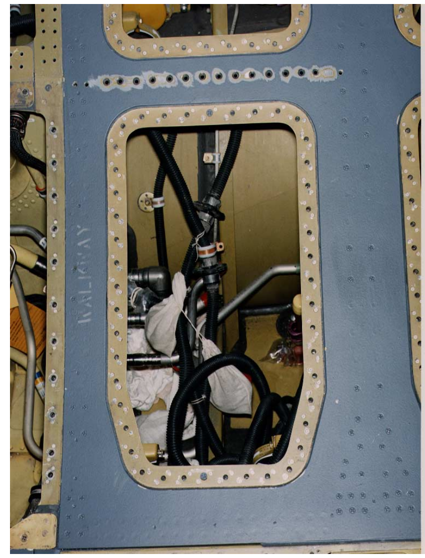


Figure 15
Panel in F-16 Fuselage Skin

The proposed repair included removing the riveted nut plates, cold expansion of the attaching nut plate rivet holes and installation of an expanded ForceTec rivetless nut plate. Following an extensive coupon and component test program where the specimens were tested at the severe F-16 fighter spectrum load, the repaired configuration showed a four times life improvement over the initial configuration. Additional testing of simulated access panels with ForceTec installed originally, the life of the specimen exceeded 12 times the life of the original riveted configuration [6]. This economical structural life enhancement modification is now specified for this particular fleet of aircraft.

Review of FTI Processes Used to Repair Aging Aircraft Structures (and New Production)

The summary list of applications shown in Attachment A is intended to give examples of hole-related structural problems, and the FTI process/product used to overcome or resolve these problems. The list is by no means complete, as the technology is used in an extensive number of actual aircraft applications to enhance the fatigue life of a structure, possibly extend inspection intervals, or provide terminating repair action. In many cases the processes are being used to repair previously "un-repairable" primary and secondary structure and components such as the C-130 Rainbow Fitting and engine mounting struts. New uses for the technology are continually evolving as needs arise. Where indicated in the list, there is ongoing research and development to adapt or apply the product/process to a particular application.

Summary

With the shrinking defense budgets and cost of procuring new weapons systems continuing to escalate, it is necessary to prolong or extend the life of existing aircraft fleets. Structural fatigue along with corrosion are two of the primary life-limiting causes of structural replacement or retirement. Fatigue cracks originating at holes can be effectively eliminated by pre-stressing the material surrounding the hole in compression using the split sleeve cold expansion method. Over many years of successful application extending the fatigue and crack growth lives of numerous aging military and commercial aircraft, the process can be confidently used to overcome hole-related fatigue problems. Furthermore, the use of expanded bushings and rivetless nut plates, derived from the cold expansion methodology can be used to repair damaged or defective holes in existing structures to generate terminating repairs and thereby, avoid costly structural replacements. The cold expansion processes can effectively and economically be used to sustain an aging aircraft fleet by extending the fatigue life and damage tolerance of the structure without compromising structural integrity, airworthiness or the operational role of the weapons system platform.

References

[1] Phillips, J. L., "Sleeve Coldworking Fastener Holes," Vol. I, 1974, Air Force Materials Laboratory report AFML-TR-74-10.

[2] Clark, G., "Modeling Residual Stresses and Fatigue Crack Growth at Cold Expanded Fastener Holes," Fatigue Fracture of Engineering Materials Structures, Vol. 14, No. 5.

[3] Petrak, G. J. and Stewart, R. P., "Retardation of Cracks Emanating from Fastener Holes," Engineering Fracture Mechanics, Vol. 6, Pergamon Press, 1974, pp. 275-282.

[4] Armen, H.; Levy, A.; Eidenoff, H.L., "Elastic-Plastic Behavior of Cold worked Holes," Proc. 24th AIAA/ASME/ASCE/AHS Structures, Structural Dynamics and Materials Conference, May 1983 (AIAA Paper 83-0865-CF), pp. 1-11.
[5] Reid, L., "Split Sleeve Cold Expansion as a Rework Process for Previously Cold Expanded Holes," International Committee on Aeronautical Fatigue (ICAF) Conference, 1993.
[6] Ransom, J., et al; "F-16 Fighting Falcon Upper Fuselage Skin Fatigue Life Enhancement," USAF Aircraft Structural Integrity Program Conference, 1999.

Questions

The following questions with answers were presented at the conclusion of the paper.

Q1 How does quality control affect the protection afforded by cold working? (C. Smith)

A1 The cold working process, while critical in its application, is relatively insensitive to minor manufacturing discrepancies such as hole finish. The process tolerances are also within normal manufacturing tolerances for most handheld tooling operations. Within the system of cold working tooling for a particular hole size, Fatigue Technology Inc. would supply "checking fixtures" that quickly verify correct starting hole tolerance, verify correct hole retainer expansion (enlargement) for the application, and also a checking GO/NO-GO gauge to periodically check that the mandrel major diameter is within tolerance for the process. If these process quality checking/verification fixtures are used, then correct hole processing is assured.

Q2 What percent of manufacturers are using this process during production? (J. Komorowski)

A2 Virtually all manufacturers of commercial transport and military aircraft in the western world use the process selectively for meeting design fatigue life goals, structural weight reduction or damage tolerance objectives. A large number of commuter aircraft and helicopter manufacturers also use the process. The ForceMate bushing installation method is extensively used in helicopter rotor assemblies and also across the board for engine and stores pylons, landing gear and wing attach fittings, and also for repairing damaged holes in these and other structural locations.

Q3 How extensive is the use of the ForceTec rivetless nut plate system and for what applications? (F. Grimsley)

A3 The ForceTec system is being used both in production and repair/replacement of existing fatigue-prone/damaged riveted nut plate installations as briefly mention in the paper. This nut plate is the first to be approved for primary structural attachments and major sub-assembly connections. The repair on the F-16 is being effectively carried out at depot level and a number of other field repair installations have also been done, on both commercial and military aircraft.

It is also worthy of note that on the F-16 application the installations are being done without the need to provide secondary sealing after installation. The sealing qualities of ForceTec were thoroughly tested to cover installation in fuel tank locations on the F-22. This feature saves rework time and cost, and omits a particularly messy post-sealing operation.

Attachment A
Summary of FTI Processes Used to Repair Aging Aircraft Structures (and New Production)

Structural Problem or Application	FTI Process	Mature	Requires R&D
Hole-related fatigue —Open or filled hole —Non-circular holes —Short edge margin (e/D) condition —Multiple-material stack-up (includes different materials) —Attachment lugs —Close proximity holes —Damage tolerance improvement —Increasing inspection interval	Split Sleeve Cold Expansion™ (SsCx™)	Yes	No
Temporary stop drill repairs	StopCrack™	Yes	No
Mis-drilled holes Installation of interference fit bushings	BushLoc® BushLoc	Yes Yes	No No
Bushing of holes —Repair of holes —Fatigue life enhancement of lugs —Improve damage tolerance of lugs —Installation of high interference fit bushings —Rework/resizing of holes in lugs —Elimination of bushing migration —Improved corrosion resistance of bushed holes —Elimination of vibration-induced bushing problems —Replacement of bushings in wing attach lugs	ForceMate®	Yes	No
Blind nut plate installations —Increased fatigue life of nut plates —Blind attachment of primary structures —Repair of fatigue-prone attaching structure with riveted nut plate attachment —Installation of blind threaded insert	ForceTec® DuraLoc	Yes Yes	Ongoing R&D Ongoing R&D
Mounting of components/systems —Attaching hydraulic hoses/electrical wiring —Mounting electrical connectors (i.e., cannon plug) through structures	 ForceTec/DuraLoc with stud/standoff Adaptation of ForceMate	 Yes Yes	 Ongoing R&D No
Repair of components/structure —Repairing corroded/cracked/oversize mounting holes (pylons/struts, etc.) —Repairing cracked access panel mounting —Repairing wing fold transmission attaching hole (corrosion/fatigue) —Installing blind threaded insert —Repairing helicopter blade attachment —Fatigue of hydraulic ports in actuators —Repairing aircraft wheels —Repairing wheel fuse plug holes	 ForceMate ForceTec ForceMate DuraLoc ForceMate SsCx/ForceMate Split Sleeve Cold Expansion Split Sleeve Cold Expansion	 Yes Yes Yes Yes Yes Yes Yes	 Ongoing R&D for new applications No Ongoing R&D No No

Structural Problem and Application	FTI Process	Mature	Requires R&D
Composite panel repairs —Resizing damaged holes —Increased durability of load transfer joint —Increased resistance to lightning strike —Blind nut plate installation —Hole reinforcement —Realignment of mis-drilled holes	 GromEx® and ForceMate GromEx and ForceMate GromEx and ForceMate ForceTec/DuraLoc ForceTec/GromEx ForceMate	Yes	Ongoing R&D
Engine Components —Fatigue life improvement of holes —Repair/resizing blade attach holes	 Split Sleeve Cold Expansion ForceMate	 Yes Yes	 No Ongoing R&D

Recognition and Correction of Sonic Fatigue Damage in Fighter Aircraft

Mr. Salvatore L. Liguore, Dr. Theodor H. Beier
The Boeing Company, PO Box 516
St. Louis, MO 63166-0516, USA

Abstract

High performance aircraft that fly aggressive low altitude or high-angle of attack maneuvers will experience high acoustic loading, especially in the presence of external stores. These high acoustic loads can lead to rapid sonic fatigue in thin gage metallic structure. As a means of prolonging fatigue life, while at the same time restoring structural load carrying capability in damaged structure, composite bonded repairs which incorporate damping layers have been developed. These damped composite bonded repairs are installed on F-15 flight test aircraft. This paper describes ways to recognize sonic fatigue damage, and the design and flight test results of representative damped patches.

Introduction

Military services spend millions of dollars annually to repair cracks in fighter aircraft. These fatigue failures result in significant system down time and reduced operational readiness. Fatigue problems can be divided into two categories: primary structure and secondary structure.

Primary structure cracks are typically the result of maneuver and other static loading which require bolted metal doublers to reinforce the damaged structure. These repairs can be complex, costly, and heavy. When a bolted repair is not possible, replacement of the cracked component can bring additional complications and cost.

An alternative life enhancement technique for primary structure is the bonded composite patch. These patches offer a more efficient way of repairing cracked structure by avoiding drilling new holes and offer weight and cost advantages over bolted repairs. Composite bonded repair technologies have been proven effective on transport aircraft [1-4]. However, this technology is just starting to be applied to fighter aircraft [5-11], where dynamic environments play a more prominent role in the cause of fatigue damage.

Secondary structure failures are often the result of sonic fatigue, which is defined as cracking due to dynamic response from fluctuating acoustic pressures. Structures, which are susceptible to sonic fatigue include, large, flat or thin panels, panels with stress concentration factors (Kt) and structural surfaces exposed to high noise, high temperature, and high mean static stress. These loads are ever increasing as a result of:

- More agile aircraft that can fly at high angles of attack and use vortex impingement to maintain controlled flight.
- More powerful and responsive engines that exhibit higher inlet acoustic levels, exhaust velocities, and exhaust temperatures, plus the ability to perform rapid throttle changes.
- Carriage of fuselage mounted stores and pods which generate turbulent wakes and shocks during high dynamic pressure (Q) flight.
- Blast pressures from large caliber, rapid fire cannon.
- Oscillating, turbulent flows in weapons bays and their wakes.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

On fighter aircraft, secondary structures are lightly loaded during maneuvers, but they still experience cracking primarily due to dynamic loads from buffet, flow-induced noise, shock/boundary layer interaction, and turbulence. Acoustic loads are highest during high speed and low altitude missions for the lower surface and during high angle of attack maneuvers for the upper surface. On the lower surface, turbulent flow forms around external stores, Figure 1, and will be used as an example source of damage is discussing repairs. This noise is characteristically high in frequency (200 to 1000 Hz) and broad band in nature as will be shown in Figure 4.

For these types of damage, the addition of damping to the structure is known to be an effective solution. Furthermore, combining adhesive bonded composite patches with damping treatments provides robust and effective repair designs for both damaged, and as yet undamaged structure.

Figure 1. On the Lower Surface, Turbulent Flow Forms Around External Stores

Sonic Fatigue Damage Recognition

Sonic fatigue damage may be recognized by the type of structure suffering damage and by the location of the damage. The majority of structure that cracks due to the acoustic loads are thin gage metallic structure; such as, fuselage skins, access doors, non-load carrying fairings, leading edges, and secondary support structure; such as, stiffeners, stringers, shear clips, and brackets. In thin gage metallic structure, cracking tends to occur at stress discontinuities arising from chem-mill radii or at fastener holes. The out-of-plane bending response of a panel causes the highest stresses and bending moments to occur at or near the edges of the panels where such features and support structure are located. The critical locations are those which experience the flow effects from the sources enumerated above. They include light panels in engine inlets, behind engine exhausts and inlets, near gun muzzles, within weapons bay cavities, or in the wakes of turbulence producing features.

For maneuver load (or static) fatigue considerations, increasing part thickness to reduce stresses can be a solution. However, for dynamic (or sonic) fatigue problems this approach is generally ineffective because reduced stress is usually accompanied by increased stiffness, which tends to move fatigue problems into adjacent structure. For sonic fatigue, the addition of damping to the structure is known to be an effective fix. Furthermore, coupling adhesive bonded composite patches with damping treatments provide robust and effective repair designs.

Three examples are offered to illustrate these concepts.

(1) Chem milled inlet skin panels of 0.05 in. thickness began cracking after 60 hr of service. The skin was increased to 0.071 without chem.-mill, and the cracks subsided until a more powerful engine with different fan characteristics was installed. Then, stringer flanges began cracking within 250 hr of the change. The stringers were changed to titanium, and within 200 additional flight hours, the rivets attaching the skins to the stringers began failing. This is a typical example of chasing the damage as a result of selective structural strengthening.

(2) A small fuselage skin panel, Figure 2, incident to the engine exhaust plume exhibited cracks after 150 hr. The cracks were stop drilled and small riveted doublers were applied. Within another 200 hr the cracks were appearing from beneath the doublers. Other aircraft were repaired with larger riveted doublers containing a myriad of "field rivets." In a short time cracks began to occur in the doublers at the rivets. This example typifies the inability of mechanically fastened doublers to arrest sonic fatigue cracking.

(3) After installation of a blunt ended pod beneath the fuselage, aircraft began suffering skin cracking and door damage after high Q sorties. After one particularly demanding occurrence, an aircraft returned with the damage shown in Figure 3. This is an example of the damage that can be caused by turbulent wakes and shocks.

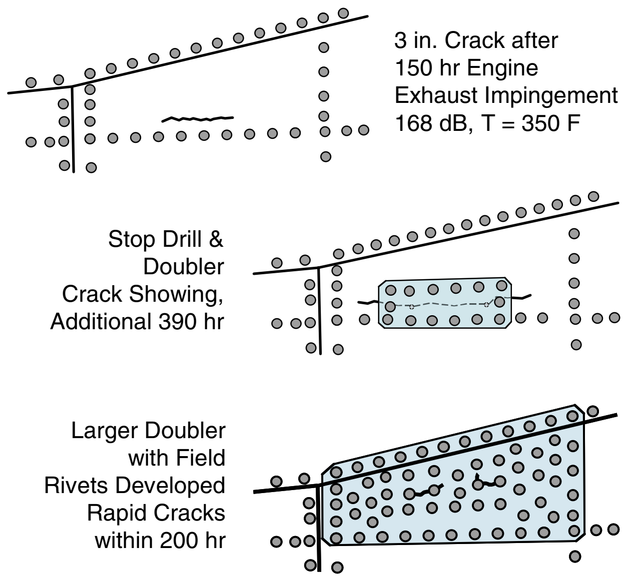


Figure 2. Failed Traditional Sonic Fatigue Repairs

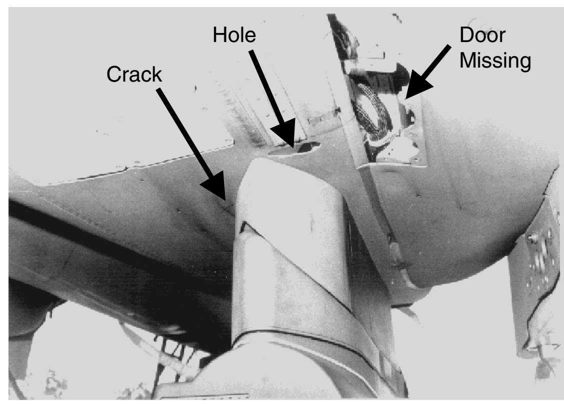


Figure 3. Sonic Damage After Prolonged High Q Flight

Typical Acoustic Loads

The increase in the acoustic spectrum on the fuselage skin at the critical locations from the addition of the pod may be seen in Figure 4. It may be noted that the spectrum levels are high over a wide frequency range. Therefore, strengthening the panel, which will increase its natural frequency, will not help, and in fact a change from 300 Hz to 500 Hz would be detrimental.

Other excitation sources mentioned herein have levels of similar magnitude. Engine exhaust plume spectra may be in the 165 dB to 170 dB range, and relatively flat over a wide range of frequencies. Acoustic spectra from engine inlets and weapons bays will

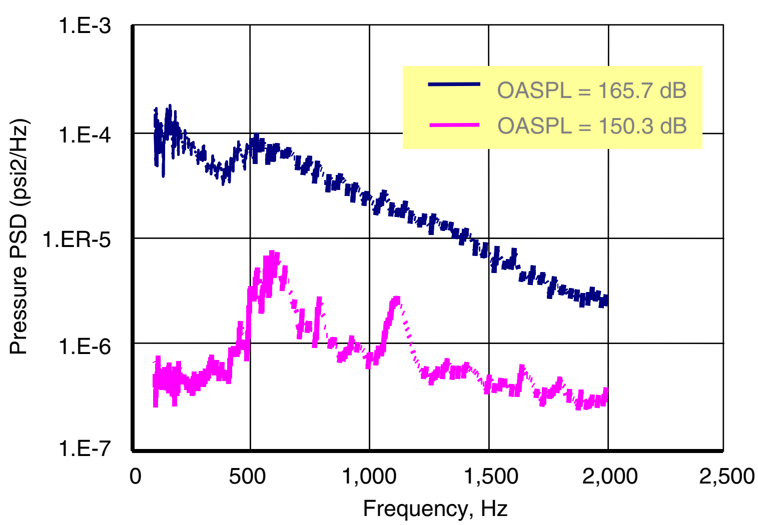


Figure 4. Incident Sonic Spectrum, With and Without Stores

have high broad-band levels, and in addition contain pure tones of very high amplitude. However, it will be impossible to tune the structural frequencies to avoid the tones because they change frequency with engine speed in the inlet case, and with Q in the weapons bay case.

Sonic Fatigue Repair Types

There are no special repair guidelines or maintenance procedures for sonic fatigue damage. A traditional repair using a fastened doubler is still the repair of choice at all levels of maintenance. The repair procedure involves stop drilling the end of cracks or the complete removal of the damaged area. The doubler is then matched drilled to the underlying structure. A sealant is spread over the doubler and the repair is bolted in-place on the aircraft. Doublers are effective in increasing the stiffness of the subject panel. This increased stiffness is often sufficient to prevent the repaired panel from further cracking if the panel were loaded in plane.

Reinforcement - The problem with repairs that involve stiffening only is that fatigue damage often appears on adjacent structure, because the dynamic energy is still present. Doublers mass load the sub-structure and cause higher dynamic bending moments at the panel edges, which may change the response of near-by panels. See Figure 5, which shows the reduction ratio in stress for the patch application as a function of doubler thickness. It may be noted that as the doubler thickness increases, the reduction ratio at the panel edge decreases below 1.0, which indicates an actual stress increase. For example, if a panel is repaired with a doubler of equal thickness, the stress at the panel center is reduced, but the stress ratio at the edge is 0.75, which is a stress increase of 33%. This will likely cause the failure to migrate to the substructure.

Another reinforcement approach involves replacing the failing panel with a thicker one, or a panel of a stiffer, more fatigue resistant material. The result will likely be the same with failures migrating to the substructure as has been noted in one of the example cases in a previous paragraph.

Damping - Damping provides a means for reducing the amplitudes of vibration through dissipation of energy. In built up metallic structure, the damping present is attributable to the minute slippage between fastened assemblies and through internal hysteric material damping. One method to enhance damping in structures is to introduce viscoelastic materials to the assembly. The most efficient form of viscoelastic damping for thin sheet metal structures is constrained layer damping[12].

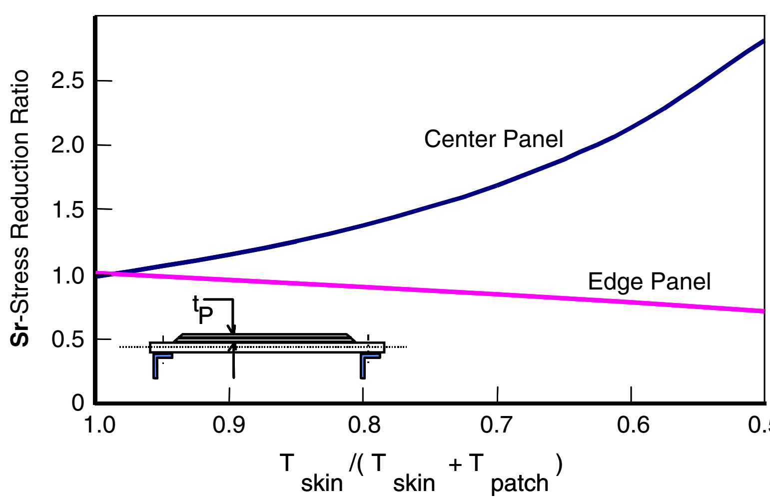


Figure 5. The Effect of Skin Reinforcement on RMS Panel Response

In linear dynamics systems, damping is inversely proportional to the response. Hence, as the damping increases the peak response decreases proportionally. Under broad band random vibration, the response is inversely proportional to the square root of the damping. Hence, the damping would have to be quadrupled in order to decrease the root mean square (RMS) response by 50%. This relationship is shown graphical in Figure 6. Referring to the figure, if the damping is increased by a factor of 6 to 7, a factor which can be easily achieved, the stress reduction in the panel is 2.5 as with the doubler. Unlike the simple doubler, the stress at the edge is also decreased by a similar amount.

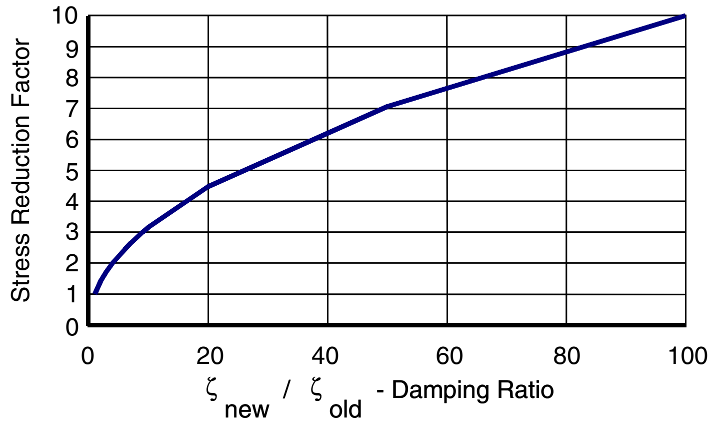


Figure 6. The Effect of Damping on Single Mode RMS Response

Bonded Repairs

The discussion has shown that damping is effective in lowering loads in and around the panel, where a doubler (or patch) alone reduces stresses in the panel center, but can increase load going into the substructure. These conclusions led to the investigation of composite bonded patches that incorporate damping. Damping reduces the adverse effects of doublers on substructure while continuing to restore load-carrying capability. A complete test program was developed for investigating the effectiveness of composite bonded patches as repairs for sonic fatigue damage. In reference (5), the first phases of this test program were described. These were tests for the development of damped repair technology, including coupon beams, sub-element vibration, and acoustic panel tests. The next phase of development involved the application of patches to high response skins and flight testing on fighter aircraft.

The process of designing bonded repairs must take into account the durability of the damaged structure, the repair, and the adhesives used to attach the repair to the structure. Millions of cycles can occur in a few flights for acoustic loadings and current state-of-the-art repairs for maneuver load damaged structures typically do not account for sonic durability. They are usually based only on restoring static strength. Maneuver load repair approaches can be inadequate for acoustic loaded structures where static loads are often much less significant and cracking typically occurs in a short period of time, often before the airframe has experienced 500 flight hours.

Installing damped bonded repairs to improve damping characteristics and to restore structural stiffness introduces challenges. Issues associated with surface preparation, bondline integrity, and corrosion protection must be addressed. Access and sealing must be considered to allow using vacuum bag and heat curing. Post repair NDE must be able to verify integrity as well as monitor repair performance throughout the life cycle.

Another consideration is cost. Although, detailed cost studies have not been performed, it has been suggested that the replacement of sheet metal parts is more cost effective than installation of bonded repairs. However, a replaced part will still have the fragility and will be prone to fatigue. Hence, standard composite bonded repair methods need to be altered to make them more attractive for application to thin gage sheet metal panels.

Some recent investigations into damped repairs can be found in Ref. (5-7). The use of damping in composite bonded repairs has been given more attention in recent years. As part of the Durability Patch program [7], the Air Force Research Lab has contracted with Boeing and CSA Engineering to develop and test damped composite patches suitable for high-cycle fatigue life extension. Also, as part of this program they have developed a low cost flight instrumentation module for diagnosing fatigue problems. The device, called a Damage Dosimeter, has a programmable interface and can be used to measure strains under dynamic conditions for long term monitoring. Add-on damping and composite bonded repairs have also been investigated at Aeronautical & Maritime Research Laboratory (AMRL) in Australia [6] for use in the outer skin of the engine inlet on RAAF F/A-18 aircraft. Their work has shown the feasibility of using a damped repair concept.

A just completed test program at AFRL, and supported by Boeing and AMRL, has substantiated mush of the discussion of this paper 7. Cracked panels were patched using various concepts, and the acoustic exposure continued to monitor crack growth. The results showed the addition of damping to be the best method of repair. Constrained layer damping applied to about 80% of the panel area performed the best, but back-side access is required for installation, and a patch is required for the crack in addition. Of the front access-only repair concepts tested, patches of the type described in the remainder of this paper performed best.

The demonstrated technology as described in this paper is transition ready for fleet wide use. Development has progressed through tests and studies to make the incorporation of damping suitable for bonded repair applications.

Structural Demonstrations

As a demonstration of the damped patch technology, two structural applications were chosen that had a history of sonic fatigue due to their proximity to external stores. The first was a center fuselage chine door, and the second was a lower surface skin panel. The latter panel was instrumented with sensors to monitor the structural response. The flight tests will be discussed below.

Center Fuselage Chine Door

On June 17, 1998, an F-15 flew for the first time with a damped composite bonded patch installed on the center fuselage chime door. The patch is a demonstration of pre-emptive type repair technology suitable for sonic fatigue prone structures. This door, and other similar ones, have had a history of damage including skin cracks, latch wear, and hinge fitting cracks. Figure 7 shows the door and patch prior to bonding and as installed on the aircraft. The patch consists of Textron Boron/Epoxy 5505 with a 3M adhesive integrated into the laminate. The patch was then bonded with paste adhesive onto the surface prepped door and cured in an autoclave for a full bonding. This door has been operational on a Boeing/USAF flight test aircraft for over two years.

Figure 7. Center Fuselage Chine Door w/ Damped Patch

Lower Surface Skin Panel

The second demonstration is a lower surface skin panel on the F-15, Figure 8. This panel is part of the lower nacelle skin assembly and experiences high dynamic response during low altitude and high-speed maneuvers. The panel is made of 2024 Al. It is 0.071 inches thick with chem-milled pockets to 0.060 inches and 0.044 inches, and has experienced cracks in the former and in the panel chem-mill boundary. According to Figure 5, this panel is a good example of how a doubler could aggravate the problem if not properly designed to lower loads at the panel edge.

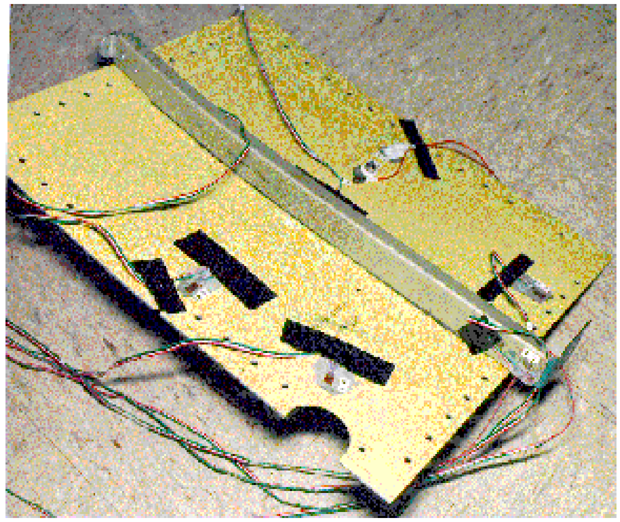


Figure 8. Lower Nacelle Skin Panel Removed from Aircraft Showing the 487.3 Former

The purpose of the damped doubler is to reduce loads in the skin and in the former through damping. The design was verified through finite element analysis. The damping performance of the patch was determined through modal strain energy analysis[14]. The panel was analyzed using random response analysis to determine overall RMS loads reduction, Figure 9. The contour plot shows how the stresses in the skin panel are expected to change. Large reductions are expected in the skin panel with smaller reductions at the edges. The modal damping predicted by the analysis was on the average of ζ = 5% for the lowest 5 modes at the temperature of interest, which is an increase of over 4.

A composite patch system was designed with integral damping treatment, as a simple and low cost retrofit, two 10 by 5-inch rectangular patches of boron/epoxy with integral damping. The complete laminate contains Textron Rigidite Boron 5505, and a damping pack comprised of damping and constraining layers (patent pending). The final patches are shown as installed on the aircraft, Figure 10.

Normalized Principle Stress Reductions:

Location	Ratio (BL w/Patch to BL)
Lug Skin	0.45
Chem-Mill	0.4
Shear Clip	0.6

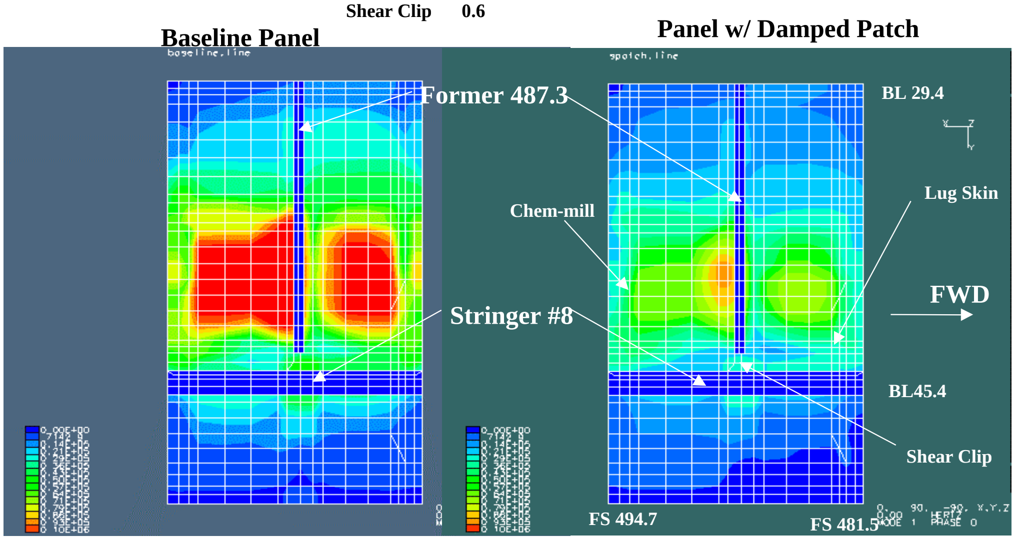


Figure 9. Random Vibration Analysis of Panel with and without Patch

Figure 10. Patches Installed on A/C Prior to Painting -- Looking Aft

Flight Test Program and Results - For the flight test, instrumentation was installed to interpret the change in the dynamic response of the panel. This instrumentation includes strain gages and accelerometers for measuring response, pressure and temperature transducers for measuring the environment. Flight-tests and modal surveys were conducted before and after patching. A similar flight plan was used for both the before and after tests, and similar external store configuration was used.

The initial flight tests established the baseline response and environmental factors such as temperature, acoustic spectrum levels, and quasi-static loading conditions. For instance, the temperature varied 20-F to 105-F. The absolute static pressure varied from 5-psia to 18-psia. The comparison of patched to unpatched response was made with the strain gages and accelerometers. On an overall basis, the acceleration showed the most dramatic response reductions for the patch configuration. These PSDs plots are shown in Figure 11.

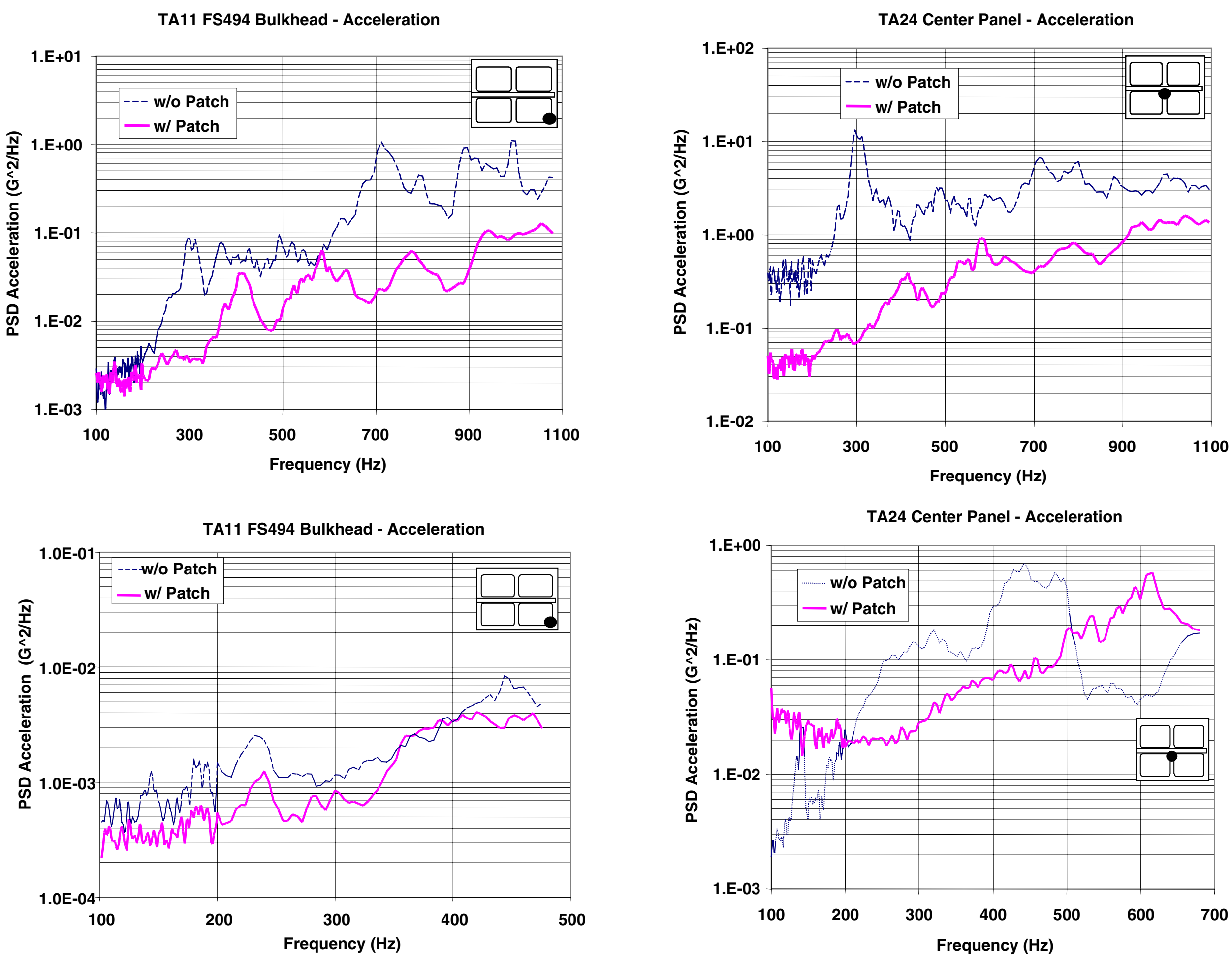


Figure 11. Comparison of Acceleration PSDs w/ and w/o Patch
(Note: The top figures are w/ external stores and the bottom figures are w/o external stores.)

For the clean aircraft (w/o stores) configuration, the center-panel response drops dramatically with the damped patch. The loads going into the bulkhead to which the panel attaches at FS 494.7 are noticeably reduced at the frequencies corresponding to the panel modes. Similarly, for the w/ stores configuration, the center-panel response drops dramatically with the damped patch. The loads going into the bulkhead to which the panel attaches at FS 494.7 are noticeably reduced at the frequency corresponding to the panel modes. On a closer examination, the RMS values for different frequency bands is listed in Figure 12. Note that better response reduction occurs in the 200 to 1000 Hz band than in the overall.

Measurand	Frequency Range (Hz)	RMS Acceleration G_{rms}		Percentage Difference
		W/o Patch	W/ Patch	
TA24	0-200	1.2	3.6	-198.39%
w/o stores	200-1000	4.5	3.2	28.94%
	1000-2000	15.1	23.0	-52.35%
	overall	17.6	24.1	-36.81%
TA24	0-200	8.8	3.4	61.41%
w/ Stores	200-1000	16.2	5.1	68.75%
	1000-2000	71.5	40.9	42.85%
	overall	78.9	42.2	46.52%
TA11	0-200	0.7	0.8	-1.97%
w/ Stores	200-1000	2.2	1.2	42.04%
	1000-2000	25.6	10.4	59.35%
	overall	26.0	10.7	58.70%

Figure 12. Comparison of RMS Acceleration Loads w/ and w/o Patch

Fleet Management Considerations

The types of damage described herein usually manifest themselves rather soon after the configuration begins full envelope flying, or after a change is mission, payload, or engine occurs. Further, it is typically not an isolated case, but will show up on most of the fleet within several hundred additional flying hours. For this reason it is prudent to patch the entire fleet as soon after a trend is noted as is practical. With this approach, the damping may be applied to the critical area, thus avoiding future damage with the addition of minimum weight and complexity. A patch will require more plies to restore strength after cracking initiates. This type of “pre-emptive” repair is recommended.

Design and application of a patch uses the same methods as other composite patching of metal structure, which may be found in the current literature. For the applications reported herein, the fiber portion of the patches was boron. However, carbon or glass fiber could be used with proper care. Surface preparation involves removing all existing paint, grit blasting, and application of silane and a primer. It is recommended that the patch be pre-cured and then bonded in place in a separate process using an adhesive that cures with only a modest temperature elevation.

The choice of the specific damping material to be used in the application depends on the temperature range to which it will be exposed. Actually, two temperature ranges should be considered; (1) a wider range within which the patch must survive without suffering brittle fracture or creeping, and (2) a narrower range within which the damping material must perform. The wider range for fighter aircraft is between –40C and 95C, whereas the performance range is between –20C and 50C. Vendor data is available for a substantial number of materials to aid in selection.

Summary and Conclusions

Rapid fatigue damage can occur in light weight structure that is subjected to high sonic loads. This type of cracking is identified by considering the type of structure suffering damage, and its location in an area of high sonic loading. It is important to identify the damage as sonic fatigue because the type of repairs typically employed for fatigue failures are not effective for sonic fatigue. The use of fastened metallic doublers or part replacement of thicker gage will likely chase the damage to adjacent locations. Replacing the parts with those of a more robust material will have the same result unless the adjacent structure and fasteners are of sufficient strength and durability.

Damped composite bonded patches have been developed for sonic fatigue applications that are low cost and easy to install. These patches are ideal for repair of cracked structure; as well as, being useful as preventative a retrofit against fatigue. General considerations have been discussed for the importance and effect of adding stiffening and damping to solve sonic fatigue. The flight test program discussed herein, has validated the applicability and performance of the damped patch concept as a long term solution for sonic fatigue damage.

Acknowledgments

The worked performed on this program was sponsored by The Boeing Company under Internal Research and Development funding.

References

[1] D. Hammond, “The C-141 Weep Hole Repair - An Update,” Air Force 3rd Aging Aircraft Conference, 26-28 September 1995.

[2] R. Fredell, K. Guyt, D. Conley, “Design Development of a Bonded Composite Fuselage Repair for the C-5A,” 1995 USAF Structural Integrity Program Conference, 28-30 November 1995.

[3] D.O. Hammond, J.B. Cochran, “C-141 Composite Material Repairs to Metallic Airframe Components,” Proceedings of the 1992 USAF Structural Integrity Program Conference, WL-TR-93-4080, September 1993.

[4] J.B. Cochran, T. Christian, D.O. Hammond, “C-141 Repair of Metal Structures by Use of Composites,” Proceedings of the 1987 Aircraft/Engine Structural Integrity Program (ASIP/ENSIP) Conference, AFWAL-TR-88-4128, June 1988.

[5] Liguore, S. L., Perez, R., and Walters, K., “Damped Composite Repairs for Acoustic Fatigue,” Presented at the 3rd AIAA Aeroacoustics Conference, May 12-14, 1997 in Atlanta GA.

[6] Chiu, W.K. and et. al, “Life Extension of the F/A-18 in-take nacelle using add-on dampers,” SPIE Vol. 3326, #0277-786X.

[7] Rogers, L.C. and Searle, I.R., "Durability Patch: repair and life extension of high cycle fatigue damage on secondary structure of aircraft," DOD/FAA/NASA Conference on Aging Aircraft, July 8-10, 1997, Ogden UT.

[8] L.R.F. Rose, K.F. Walker, A.A. Baker, E.S. Wilson, “Repair Substantiation for a Bonded Composite Repair to F-111 Lower Wing Skin,” Air Force 4th Aging Aircraft Conference, 9-11 July 1996.

[9] J. Mazza, “Repair of Fatigue Cracks in Dutch F-16 Aircraft with Bonded Boron Patches,” Air Force 4th Aging Aircraft Conference, 9-11 July 1996.

[10] J.M. Kelly, “F-4 Boron/Epoxy Repair Patch Double Lap Shear Tests,” McDonnell Douglas Laboratories Technical Memorandum, 28 March 1995.

[11] J. Colvin, “Stiffness Enhancement Using Bonded Composite Skin Doublers Solved F-15 Vertical Tail Buffet Problems,” Proceedings of the 1992 USAF Structural Integrity Program Conference, WL-TR-93-4080, September 1993.

[12] J. Soovere, Aerospace Structures Technology Damping Design Guide, AFWAL-TR-84-3089.

[13] S. Liguore, “Retrofit Damping Concepts for Fuselage Skin Structure,” McDonnell Douglas Corporation, Report Number MDC 95X0026, 8 September 1995.

[14] L. Rogers, C. D. Johnson, D. A. Keinholz, “The Modal Strain Energy Finite Element Analysis Method and its Application to Damped Laminated Beams,” 51st Shock and Vibration Symposium, San Diego, CA., 1980.

[15] R. Gordon, J. Hollkamp, S. Liguore, “An Experimental Investigation of Damped Repairs for Sonic Fatigue Cracking,” 5’th Joint NASA/FAA/DoD Conference on Aging Aircraft, Kissimmee, Florida, September, 2001.

This page has been deliberately left blank

Page intentionnellement blanche

SUSTAIN™
Sustainment Strategy for Avionics Information Needs

Dr. Eric S. Sjoberg, Ms. Linda L. Laughlin
Dr. J. Powers Garmon, Ms. Molly J. Gary
Georgia Tech Research Institute
400 W. Tenth St., Atlanta, GA 30332-0801, USA

Mr. David Lasater and Mr. Samuel Calloway
Warner Robins Air Logistics Center/LFEFA
296 Cochran St., Robins Air Force Base, GA 31098-1622, USA

Introduction

The primary directive for a military systems sustainer is to provide support to the warfighter by insuring that his weapons system is operational and available. That requirement is the bottom line, but there are myriad ways of accomplishing it at vastly different costs. Combined with the fact that military weapons systems are being called upon to remain in service far past their design lifetime, increased importance is placed on the system sustainer to make correct decisions, especially considering the shrinking military budgets. There is a wealth of data available for the sustainer to use in making decisions, including historical reliability and cost, spares availability, future planned use and, perhaps most importantly in the near term, component obsolescence. An efficient, mechanized method of analyzing all the data available to the sustainer is needed to help him, or her, make correct decisions at the correct time.

Component obsolescence is not a problem for military avionics systems…until that component threatens mission readiness. First generation component obsolescence tools evaluated individual component obsolescence in a vacuum. Second generation tools monitor component obsolescence at a board, box, or even system level and provide an estimate of "system health" and may even allow sharing of information across systems. Neither of these approaches answers the basic question of when an obsolete component will affect mission readiness. Without that information, it is impossible for the sustainer to make the best decisions. For example, there may be a severe obsolescence problem on a particular board of an aircraft radar system. But if that board exhibits good reliability and there are adequate spare assemblies available, then addressing this particular obsolescence issue may be postponed, perhaps indefinitely.

What is required is a true sustainment tool that allows implementation of a "just-in-time" sustainment approach, by consolidating all the required information into a single automated tool which displays data in a concise and informative manner. The tool should identify and prioritize the sustainment actions required and also tell the sustainer when each action must occur in order to avoid any compromise to mission readiness. The sustainment actions must be identified far enough in advance so that they can be planned for, budgeted for, and implemented before any aircraft are grounded. The following paragraphs describe the characteristics required of such a tool and the program instituted by the Warner Robins Air Logistics Center to develop such a program.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

Required Characteristics of a Sustainment Software Tool

The minimum requirements for a sustainment software tool are that it should:

- Be **relevant** to the needs of the user
- Perform **accurate** data analysis
- **Completely** consider all relevant information
- Perform **timely** analysis of up-to-date data
- Be **user friendly** in its data entry and presentation

Additional implementation concerns include modular software organization and flexible data analysis capability. Each of these areas is discussed in the following sub-sections.

Relevance

The sustainer of a military weapon system provides a service to the user of that system and, often, funding for that service is provided by the user. It is incumbent upon the sustainer to base decisions on the needs of the user. Therefore, for a military weapon system, the primary metric for any sustainment decision must be mission readiness. A secondary metric is the cost to maintain a given level of mission readiness. Decision metrics based solely on cost are very likely at some time to overlook a significant mission readiness driver because it does not provide an adequate return on investment.

Accuracy

The sustainer must believe the results presented by the analysis tool and, in turn, his superiors and the users must also be convinced that the tool is providing accurate data. In order to meet this requirement, the results must be traceable back to the input data and most importantly, the input data must be correct. “Officially approved” databases should be used wherever possible as the source for defendable, accurate data. If “official” data bases do not exist, then historically accurate and widely used data sources should be selected.

Completeness

Consideration of all aspects of system sustainment is required for accurate results and is the most difficult part of developing a sustainment tool. A complete sustainment tool must have the following attributes:

1. Sustainment issues must be addressed from the component level up to no lower than the system level (i.e., radar system, EW system, etc.) and preferably to the platform level for all equipment for which the sustainer is responsible.
2. All variants of each system and their interchangeability must be included.
3. Spares at all levels (i.e., component, board, box, etc.) must be considered including the assembly indenturing.
4. Component obsolescence must be determined.
5. Repair history to the component level must be available.
6. Force structure effects should be included.

As mentioned above, these data should be obtained from “official” databases to insure data accuracy.

Timeliness

There are two aspects to timeliness in this setting. First, the data must be up-to-date so that the analysis results are current. Second, the analysis algorithms must be able to accurately predict up to five years in the future and farther than that with decreased fidelity. And further, the analysis must be performed in an efficient manner so that data latency is minimized.

User Friendliness

If the sustainer will not use the software, then it is of no use. The two main requirements in this regard are first, since there is a very large amount of data to be analyzed, the data must be available electronically and the data input and analysis must be largely automatic. And second, the processed data must be presented in an unambiguous and easily understood manner.

Implementation Considerations

The benefits of modular software have been proven. If the sustainment tool is destined for more than one application, then some customization for each user should be anticipated. With modular software, additional capabilities can be added to a baseline capability at minimal expense so as to provide only those capabilities required by the user. The tool should also be flexible to allow “what-if” scenarios to be postulated and explored by the sustainer.

SUSTAIN™

The SUSTAIN™ (Sustainment Strategy for Avionics Information Needs) program is a comprehensive software sustainment tool with the above characteristics. Its development is being sponsored by the Warner Robins Air Logistics Center F-15 Avionics Hardware Section (WR-ALC/LFEFA). In 1989 this group at Warner Robins began an aggressive program to combat integrated circuit (IC) obsolescence in the F-15 avionics. That program has been in the forefront of obsolescence resolution activities in the USA and has a proven history of success. As their program evolved and matured, LFEFA recognized the need for a comprehensive sustainment tool to add to the AVCOM component obsolescence evaluation tool. The Georgia Tech Research Institute has been working with WR-ALC/LFEFA since 1989 on various sustainment activities for the F-15 avionics and is currently developing SUSTAIN™. This tool is being developed to support those avionics systems unique to the F-15 for which WR-ALC/LFEFA has sustainment responsibility. Electronic warfare systems, jet engines, and common items, such as radios, are not currently included in SUSTAIN™. Specialized contractor repair items are also not included because, in many cases, LFEFA does not have the insight into the components of those devices to effectively support them.

The SUSTAIN™ concept is quite straight forward, but the implementation is necessarily somewhat complicated. Real historical data are used wherever possible to calculate the required parameters as noted below. The only estimated information is the predicted component obsolescence. It is assumed that the component usage rates measured today are predictive of the future usage rates and are used to predict required sustainment actions.

The basic functional modules that comprise SUSTAIN™ include the mission readiness function, the sustainment action function, the sustainment cost function, and the technology insertion function. Additional potential capabilities include a sensitivity analysis function and a reliability analysis function. The following paragraphs describe the above functions including the databases utilized and examples of output plots.

Mission Readiness Function

This function is the heart of this sustainment concept. Two metrics for determining mission readiness are computed by SUSTAIN™; the mission incapable (MICAP) vulnerability versus time and the mission degradation versus time. The main difference between these metrics is that MICAP vulnerability is a system view while mission degradation is a lower level look at the assemblies that make up the system. In order to compute these metrics, detailed knowledge of the system, its components, assembly interchangeability, spares at all levels, force structure, component repair information, Defense Logistics Agency (DLA) inventory and usage rates, and component obsolescence are required. Table 1 lists the data required and the source of the data. All data sources are updated on a quarterly basis.

Table 1. Mission Readiness Data

Required information	Data Source
System structure, indentured parts list and interchangeability	Illustrated Parts Breakdown T.O. Contained in AVCOM (MTI Inc.)
Component obsolescence	AVCOM (MTI Inc.)
System spares and requirements	Express (USAF)
Component inventory and usage rates	DLA (DoD)
Depot repair data	Avionics Repair Knowledge Base (Warner Robins ALC) Work Documents and MPS System (Ogden ALC)
Force Structure	TBD

System Structure

Information on system assemblies including the indentured parts list and assembly interchangeability is taken from the official USAF Illustrated Parts Breakdown Technical Order. This information, for the F-15, is contained electronically in the AVCOM component obsolescence tool. This proven tool was developed by Manufacturing Technology Inc. (MTI) and has been in use by the F-15 sustainers for several years.

Component Obsolescence

AVCOM also contains the current status of component availability and predictions as to future availability of approved components. Only those components currently listed as obsolete are included in the MICAP vulnerability evaluations.

System Spares and Required Inventory

Express is an official USAF database that summarizes data from several other databases. It is used by SUSTAIN™ to supply the number of spare assemblies at all levels (LRU, SRU, sub-assembly, etc.). The assemblies are divided into several categories, including those that are either serviceable, repairable, in-transit, non-repairable, etc. The repairable category may then require further subdivision if an assembly is awaiting parts (AWP) that are obsolete and unobtainable. That determination is made from data obtained from the facility at which that assembly is repaired. Also itemized in Express are two required spares quantities for each assembly in the system: the peace-time operating spares (POS) level and the war readiness materiel (WRM) level. These spares quantities are official USAF determined values. The POS level is the lower of the two spares levels and is a minimum spares quantity required to support world-wide peacetime operations. The WRM level is no less than the POS level and contains additional spares required in the case of a conflict situation.

Component Inventory and Usage Rates

“The Defense Logistics Agency is a logistics combat support agency whose primary role is to provide supplies and services to America's military forces worldwide.” (http://www.dla.mil/) For the F-15 application, DLA has been contracted to provide information on all components (primarily integrated circuits) of interest to this aircraft. For a set of national stock numbers (NSNs), DLA provides the quantity on hand and the historical demand rate for that NSN over the last eight quarters in electronic format. These data are updated quarterly. Note that the demand rate is for all organizations that order through DLA and includes more than F-15 usage. This information is used to forecast for how long DLA will be able to furnish components for the systems under study.

Depot Repair Data

The F-15 unique avionics are primarily maintained at two ALCs; Warner Robins and Ogden. To our knowledge, a USAF-wide depot repair database does not exist, so local databases are used to capture repair history. For the SUSTAIN™ application, repair history of all systems down to the component level, including the reference designation, is desired. This allows the program to determine the F-15 unique usage rate for each component in the systems under consideration. Additionally, information on AWP assemblies may be analyzed to determine if an assembly is waiting for an unobtainable component. If that is the case, then that assembly should be eliminated from the spares inventory, but if the assembly is waiting for an available component, then it will be counted in the spares inventory.

Force Structure

SUSTAIN™ assumes that the current failure rates per flying hour will be constant into the future. As the number of aircraft, or the number of flying hours per aircraft change, then the total number of failures for a component on a board will change. The force structure numbers are based upon USAF Air Combat Command estimates.

Mission Readiness Computation

SUSTAIN™ determines the impact on mission readiness of unobtainable components in the following manner. The date at which components are predicted to become obsolete is predicted by AVCOM. On this date it is assumed that the component can no longer be acquired from a commercial vendor. The DLA inventory and usage rate of that component is then used to determine when the DLA inventory will be exhausted. At that point in time, any future failures of that component in the system under consideration are considered to be non-repairable. The impact of that unavailable component on every assembly in the system is then determined based on the historical depot repair history and anticipated force structure. Then, as future failures are expected to occur, the spare assemblies are drawn down to repair the systems. All interchangeable assemblies are included in the analysis. When all the interchangeable assembly spares are exhausted, then next higher level assemblies are cannibalized for the required assembly. Additional spares (of different assemblies) resulting from the cannibalization are added to the available spares as appropriate. This process is continued all the way up to the system or aircraft level for all systems under consideration. When the number of spare assemblies drops below the WRM level then a mission degradation situation is predicted in which some aircraft will be grounded for a period of time (a temporary MICAP) due to an insufficient spares supply. When the number of spare assemblies drops below zero (i.e., the number of operational assemblies is less than the number of aircraft), then a permanent MICAP will occur. Figures 1 through 3 present typical plots of the mission readiness evaluation for a notional avionics system. A single line-replaceable unit, L-12, is primarily responsible for System 1 depletion, as depicted by Figure 1. Shop-replaceable unit slot S-122A,-122B is responsible for L-12 depletion, as seen in Figure 2. The plots shown in Figure 3 reveal that five microcircuits contribute to slot S-122A,-122B depletion. Three microcircuits, P-36, P-37, and P-39 are identified as depletion drivers. MICAP analysis indicates that extended MICAPs due to System 1 can be deferred for several years if sustainment action is taken for the three aforementioned parts.

Sustainment Action Function

The sustainment action function is intended to assist the sustainer to identify the part-level actions required to maintain the system. Those components identified by the mission readiness function as causing MICAPs or mission degradation within the analysis period are targeted for action. In addition, components that are predicted to become obsolete are included in the analysis. Thus, depending on the time frame under consideration, most, if not all, components may be targeted for sustainment actions. This module accounts for uses of redesigned components in each application in the system under consideration and across systems on the platform. Thus, redesign/replacement of an obsolete IC on one board will solve that same IC's obsolescence problem in all applications. An "Action Code" is assigned to each component indicating the urgency with which the component obsolescence must be addressed.

Historical sustainment action information and a part solution/cost matrix supplement the above data. Based on the type of component that is unobtainable (digital, analog, ASIC, hybrid, etc.) a specific type of sustainment action is recommended which could vary from a relatively minor form, fit, function and interface (F^3I) component replacement to a major redesign of a hybrid. Thirty different sustainment actions are currently supported by SUSTAIN™, plus flags for insufficient component data. Information on the component type is contained in AVCOM and

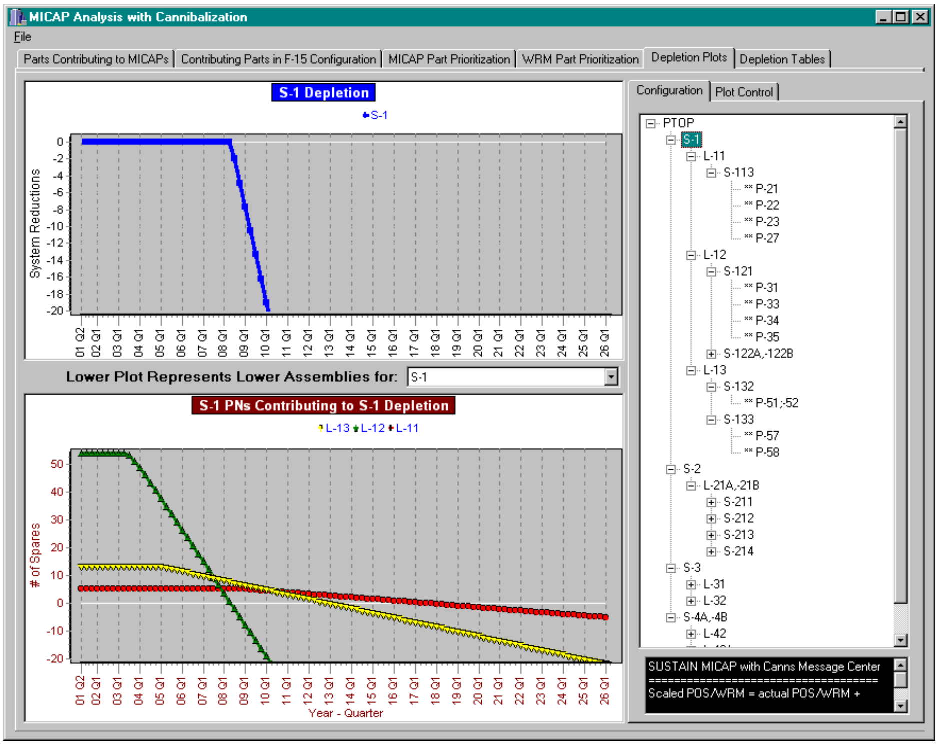


Figure 1. System- and LRU-level Depletion Curves

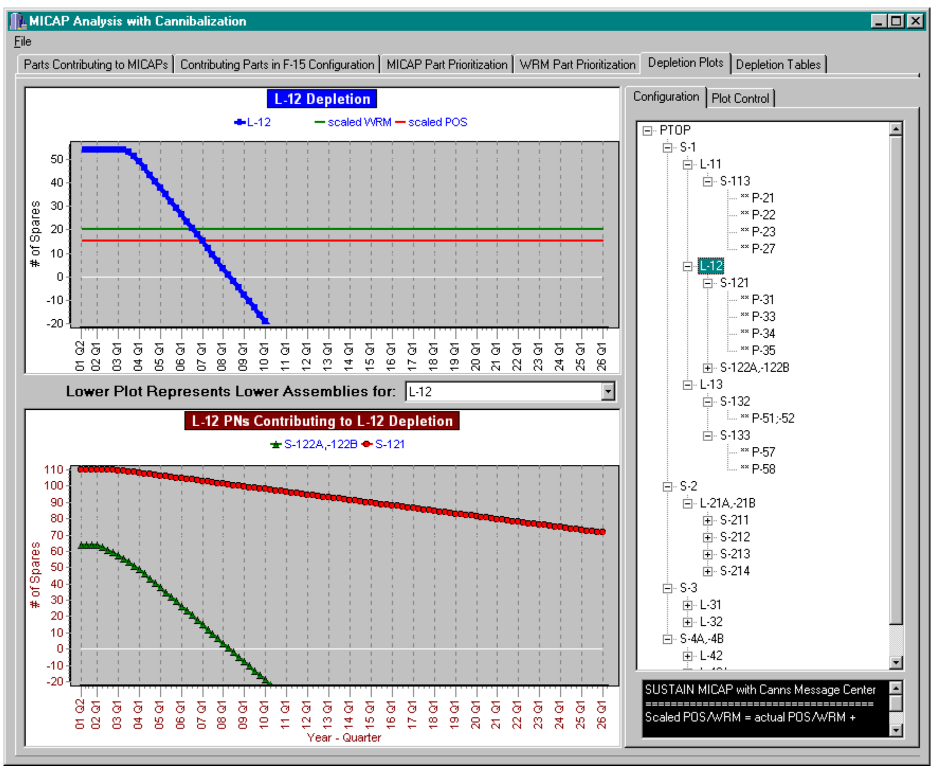


Figure 2. LRU- and SRU-level Depletion Curves

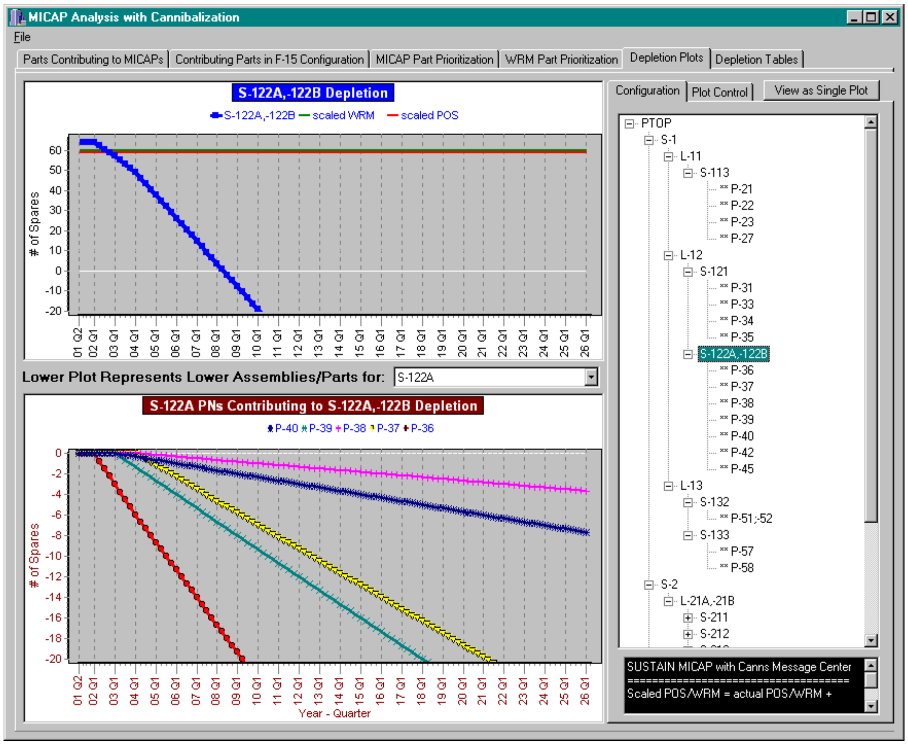


Figure 3. SRU- and Microcircuit-level Depletion Curves

the recommended action is obtained from historical data on sustainment actions for similar components. The part solution/cost matrix contains data on the costs and schedule for performing each type of sustainment action and is also based on historical data.

Additional electronic data that would enhance the sustainment action function includes refined information on aftermarket suppliers and their inventory of components or IC die and information on life-time buy opportunities. Some of this data is currently used by SUSTAIN™, but automating the updating of this data could be quite important in determining the best component sustainment action.

Sustainment Cost Function

The sustainment cost function in SUSTAIN™ is strictly defined as the sum of two parts: the costs associated with assembly returns to the depot for repair (referred to as “not repairable this station” or NRTS assemblies) and the predicted costs of F^3I component obsolescence resolution. No other costs are included in this calculation. This cost metric therefore captures the costs associated with the reliability of each NRTSed assembly, all depot costs (manpower, test equipment, overhead, etc.), and obsolescence effects. Specific costs not captured by this approach include manpower at the O and I (organizational or flight-line and intermediate) level shops, costs of test equipment at the I level shop, cost of can-not-duplicate (CND) failures at I level, and repairs accomplished at I level.

The NRTS costs are determined by the number of NRTS actions for each assembly returned to the depot for repair and the exchange cost for that assembly which is obtained from standard government databases. Obsolescence resolution costs are obtained from the Sustainment Action function.

Technology Insertion Function

The technology insertion function assists the sustainer to optimize sustainment actions. It automates the task of comparing sustainment solutions at several levels, i.e., component, board, assembly, box or system. This function examines the component obsolescence of each assembly in the indentured assembly list and correlates those projections to determine when it might be more cost effective to replace a higher level assembly rather than a component. For example, if it is predicted that several components on one board will become unobtainable in a short span of time and replacement of those components will cost $2 million, but redesign of that board would cost $1 million, then it might be more cost effective to redesign at the board level. Other considerations that SUSTAIN™ takes into account include the impact on other assemblies of component redesign, failure rates, the risks of each potential solution and the estimated reliability of each solution. Assumptions that are made by SUSTAIN™ include that the redesigned assembly is F^3I and that the impact on I level and depot test equipment is minimal.

Additional Capabilities

Sensitivity Analysis Function

The sensitivity analysis function allows the user to determine the sensitivity to certain actions by postulating “what-if” situations which are then analyzed in a normal fashion by the SUSTAIN™ software. For example, if an obsolescence resolution action were postponed, or if the force structure were changed, then the program would provide the capability to quickly evaluate the impact of that action in terms of mission readiness. This feature could also be very useful to maximize mission readiness while operating within budget constraints.

Reliability Analysis Function

This capability elevates the functionality of SUSTAIN™ to become a complete one-stop sustainment tool. This function would identify those assemblies that are maintenance drivers from either a failure point of view or from a maintenance man-hours perspective. To accomplish this capability, maintenance data from the intermediate repair shop and from the depot (and potentially organizational level) would be gathered to determine the number of aircraft removals, the I-level CND rate, the NRTS rate of each assembly and by correlating serial numbers or work order numbers, the depot repair action. Reliability statistics can then be easily computed and reliability drivers determined. These data may then be used to get an estimate of the true cost of ownership of each assembly studied. By then correlating the reliability, cost of ownership and mission readiness information, the best sustainment decisions may be made.

Conclusions

SUSTAIN™ is being developed to be relevant to the needs of the user, and to provide accurate and complete data analysis in a timely and user friendly manner. The unique feature of the concept is that it takes advantage of the large amount of data available to the sustainer and processes that data in a manner that is most useful to him or her.

Relevance is insured by using mission readiness as the primary metric for evaluating potential sustainment actions. Additional information comes from the automatic calculation of the sustainment cost for each assembly under consideration.

Accuracy of the output is guaranteed if the analysis algorithms are correct and the input data are correct. By using only government approved or acknowledged databases, then the data accuracy is as good as can be obtained.

Completeness of the analysis is obtained through examination of the complete indentured parts list for each system included in SUSTAIN™ and through inclusion of all data relevant to the sustainment process.

Timely data is insured through quarterly data updates from each data source. Projections of sustainment actions from 5 years to 25 years are automatically generated.

User friendly, unambiguous output data format is a main goal of SUSTAIN™. Data updates are also performed automatically on data provided electronically.

The initial capability of the program was demonstrated in the spring of 2001 and complete capability is expected in mid-2002.

A USN Strategy for Mechanical and Propulsion Systems Diagnostics and Prognostics, Life Usage Monitoring and Damage Tolerance: Applications to Aging Aircraft Problems

Andrew Hess
(Tel: 301-757-0477, Fax: 301-757-0562
E-mail: HessAJ@navair.navy.mil

William Hardman
Tel: 301-757-0508, Fax: 301-757-0562
E-mail: HardmanWJ@navair.navy.mil

Rebecca Ahne
Tel: 301-757-0520, Fax: 301-757-0562
E-mail: AhneRL@navair.navy.mil

Leo Fila
Tel: 301-757-0517, Fax 301-757-0562
E-mail: FilaLT@navair.navy.mil

Naval Air Warfare Center Aircraft Division
Propulsion and Power Dept.
22195 Elmer Rd. Bldg. 106
Patuxent River, MD 20670-1534, USA

ABSTRACT

A US Navy strategy has been generated to develop and demonstrate diagnostics, prognostics, health management and life management for propulsion and mechanical systems. How this overall strategy has evolved and the current status will be presented. The SH-60 platform was selected for the first proof-of-concept effort to develop, demonstrate, and integrate available and advanced mechanical diagnostic technologies for propulsion and power drive system monitoring. Included in these technologies were various rule based and model-based analysis techniques that were applied to demonstrate and validate various levels of diagnostic and prognostic capabilities. These will be discussed and updated. Using past "seeded fault" tests as case examples, various diagnostic methods were used to identify the faults, and various means of applying prognostics, health management and life management are discussed. Other more recent examples of "seeded faults" and related tests will also be discussed as case studies, demonstrating various degrees of diagnostic, prognostic, health management and life management capabilities. Relative rating of the performance of some of the different analysis techniques evaluated will also be discussed. As used in this paper, prognostics is the capability to provide early detection of the precursor and/or incipient fault condition to a component or subelement failure condition; and to have the technology and means to manage and predict the progression of this fault condition to component failure. The benefit of this prognostic approach is increased safety and decreased maintainability costs over the aircraft life cycle, enabling better management of both existing and potential aircraft system faults. This prognostic philosophy will be further embellished, using examples from past and more recent "seeded fault" databases; to define accomplishments and to discuss additional needed demonstration requirements. Multi-variate analysis, reasoners, and information fusion requirements and approaches will also be discussed. Finally, very recent accomplishments, status and future planned efforts for the USN Helicopter Transmission Test Facility (HTTF) and other related test resources will be presented.

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 to effect 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. The need to accurately identify and diagnose developing faults in mechanical systems is central to the

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

ability to reduce mechanically induced failures and excessive maintenance.

The U.S. Navy would clearly benefit from a reliable state-of-the-art diagnostic capability on-board rotary wing aircraft. An advanced prognostic capability would provide even further benefits. 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.

The evolution of automated diagnostic systems for helicopter mechanical systems has been greatly advanced by the Navy in a 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) was carried out using both an iron bird test stand and SH-60B/F flight vehicles.

While diagnostic capabilities to detect a specific component failure event are relatively straight forward; prognostic capabilities are less well developed and can have a much larger payoff. Any system considered for fleet-wide implementation should have both capabilities. Any program to demonstrate and validate diagnostic capabilities must also address some degree of prognostics. This program attempts to do both.

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.). Several tests have been run in the Navy's spin pit facility and Helicopter Transmission Test Facility (HTTF) at Patuxent River Naval Air Station, and these tests will be discussed in this paper along with their applicability to aging aircraft. Also, a discussion of future planned events will be presented in this paper, along with the role they play in mitigating problems traditionally experienced in an aging

OVERVIEW OF HIDS

In 1993, the NAWCAD awarded a competitive contract on the Broad Agency Announcement to Technology Integration Inc. (Now Goodrich Corporation) for two functionally equivalent integrated diagnostic systems. One system was configured for rack mounting in the HTTF and the other is flyable ruggedized commercial grade hardware. The design uses an industry-standard open architecture to facilitate modularity and insertion of new hardware and software. The system is comprised of 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 subsystem. System architecture and data flow is shown in Figure 1. Though not a production type unit, the vibration acquisition system is essential to acquire the raw data necessary to substantiate the diagnostics technology and obtain enough knowledge to write the minimum acceptable production specification.

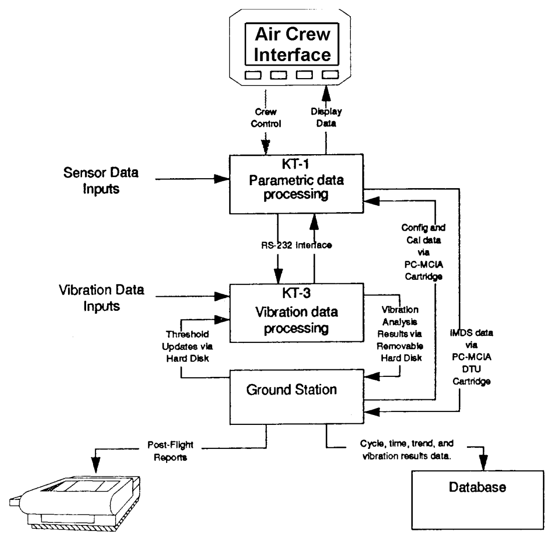


Fig. 1. Diagnostic System Architecture.

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 the engine Power Performance Index (PPI). The PPI is a fourth order best fit curve representing an engine degraded 7.5% from the specification line, and can provide a warning to the pilot when an engine has degraded due to salt ingestion, sand erosion or other foreign object damage (FOD).

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 detection with low false alarm rates. Vibration data recorded at both Trenton, NJ and Patuxent River, MD used 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:

Fig. 2. Central Display Unit.

1. 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 database.

2. 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 sample, 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.

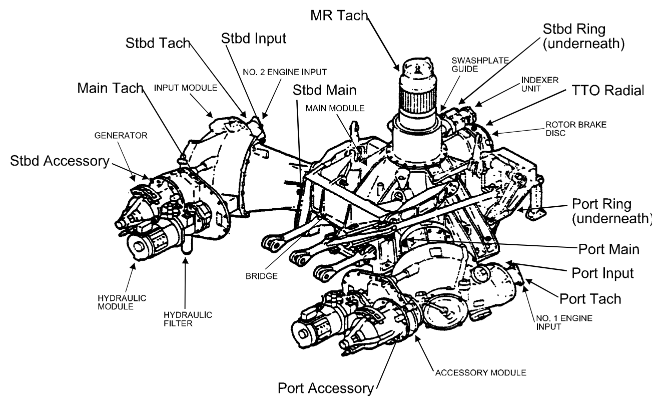


Fig. 3. Accelerometer Locations on Input and Main Modules.

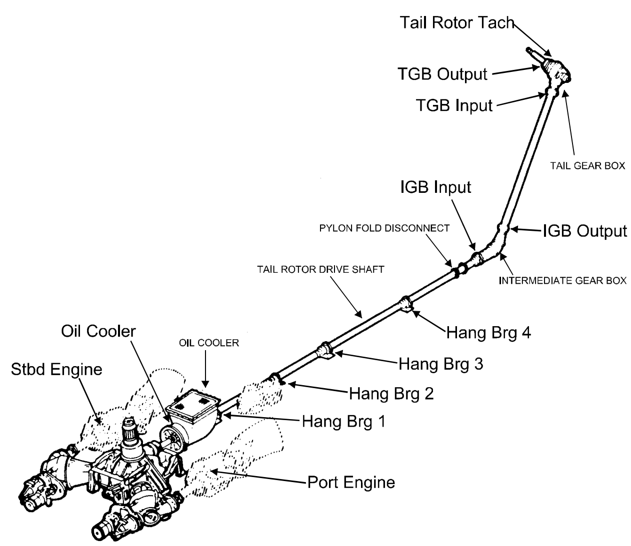


Fig. 4. Accelerometer Locations on the Tail Drive System.

3. Capability for on-board processing. All gears, bearings and shafts are analyzed and the diagnostic results are written to the aircraft parameter data file 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 exceeding preset limits, 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 in response to 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.

Vibration Based Prognostics: Though it is often difficult to separate diagnostic and prognostic performance in a seeded fault program such as this, one of the by-products of this testing was the demonstration of the potential and performance of prognostics.

As a working definition for this paper: prognostics is the capability to provide early detection of the precursor and/or incipient fault condition to a component or sub-element failure condition; and to have the technology and means to manage and predict the progression of this fault condition to component failure. The early detected, “incipient” fault condition, is monitored, “tracked”, and safely managed from a “small” fault as it progresses to a “larger” fault, until it warrants some maintenance action and/or replacement. Through this early detection and monitoring management of incipient fault progression, the health of the component is known at any point in time and the future failure event can be safely predicted in time to prevent it.

Applying many of the same algorithms and techniques used for vibration based mechanical diagnostics, a significant degree of component failure prediction and prognostics was demonstrated during these tests. Often the extrapolation of vibration frequency data, statistical parameters and/or diagnostic indicator trends is the technique used to enable failure prediction. It is of course key to have sensors, algorithms, and diagnostics indicators (or indices) that are sensitive and accurate enough to “see” the precursor or incipient “small” component fault. It is equally important to have a reliable experience database with examples of similar types of “faults” so that the failure progression rate is understood. Using this experience database knowledge and the understanding of various types of failure progressions will enable the intelligent settings of alarm thresholds. It is envisioned that in most cases, the alarm thresholds for safety-of-flight (cockpit warning) will be significantly higher than for maintenance. Establishing these alarm thresholds is a very necessary step in implementing future failure event prediction and enabling prognostics. Without the benefit of an extensive experience database of actual component failures with fault progression data and/or a comprehensive “seeded fault” trials as the SH-60 HIDS program, establishing these alarm thresholds is virtually impossible.

Rotor Track and Balance: The ROTABS system promises to negate the need for on-board trackers and utilizes higher order mathematics and a significantly larger data set to resolve the adjustments required to keep the rotor system in track and balance.

A continuous monitoring of the in-flight rotor track and balance condition will alert the maintainers of out-of-limit conditions that, among other things, will result in high vibration stress conditions. By keeping the rotor system in a “better” track and balance condition, overall vibration levels on all aircraft structural components and subsystems will be reduced. This could significantly increase the life of many of these aircraft structural components and subsystems. In particular, avionics systems could see a large improvement in life. This capability alone would positively impact several damage tolerance issues on aging aircraft.

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 database 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.

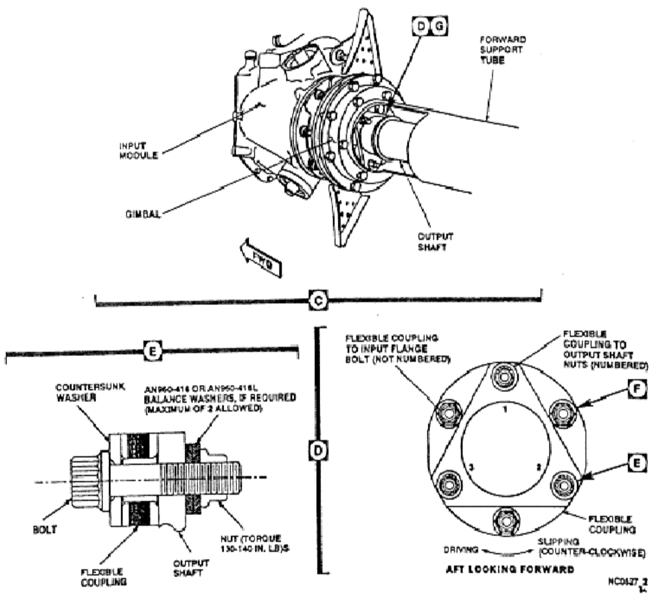


Fig. 5. High Speed Shaft Interface.

DETECTED FAULTS

To obtain this vision of vibration based diagnostics and prognostics, several seeded fault tests have been run to correlate vibration signatures with certain failure conditions. Below is a description and brief discussion of these tests and results.

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 5) has been a problem area, where the difficult to inspect Thomas Coupling disc pack has suffered several failures. Other aircraft have suffered similar catastrophic failures. Figure 6 shows the engine high speed shaft (with cracked Thomas Couplings) that was removed from the fleet and tested at Trenton. The HIDS system detected the fault and isolated it to the starboard side. This provides a rationale for providing a cockpit alert for critical, rapidly degrading components.

Fig. 6. Cracked Thomas Coupling.

One particular test article was a Coast Guard HH-60J main transmission input module that emanates high vibrations at half of the gear mesh frequency. The Navy has encountered a few incidents of half-mesh input modules, where every other tooth of a semi-hunting mesh is highly loaded. Since both the pinion and gear have even numbers of teeth, wear occurs at a much faster rate. Moreover, aircraft with these half-mesh input modules have a history of rejecting engines because of power turbine shaft wear and resultant cockpit torque indication errors. The Coast Guard rejected one engine on the subject aircraft because of the torque indication problem. The cause of half-mesh anomaly is believed to be gear profile errors introduced in the machining process.

The objectives of the test were to exercise the input module in a highly controlled and instrumented environment to

- develop a reliable method for the Coast Guard to identify half mesh modules using their field vibration equipment,
- determine if Navy tests could be conducted at lower torque than the current 75% requirement, making the test compatible with shipboard operations,
- test a novel fix, indexing the pinion by one tooth thereby changing mating teeth in mesh, and
- return the asset to service if within acceptable vibration limits.

All of the objectives of the test were successfully accomplished, with the exception that the material condition of the test article precluded a return to service. Prior to initial test run, inspection of the mating gears via the input module inspection port revealed wear patterns and spalling, confirming high loading of every other tooth in the gear mesh (see figure 7). The degree of spalling was unexpected, and required the asset to be overhauled. However, the pinion was still indexed to determine whether vibrations at the half-mesh frequency could be brought to within acceptable limits. As exhibited in figure 8, the vibration was reduced well below the Navy limit of 0.15 IPS, and would have allowed the asset to be returned to service. Reliable limits were developed for Coast Guard field vibration equipment by comparing measurements from several vibration monitoring systems. The chart also shows that a low 40% torque, such as required for single-engine flat-pitch operation, provides similar detection capability as for higher torques. With the real-time monitoring capabilities of the Integrated Mechanical diagnostic System Health and Usage Monitoring System (IMD HUMS) about to enter the fleet, detection of aircraft system faults such as this half-mesh anomaly are automated and performed every flight, lowering operational costs and increasing safety.

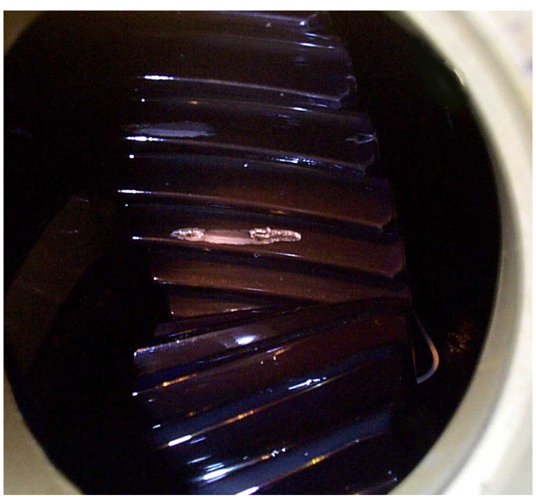


Fig. 7. Half Mesh on Coast Guard HH-60J.

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.

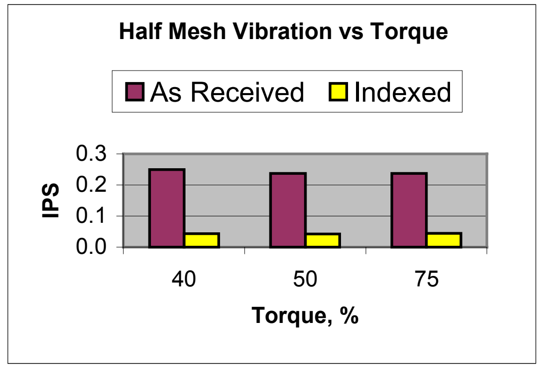


Figure 8: Results from Half Mesh Test

A means used in the helicopter community to promulgate this type of investigation (seeding a fault) is to weaken a gear 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 helped to determine optimum notch placement. Figure 9 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.

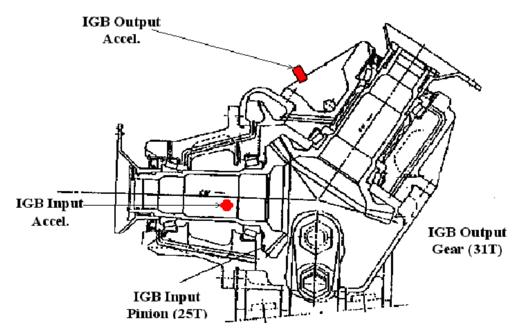


Fig. 9. SH-60 Intermediate Gearbox Cutaway.

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 10 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 large web crack extending to the bearing support diameter. No indication from the gearbox chip indicator was observed.

Fig. 10. Cracked Intermediate Gerabox Pinion.

A review of the diagnostic results shows the model-based algorithms successfully detect the presence of the gear tooth fault. After indicating a healthy gear for roughly 267 minutes (most acquisitions were 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. 2 discusses indicator results of another pinion tooth fault). Test results illustrated an EDM notched tooth behaves much like adjacent teeth until the part exhibits fatigue and a crack develops. The crack effectively weakens the tooth in bending, 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 11 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. 11. Main Transmission Input Pinion Crack.

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.

The HIDS system also attempted to 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. Actual "Component Condition" indicator, and two gear health indicators which determine the component condition have been identified in the analysis. Early warning of a local gear tooth anomaly is provided by Residual Kurtosis, and Residual Peak to Peak continuously elevates as the gear tooth crack propagates to a severe condition. These indicators could therefore be integrated into the diagnostics package as early warning and impending failure indicators respectively.

Diagnostic system sensitivity to defects and faults in tail drive shafts and bearings was also evaluated. 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 airframe 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 12 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. A fault would clearly exhibit itself by strong tones at frequencies specific to the inner race defect frequency and also at shaft speed. By comparison, fault-free hanger bearings would not generate bearing defect frequencies. Observing the spectral frequency the former was observed, further lending confidence to the algorithms.

Post test inspection of the bearing revealed that the inner ring was fractured as shown in Figure 13. 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.

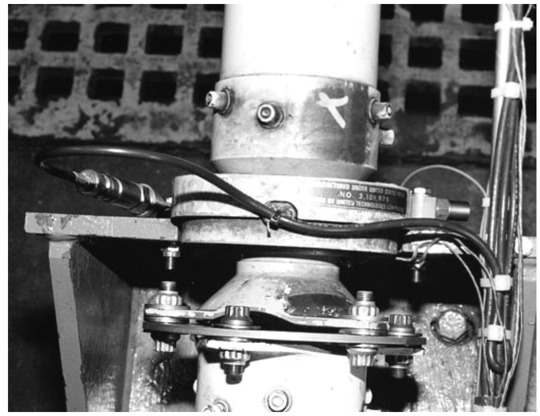


Fig. 12. Hanger Bearing Assembly.

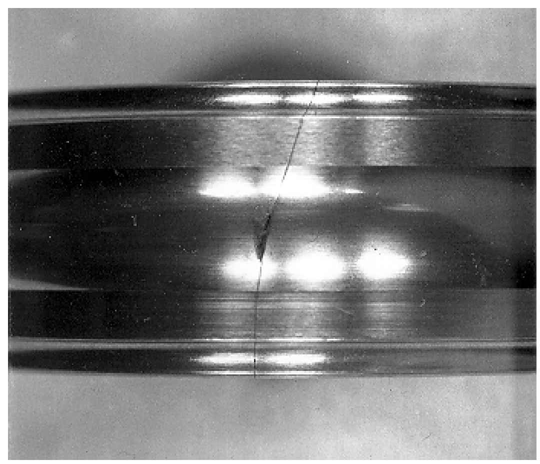


Fig. 13. Post-Test Condition of Hanger Bearing.

The diagnostic system sensitivity to bearing defects in gearboxes was also evaluated. 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, suggesting it would be difficult to detect. Figure 14 illustrates the SH-60 main transmission system and respective vibration accelerometer locations. The Figure 15 fleet rejected component was installed in the HTTF starboard location. The starboard main condition indicator toggles into the alarm position when the fault is implanted and reverts back to the okay position when the fault is removed. 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.

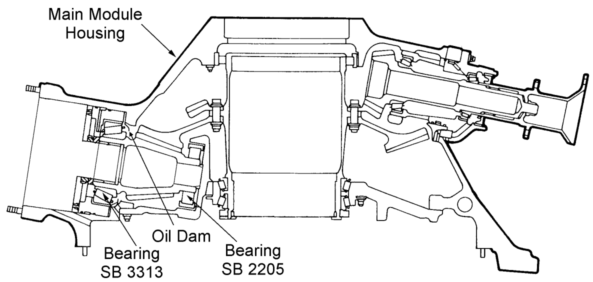


Fig. 14. Locations of SB-2205 and SB-3313 Bearings in the Main Module.

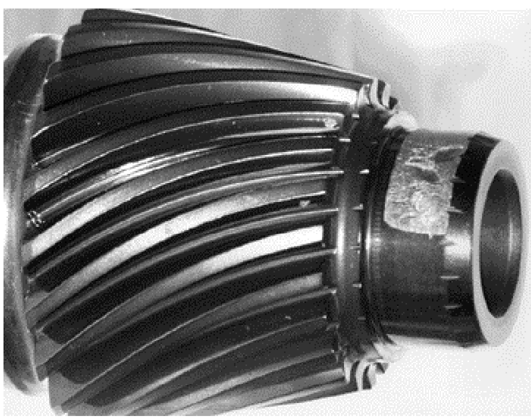


Fig. 15. Main Module Input Pinion with Spalled Integral Raceway Bearing SB 2205.

Prognostics could effectively be applied to the failure of this component. The SB2205 fault progresses in a repeatable manner from a small, localized spall into a larger one that will eventually encompass a good portion of the inner race diameter. At this point, the chip detector will provide an indication of a failure somewhere in the gearbox with no indication of fault location or severity. On the other hand, the model-based bearing indicators identify the presence of the fault early in this process. By carefully tracking the progression of this fault by utilizing the algorithmic indicators, maintenance and mission planning can be conducted in an effective manner, and unscheduled downtime can be effectively reduced.

Evaluation of variability of data across flight regimes (including torque and weight variations) was also conducted. There is considerable difference in the vibration 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.

Evaluation of sensor placement sensitivity for the various defects was performed. 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. In fact, the enveloped kurtosis of the stbd input sensor does not respond to the fault, whereas the port main sensor does. Based on results from this test, the port main sensor was then mapped as the secondary sensor for the stbd SB 2205 bearing.

This program also undertook an evaluation of 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. Other similar tests (some naturally occurring) were recorded. Gearbox gear pattern shim surveys were also performed. Test results are pending data review.

A seeded fault data library that can be used to evaluate systems in the future without repeating the test program was developed. 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 IMD HUMS initiative, the

HIDS data is being distributed to various institutions to develop and evaluate transmission planetary system gear and bearing algorithms.

As many currently available propulsion and power drive system diagnostic technologies as possible were evaluated in HTTF, and their relative measures of effectiveness were assessed. Engineering evaluation testing of Stress Wave Analysis, Electrostatic Engine Exhaust Monitoring, Inductive Oil Debris Monitoring, Quantitative Oil Debris Monitoring, Optical Oil Debris Monitoring, Laser Interferometer, 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 16). 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 15 main transmission input pinion with a spalled integral bearing raceway was used as a tool to generate debris for the evaluation. This test 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.

The HIDS program also undertook a comparison of 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 NAWCADPAX. 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 set 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. The fault 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.

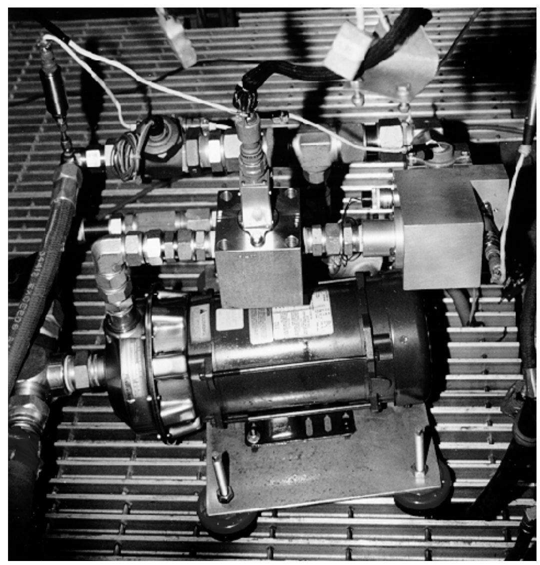


Fig. 16. Test Rig for Oil Monitoring Evaluation.

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 12 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 the HTTF for installation and continued testing in a test cell environment to provide a comparison to flight test 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 17. 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. This is an outstanding success story, and a testament to the work being performed under this program.

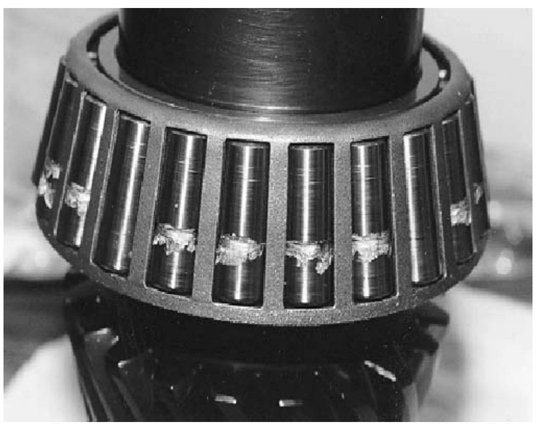


Fig. 17. SB 3313 Removed from PAX Aircraft.

Diagnostic results were categorized with respect to aircraft flight regime to define optimized system acquisition and processing requirements. A great deal of scatter was found in the value of the faulted bearing indicator. This is due to the differences in flight regime and torque. Discrete frequency excitation levels are a function of load, and a determination of what regimes produce satisfactory results is needed.

The diagnostics ability to reduce component “false removals” and trial and error maintenance practices was demonstrated. Several fleet removed components were tested and found to be fault free. Four hydraulic pumps removed for oil pressure problems were found to operate normally in the HTTF. 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.

Methods that reduce false alarms and improve component condition assessments were demonstrated. Numerous indicators have been developed to quantify health of the drive train components. Rather than use each of these indicators in isolation, utilizing data fusion can derive additional benefit. Previous multiple sensor data fusion techniques have had great success in fault detection and classification. An automated data fusion technique currently under investigation is Hotelling's T2 Multivariate Analysis. This technique combines multiple indicators into one composite indicator. The composite indicator has been shown to increase robustness of condition calls because it changes by orders of magnitude in the presence of a fault. In addition, a reduction in false alarm calls is produced by establishing tighter control limits by taking advantage of the underlying correlation among the indicators.

In order to select the indicators that produce a more robust response, a goodness of fit test is being employed to ensure that the assumption of normality is not being violated on baseline data. All indicators not falling within the multivariate normal distribution are dropped from consideration. A correlation study is performed to further select indicators with favorable relationships. The indicators showing the strongest change in correlation between fault and baseline data are used in the T2 analysis.

The advanced statistical quality control technique has been applied to the HTTF crack propagation data and compared to current component condition call indicators. A preliminary study produced good results and will be reported under a future NAWCAD report. This technique has provided a more robust classification of the fault with a large reduction in false alarm calls. Alternative methods exist which yield a more robust estimate of the in-control parameters and this would further decrease false alarm rates while preserving the responsiveness of the T2 analysis to faults.

APPLICATION TO AGING AIRCRAFT

Diagnostics and prognostics are applicable to all phases of an aircraft’s service life. As the aircraft ages, failures can be driven by multiple failure mechanisms. Experience has shown that improper maintenance, re-use of parts, rework tooling, build variability, exposure to new operating tempos and environments, and changing mission profiles can contribute to new failure modes. In addition, aircraft are often adapted to roles for which they were not originally designed, incurring increased gross weight requirements, and operated longer than originally planned.

Model-based approaches are particularly suited to address these problems as they monitor for deviations from what is considered normal without requiring extensive data collection and training efforts. That being said, seeded fault tests jump-start the maturity of the diagnostics and prognostics by providing real-world examples of common or safety critical failure modes in a controlled and highly instrumented environment. This fault database can then be used to set limits intelligently and reduce false alarms. This approach is preferred, when feasible, to maturing a system based on field failures.

Engine degradation models such as the PPI provide information on the engine gas path state of health. During the Gulf War, military engines suffered severe erosion as a result of the fine sand characteristic of the region. The model, if employed, would have highlighted increased rates of performance degradation, and provided an increased measure of safety by triggering an alert when performance had dropped below an acceptable level. The model also enables optimization of maintenance, as trending will allow forecasting of when maintenance will be required and provides for scheduling the maintenance versus aborting the mission.

CONCLUSION

Effective diagnostics and prognostics are essential to the operation of all aircraft types, and are key to increasing safety and reliability while reducing maintenance costs. As aircraft age, new failure mechanisms are discovered. Model-based diagnostics and prognostics approaches are particularly suited to address these problems as they arise, because they do not require extensive data collection and training efforts.

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

The Need for a Systems Engineering Approach for Measuring and Predicting the Degradation of Aging Systems and How It Can Be Achieved

William Robinson
Gisele Welch
Gary O'Neill
Georgia Tech Research Institute
Logistics and Maintenance Applied Research Center
925 Dalney St
Atlanta, Georgia 30332-0834
USA

1.0 Introduction

This paper will explore the need for a comprehensive approach to measuring, and predicting, degradations in aging NATO aircraft and use of these predictions in a 'systems approach' to solving the challenges faced in supporting these aircraft. Various groups within the NATO countries have already accomplished significant progress in this area, so this paper is an attempt to refine a more global process that will provide the most useful results in the least amount of time. We believe that the appropriate application of both emerging and seemingly unrelated technologies, coupled with a systems engineering management approach, may provide acceptable results.

2.0 Defining the Real Problem(s)

As stated in the theme for this meeting, the problem being addressed is aging aircraft and how to best minimize the effect this situation has on NATO countries. This is the ultimate, high visibility, problem to be solved. In reality, though, the 'Aging Aircraft Problem' is a series of smaller scale, inter-related issues. This reality demands that a 'systems approach' be used to formulate the specifics of the problem and define the successful path to resolve those issues.

The more complex problem facing NATO is how does a team of countries work together to develop a program that allows the inter-related problem elements to be solved in an effective manner that provides each country with measurable results in lowering the burden of repairing and maintaining an aging fleet. At the heart of this problem is the need to understand the degradation processes involved and the need to predict the future effects of degradation in a cohesive manner that provides effective insight to the potential solutions.

Managers in organizations, such as NATO, that are forced to deal with the problems associated with aging systems must not only focus on solutions; but also determine how to implement a process that will provide solutions in the manner which achieves a cost effective solution while maintaining necessary operational capability. Total success is unlikely. Optimizing the return (safety, availability, minimized operational costs, mission capability) on investment (funding, personnel, time, political posture) is the best that can be accomplished and should be the goal. The reality of the situation is that there is more "return" needed than there is

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

"investment" available. It is the manager's job to attempt to provide a balanced solution to optimize return on investment.

3.0 Systems Engineering

The development and use of the systems engineering process for military development programs, began in the mid-1950s on ballistic missile development programs, and expanded world wide through the 1960's. Programs that benefit the most from a systems engineering approach are typified by the following characteristics;

- Large teams are needed to develop the solution or system.
- Personnel resources are highly specialized.
- Many different organizations are involved.
- Participating Contractors and Government organizations are located throughout the country or world; making communications, coordination, and interfacing difficult..
- Many related problem elements are being solved concurrently.
- Operational and logistic support requirements are very complex.
- Time to develop a solution or product is constrained.
- Solutions are dependent on the successful transitioning of advanced technologies.

Systems engineering is both a technical process and a management process. Systems engineering is a methodology or process by which expert knowledge is applied to:

- Transform an operational need into a description of the system performance parameters (commonly known as requirements).
- Development of a system configuration that will achieve performance parameters.
- Integrate related parameters and insure compatibility of all physical, functional, and program interfaces in a manner that optimizes total system performance compliance.

Since most of this audience is made of individuals working directly in the aerospace industry or closely related areas, you will be or have been involved with the systems engineering approach for development of aircraft or aircraft sub-systems.

Traditionally, the systems engineering process has been applied to technical development programs in which a hardware or software system was being developed. However, we are proposing that this methodology might also be successfully applied to the development and optimization of the system, or model, for predicting the future effects of degradation. The characteristics of a solution to the NATO Aging Aircraft problem match the characteristics listed above and like most development programs these days, also involves international politics, budgetary constraints, and limited personnel resources. In short, this is a 'textbook example' of a challenge that would benefit from Systems Engineering Processes.

4.0 Basic Systems Engineering Process

Without a flexible, but rigorous approach to solving a complex problem, funds, time, and personnel can be wasted either by solving the "wrong" problem, developing an incomplete solution, or over-developing a good solution. Since the parameters that affect the problem definition are often dynamic in the real world, we need a process that is adaptable to changing requirements, yet structured in a way that minimizes lost effort. The systems engineering process uses the following structure:

(a) Define the requirements or needs that the solution should fulfill
  - Define end-user requirements (top-level global requirements)
  - Perform functional analysis to divide top-level requirements into smaller elements and to determine alternate means of achieving the top-level requirements.
  - Define the inter-relationship between the requirements, if possible.

(b) Develop concept designs or plans that will satisfy the requirements.

(c) Evaluate the proposed concepts and decide on most promising approach(s)
  - Perform trade studies to identify weaknesses and risks
  - Evaluate and optimize to eliminate weaknesses and minimize risks
  - Quantify compliance of concepts relative to top-level requirements
  - Chose 1-3 concepts to more fully develop

(d) Fully develop the concept(s) chosen in the previous step.

(e) Verify that the system or program meets the top-level requirements.

Steps (a) through (c) are iterative as shown in the diagram of Figure 1.

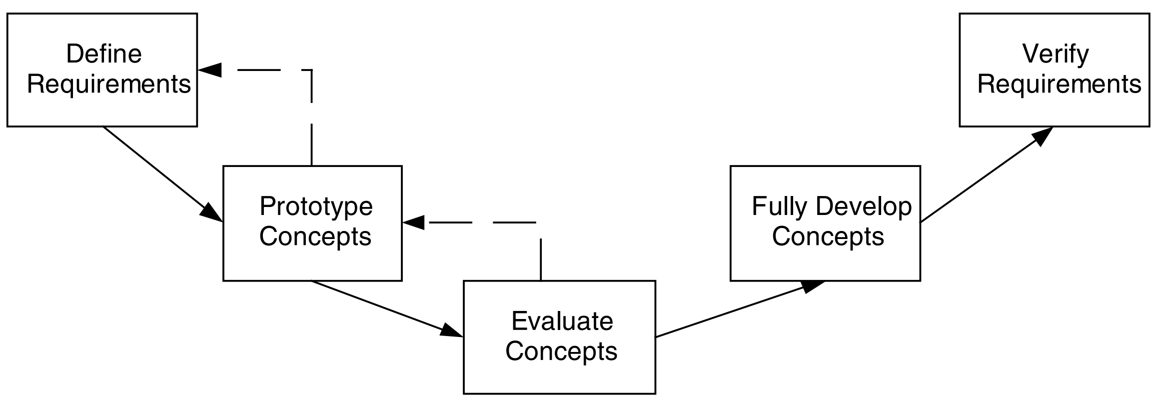


Figure 1. The Systems Engineering Approach

Most people and organizations developing new products and solving day-to-day problems use the above process, or a modified version of it, because it is a natural process to follow. What is sometimes lacking is a disciplined and systematic framework for quantifying and documenting the various steps, resulting in a less structured process that allows the results to be influenced by chance, limited or irrelevant knowledge and experience, intuition, or other factors.

5.0 Top-level Requirements

A rigorous systems engineering process will provide acceptable results in meeting technical requirements. We must also realize that there are other non-technical requirements that can be ultimately more important because they often decide the perception or degree of success or failure of the project by other stakeholders, such as the legislature or public opinion

As stated by others working on the aging aircraft problems over the past 5-7 years, examples of the top-level requirements that pertain to the aging aircraft problem are shown in Figure 2. We have added two requirements that are often missing from the requirements list: (a) time given to "solve" the problem and (b) funding available to "solve" the problem. These last two requirements are the most important for those managing the process that will develop a solution.

Top-Level 'System' Requirements
Maintain Flight Safety Extend Aircraft Service Life Maintain or Increase Aircraft Availability Maintain Mission Capability Reduce or Constrain Total Operating Cost Permit Sub-system Moderation Cost of Improvement Program Minimum Impact on Aircraft Availability

Figure 2. Top-level Aging Aircraft requirements

All requirements, whether top-level or those that are derived from the top-level requirements, must be verifiable either through analysis, test, or a combination. Examples of verifiable requirements might include: Accident rate of less than 1 aircraft loss per 100,000 flight hours: Availability greater than 90%; FMC rates above 90%; and growth in cost of ownership less than some baseline amount.

6.0 Modeling the Aging Aircraft System

The aging aircraft system can be modeled using a three tier modeling architecture, as shown schematically in Figures 3 and 4. The quantifiable outputs of the top-level model will be used to determine compliance with the top-level requirements discussed in Section 5.0.

The next lower level of model development would represent the sub-systems and components of the aircraft in terms of their contributions to the high level quantities. As examples, the second tier models might address; how engine failures affect the accident rate, availability and FMC rates; how does maintenance on the system affect availability, FMC, and ownership cost.

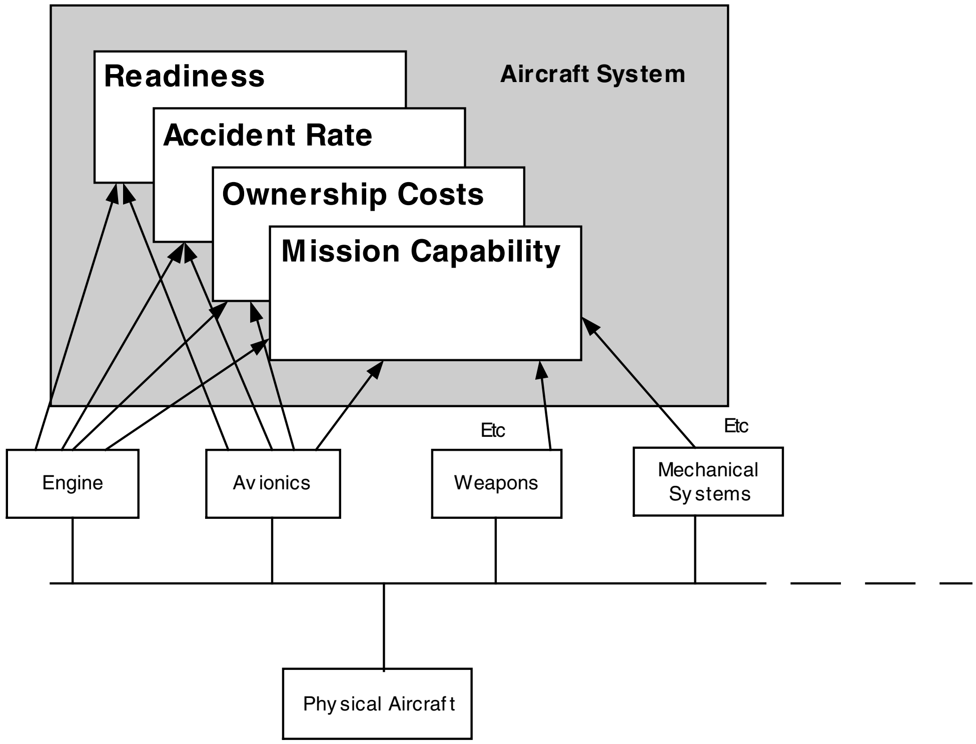


Figure 3: Aircraft System and Sub-System Models

The third level of this systems model would represent the effects of operation (including changes to original employment assumptions), age related damage mechanisms and other failure mechanisms (improper maintenance, non standard material, etc) and how these relate to the aircraft sub-systems and components

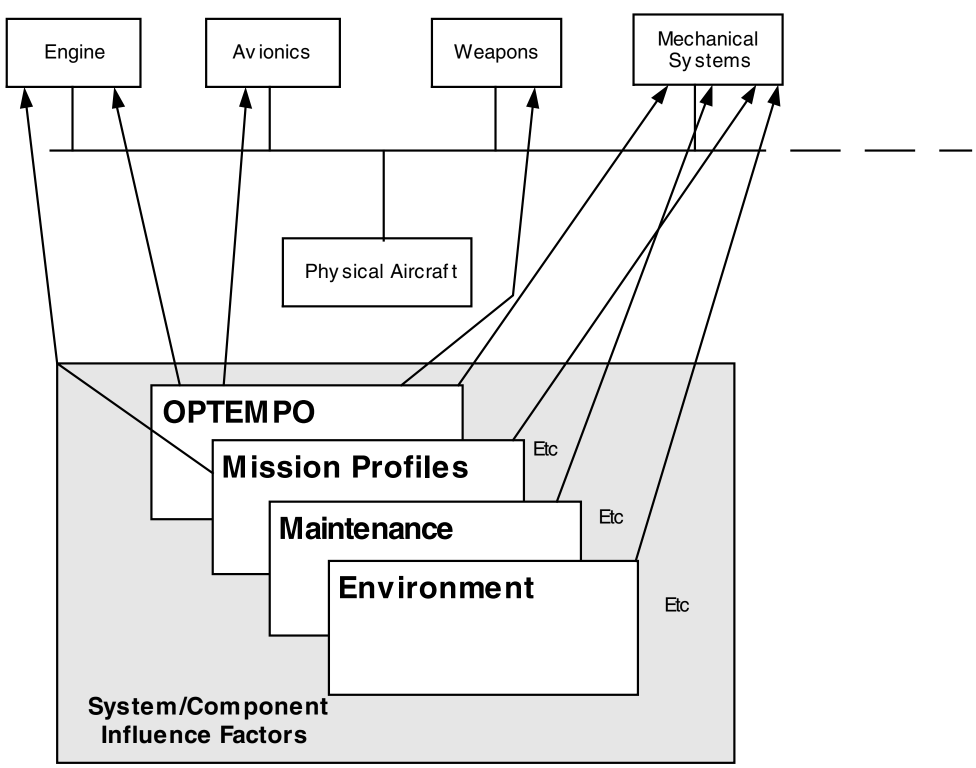


Figure 4: Tier Three Models

7.0 Functional Analysis

The 3-tiered model above will provide a powerful tool for analyzing the optimum ‘system’ configuration and where improvements need to be made in order to meet the top-level requirements. By using this architecture to flow requirements down to the third level and by flowing capabilities up to the top-level, an iterative process can be developed that will identify weak areas that need improvement. This type of modeling can also be used to perform sensitivity analysis to determine where the most return on investment can be realized, where technology insertion may have the biggest benefits, and to help identify and quantify risk.

The lowest tier of the model above is intended to describe damage mechanisms that degrade the system or component level operation of the aircraft. From these descriptions, a prediction of the effect on the aircraft as a total system can be forecast. From this forecast, the damage mechanisms that contribute to the greatest degradation of the aircraft as a system can be assessed. From that assessment, decisions as to what actions to take with respect to those mechanisms can be optimized: Just as with human health, the treatment for various factors attacking an aircraft can be balanced when taken together as a whole.

Once the individual models are developed and validated in each of the two lower tiers, the top-level model can be assembled and validated, by combining the elements according to their inter-relationships. After validation of the top-level model, it can be used with a range of statistically valid input parameters for the lower tiers that relate to the “real-world” and the range of results can be analyzed. From the analysis, a decision can be made, selecting the combination that achieves the best mix of desired outcomes. The result is a set of requirements, each of which has a quantifiable range of acceptable values.

From this set of balanced requirements, a set of concepts can be developed which address the requirements. For example, if reducing the amount of stress corrosion cracking is deemed necessary to raise the availability and lower the maintenance manhours of a particular aircraft, then several concepts which are focused on stress corrosion cracking could be developed and tested, with the most effective means chosen on the basis of defined metrics.

As the systems engineering process continues, the models developed during the functional analysis phase will become very important in quantifying and minimizing risk, and optimizing return on investment. The solutions chosen will be more credible and justifiable because they were obtained in a rigorous manner based on facts are quantifiable and were validated with knowledge and experience. Other important values that arise from the development of these models are: the ability to quickly review the effects of changes that may occur over time, and the ability to modify the overall design to meet the influences of a dynamic world.

8.0 Prediction of Damage Mechanism’s Impact on the Aircraft as a System

As we’ve discussed above, the proposed 3-tiered model contains the effects of damage mechanisms at its lowest tier. To provide meaningful results for the high level model of aircraft characteristics, the lower tiers must have comprehensive, high fidelity quantities serving as input parameters. High fidelity models are usually the fastest and least expensive tools for predicting future degradation rates under various influential parameters.

Therefore, the key to understanding the overall issues regarding aging aircraft is to first understand the lower level parameters. The first element is the present condition of the fleet. Next, we must analyze the rate of degradation under various realistic situations, and then assess the extent of the problem if the degradation is left unchecked. Once the problem is defined in terms of meaningful quantities versus time, concepts can be developed and implemented to slow-down or arrest the degradation. Modeling the degradation mechanisms also permits sensitivity analyses to be performed, which will demonstrate the parameters having the largest effect on degradation, guiding the selection of the parameters that need the highest priority for solution.

The two dominant areas that require the most attention for aging aircraft are structural degradation due to corrosion enhanced cracking and electrical power or signal wiring degradation, due to insulation/shielding failure and conductor open-circuit failure. Both of these problems are expensive and time consuming to fix. So it is important to understand the present extent of degradation and apply modeling techniques to determine the rate of degradation.

To determine the best modeling approach for degradation prediction of NATO aircraft affected by structural corrosion damage, the following process should be considered:

- Develop database that includes NATO inventory information.
- Add historical corrosion information to database.
- Develop "Repair Priorities Algorithm" to determine initial priorities for repair,
- Add information on corrosion measurement techniques to database.
- Perform corrosion degradation measurements to establish corrosion baseline, and add corrosion measurement data to database.
- Using corrosion data, develop a "Corrosion Degradation Model" to predict future corrosion related degradation.
- Update corrosion prediction algorithm to determine optimum measurements to use on highest priority aircraft and locations.
- Use predictions to help modify/define top-level requirements, funding efforts, and future plans.

This process is dynamic and may require iteration as new knowledge is gained. This process is shown graphically in Figure 5.

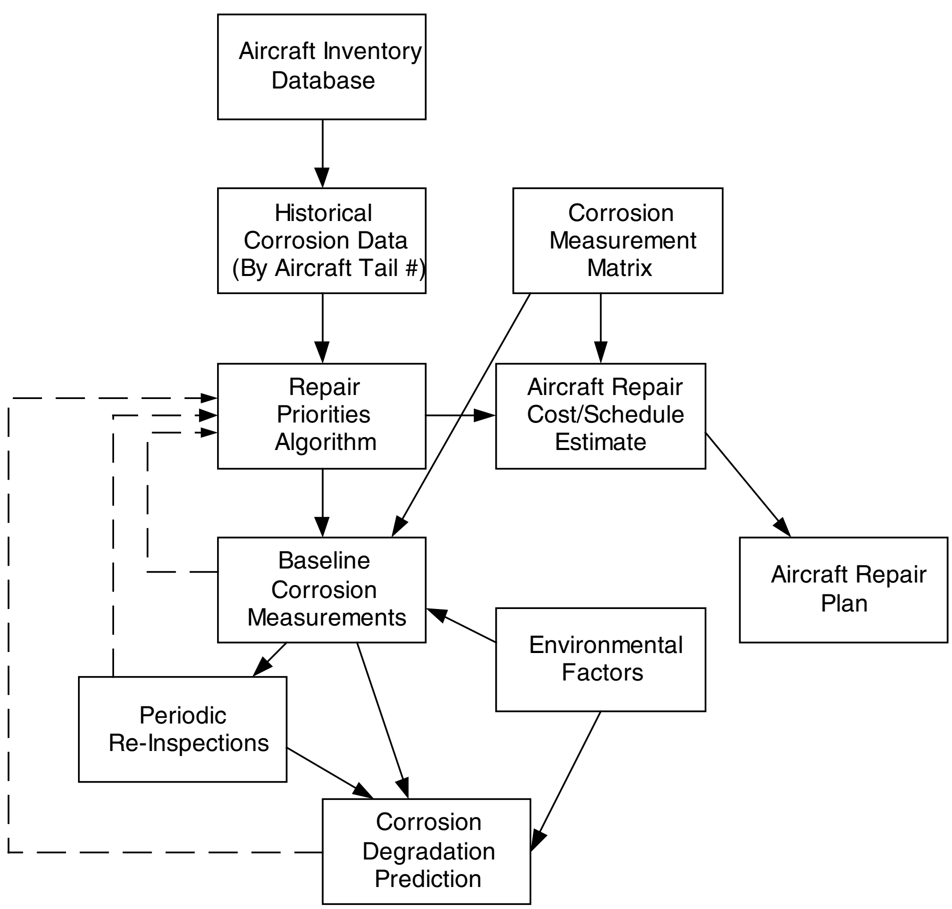


Figure 5. Measurement and Prediction Process

Each block in the diagram will be discussed in the sections that follow.

9.0 NATO Inventory Information

First, a database containing the various types of NATO aircraft must be established. The database should contain the number of each type of aircraft in service, aircraft manufacturer, countries owning the aircraft, and any other information that might be pertinent to later decisions regarding which aircraft should be repaired and the repair timeline. Such a database probably already exists in some form, but probably does not contain all the needed information to develop a useful algorithm to determine optimum deployment of resources for the task of repairing structural corrosion damage.

It is also important to begin establishing realistic funding profiles for repair of each type of aircraft. While current funding profiles may have been previously generated based on incorrect or non-applicable assumptions, starting with these projected funding amounts and timelines is helpful as an initial baseline. As the rest of the process is completed and iterated, each nation and NATO as a whole may see how changes and redistribution in the funding might produce better overall results.

10.0 Corrosion History Information

Previous inspection and repair records, other relevant maintenance history, structural susceptibility information, and environmental data, along with any other structural or corrosion related information that might add to the knowledge base should be added to the database for each tail number. Just as previous funding timelines are useful as a baseline, past technical information may also be of some help in establishing a baseline of the condition of the aircraft, even if it contains some small fraction of incorrect data.

Care must be taken to strive for consistency of format and accuracy for this information. The existing data may be in different formats for the same types of aircraft and from nation to nation, or vary from aircraft type to type. The amount and usefulness of the data to an overall model must be carefully assessed. Extreme caution should be used when trying to rely on previous data to indicate the degree of corrosion degradation or extrapolating the rate of degradation. The cost of obtaining and reformatting this data should be analyzed, as it may be too expensive to make it worthwhile in some limited cases.

11.0 Repair Priorities Algorithm

Using the information in the database along with expert knowledge, an algorithm can be developed to help determine the priorities with regard to which aircraft and which structural damage type should be given highest priority. This algorithm will be called the Repair Priorities Algorithm. The algorithm developed for this analysis must take into consideration all of the important aspects of the issues previously stated in Figure 2. In addition, the algorithm must take into consideration the time line of degradation, estimated repair cost and schedule, and overall funding profiles.

12.0 Environmental Factors

The environment plays a key role in all aging aircraft related damage mechanisms. Duration of exposure to high humidity salt air, sand, heat and corrosive gases are examples of factors that influence the aging process. Exposure histories of individual aircraft, as well as algorithms based on fleetwide experience are essential to damage prediction.

13.0 Corrosion Measurement Techniques

Many different types of corrosion detection and measurement techniques have been developed over the last 5-10 years. Many of the techniques used today were originally developed for detecting and measuring other types of structural flaws and have been modified to address the peculiarities of corrosion and corrosion-induced failures. No technique works well in all situations; but for almost every measurement condition, an accepted technique is optimum. Researchers in the various NATO countries are developing new measurement techniques and these new techniques need to be reviewed, tested, and compared with the older techniques and with future measurement objectives. Not only is measurement accuracy important, low false data rate, speed of measurement, and costs of measurement are important factors in the overall model.

Once the aircraft types and corrosion problem areas have been prioritized, a matrix of measurement techniques applicable to each type of aircraft and problem area can be created and added to the database. The measurement information should include effectiveness, performance time, and cost metrics that can be used to determine the most effective measurement techniques to use for each problem area. Adding this measurement information to the database will guide the choice of the best corrosion measurement process to align with the results of the Repair Priorities Algorithm. This combination will provide useful cost data for the measurement process needed to support the plans that result from the Repair Priorities Algorithm.

14.0 Corrosion Degradation Measurements

Having derived the most pressing corrosion priorities and the best measurement for assessment of system degradation, a plan to develop fleet baseline data and periodic updates of degradation can be developed with participation, at some level, by all NATO countries. In addition to serving as a technical measurement guide, the plan is useful as a management tool. The measurement plan should include a recommended data formats, calibration techniques, and other technical information needed to produce data that is both technically useful, consistent, and tailored for improving the Repair Priorities Algorithm.

As new corrosion measurements are taken, the data should be periodically added to the database. Management of the data and database should be addressed in the measurement plan. The plan should address what data should be taken, the format of the input data, where it will reside, who will input data to the database, how data will be input, who will be responsible for management of the database, and how the database management will get funded.

Management of the database, while important, is not the primary goal here and should not be an excuse to generate a bureaucracy. If designed properly, it should be easy to input data, easy to manage the data, and user friendly for both the field personnel collecting the data and the algorithm developers using the data.

15.0 Initial Model to Predict Corrosion Related Degradation

As the new data is added, the Repair Priorities Algorithm should also be periodically updated and re-calculated to confirm or modify the measurement approach, measurement funding priorities, and the usefulness of the data being captured.

In addition to updating the Repair Priorities Algorithm, a new model should be developed that takes into consideration the baseline corrosion degradation and predicts how the corrosion and related structural problems will change with time. We will call this new model the “NATO Corrosion Degradation Model”. A number of people/organizations are working on similar models or parts of this model. However, to my knowledge, no one has tailored a corrosion degradation model to the NATO aging aircraft fleet.

Among the model’s input parameters will be the corrosion degradation measurements being made under the measurement plan, effectiveness of repair actions, effectiveness of operational changes, and effects due to changes in mission. The model will allow the user to determine how the level and rate of degradation is affected by varying the input parameters, thus allowing the parameters to be optimized and allowing sensitivity analysis to be made for the various input parameters.

As a side note, one of the important technology areas that has been pursued in recent years is to develop techniques and processes to provide accelerated corrosion degradation under controlled conditions. These techniques may provide an important tool in the development of degradation models, if the techniques can be proven to be reliable.

16.0 Update Corrosion Prediction Model

Just as with the Repair Priorities Algorithm, the NATO Corrosion Degradation Model should be reviewed and updated as more knowledge is gained about the corrosion process and how it is influenced by various parameters. Again, as with the measurement database management plan, management and upgrade of this model is an enabling goal, not the ultimate goal; which is to fix the problem. So, the model should be developed just to the level required to give the answers needed to fix the problem.

17.0 Wiring Degradation Modeling Process

A process similar to the one outlined above for corrosion should be constructed for the other dominant aging mechanism: aircraft wiring degradation. Beginning with the aircraft inventory database, historical data on wiring degradation and inspection techniques, a Wiring Improvement Algorithm can be developed to determine the best approach to solve the wiring issues on an aircraft type basis. Once the repair priorities are established, a measurement plan to assess the baseline condition of the aircraft can be updated with periodic re-inspections. From that process, a refined degradation prediction model can be developed, and updated in the same way as the corrosion prediction model.

It may be useful to develop both prediction processes in parallel, to minimize the out of service time of the aircraft during baseline inspections and repair.

18.0 Developing the Concepts

Once models are in place and validated, various concepts (designs) can be developed to satisfy the requirements. Once a concept has enough structure and definition, it can be modeled and evaluated using the same process that developed the overall model. Trade studies can be made during this phase, risks identified, and risk assessments made. Evaluation of the concepts will indicate if the requirements can be met, or the existence of any shortfalls. . If the concept produces results below the requirement, the concept can perhaps be modified to increase the capability. If the evaluations show more capability than needed to satisfy a requirement, perhaps the concept can be modified to decrease its capability by decreasing cost, time, personnel resources, or other factors.

When all of the concepts have been evaluated and the degree of compliance established for each concept, a decision can be made on how to proceed with further concept development. Depending on funding, time availability, and other issues, more than one concept may be taken to the next level of analysis or implementation. It is unlikely that all concepts will meet all the requirements equally. Usually, the decision to proceed with a concept is straightforward. In many cases only one concept plan should be developed. If none of the concept plans meet the requirements, the project should be dropped, the requirements re-analyzed, or more clever people should be employed.

19.0 Development of the Plan(s)

After selecting the best concept(s), based on quantifiable requirements and capabilities, a plan of action and milestones to develop and implement testing of the concept(s) can be written to whatever level is required for successful proof. As with any complex plan or program, the work should be reviewed periodically to insure all resources are being applied to completing the plan(s) and that all elements of the team are working toward the same goals.

In any dynamic environment, the plan(s) may require modification as requirements change, as resources change, or as second-order problems arise during detailed development. While these changes are a nuisance and sometimes frustrating to deal with, the previous modeling of requirements and capabilities will at least allow the program managers to understand the effects of the changes and how to optimize the outcome. These tools will often allow the program managers to better justify requests why more funding might be needed, the impact of potential funding cuts or program time slips.

20.0 Validation of the Final Concept Implementation Plan

Prior to full implementation of the proven concept, a “sanity check” should be made on the plan to insure it agrees with past experience and knowledge. Management should review the plan. This would be the equivalent of a Critical Design Review (CDR) for equipment or software development programs.

21.0 Summary

This paper has attempted to show how a systems engineering methodology can help program managers from different NATO nations work together to develop measurement and prediction models that can be used to optimize financial and personnel resources in the quest to satisfy adequate operating requirements for the aging NATO fleet. The object of this approach is to keep management focused and coordinated on the end goals (top level requirements) and the process that will optimize the trip from the present situation to attainment of the goals.

It must be stressed that the systems engineering methodology provides tools that will allow reasonable requirements to be defined in a verifiable, quantifiable manner. The authors believe that it is far worse to have requirements that are too stringent, than to have requirements that are slightly lax. Requirements that have been arbitrarily set too high due to lack of knowledge or lack of test data will waste financial and personnel resources and this waste can never be recovered. Any safety margins or “padding” put into a requirement must be based on variances that can be proven as a result of test results or rigorous analysis.

F-111 Sole Operator Program: Maintaining the Structural Integrity of an Ageing Fighter Aircraft

Kevin Watters, Francis Rose, Manfred Heller, Paul Callus
Defence Science & Technology Organisation
Aeronautical and Maritime Research Laboratory
GPO Box 4331
Melbourne, Victoria, Australia 3001

Bill Madley
Aerostructures Australia
Level 14/222 Kingsway
South Melbourne, Victoria, Australia 3205

Paul Connor
Royal Australian Air Force
ASI-DGTA
Directorate General Technical Airworthiness
B-2-South
RAAF Williams
Laverton, Victoria, Australia 3027

Introduction

The F-111 is a dual-seat, supersonic, variable-geometry-wing strike aircraft that has enjoyed RAAF and USAF service for the last 30 years. Approximately 550 aircraft were built between 1964 and 1973, with model variants from F-111A to F-111F. RAAF operations commenced in 1973 with the purchase of 24 F-111C aircraft and are planned to continue to 2020. Due to the age of the fleet and its changed circumstances following the retirement of the USAF F-111 fleets, a special program was initiated by the RAAF [1], to step back and take a fresh, holistic look at its F-111 aircraft structural integrity (ASI) management. It was titled the 'F-111 Sole Operator Program' and the aim was to ensure that all the required capabilities and data were in place to support the structural integrity of the aircraft to 2020.

The Sole Operator Program (SOP) sits in a framework of broader RAAF initiatives to ensure the supportability of all F-111 systems to the planned withdrawal date. This paper describes both the F-111 SOP and the broader RAAF initiatives on F-111 structural integrity. It is a comprehensive program tailored to management by safety-by-inspection, and encompasses identification of all potential structural degradation sites and all relevant degradation mechanisms, analysis of fatigue and corrosion growth rates, developing and characterising NDI techniques, and developing repairs, modifications and part substitution technologies. Corrosion prevention and control is also a key element of the program.

The F-111 SOP includes extending existing capabilities through applied research to address particular F-111 ASI issues associated with its special materials such as D6ac steel, 7079 T651 aluminium alloy and bonded aluminium honeycomb panels. Two highlights of this research (viz. rework shape optimisation and fibre composite replacement panels) are presented here, following an overview of the overall SOP.

RAAF F-111 Fleet

The RAAF F-111 fleet currently comprises 13 F-111C, 4 RF-111C (C models modified for the reconnaissance role), 4 F-111A (attrition replacements) and 14 F-111G (used ex-USAF aircraft purchased to extend the fleet). The F-111 role in RAAF service is strategic strike and reconnaissance. With near 30 years of operations completed, most RAAF aircraft have

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

completed between 5,000 and 7,500 AFHRS[1]. On withdrawal, RAAF aircraft will have experienced almost 50 years service and 10,000 AFHRS. The F-111 has enjoyed an excellent safety record in RAAF service and is highly regarded with respect to its military capability and flexibility.

Specifically, the F-111 is managed on a Safety by Inspection (SBI) basis, which is underpinned by the Durability and Damage Tolerance philosophy. This is implemented through the completion of Durability and Damage Tolerance Analyses (DADTAs) for the critical structural locations, or DADTA Items (DIs). The RAAF has implemented a dedicated DADTA program accounting for the structural differences and based on RAAF F-111 usage.

Capability Options Study

In the mid 1990s the capability planning area of the RAAF carried out a study of F-111 supportability and replacement options. The costs of supporting all F-111 systems were accounted. The study concluded that the continued operation of the F-111 was clearly the best option to sustain at a reasonable cost the strike capability for Australia's defence.

Impact of US Fleet Retirement

The RAAF relied heavily on the USAF and the OEM (Lockheed Martin, LMAero, Fort Worth) for F-111 ASI support. The USAF had significantly larger fleets of F-111s and their fleet leaders were generally at higher hours (~ 7,500 AFHRS) than the RAAF fleet leaders (~ 6,000 AFHRS, before the G model acquisition). The USAF generally found the problems first and developed engineering solutions in conjunction with the OEM, which were made available to the RAAF. The RAAF also directly tasked the OEM for support on RAAF-specific F-111 ASI problems. A good example of this is the cold work modification of the overwing longerons which was developed by LMAero for the USAF and is currently being applied to the RAAF fleet.

With the withdrawal of the last USAF F-111 fleet in 1998, the RAAF became the sole operator of the aircraft type and it would not be long before the RAAF fleet leaders overtook where the USAF fleet leaders had finished. It was also recognised to be not viable for the small RAAF fleet to sustain the full OEM support for the F-111.

Another significant impact of the USAF fleet retirement was the closure of their F-111 maintenance facilities that the RAAF previously had access to. These included the cold proof load test[2] (CPLT) facility and the bonded panel repair and rebuild facility. In short, with USAF involvement, the RAAF was able to assure airworthiness and good aircraft availability without large investments in support infrastructure.

Structural Management Initiatives

Spares Purchase

With the winding back of OEM support, the RAAF completed a dedicated program to identify LOT spares requirements, and to make purchases where continued supplies could not be guaranteed. The purchases not only included specific manufacture runs where necessary, but also included the purchase of ex-USAF spares inventories. Further, with many serviceable USAF aircraft retired, a number of aircraft and components have been earmarked as future spares sources.

---

[1] Actual flight hours

[2] Cold proof loading test (CPLT) is a periodic proof loading testing program performed on the F-111 structure in a special facility to confirm the absence of flaws above a small critical size. This then clears the aircraft for a further period of safe flight. In a CPLT the aircraft is first cooled to –40 ºF (–40 ºC) to embrittle the D6ac steel structure and then load cycles of –2.4g and +7.33g at 56º wing sweep angle, and –3.0 g and +7.33 g at 26º wing sweep angle, are applied.

Data Study

To continue in the Sole Operator environment, the RAAF identified early that a significant increase in Australian data holdings would be required. A dedicated data study team was established, with the team tasked to retrieve all appropriate data holdings from both the USAF and OEM. This is a substantial task that aims to provide a basis for the required indigenous capability to support all the aircraft systems, not just the structural aspects.

Sole Operator Program

The RAAF Aircraft Structural Integrity section (ASI-DGTA) identified that to bridge the significant gaps between the current ASI Program and the mature ASIP required in the sole operator environment, a dedicated program was required. The Sole Operator Program (SOP), which is the focus of this paper, has been established to achieve this. The SOP was established by critically reviewing all the elements of an ASIP, identifying the shortfalls in the current RAAF F-111 ASIP and forming appropriate tasks to address these shortfalls. Importantly the SOP recognises the important roles of the RAAF, DSTO and Australian industry in the ASIP.

CPLT Facility

An integral component of the RAAF F-111 SBI program is the completion of a CPLT on each aircraft at 2000 AFHR intervals. The completion of CPLT was identified as a continuing requirement through to PWD. As such, a dedicated CPLT facility has been established at RAAF Amberley to replace the USAF facility to which the RAAF previously had access. This is new CPLT facility at a cost of A$30m was commissioned in July 2001.

Bonded Panels Repair

During USAF F-111 operations, all overhauls of RAAF F-111 bonded panels were completed by the USAF at Sacramento. With the USAF F-111 retirement, the dedicated bonded panel facility has also been withdrawn, with effect 1999. Accordingly, the RAAF initiated a program to purchase and overhaul a significant number of bonded panels prior to the facility closure. This overhaul program combined with the additional supply of panels from stored USAF aircraft is aimed to address the long term requirements; however, there is a real concern that this supply will still not be sufficient to achieve PWD for all panels.

Key Structural Issues

D6ac Steel

D6ac steel is used in the highly loaded areas in the F-111 airframe. These include the wing pivot fittings, the wing carry-through box, the overwing longerons and some key fuselage frames. The material has a short critical crack length under the F-111 loads and is highly susceptible to corrosion causing pitting. It is inspected using magnetic rubber inspection (MRI) for known and inspectable crack sites and using CPLT for uninspectable sites and general global inspection.

There have been many instances of fatigue cracks detected in D6ac steel structure in RAAF and USAF service and a number of structural failures have occurred during CPLT. Key issues for D6ac steel part management are modifying known problem areas to achieve adequate durability and relieve the MRI inspection maintenance burden, making conservative allowance for crack initiation from corrosion pits, and setting the CPLT inspection interval.

Bonded Panels

Bonded panels comprise most of the internal and external panel structure of the fuselage as well as the horizontal stabilators and other control surfaces. They consist of inner and outer aluminium alloy skins with aluminium honeycomb bonded between them. The panels suffer from corrosion and disbonding, largely due to moisture ingress. Their maintenance is very expensive and one of the major contributors to the overall F-111 maintenance cost. Key issues for bonded panels are NDI, damage significance at an individual panel level and at a global structural level (multi element damage), repair limits, repair and rebuild processes and facilities and spares and replacement options.

7079 Aluminium

Most of the F-111 forward fuselage frame, beam and longeron parts are made of 7079 T651 aluminium alloy. This material is highly susceptible to stress corrosion cracking (SCC), and widespread SCC has been detected in RAAF and USAF aircraft. The key issues for SCC are prevention and control, structural significance, multi site damage, difficult access and repair complexity.

Long Wing Fatigue Substantiation

The fatigue substantiation of the RAAF long-wing models of the F-111 is not adequate. The manufacturer's tests comprised a fighter spectrum applied to a short-wing model and a benign bomber spectrum applied to a long-wing model. The RAAF requirement is more like a fighter spectrum applied to a long wing model. In addition, the manufacturer's tests did not include CPLT loading which has a profound effect on fatigue performance. For these reasons, critical service cracks which have already occurred in the wing pivot fitting and at an outer wing station, were not adequately revealed in the manufacturer's tests. The concern is that there may be other such locations in the wing which may come to light in the next 20 years service.

There is some (but less) concern for the fatigue substantiation of the fuselage. The known fatigue problems in the fuselage are in D6ac steel parts and the strategy is to improve their management to ensure safety and durability and save maintenance costs. There is more redundancy and fail safety in the fuselage structure and the highly loaded D6ac steel parts are monitored by the CPLT. The key degradation mechanisms for the fuselage are anticipated to be environmental (corrosion, SCC and bonded panels degradation).

Ageing Aircraft

The F-111 can be considered to be an aging aircraft. It's calendar age is currently 30 years and will be 50 years at retirement. It's original design life was 4,000 hours, although the manufacturer's testing to 40,000 hours provided a fatigue substantiation of 10,000 hours life. The RAAF fleet will be approaching 10,000 hours at the time of retirement. Typical aging aircraft conditions such as widespread fatigue damage may not be relevant to the F-111, but nonetheless it is appropriate to audit the fleet condition to identify any relevant concerns.

Maintenance

Maintenance is a key issue for F-111 fleet viability. The F-111 incurs twice the hourly cost to operate as the next most expensive RAAF aircraft (the F/A-18). Cost of maintenance is the key operating cost driver, and structural maintenance is a significant proportion of it. Aside from cost, maintenance issues also have a big impact on the operational availability of a small fleet. Containing maintenance costs and times, as the aircraft ages over the next 20 years of service, is a vital requirement.

RAAF ASIP

The ASI Sole Operator Program has its roots in the fundamental principles behind Structural Integrity Management in the RAAF. These principles have been prepared from a holistic approach, which means that they are equally applicable to shared and sole operator environments. As these principles form the framework on which the requirements of the ASI Sole Operator Program is based, an initial examination of them is required.

The broad objective of RAAF ASI management is to enable air operations to be conducted within an acceptable level of risk of structural failure of aircraft to their planned withdrawal date (PWD). The attainment of the broad objected requires the following outcomes be achieved:

- constraint of risk of structural failure of the aircraft to an acceptable level,
- achievement of planned rates of aircraft availability,
- avoidance of the unforecast cost of refurbishment, and
- achievement of the PWD.

The desired outcomes provide a spring board for the generation of strategies through which these outcomes are achieved. These strategies must reflect a life cycle approach, with their application tailored to meet the requirements of the initial concept phase, acquisition phase, and in-service phase.

These strategies are implemented through the ASIP, which identifies relevant agencies, activities and resources necessary for the achievement of the RAAF airworthiness objective. From these strategies a list of ASIP management requirements fall out. These requirements represent a framework about which the ASIP is built up. These requirements are summarised as follows:

Design Support Network	design documentation
fatigue testing and analysis (including follow on testing)	structural life assessment
usage monitoring	operational loads monitoring
structural condition monitoring	structural teardown program
configuration control (structural repair and inspection records)	structural degradation and control program
management of aging aircraft	

The F-111 Sole Operator Program ensures that the F-111 ASIP satisfies the ASI management requirements such that the nominated outcomes can be achieved.

Sole Operator Program

Participants

The engineering support originally provided by the OEM (LMAero) had to be found through different organisations, preferably in-country. This was accomplished by preserving the LMAero capability in the short term (three to five years), and using that time to transition the OEM technology and engineering capability to an in-country Design Support Network (DSN) consisting of the RAAF, DSTO and industry (Aerostructures).

LMAero Tasks

At the outset of the Sole Operator Program it was decided that the best way to develop the capability in Australia that was then solely resident in the OEM (LMAero), was to send people to the US to work on F-111 tasks with the OEM staff. Seven engineers from DSTO and Aerostructures were attached to the US in stages to work on the five tasks listed below.

Task 1 - Internal Loads Model
Task 2 - Increased Scope of RAAF DADTA Study
Task 3 - Structurally Significant Items (SSI)
Task 4 - METLIFE[3] Extension Work
Task 5 - F-111C Load Equations Review

The five LMAero tasks were completed in 2000 and delivered very useful engineering reports, models and data, in addition to the capability development. Also, the LMAero DADTA software was made available to Australia for ongoing use on F-111. LMAero displayed excellent corporate citizenship in helping Australia to make this transition.

[3] METLIFE is a Lockheed Martin proprietary crack growth analysis code.

Fleet Condition Audit

In early 2000, a thorough review of the current condition of the fleet was undertaken and provided the baseline upon which improved estimates of the projected fleet condition can be made. The fleet condition audit included the collection of maintenance data and the analysis of that data with the purpose of establishing trends. Condition data was obtained from various maintenance records from 1977 to 1999 with a total of 8507 documents included in the audit, involving 875 part numbers, and leading to 7700 entries into the database. The significant damage types on the RAAF F-111 fleet were cracking (28%), corrosion (11%), disbonds (24%) and mechanical damage (19%), and 21% of all defects were listed as having multi-site damage. The condition audit database was subsequently integrated into the F-111 Structures Information System database.

Corrosion Characterisation, Prevention & Control

An important element of the F-111 SOP is research to characterise the corrosion environment of the F-111 and the locations and rate at which corrosion can be expected to develop in its airframe. Environment sensors have been placed at the flight line and in the maintenance hangar to measure the external environment, and in equipment bays in the aircraft to measure the internal environment. Eighteen months of external environment data has been collected, but the internal environment monitoring has just started.

A laboratory test program on corrosion is in progress. It has three main areas of investigation: rate of corrosion pit development in D6ac steel with simulated imperfections in coatings; hydrogen embrittlement characterisation in D6ac steel (due to brush cadmium plating or corrosion); and use of washing detergents and corrosion preventive compounds for prevention and control of stress corrosion cracking in 7079 aluminium alloy.

Databases

CDRMS Database

The RAAF have developed the Configuration Data Recovery and Management System (CDRMS) database to store all known F-111 engineering data. F-111 engineering data collected from organisations in the United States by the RAAF Data Study Team is being scanned and stored in the CDRMS. Every piece of data is indexed against configuration item (CI), which is the lowest level at which the F-111 structure is managed (from an engineering point of view). Where applicable, information is linked to part number. CDRMS is a Microsoft Sequel database with web browser user interface.

F-111 Structures Information System

The F-111 Structures Information System database was developed by Aerostructures to enable parts/component, engineering and RAAF condition data on the F-111 structure to be stored and subsequently accessed for development of repairs, modifications and for other engineering investigations. The SIS database stores parts data (part name, part number, effectivity, material, etc), engineering data (design data, load/stress, fatigue data, etc) and condition data (damage type, location, time, part number, etc). The F-111 SIS is a Microsoft Sequel database with web browser user interface (Figure 1).

AMRL Teardown Database.

The F-111 Teardown Database was developed by AMRL to store the information obtained from examination of each part removed from the F-111 teardown aircraft. Information stored in the database includes a digital photograph of each part, all NDI reports and any other test results. This database provides the ability to pictorially drill down through the structure to identify specific parts.

The database uses a Microsoft Access database but is planned to migrate to Microsoft Sequel database. Cold fusion web server software was used for the web browser interface.

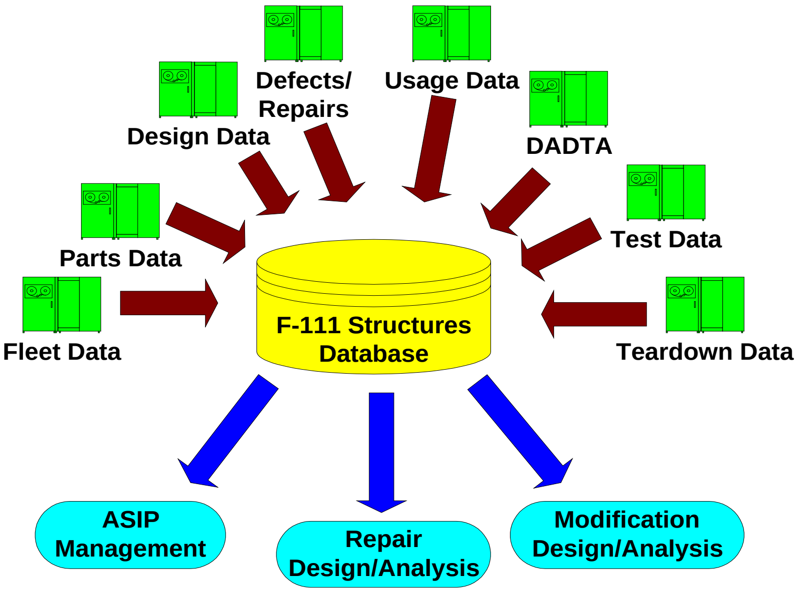


Figure 1: F-111 SIS Database

Future Direction

The CDRMS, F-111 SIS and the AMRL teardown databases have full part number identification to enable them to be cross-queried and easily linked (Figure 1). These databases provide extensive F-111 structures information, which will provide the basis for the continuing structural integrity management of the aircraft.

Fuselage Teardown

Following on from the fleet condition audit and the Significant Structural Items task at LMAero, the final leg of the strategy to identify all the critical locations in the F-111 structure was to conduct teardown inspections of a high-time fuselage and wing. A RAAF fuselage could not be spared so a retired USAF fuselage was acquired that was as similar as possible to the RAAF fuselages in configuration, usage and environmental exposure. The teardown of that fuselage is about 50% complete and involves the disassembly and visual inspection of all parts. The set of the parts deemed to be potential significant damage sites (largely based on the SSI task results) is then being subject to paint stripping, NDI (eddy current, ultrasonic, magnetic particle and X ray) and fractographic examination. The left/right symmetry of the aircraft is being used to limit the number of parts examined in detail.

The fuselage teardown involves a team of 12 fitters, technicians and fractographers and some professional supervision. It has been in progress for 18 months and will take a further 18 months to complete. Total cost is estimated to be of the order of A$5m.

Wing Test & Teardown

A RAAF wing was retired from service with 5,500 accrued flying hours and will be subjected to a full teardown inspection. Prior to the teardown, the wing is being subjected to a fatigue test to enhance any damage from service and ensure that all potential damage sites are revealed in the teardown. The total service plus test hours on the wing is currently 10,000. The test will be taken to at least 20,000 hours and possibly to 40,000 hours as time permits. The test is being done to a representative RAAF spectrum and includes the CPLT loading at the current RAAF interval.

External Loads

The external loads task at LMAero showed that the LMAero load equations for the F-111 were soundly based on flight test data to give accurate load components at a set of wing and fuselage reference stations. Follow-on work in DSTO is a re-processing of the LMAero flight test data to cover the full RAAF F-111 flight envelope. Also, detailed pressure distributions are being developed by CFD analysis to produce a full-field picture of the external loads. Automated software has been developed to pull this loads information together as input to the FE internal loads model of the airframe.

Finite Element Modelling

Internal Loads Model

The deliverable from one of the LMAero tasks was an FE Internal Loads Model (ILM) of the F-111C airframe. The model comprises 305,000 elements and 1.8 million degrees of freedom and runs in NASTRAN. Unfortunately, it was found there was little strain data in existence to validate this model, so a full-scale strain survey is to be conducted on a RAAF F-111 in the CPLT facility. The test aircraft is currently being instrumented with about 700 strain gauge elements. The ILM will be correlated with the strain data and adjusted as necessary. Then, in conjunction with the automated external loads input, it will comprise a virtual turnkey system to plug in a flight condition and determine the internal loads in the structure, which can be input as boundary conditions to a fine-grid sub-structure stress model.

The ILM will provide a key capability on which all future DADTA work and repair and modification design will be based. The complete external loads and calibrated ILM capability comes at a cost of about A$5m, but is deemed to be a very worthwhile investment.

Fine-Grid FE Models

Fine-grid FE models are currently being developed for a number of known problem D6ac steel parts in the F-111 fuselage. These are the 496 nacelle former frame, the 770 bulkhead and the overwing longerons. When loads for these models are available from the calibrated ILM, accurate stresses will be calculated in the parts for input to a fresh DADTA. Revised modification actions and inspection intervals should result in savings in future maintenance costs.

F1/F2 Fuel Tanks (Stress Corrosion Cracking)

A sub-structure FE model of the F1/F2 fuel tank area of the forward fuselage is under construction. This model will be used to evaluate the structural significance of widespread stress corrosion cracking. The fuselage teardown data and compiled data from USAF and RAAF maintenance records will be used to identify in detail all the SCC locations. Multi site damage scenarios will be investigated. Economic repair designs will be developed using this model in conjunction with local detailed FE stress models. It is envisaged that a major inspection and refurbishment of the F-111 forward fuselage will be undertaken as a one-off fix to the SCC problem. There is a large coupon test program in progress to characterise SCC in 7079 aluminium and develop prevention and control measures. These will be applied during the refurbishment program.

DADTA Capability

In the past, DADTAs of the F-111 have been performed by the OEM for the RAAF. A significant element of the Sole Operator Program has been to transfer that capability to Australia, firstly in DSTO and then transitioned to industry (Aerostructures). Two of the LMAero tasks were aimed at DADTA capability transfer, and the LMAero DADTA software has been made available to Australia for ongoing use on F-111.

In addition to acquiring the LMAero DADTA capability, DSTO has worked collaboratively with LMAero to further develop the capability. Research areas have been multi site crack initiation and coalescence and crack growth through notch plasticity fields, and the results have been coded into the LMAero DADTA software.

Corrosion Modelling

Aside from the SCC problem, the main corrosion concern for the F-111 is for pitting corrosion to initiate fatigue cracks, and D6ac steel is the material of most concern. The experimental work to characterise the rate of corrosion pit development has been mentioned. Another large experimental program is to establish the equivalent crack size (ECS) of a corrosion pit in D6ac steel. Corrosion pits of various depths were induced in test coupons which were then subjected to both constant amplitude and representative spectrum fatigue loading to a range of stress levels. The corrosion pits were also measured microscopically. Good results have been obtained from this work to establish a correlation between a corrosion pit morphology metric (depth × aspect ratio) and ECS. The next stage is to integrate the corrosion pitting rates and ECS into a crack growth analysis to develop a DADTA covering the scenario of fatigue induced by corrosion.

Bonded Panels

The main issues with bonded panels are their maintenance cost and whether the RAAF has enough spares to last to 2020. The research on developing better NDI techniques, understanding the impact of damage on strength, understanding the residual strength of aged but seemingly undamaged panels and developing better repair methods will save unnecessary maintenance and reduce necessary maintenance costs. The program to develop technology for cost effective substitution of composite material panels (described below) will mitigate against any shortfall in spares.

NDI Research

Some F-111-specific NDI research is in progress to refine techniques for bonded honeycomb panels. The options being investigated are through-transmission ultrasonics, thermography and high resolution digitisation of X rays. A large field trial has been conducted to characterise the probability of detection of magnetic rubber inspection because of its key impact on inspection intervals for D6ac steel. Specimens were cracked in the laboratory and sent to RAAF NDI technicians to inspect. The results have been analysed but have flagged the need to gather some further trials data to resolve some anomalies.

Risk Analysis

DSTO is currently building its capability to perform risk analyses. It is planned to carry out a risk analysis of the F-111 once the data from the teardown inspections is available. The aim will be to determine any appropriate adjustment of DADTA inspection intervals to allow for widespread flaws and preserve acceptable risk-per-flight levels.

Life Extension

Some life extension measures have already been taken for the F-111 to deal with known life-limiting issues. Two of them are described in detail below. They have been very successful and have the potential for more general application.

Wing Pivot Fitting Optimisation

The wing pivot fitting (WPF) is a primary structural component in the F-111 wing and has been the site of fatigue cracking in service and structural failures during cold proof load testing (CPLT). It is essentially a box structure with integrally forged stiffeners that transmits concentrated wing loads into the wing carry through box through a pivot pin. The main locations at which cracks have occurred are stress concentrating geometric features in the WPF upper plate stiffeners, known as stiffener runouts (SROs) and fuel flow vent holes (FFVHs). Typical blueprint geometries, and associated elastic stresses, at idealised CPLT loading, are shown in Figures 2 and 3, for the SROs and FFVHs. The unusual occurrence of fatigue cracking at these features in the upper plate of the wing (where the in-service flight loading is compression dominated), is attributed to the presence of residual tensile stresses which are caused by localised compresive material yielding during CPLT. Such fatigue cracks jeopardise the structural integrity of the wing and must be strictly managed in service. This imposes a costly maintenance burden. Also, the aircraft availability is reduced and at current estimated crack growth rates the planned withdrawal date of 2020 may not be achieved. Hence, any measure that allows extension of this interval can potentially produce significant benefits to the RAAF.

Historically, such critical stress concentrators have been reworked to remove the damaged material, where the rework shapes consist of circular arcs and/or straight-line segments. The design of these ‘traditional’ reworks has typically been undertaken using trial-and-error finite element (FE) analyses. While such traditional shapes, as shown in Figures 2 and 3, provide for crack removal, typically they do not provide significant reductions in stress concentrations, and hence further cracking usually occurs.

Hence, in the present work, precise free-form optimal shapes are determined using a finite-element-based gradientless shape optimisation procedure that has been developed in AED over recent years [2-6]. For the F-111 application, these optimal rework shapes were developed using a two stage method, where preliminary work was carried out using fully automated 2D FE optimisation analyses, with subsequent refinement being completed in a semi-automated manner using a complex and large scale 3D FE model, [7, 8]. Here the objective has been to minimise the peak compressive stresses during CPLT, which would therefore also minimise the resultant residual tensile stresses. As indicated in Figures 2 and 3, the optimal shapes typically provide a predicted 30% – 40% reduction in peak compressive elastic stresses (at the critical locations of approximately; x = 28 mm for the SROs and $\theta =$ 320 degrees for FFVHs) as compared to the traditional rework shapes. It can also be seen that the reductions are about 50% as compared to the blueprint geometries. A number of important issues have been addressed in the present practical problem, including: reduction of multiple stress peaks around the hole boundaries (both tensile and compressive); use of higher-order finite elements for efficient robust stress prediction; accounting for the effect of size constraints on the optimal shapes; and assessing the robustness [9] of the idealised optimal shapes to perturbations away from idealised conditions, such as those due to potential manufacturing errors.

As part of an associated validation program, the precise shapes have been manufactured in two full-scale static test wings using an advanced electro discharge machining procedure [10]. Experimental elastic strain measurements for the optimal shapes compare very well with the FE predictions [11]**,** (ie less than 9% error in peak strain). Further dynamic wing tests are currently underway in order to determine the damage tolerance and durability of these optimal shapes. Based on the successful results to date, fleet-wide implementation of the optimal reworks is scheduled to commence this year.

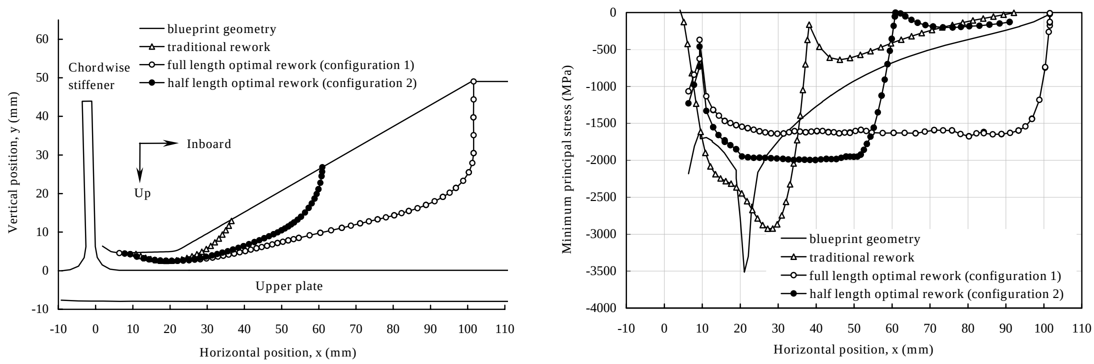


Figure 2: Comparison of blueprint, traditional and optimised rework shapes (shown inverted) and elastic boundary stress distributions at CPLT loading of +7.33g for SRO#2.

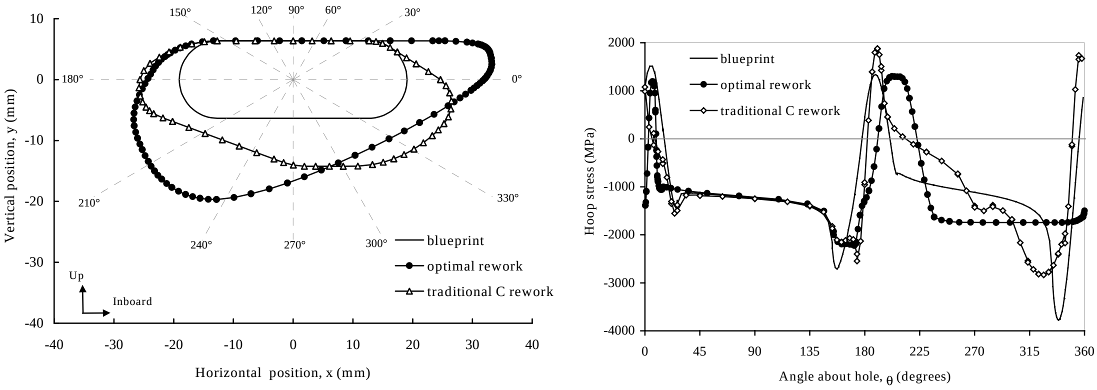
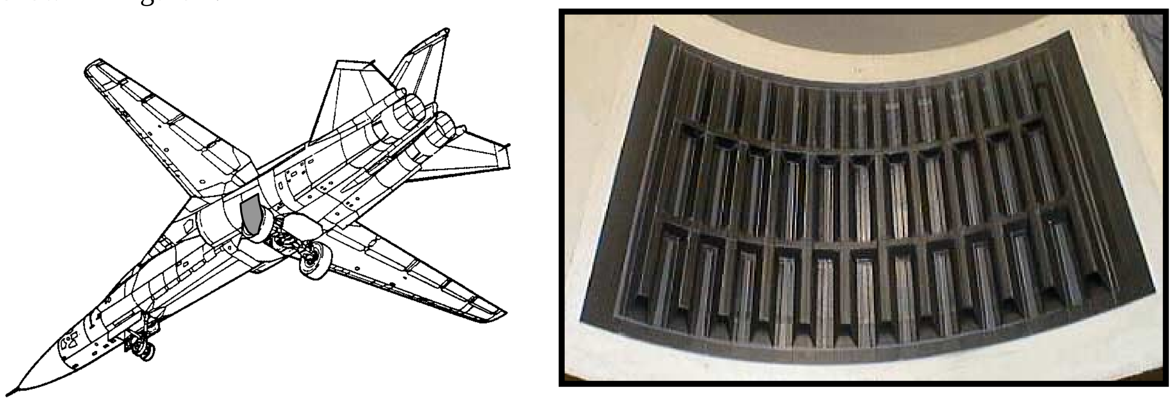


Figure 3: Comparison of blueprint, traditional C and optimised rework shapes and elastic boundary stress distributions at CPLT loading of +7.33g for FFVH 13

Bonded Panels Replacement

Each F-111 contains hundreds of bonded metallic panels, many of which require regular maintenance and/or replacement. This is very expensive. DSTO, in collaboration with the Cooperative Research Centre for Advanced Composite Structures (CRC-ACS), are therefore developing the replacement panel technology. This is a generic approach where costly-to-support metallic components, such as these panels, are replaced with more durable and cost effective composite panels. This technology is being developed and validated through the design, manufacture and certification of a demonstrator replacement for F-111 Panel 3208. The location of this panel on the aircraft and a photograph of the composite replacement, are shown in Figure 4.



(a) (b)

Figure 4: (a) Schematic diagram of F-111 showing the location of panel 3208, and (b) demonstrator replacement for Panel 3208

Design

In this work the F-111 original equipment manufacturer (OEM) stress notes were used to determine the load envelope and critical design cases for Panel 3208. Work is ongoing to develop a true reverse engineering capability, where an equivalent panel design may be determined without need for OEM data. A panel of equivalent stiffness in all modes chosen from standard composite panel solutions. The strength envelope for this replacement was superimposed on the load envelope for the metallic panel. The design was revised until the load envelope was covered fully by the strength envelope of the replacement panel. This was the preliminary panel design.

An important factor in design of the replacement panel is the panel-to-fuselage attachment. A technique for laminating thin stainless steel or titanium shims into the edge of the panels was developed, but was not needed for the trial panel because bearing stresses were low.

The effect of differences in stiffness and thermal expansion coefficient between the original panel and the preliminary panel design was then assessed. Simplified FE models of the panel and local sub-structure were generated to verify that these effects were not detrimental. For example, because of strength requirements, the shear stiffness of replacement Panel 3208 was 70 % higher than that of the original. FE models showed that this elevated the shear flow in adjacent panels by less than 5 %, which was considered acceptable.

Manufacture

A wide range of configurations was evaluated. The best compromise for cost, structural integrity and ease of manufacture was the top-hat and z stiffened construction shown in Figure 4 (b).

The first part of manufacture was to obtain the surface coordinates of the original surface. These were captured using photogrammetry because it could be performed in-situ, on a wide range of panel sizes, with a minimum of disturbance to aircraft operations.

The need to minimise tooling and curing costs led to the selection of commercially available medium temperature moulding (MTM) composite fabric. Parts laid-up with this material are initially oven-cured under vacuum at around 80 °C, released from the mould then post-cured at 180 °C as a free-standing part. This eliminates the need for an autoclave and allows the use of cheap tooling materials. The properties of these MTM composites are relatively poor under hot/wet conditions so the number of plies used in the replacement panel was increased beyond that which would be used for an autoclave curing part. The additional weight was not a problem since all-composite replacement panels will be significantly lighter than the original, 6 kg compared to 9 kg in the case of Panel 3208.

Certification

A part building block approach to certification was used, with testing performed at the coupon, sub-component and generic full-scale level. Consistent with the requirement to minimise costs, only the essential testing was included. This testing was aimed at establishing conservative design allowables for the new materials and validating the analysis/assumptions made in design.

Coupon testing was only performed to establish a conservative environmental knockdown factor for the sub-component test. The interlaminar shear test was chosen since it is known to provide the highest knockdown and is a simple test to conduct.

Representative sub-component test specimens were impacted at critical locations then tested to failure in shear and compression. This generated the static failure strain design allowables. Damage tolerance was demonstrated in a simplified program of fatigue testing of impacted sub-component specimens.

It is proposed that the final proof-of-structure be demonstrated analytically, rather than through full-scale testing as done with traditional certification approaches. Tests will be performed on instrumented flat and single curvature, close to full-scale, generic panels to demonstrate the validity of the FE models used to design the replacement panels.

This approach will be validated when an instrumented demonstrator Panel 3208 is fitted to an F-111 C aircraft and subject to a ground strain survey. The predictions made by the FE models will be compared to the strains observed during this test.

Conclusion

The SOP is currently in the fourth year of the eight year program. The majority of infrastructure development tasks, such as the establishment of the necessary testing and teardown infrastructure and database development tasks, and a considerable proportion of the data transfer tasks have now been completed. Additionally, a number of critical capabilities have been successfully transferred from Lockheed to the Australian DSN, in particular an F-111 DADTA capability now exists in Australian industry and the Internal Loads Model is now resident at DSTO.

The F-111 Sole Operator Program was a response to changed circumstances for the RAAF F-111 supportability. It forced the development of in-country capabilities. The total cost of the F-111 SOP is estimated as A$25m. Its prime purpose was to ensure continued safe operation of the RAAF F-111 fleet for the remaining 20 years of service. However, it is expected that the program will pay for itself several times over in maintenance savings, without even having to be justified in terms of delayed replacement acquisition savings. As such, it shows that for any aircraft fleet, it can be worthwhile to step back and take a fresh holistic look at its management and identify areas where research leading to enhanced support capability can have a significant pay off.

References

1. Agius, J., Connor, P., Madley, W. and Watters, K. C., (1998) *Royal Australian Air Force (RAAF) F-111 Aircraft Structural Integrity Sole Operator Program,* Proceedings of the 1998 USAF Aircraft Structural Integrity Program Conference, San Antonio, Texas, 1-3 December 1998.
2. Kaye, R., and Heller, M., (1997) *Structural Shape Optimisation by Iterative Finite Element Solution,* DSTO-RR-0105, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.
3. Heller, M., Kaye, R. and Rose, L.R.F., (1999) *A Gradient-less Finite Element Procedure for Shape Optimisation,* Journal of Strain Analysis, 34(5) 323-336.
4. Waldman, W., Heller, M. and Chen, G., (2000) *Optimal free-form fillet shapes for tension and bending,* (accepted for publication in the Journal of Fatigue).
5. Heller, M., Rose, L.R.F. and Waldman, W., (2000) *Rework shape optimisation for two closely-spaced holes,* (draft for publication in the Journal of Fatigue).
6. Heller, M., Waldman, W., Kaye, R., and Chen, G., *Developments in local notch rework shape optimisation for life extension,* Australian Fracture Group Symposium on Structural Integrity and Fracture, University of Technology, Sydney, pp. 128-139, 29-30 June 2000.
7. Heller, M., Burchill, M., McDonald, M. and Watters, K.C., (2001) *Shape Optimisation of Critical Stiffener Runouts in the F-111 Wing Pivot Fitting,* DSTO-TR-1120, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.
8. Heller, M., Burchill, M., McDonald, M. and Watters, K.C., (2001) *Shape Optimisation of Critical Fuel Flow Vent Holes in the F-111 Wing Pivot Fitting,* DSTO-TR-119, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.
9. McDonald, M., Heller, M., Goldstraw, M. and Hew, A., (2001) *Robustness of the F-111 Wing Pivot Fitting Optimal Rework Shapes,* DSTO-TR-1121, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.
10. Weller, S., and Koning, R., (2001) *F-111 Wing Pivot Fitting Optimisation: Summary of the Manufacturing Process,* DSTO-TN-0349, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.
11. McDonald, M., (2001) *Validation of Optimal Reworks for the F-111 Wing Pivot Fitting by Full-Scale Strain Survey,* DSTO Technical Report, File M1/9/845, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

This page has been deliberately left blank

Page intentionnellement blanche

Research of Extension of the Life Cycle of Helicopter Rotor Blade in Hungary

M. Balaskó[1], G. Endrőczi[2], J. Veres[3], Gy. Molnár[3], F. Kőrösi[4]
[1]KFKI Atomic Energy Research Institute
H-1525 Budapest, P.O.Box 49. Hungary
[2]KFKI Research Institute for Particles and Nucleus
H-1525 Budapest, P.O.Box 49, Hungary
[3]Logistic Center of Hungarian Air Force
H-1885 Budapest, B.O.Box 25, Hungary
[4]Szent István University, Institute of Environmental
and Land Management, H-2100,
Hungary

Abstract

Combined measurements have been carried out at the Budapest research reactor, where the dimension of the radiography station was extended for the purpose the control the condition of helicopter rotor blades in the different period of their life time. High resolution radiography pictures were taken to find anomaly in the distribution of resin materials at the core-honeycomb-hull interfaces, failure at the "adhesive filling" and possible bondline flaws. Parallel to the radiographic visualisation vibration tests using the method of statistic energy analysis (focused on damping and energy distributions and propagation) served for control of dynamic behaviour of different aged structures. As a result of the work it is suggested that the combined application of the neutron-, X-ray radiography and vibration diagnostics might be a very useful method for the condition monitoring of helicopter rotor blades and other similar composite structures.

1. Introduction

The safe life testing of the rotary wing aircrafts, especially those of the rotor blades, is of paramount importance. The declaration of a structural failure that can grow to the point where structural integrity is affected comprises a central core. In this process the composition inspection, monitoring the rate growth of the defect in relation to the total flight hours are essential. The above demands underline the necessity in testing and applying new non-destructive testing (NDT) methods for inspection in service.

As a consequence, for the composite structure investigation of the rotor blades, three NDT methods: (i) Dynamic Neutron Radiography (DNR) (ii) Dynamic X-ray Radiography (DXR), and (iii) Vibration Diagnostics (VD) with Statistical Energy Analysis (SEA) were semi-simultaneously applied [1]. The used three NDT methods give complementary information upon the investigated object. The DNR is capable of investigating the irregularities in the fibre-glass/epoxy honeycomb structures. The details of metal parts and contactor of the heating elements are shown by DXR. The SEA and other VD methods are suitable for identifying small changes of structural damping. The experiments were performed at the Dynamic Radiography Station (DRS) of the Budapest (10 MW) research reactor where the necessary development, regarding the extension of the dimensions of the investigated object, was carried out. As a result the working place can inspect 10 m long and 0.7 m wide targets by neutron and X-ray radiography and by vibration diagnostics.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

The main goal of the present study was to demonstrate the applicability of DRS in revelling the defects of the composite structures of helicopter rotor blades for Hungarian Air Force. In describing the defects the terms used by [2] for classifying the structural integrity and aging for helicopters were followed upon.

2. Methods

2.1. Radiography

Neutron radiography utilizes transmission of radiation to obtain information on the structure and/or inner processes of a given object. The basic principle of NR is very simple [3]. The object under examination is placed in the path of the incident radiation, and the transmitted radiation is detected by a two-dimensional imaging, as is illustrated in Figure 1. The NR

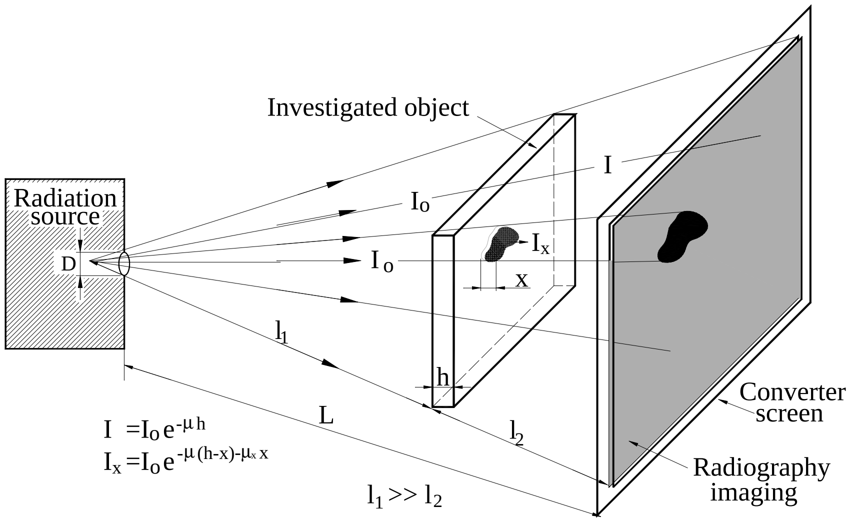


Fig. 1. General principle of radiography

arrangement consists of a neutron source, a pin-hole type collimator which forms the beam, and a detecting system which registers the transmitted image of the investigated object. The most important characteristic technical parameter of an NR facility is the collimation ratio *L/D* where *L* is the distance between the incident aperture of the collimator and the imagintg plane, *D* is the diameter of the aperture. This important parameter describes the beam collimation and will limit the obtainable spatial resolution by the inherent blurring independently from the properties of the imaging system. This unsharpness U_{beam} can be related to the distance between the object and the detector plane l_2 and to the *L/D* ratio

$$U_{beam} = \frac{l_s}{L/D}$$

Two opposing demands have to be taken into consideration when planning a radiography arrangement: if *L/D* is large then the neutron flux Φ_{NR} at the imaging plane is relatively weak but the geometrical sharpness is high, and vice versa.

$$\Phi_{NR} = \frac{\Phi_s}{16(L/D)^2}$$

where Φ_s is the incident neutron flux.

In radiography imaging the attenuation coefficient μ is a crucial parameter. The transmitted intensity of the radiation, *I*, passing through a sample with an average transmission of μ can be written as

$$I = I_o e^{-\mu h}$$

Where I_o is the incident intensity and *h* is the thickness of the sample. If there is any inclusion (inhomogeneity, inner structure) in the sample of thickness *x* and transmission μ_x then the transmitted intensity, I_x is given as

$$I_x = I_o e^{-\mu(h-x)-\mu_x x}$$

If the value of μ and μ_x are different from each other then the presence of the inclusion will provide a contrast in the radiography image.

The attenuation coefficient vs. atomic number is plotted in Figure 2 for neutron radiation and for gamma- and X-rays. Its value depends on both the coherent and incoherent scattering and on the absorption properties of the element(s). For neutrons μ, does not show any regularity as a function of atomic number, and for some of the lightest elements (H, B, Li) the attenuation coefficient is by two orders of magnitude greater than the corresponding parameter for most of the technically important elements, such as Al, Si, Mg, Fe, Cr. This fact is of practical importance, viz. neutrons penetrate almost all metals used for construction purposes with little loss in intensity; in contrast they are considerably attenuated in passing through materials containing hydrogen, such as water, oil or several types of synthetics. On the other hand in the case of X-ray and gamma radiation, this dependence may be characterized by more or less continuously increasing curves.. This means that the radiation is absorbed to a great extent by heavy elements whereas it penetrates light materials such as hydrogen without significant loss in intensity.

These differences for various radiations provide the possibility to gain complementary information by using all three types of radiation together.

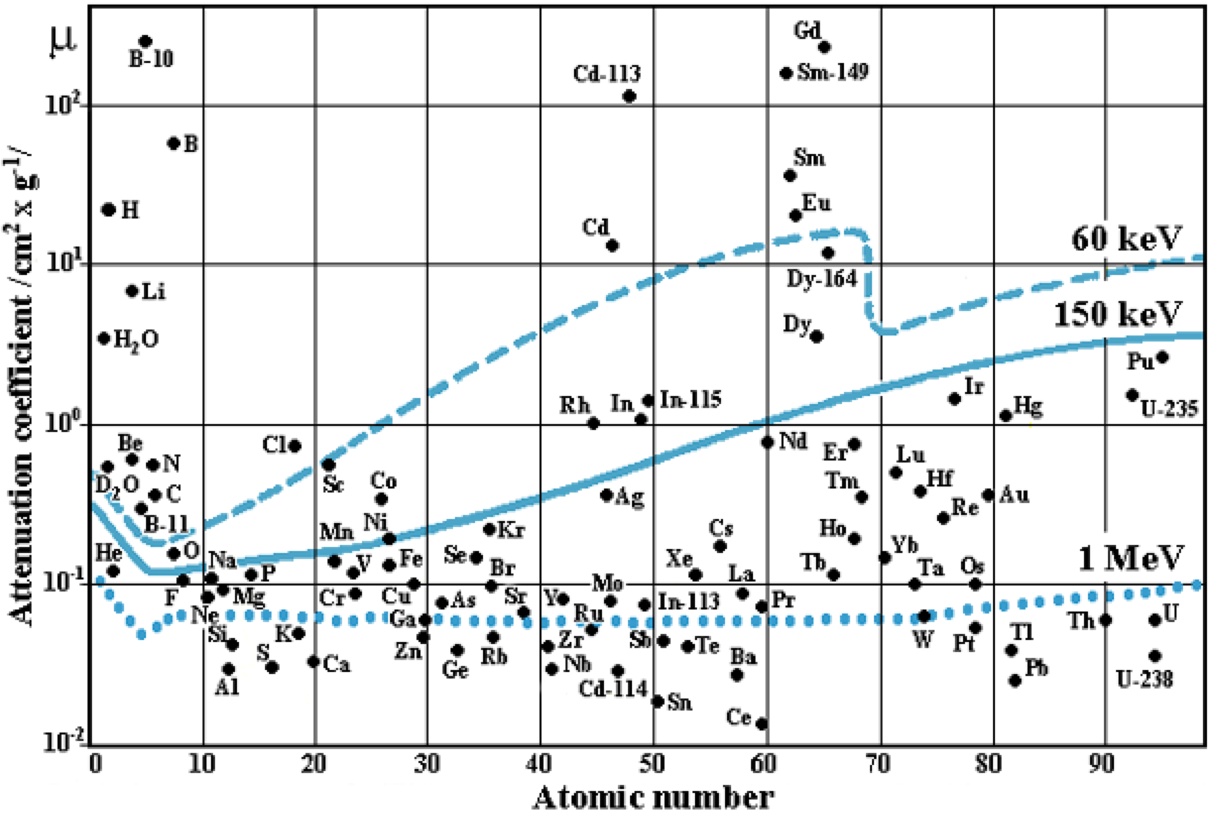


Fig. 2. Attenuation coefficient (note the logarithmic scale) of elements forneutrons (separate dots), for 1 MeV gamma-ray (dotted line), for 150 keV X-ray (solid line) and for 60 keV X-ray (dashed line)

2.1.1. Imaging techniques

In that neutrons are neutral particles a converter material - in NR generally a foil – is used to convert neutrons to another type of radiation, to enable them to be detected directly. Various detector systems are employed in NR: combinations of film and neutron sensitive converter foil, combinations of a light-emitting scintillator screen with a CCD camera and, more recently, imaging plates. Depending on the object to be investigated and the task to be solved, two basic types of NR are in use: static radiography and dynamic radiography (real-time). Both techniques provide averaged information on the investigated object in its depth. Neutron computer tomography (NCT) is a rapidly developing technique that provides information on the three-dimensional structure of a given object.

2.1.1.1. ***Static NR*** records a static picture of the object to be investigated. Even nowadays, film techniques are the most widely used. The information is not a priori obtained in digital form, but may be digitized with a scanner or densitometer. The most recent developments are the imaging plate (IP) system and the camera-based technique, both of which are now being used to a much greater extent.

The IP is a new film-like radiation image sensor based on photostimulated luminescence. It consists of a specifically designed composite structure that traps and stores the radiation energy. A polyester support film is uniformly coated with a photo-stimulatable luminescent material - barium fluorobromide containing a trace amount of Eu^{2+} as a luminescence centre ($BaFBr:Eu^{2+}$) – and it is then coated with a thin protective layer. The stored energy is stable until scanned with a laser beam whereupon the energy is released as luminescence. In the case of neutron sensitive IP the storage luminescent material is mixed with gadolinium oxide.

The camera-based system consists of a scintillator plate and either a low-light-level (LLL) video or CCD camera which records the light emitted by the scintillator. The images recorded by a CCD camera are inherently digital while those of a video camera can be recorded by video recorder or can optionally be digitized by a frame-graber. In static radiography the images recorded by the camera are integrated, and thus a static picture of good statistics may be obtained from the object.

2.1.1.2. Real-time NR is used to investigate movements inside the investigated object (flow of fluids in metal tubes, evaporation or condensation processes, two-phase systems). The imaging system consists of a scintillator plate that converts the neutrons into light which is detected by an LLL video camera with short imaging cycle or by a CCD camera. The individual images are registered and analysed on a time scale, they may be visualized on a monitor and recorded by a video recorder or by a computer. Compared with static NR this technique needs a relatively high neutron flux: at least 10^6 n cm^{-2} sec^{-1}.

2.2. Vibration Diagnostics (VD)

Parallel to the NDT measurements the Statistical Energy Analysis (SEA) as a vibration diagnostical tool were applied for the detection of structural behaviour of rotor blades.

The SEA applicable for structural identification where the modal density (amount of modes in a given frequency range) nearly constant.

The increasing development of SEA technique has been based on the early works of R.H.Lyon, P.W.Smith Jr. And G.Maidanik [4]. Prof. Lyon gave a complete summary of the theory and applications in 1975 and his work still serves as one of the most important refereences in the field.

In this theory the energy denotes the primary variable and displacement, velocity, acceleration, stress, etc. are deduced from that. The dynamical parameters of the system, the environment and responses are looked upon as statistical populations having known distributions. The structural identification is based on estimated dissipative properties using statistical variables [5].

The theory is based on the recognition that the kinetic and potential energies of similar natural modes or resonators whitin a frequency range are equal at resonacies. The energies of individual modes simply add to form the total system energy. The averaged displacements, velocities, accelerations, forces, mechanical impedances, energy losses or dampings, etc. are formulated in a system fashion to the description of a single mode resonator. Considering a group of resonators the basic expressions of responses and energies in the use of white noise excitation in a given frequency range can be formulated as follows:

$$< F^2 >_{\Delta f} = S_f \Delta f$$

$$< y^2 > = < \ddot{y}^2 > / \omega_0^2$$

where Δf - frequency band
f,ω - center frequency of a band
<> - spatial average
S_f - power spectral density of force
F^2 - mean square value of force
y - displacement

This approximation is valid, when the resonant part dominates instead of mass.

The power input to the system, averaged over source location is:

$$P_{in} = S_f \Delta f \frac{\Pi n}{2M} = F^2 < G >$$

where n - modal density
G - conductance
M - mass

It can be seen that the modal density and the average conductance are closely related.

For a single resonator the loss factor is introduced as a phase angle of the complex Young's modulus

$$E = E_0(1 - i\eta)\ ,$$

for a viscous element

$$\eta = R / \omega M\ .$$

The dissipation or damping properties of a system can be described using energy parameters. The measure of the dissipation of energy stored in the „loss-factor" which is the ratio of energy dissipated per unit time to average energy stored.

$$\eta = \frac{P_{diss}}{2 \Pi f E_{stored}}$$

The other important problem is the energy transfer between the structures. In the case of multimodal resonator systems vibration energy is transferred from one subsystem to the other by similar modes. The transferred energy is respect to the stored energies and the involved modes. For arbitrary two systems, it was found to be,

$$P_{i,j} = \omega\eta_{i,j} E_{i,total} - \omega\eta_{j,i} E_{j,total}$$

According to these equations, the SEA calculates the flow and storage of vibration energies in a complete system, identifies the structure as a series of energy storage and dissipating elements. Thus in a general SEA model the structures are idealized into an assemblage of individual subsystems having similar and significant energy storage modes.

In the model the input powers, $P_{i,in}$, resulted from acoustical noise or mechanical excitation, have not sensitivies to the state of coupling between subsystems. The dissipated powers, $P_{i,diss}$, represent the energy lost to mechanical vibration and depends only on the amount of energy stored in the subsystem. The transmitted power, Pi,j , represents the energy exchanged between subsystems. The dissipated power cannot be returned to the system. The amount of energy stored in the subsystem is determined by available modes N1...Nn within the frequency band. The following basic relationships concern the model without detailed verifications:

The power flow between subsystems is

- proportional to the actual vibration energies of subsystems
- directly proportional to the difference in decoupled energy of subsystems
- the average power flow is from the subsystem of greater to lesser energy
- equal difference of subsystem energies is either direction will result in an equal power flow (the power is reciprocal)

For the i'th subsystem the power balance is:

$$P_{i,in} = P_{i,diss} + \sum_{j=1}^{N_i} P_{i,j} \qquad i \neq j$$

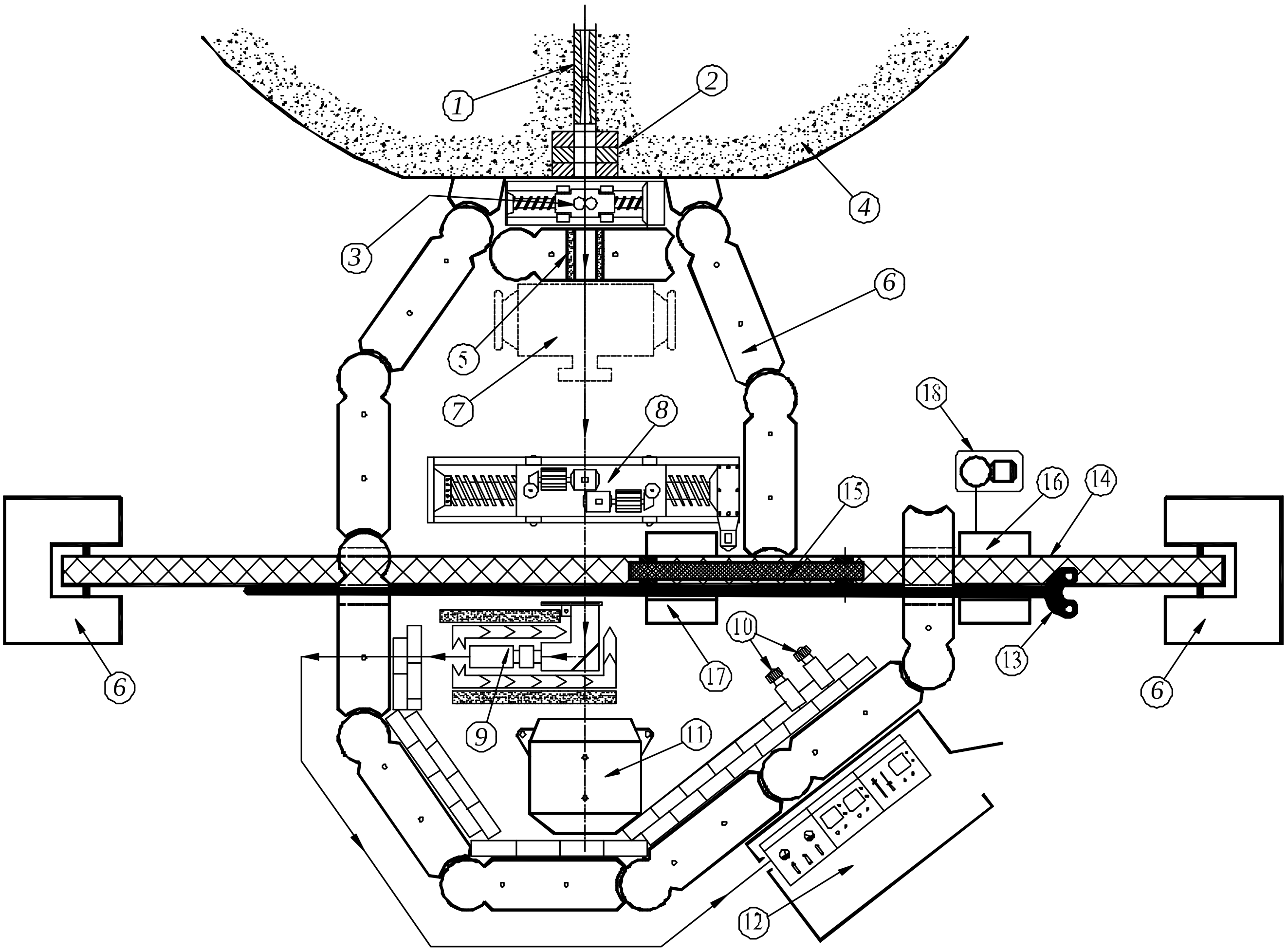
The relations between power, stored energy and loss-factors can be written in matrix wherein the structure is identified by the loss factor matrix. The loss factor matrix, the vector of energy stored and the input powers stand for the stiffness matrix and vectors of displacements and applied forces.

$$\underline{\underline{\eta}}\,\underline{E} = \underline{P_{in}} / \omega$$

$$\underline{\underline{J}}\,\underline{y} = \underline{F}$$

3. Experimental facility

Measurements were performed at the Dynamic Radiography Station (DRS) at the 10 MW Research Reactors in Budapest. The Figure 3 shows the arrangement of the DRS. Its main parameter are the follows: 10^8n.cm^{-2}.sec^{-1}, the collimation ratio (L/D): 170, the diameter of the beam: 180 mm. The X-ray generator was adjusted to 150 kV and 3 mA. The information carrying radiography images were converted into light ones by NE426 scintillator screen for neutron radiography and NaCs single crystal was used for X-ray radiography. The obtained light images were detected 10^{-4} lux low light level TV camera (ITV1122 type) and registered by S-VHS videocassette recorder [6]. For digitalization and image processing QUANTEL SAPPHYRE V.05 and IMAN a β version softwares were applied.



1. Inside collimator
2. Iron shutter system
3. Beam diameter changer and filter unit
4. Reactor shielding
5. Outside collimator
6. Biological shielding
7. X-ray generator
8. Investigated object with remote control mechanism
9. Imaging system with low light level TV camera
10. Background TV cameras
11. Beam stop
12. Control cabin
13. Rotor blade (~ 10 m)
14. Rail of the rotor blade transporter
15. Transporter of the rotor blade
16. Moistening module
17. Drying unit
18. High pressure water pump

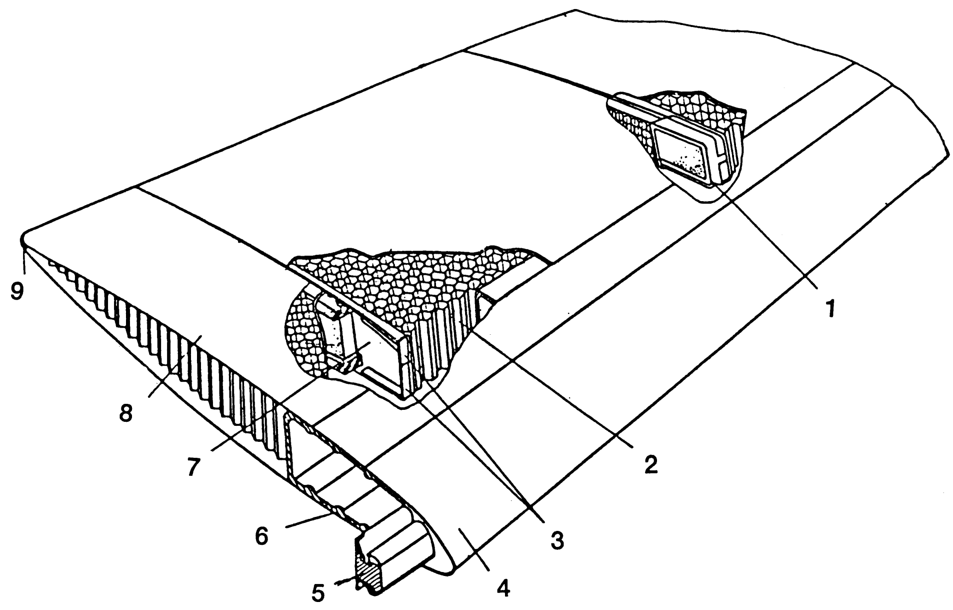
Fig. 3. Schematic arrangement of Dynamic Radiography Station at the Budapest research reactor

Fig. 4. Overview of the Dynamic Radiography experimental station

We could verify the state of rotor blades in dry and wet conditions simulating the complicated weather circumstances. The moisture was served by the Moistening module and the necessary water was supplied by the closed-circulated High pressure water pump. The overview of the experimental setup is shown in Figure 4. As it can be observed in the Figure 4, the complete rotor blade can be placed and moved in the neutron and X-ray beam. Simultaneously with the radiography visualization, on the other blade the vibration sensors were placed at given points of the blades and the damping characteristics were investigated. The registered vibration noises were analysed with a dual-channel real time frequency analyser (BK2035). In addition to the Statistical Energy Analysis measurement a small exciter table (BK4810) and an impedance head (BK 8000) used.

4. Investigated object

The lifetime extension of the rotary wings is a main achievable goal for the helicopter serving in the Hungarian Army. Majority of the helicopters, Mi 8 and Mi 24 types in the Hungarian Army's inventory are several decades old and required to continue their service even longer.
One of the most important parts of them is the rotor blade. They are made of composite structures and contain 21 pieces of honeycomb construction with many bonded surfaces. The 21 sections of the rotor blades were divided into 4 zones horizontally and 53 fields in



1. **balance seal**
2. **honeycomb structure**
3. **face-rib**
4. **anti-ice heater**
5. **anti-flatter weight**
6. **spar**
7. **rib.**
8. **skin**
9. **backside stringer**

Fig. 5. The inner structure of the rotor blade

vertically. The key part of the rotor blade comprises the aluminium alloy metal main holder bonded to the honeycomb structure as seen in Figure 5.

5. Measurement

The description of the measurement was very prudent regarding the dangerous nature of the radiation material testings both neutron an X-ray. On schedule of the inspection the first step was the VD measurement, the second step was an NR inspection in dry condition, the third step was the NR inspection in wept condition, the fourth step was the XR test and the last one was another. VD measurement again. A new (RLU) and (RL-10) rotor blades were verified by this inspection technology. We are able to declare that we did not experience any effect of the radiation techniques.

5.1. Neutron radiography

The NR is capable of detecting and visualizing irregularity and defects in composite construction. One of the main advantages of the NR constitutes that it is especially sensitive in revealing the hydrogen- (water) content and its distribution in the investigated object. To this end a special analysis was designed and carried out at DRS. In the first step the all surface of the blade was scanned with DNR in dry conditions. In the second step having the rotor blade being moistened, the scanning operation was repeated to follow the penetration sites and distribution of water in the composites structures.

The most important bond is the one that sticks together the aluminium alloy metal main holder and the honeycomb structure.

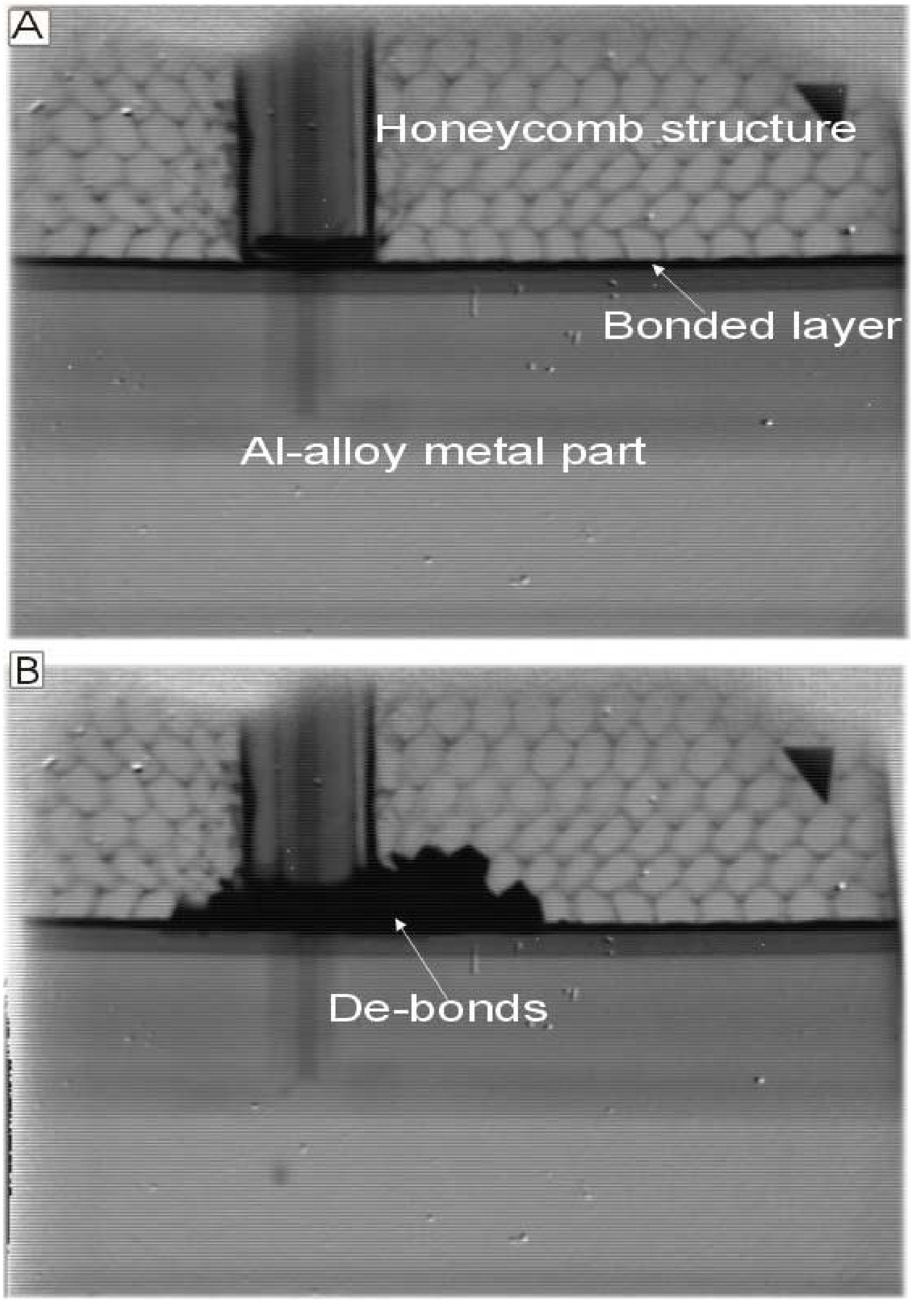


Fig. 6. DNR images of the de-bonds. “A” scanned without water “B” contrasted by water

Fig. 6/a shows this part between the 13th and 14th section. The bonded area is seen as a horizontal black line under dry condition. In the Fig. 6/b the efficacy of moistening in unveiling the damage of the gum sealing at the border of the sections by centrifugal force was explicitly demonstrated.

The other failure makes up the delamination within piles of a laminate as displayed in Fig. 7. This failure was recognized at the 4th band, 49th field of RL 10 rotor blade (Fig. 7A), with the most affected areas being displayed with curves. The Figure 7B shows Sobel-enhanced edges of the delamination, while the Fig. 7C illustrates the Fig. 7D representation of the concerned area. This type of failure may be brought about by improper surface preparation, contamination and embedded foreign matter.

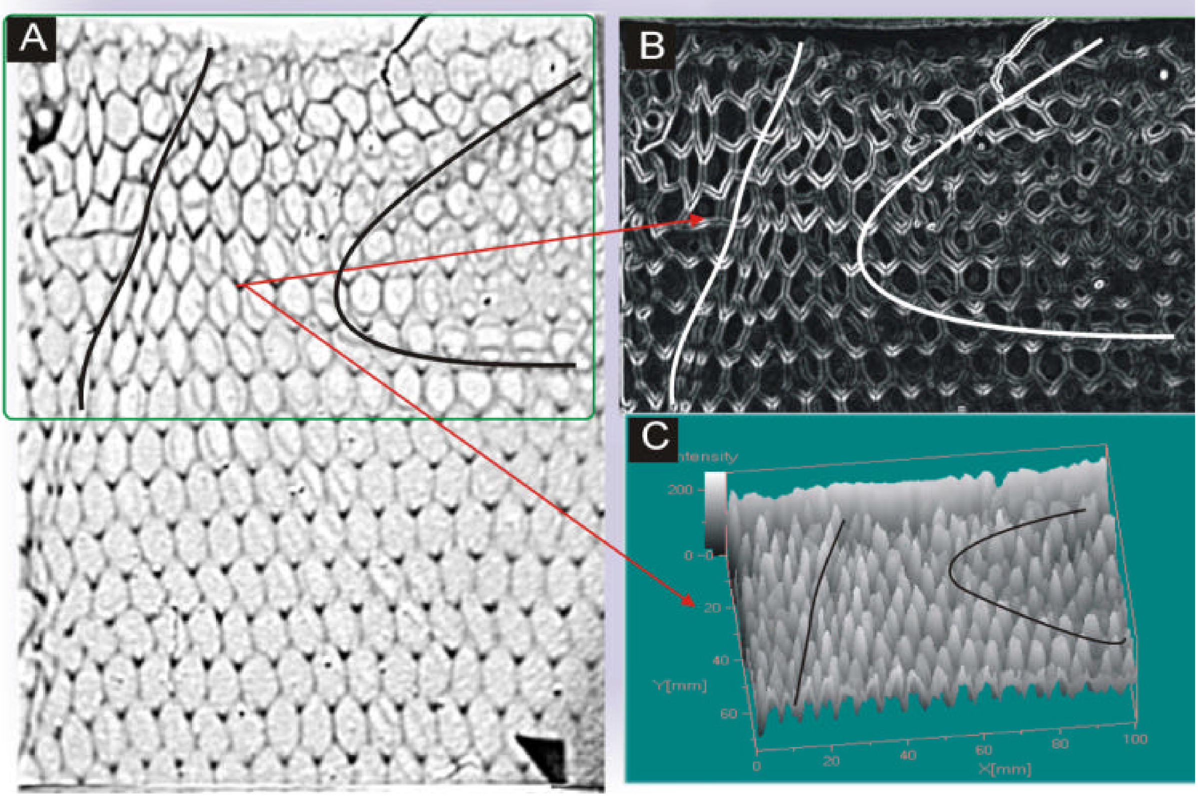


Fig. 7. Represetation of fiber misalignment. "A" shows the DNR image; "B" stands for the processed image of the marked area; "C" illustrates the 3D plot of the images. The curves demonstrate the affected area

The DNR image exhibited in Fig. 8 discloses another type of defect, i.e. resin-rich areas and resin-starved ones. Resin rich areas are localized, and filled with resin or lacking in fiber. The defects is caused by improper compaction or bleeding. Resin-starved areas are localized with insufficient resin evident as dry spots or areas. (These defect were located on the RL-10 blade: 4th band 35-36th field, 8th section.)

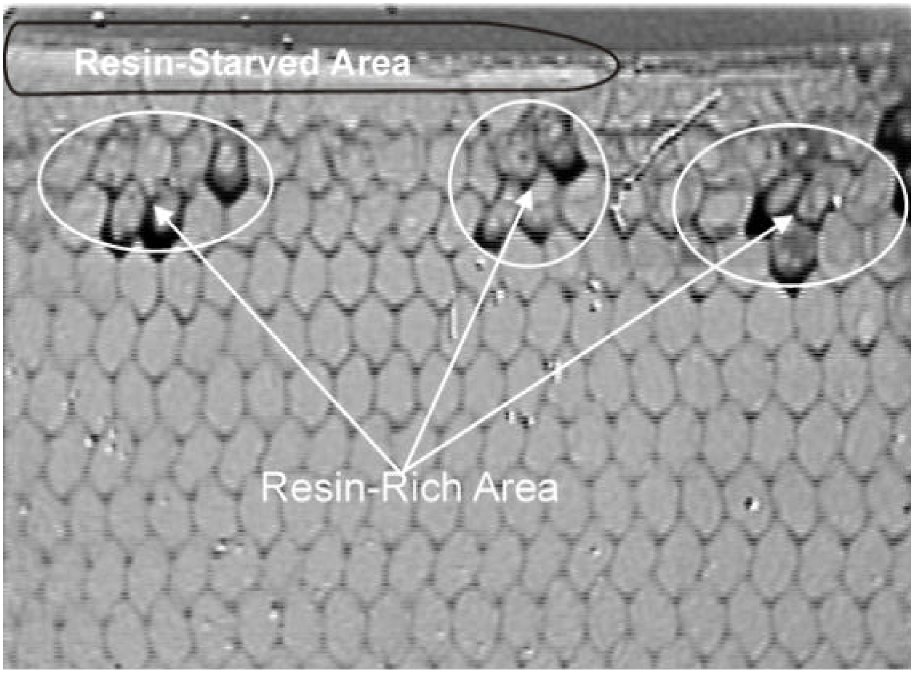


Fig. 8. Resin-rich and resin-starved areas as revealed by DNR imaging

Fig. 9 reveals honeycomb misalignment (‘A’), porosity (‘B’), and damage of the gum sealing (‘C’). Honeycomb misalignment is distortion of the piles resulting in changes from the desired orientation. These defects are due to improper lay-up and cure. Porosity (‘B’) is pockets within the bonding material. They are begot by entrapped air and gas bubbles, and caused by volatile substances, improper flow of resin and unequal pressure distribution. De-bonds (‘C’) were discussed above. (These defects were identified on RL-10 blade: 3rd . band 37th field, 16th section amid moistened condition.

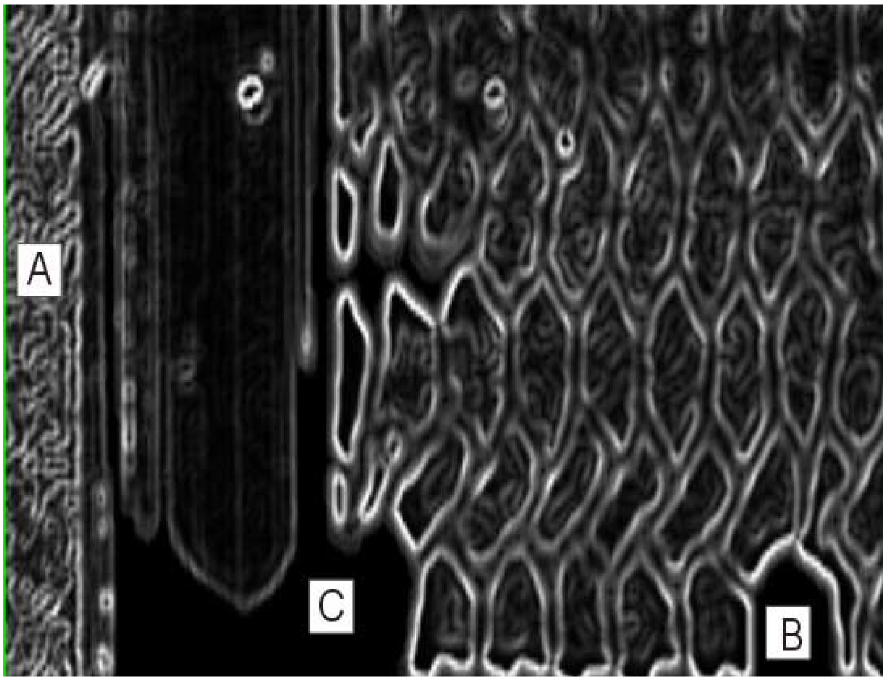


Fig. 9. Analyzed DNR image, demonstrating various defects: “A” honeycomb misalignment; “B” porosity; “C” damage of the gum sealing

5.2. X-ray Radiography

X-ray radiography is a complementary and useful tool in detecting the structural integrity of metal parts of the blades. Fig. 10 shows the heating element arrangements and their contacts on the blade. The measurement was performed on RL-10 rotor blade, iv.Band 54.field, 1st section.

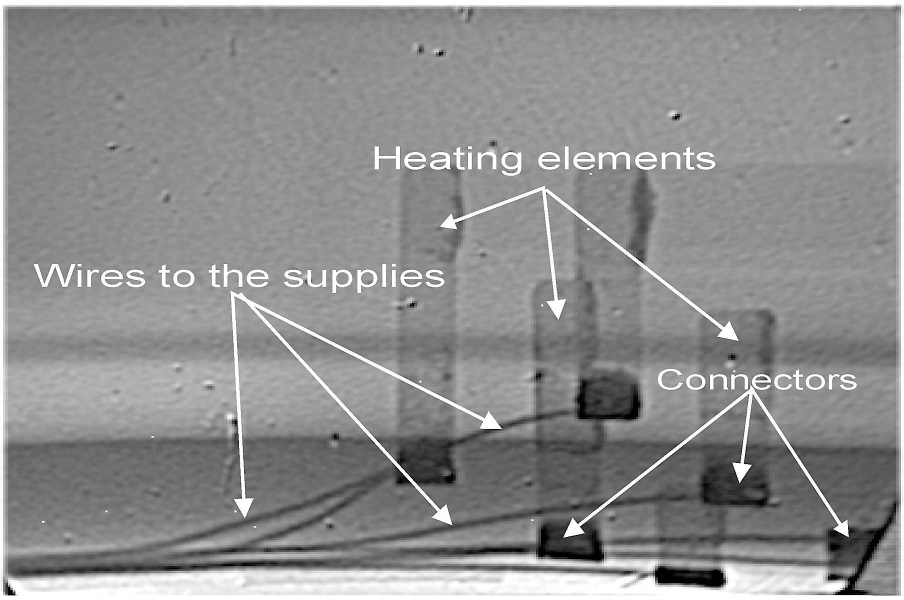


Fig. 10. Metal parts of the rotor blade visualized by XDR

5.3. Vibration Diagnostic

After the first laboratory tests series of different aged blades were measured. A simple two subsystems SEA model was used for the structural identification of rotor blades. During the measurements the sensors were placed at the 2., 5., 8., 14., 17., and 19. sectors at the border line of core and honeycomb section, at the 11. sector an exciter table with impedance head served for power input. The stored vibration energy, loss factors (proportional to damping) and their ratios were calculated in different frequency ranges. The quantity indicated on Fig 11. represent the general condition of blades.

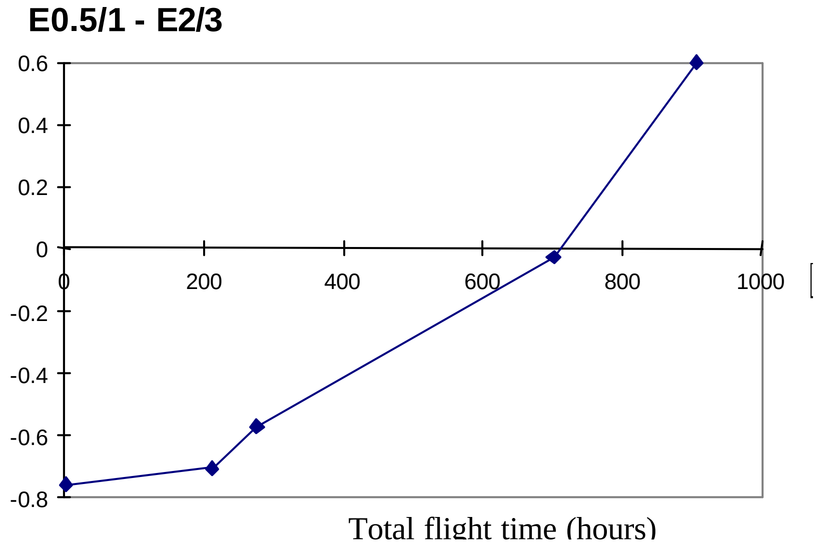


Fig. 11. Difference of stored energy between frequency bands at given flight times

Another parameter in the Fig. 12. demonstrates that distinct differences in the vibration energy collector capacity between new and old rotor blades in dry and moistened conditions. At a frequency band of 2.5-3 kHz exciting energy, the energy storage for the either the new or the old one displayed two peaks. The damping capacity for the new one was considerably lower than that for the old one. While the moistened blade exhibited no peak vs. the sensors positions. Another important observation from the SEA model that with increasing the frequency of exciting energy the energy storage capacity of the new blade's the first position, i.e. the blade end, was considerably, about 2 order, higher. It underlines the significantly less damping property of that section of the new rotor blade, as it is shown in the Fig. 13.

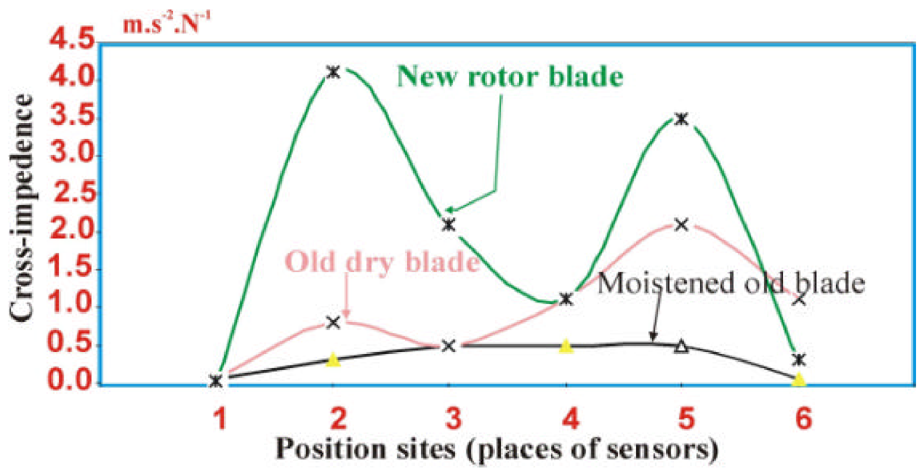


Fig. 12. The stored vibration energy of the old and new rotor blades at a frequency band of 2.5-3kHz

These evidences clearly prove that applying the two subsystems SEA models into the blade adequately indicate the differences in the conditions of the blades. According to the SEA measurement no damage to the blade was caused either by neutron or X-ray radiation during the test.

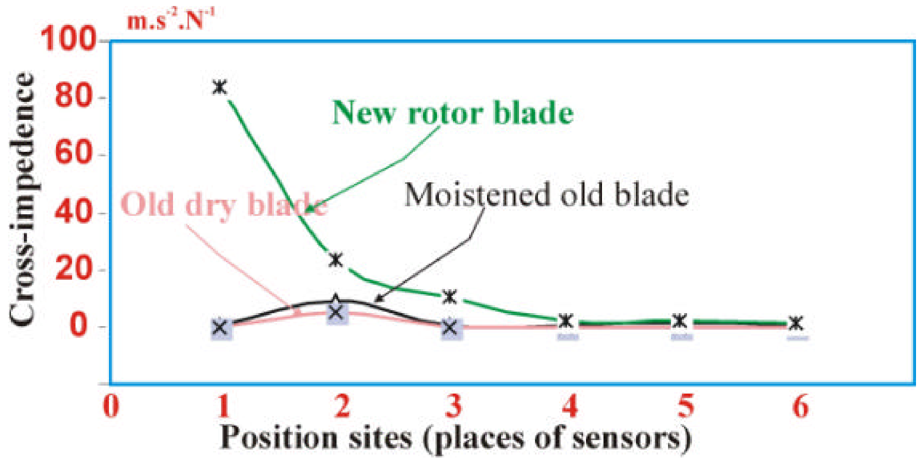


Fig. 13. The stored vibration energy of the old and new rotor blades at a frequency band of 6-8 kHz

6. Conclusions

The joint application of dynamic neutron and X-ray radiography and the statistical energy analysis proved to be applicable in the visualizing and detecting changes in the inner details of multiplayer-honeycomb structure of the helicopter rotor blades.

The defects revalued by DNR were:

Delamination
Porosity
Resin-rich areas
Resin-starved areas
Honeycomb misalignment
Damage of the sealing
De-bonds.

By using DXR the metal functioning part of the rotor blade can be inspected.

The SEA technique adequately and supplementary to the DNR and XNR characterized the stochastic amounts of micro-cracks of the rotor blades in connection with flight hours.

References

1. M. Balaskó, G. *Endrõczi, J.* Járfás, Gy. Makai, Study of complex composite-metal structure with dynamic neutron radiography and vibration diagnostics. Proc. 7th European Conference on Non-destructive Testing (ECNDT'98) Ed. B. Larsen. Copenhagen, (1998). Pp.341-348.

2. J.K. Sen and R.A. Everett, Structural integrity and aging related issuez for Helicopters, RTO/NATO 2000, ISBN92-837-1051-7, pp.5.1-5.21.
3. M.Balaskó, E.Sváb, Neutron radiography in research and development, Nukleonika 39, 1-2(1994) pp.3-22.
4. R.H. Lyon, Statistical Energy Analysis of Dynamical System, MIT Press 1975.
5. B.L. Clarkson et al, Experimental Work to Evaluate Parameters Required in the SEA Prediction Method, ESA Contract report on 4100/79/NL/PP. Rider1, 1981
6. M.Balaskó, E.Sváb, Dynamic neutron radiography instrumentation and applications in Cental Europe, Nuclear Instruments and Methods in Physics Research A377(1996)pp.140-143.

This page has been deliberately left blank

Page intentionnellement blanche

Life Cycle Cost Modeling and Simulation to Determine the Economic Service Life of Aging Aircraft

Dr. K.R. Sperry
The Boeing Company
2401 Liberty Parkway, Oklahoma City, Oklahoma 73034, USA
kenneth.r.sperry@boeing.com Telephone (405) 739-1402

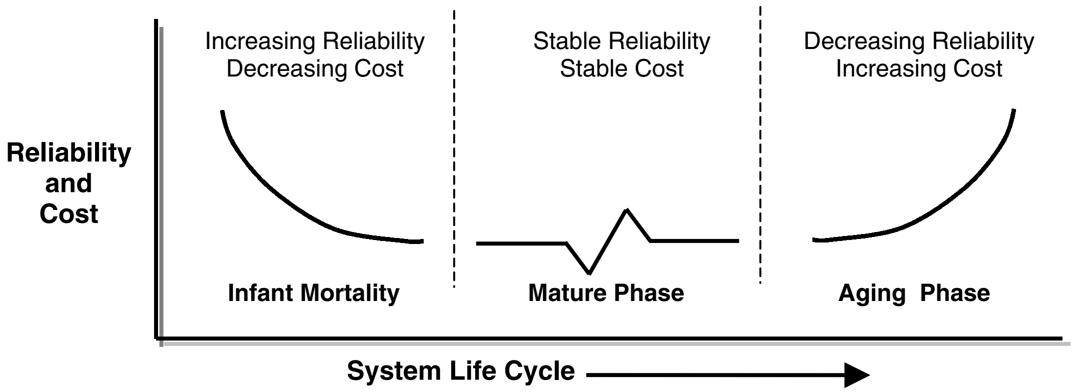
Major K.E. Burns, Ph.D.
United States Air Force
1113 Danzig Dr., O'Fallon, Illinois 62269, USA
kevin.burns@scott.af.mil Telephone (618) 229-4296

Abstract Estimating the point at which the advantages of a modern aircraft alternative exceed the economic burden of maintaining aging aircraft is very complex. This paper presents a cost estimating methodology to forecast costs associated with maintaining an aging aircraft fleet, by combining traditional Operation and Support (O&S) cost elements from a USAF AFI 65-503 CORE model, with expert analysis to quantify maintenance cost growth due to aging. The result is an Economic Service Life (ESL) model that can be used to determine the economic service life of an aircraft. The uncertainties associated with long-range forecasting are considered by combining range estimates within a Monte Carlo simulation for each critical input variable. The model's cost output then becomes a useful fleet management tool to evaluate potential fleet costs while varying annual flying hours and/or aircraft inventory and aids in the evaluation of modernization/ retirement scenarios. Cost output from the model is presented in Constant-Year (CY), Then-Year (TY) and discounted or Net Present Value (NPV) dollars to allow further economic decision analysis.

Background Cost analysts often describe Life Cycle Costs as following a "bathtub" cost curve, which is generally related to the more common reliability bathtub[1] curve. This is defined by a system experiencing early failures during the "burn-in" or "infant mortality" phase due to manufacturing and design defects that are gradually remedied. The next phase is defined by a long period of operation with stable and predictable maintenance costs during the "mature" phase. After the system reaches a certain age, defined by cycles, flying hours, or calendar years, failures and costs begin to rise during the wear-out or aging phase. This later phase is attributed to cumulative component stress, corrosion and general deterioration of the system.

[1] J.W. Langford, Logistics Principles and Applications, McGraw Hill 1995

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).



Intuitively as an aircraft ages, like health care in people, maintenance costs increase during the aging phase. For aircraft, the onset of "Aging" is defined as when a system reaches its Designed Service Objective[2] (DSO). DSO has been defined by the FAA and various aircraft OEMs as 20,000 cycles, 20,000 flight hours or 20 years, which ever comes first.

To determine the economic service life of an aging aircraft the most probable "status quo" cost forecast baseline must be compared to the cost baseline of the alternative(s). In order to project costs forty-years into the future, of an aircraft already twenty to forty years old, the analyst must quantify the cost growth of two primary areas: 1) maintenance[3] and 2) modifications. Additionally, the cost baseline of the alternative(s) must also consider cost growth in these same two areas, given their maintenance costs will similarly increase with age.

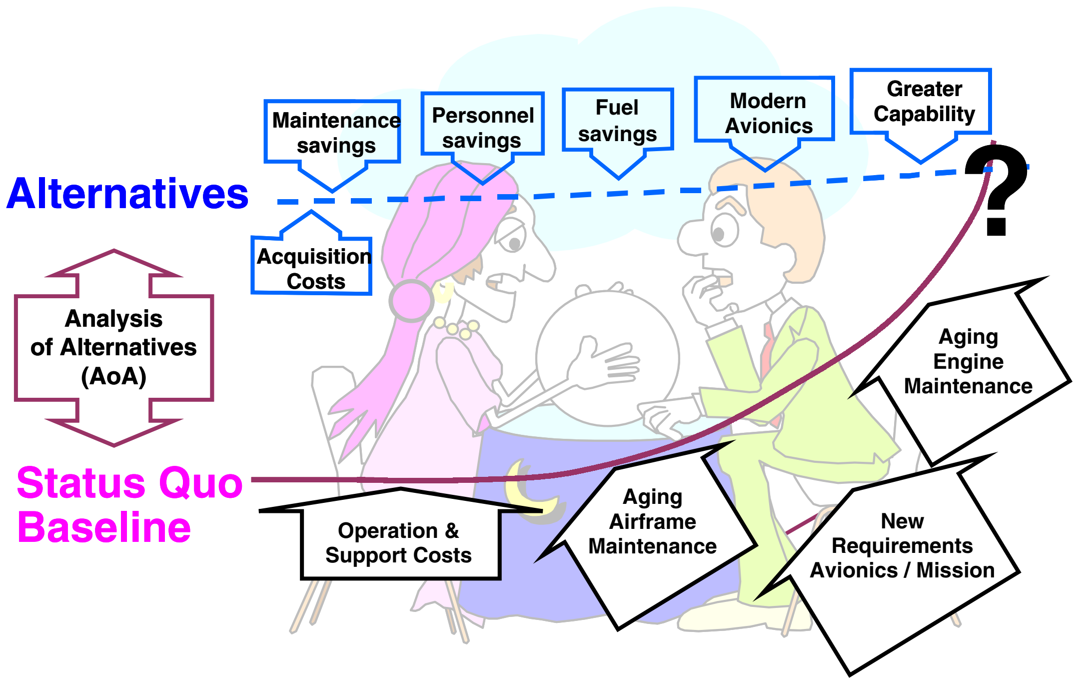
Traditional Air Force Operating and Support (O&S) cost estimating models have never been tasked to provide a forty-year forecast, especially for an aircraft already forty years old. Tasks of this nature have previously been addressed by merely replicating the current year O&S costs for forty years, and declaring this omission in the Ground Rules and Assumptions. The concept of potentially operating an aircraft for eighty years has previously not been an issue, however many designs from the 1950s are now being studied using modern Durability and Damage Tolerance Analysis (DADTA)[4] techniques. Today with a combination of advanced structural inspection techniques, limited budgets, and aggressive modernization programs, Service Life Extension Programs (SLEP) are being considered for many aircraft. The following overview of recently developed economic service life modeling techniques is offered as an evaluation tool to aid in the decision making process.

---

[2] M. Didonato, G. Swears, The Economic Considerations of Operating Post Production Aircraft Beyond Design Service Objectives. Presented at the Aircraft Heavy Maintenance and Upgrade Conference, The Boeing Company, December 4, 1997.

[3] R.C. Rice, Considerations of Fatigue Cracking and Corrosion in the Economic Service Life Assessment of Aging Aircraft, The Battelle Corporation, November 10,

[4] Dr. Hal Burnside, "Flying Longer with Confidence," Technology Today, September 1993, Vol. 14, No.3



Rather than simply answering which alternative(s) has the lowest costs, an Economic Service Life (ESL) model was developed to provide the capability to evaluate multiple and simultaneous "what-if" scenarios. This model would allow changes to both the "status quo" cost baseline as well as each competing alternative. The capability to evaluate operational changes such as the number of aircraft, annual flying hours, personnel to aircraft ratios (crew ratios), as well as changes in estimated maintenance and modification requirements are included. Capability to perform sensitivity analysis by varying both model inputs and the uncertainty associated with each model input was included. The uncertainty of each model input, as well as the model input values themselves, were developed by an Integrated Process Team (IPT) consisting of aircraft industry experts, aircraft operators, and aircraft maintainers who studied and evaluated relevant historical events. Additionally, a thorough review of relevant aging aircraft cost growth studies were evaluated and found to complement the findings and cost output established by this ESL model.

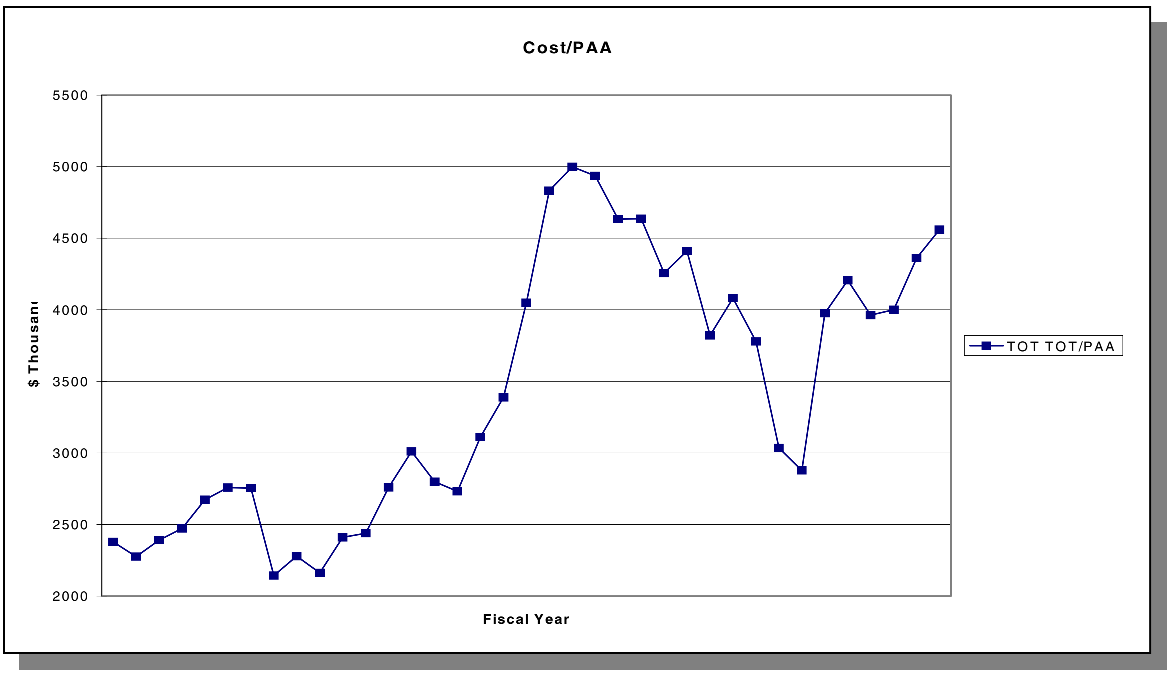
Cost analysts working with aging aircraft recognize the need to account for cost growth as a function of equipment age. Cost growth, even in a Constant Year (CY) dollar analysis is necessary to estimate the real growth in both maintenance and modification requirements.

<u>Data Analysis</u>

Cost forecasting of aging aircraft is a difficult business. Historical data can be hard to come by and "useful/relevant" historical data rarer yet. One must exercise caution however, even when good historical data are obtained. Numerous problems can arise in forecasting aging aircraft costs.

A relatively easy approach is to fit a statistical model, such as a regression model, to the historical data and then project into the future using the fitted model. The analyst makes a number of assumptions when this approach is taken. Perhaps the most basic assumption is that no underlying circumstances surrounding the aircraft system have substantially changed throughout the life of the aircraft and that there will be no significant changes in the future. Conversely, one must be assured that the processes that shaped the historical data will continue into the future.

Often this assumption is not valid. Many changes occur to an aircraft as it ages. A simple example can illustrate this point.

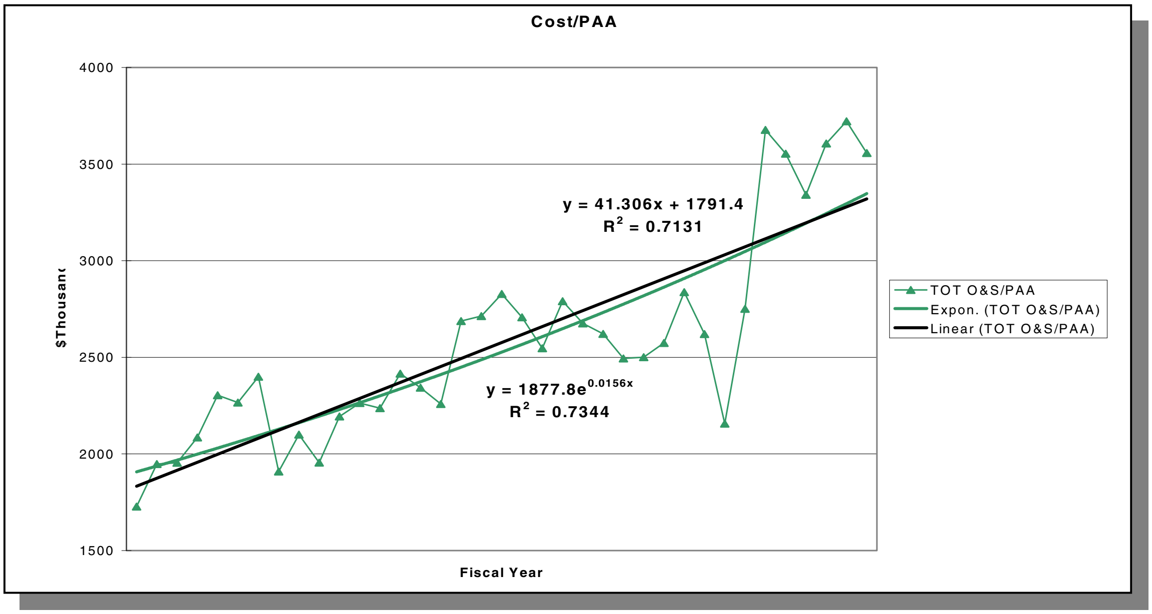


In this example, there are at least two problems with the assumption that nothing changed throughout the life of the aircraft. The first is the large "hump" approximately 2/3 into the time span indicated. This "hump" can be attributed to a very substantial modification program. The aircraft may or may not experience a similar modification again. A second problem is with the tremendous single year increase observed directly after the modification program. This increase is attributed, in part, to substantial accounting changes in how costs were attributed to the aircraft system.

If the aforementioned assumption is valid, there are still difficulties to overcome. Even if the analyst limits himself to regression models, there are important choices to make. There are at least two different methods of describing growth rates in historical data: linear and exponential. Often, the fit, as measured by R^2, the coefficient of determination, can be quite close. However, projecting out over long periods of time in the future can result in tremendous differences between a linear growth rate and an exponential growth rate.

As an example, consider the following historical data on an aircraft system.

The R^2 values for the two models are very close. It is not obvious which model fits the historical data better. If the analyst chooses to project one or the other of these models, the choice is an important one; the difference in projecting out these two models is quite



large. For this example, after 20 years the exponential growth projects an annual expenditure 10% higher than the linear growth model. After 40 years, the exponential growth projection is 25% higher.

Finally, finding a metric that is fair to compare over lengthy time intervals also poses difficulties. Analysts often recognize the need for normalizing cost data to compensate for differences in fleet sizes and/or flying hours over time. While this is understandable, it is not always so easy to do. Cost per flying hour is a common metric that is used to track cost trends over time. This metric can be misleading for certain aircraft fleets; this is particularly true for aircraft fleets that have relatively high fixed costs due to low utilization rates.

Again, an example is helpful.

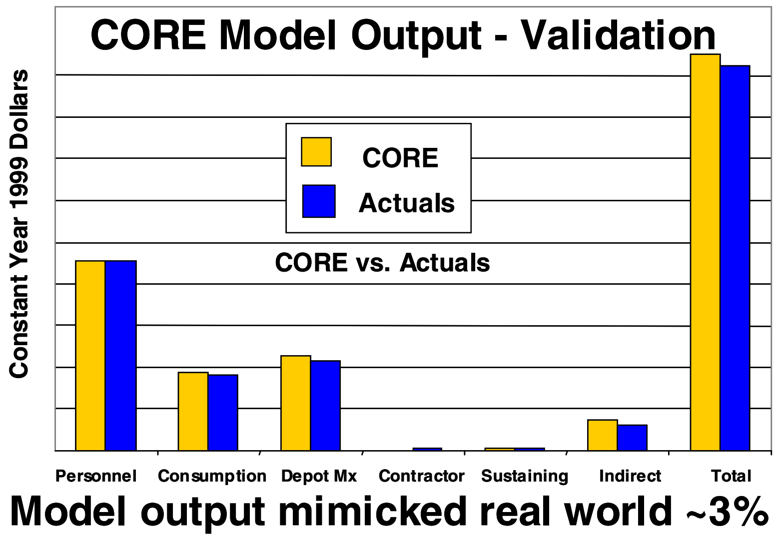
Then Year Dollars	**1996**	**1997**	**1998**	**1999**	**2000**
Total Cost $	$1,486,300,237	$1,508,306,188	$1,722,946,676	$2,141,593,513	$2,004,591,495
Total Flying Hours	213,885	209,755	210,118	212,953	175,330
Cost Per Flying Hour $	$6,949.06	$7,190.80	$8,199.90	$10,056.65	$11,433.25

In the above table, note that the total costs increase 35% in four years. However, if the analyst chooses to report cost per flying hour, note that costs increase a staggering 65%!

These types of issues and other need to be taken into account for any ESL model.

<u>Developing a CORE Model</u>

At the heart of the ESL model is a Cost Oriented Resource Estimating[5] (CORE) model. The CORE model is used by the USAF to develop Operating and Support (O&S) costs estimates. Model output can be used for either budgeting/programming exercises or Life Cycle Cost (LCC) studies. Standard model inputs are obtained from manual look-up tables, which are updated and published annually based on fact-of-life budget realities from the previous year. Additional model inputs in the form of Cost Estimating Relationships (CERs) tailored for the specific Mission Design Series are required to be developed.



CORE model output is provided in Cost Analysis Improvement Group[6] (CAIG) hierarchical cost structure. CAIG structure defines O&S costs as: 1) Mission Personnel, 2) Unit Level Consumption, 3) Intermediate Level Maintenance, 4) Organizational Maintenance, 5) Depot Maintenance, 6) Contractor Support and 7) Indirect support. The USAF CORE model was used primarily for it's capability to provide O&S costs in CAIG format which are directly comparable to cost output from the Air Force Total Ownership Cost (AFTOC) reporting system. Validation of CORE[7] model output against same year AFTOC costs is recommended to insure realistic output and model accuracy. Any significant deviations between model output and the AFTOC / ABIDES "reality check" must be explained.

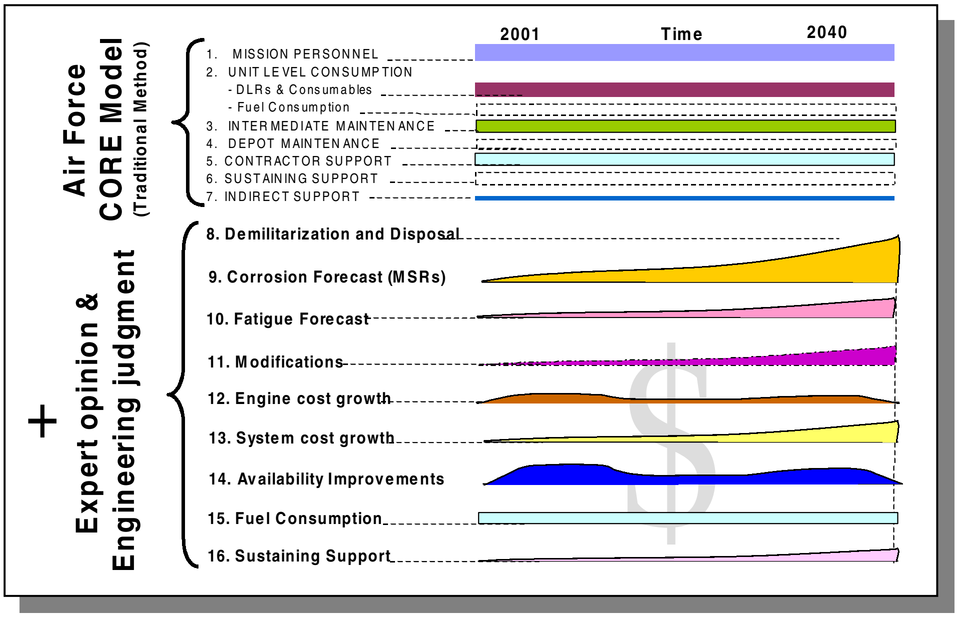
---

[5] Air Force Instruction 65-503, Attachment A54-1, 31 October 1994, http://www.saffm.hq.af.mil/

[6] The Air Force Total Ownership Cost (AFTOC) management information system Cost Analysis Improvement Group (CAIG) format identifies all costs (direct and indirect) to both CAIG elements and sub-elements and to the appropriate major system or aircraft Mission Design Series (MDS) by MAJCOM, Numbered Air Force (NAF), Unit (Wing), and Base. https://aftoc.hill.af.mil/aftocmis/default.asp

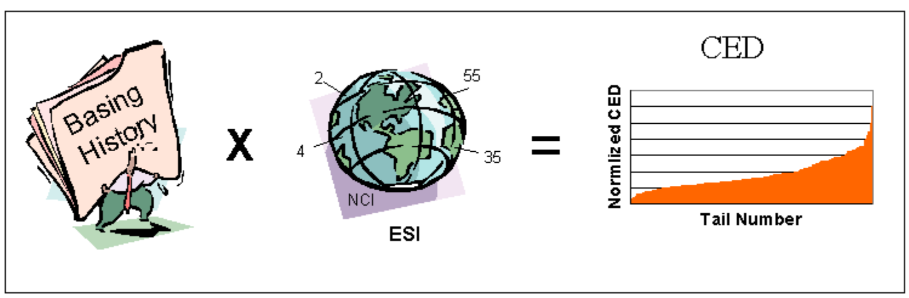
[7] "The Air Force Total Ownership Cost (AFTOC) Management Information System responds to the Secretary of Defense's Year 2000 goal for each Service to develop a system to provide senior leadership '…routine visibility into weapon system life cycle costs.' Additionally, it supports the acquisition community in meeting the Defense Systems Affordability Council direction to the Services' Senior Acquisition Executives to '…establish aggressive, time-phased TOC reduction goals.' By completion of the third phase of AFTOC development, the system will provide detailed cost information on all major weapon systems, inclusive of aircraft, space systems, and missiles. The AFTOC system, when fully implemented, will be the authoritative source across the Air Force for financial, acquisition, and logistics information." .... SAF/FM

This list of CAIG cost elements is then augmented by the addition of Demilitarization / Disposal costs, plus expert opinion and engineering judgment of "Aging related" costs. Aging maintenance costs were grouped in the major cost elements of: airframe corrosion, airframe fatigue, modifications (both structural and non-structural), engine cost, aircraft systems costs and aircraft availability improvements. An additional cost category identified as "unknown-unknowns" was also added due to the uncertainty of long-range cost forecasting.



Expert Cost Estimates

One example of expert cost estimates used to augment the CORE model is estimating the cost due to Major Structural Repairs (MSRs). MSRs due to corrosion were estimated by first establishing the relationship between an aircraft's Cumulative Environmental Damage (CED) and

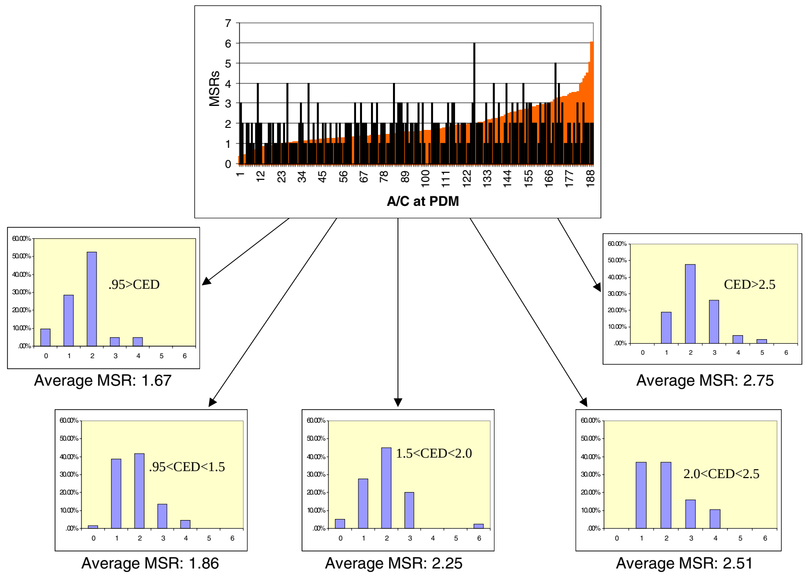


documented MSRs. Each aircraft's basing duration (in days) was multiplied by the Environmental Severity Index[8] (ESI) for the location of the aircraft. The product is an ordinal index, ranking aircraft by their exposure to corrosive environments. To validate the accuracy of this index, a cumulative MSR count based on Programmed Depot Maintenance (PDM) records were matched to specific tail numbers to calibrate the index. This matching was hampered by the relatively small population of reliable maintenance

[8] Environmental Severity Index produced by NCI

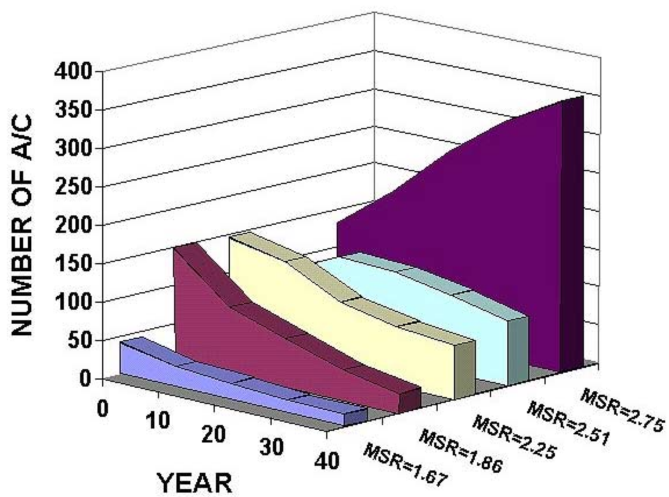
(PDM) records. The PDM data available represented only 15 years from only one of three PDM facilities.

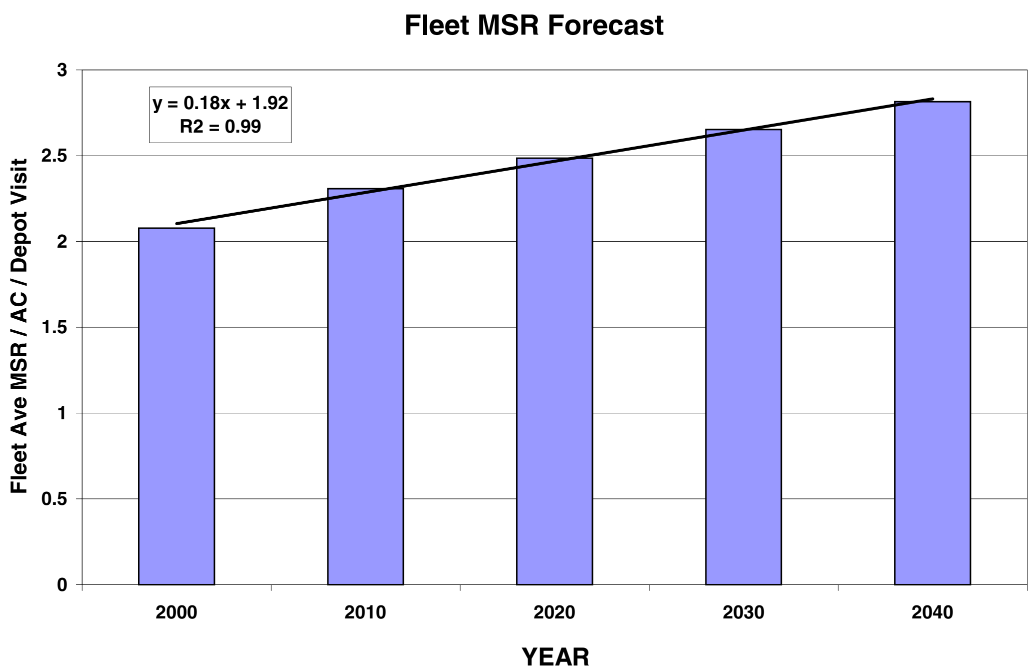
With a relationship established between calculated CED and MSRs the cumulative fleet environmental damage was then calculated by advancing the fleet's age forward in time according to the current aircraft basing assignments and forecast aircraft rotation plans. The fleet age was advanced in 10-year increments to account for the CED. Damage due to fatigue for this application is insignificant due to the very low annual utilization.



Note how the fleet population shifts from a high percentage of the fleet population in a low MSR category (average of 1.67) to a high MSR category (average of 2.75) with the passage of time.

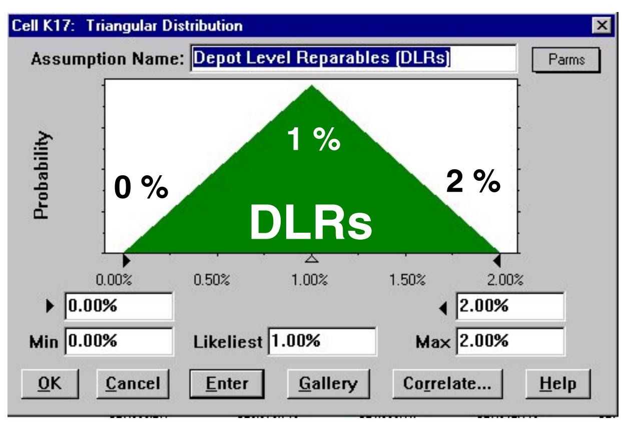
Major Structural Repairs are estimated to increase from approximately 2.0 MSRs per aircraft per depot visit (5 year interval) to approximately 2.75 MSRs per aircraft per depot visit.   This maintenance growth rate considers the fact that the MSRs that are now being experienced based on the first forty years of service, will likely not be necessary for another forty years. The repairs performed today use newer technology materials and modern installation practices with corrosion resistive properties.





Uncertainty in cost forecasting

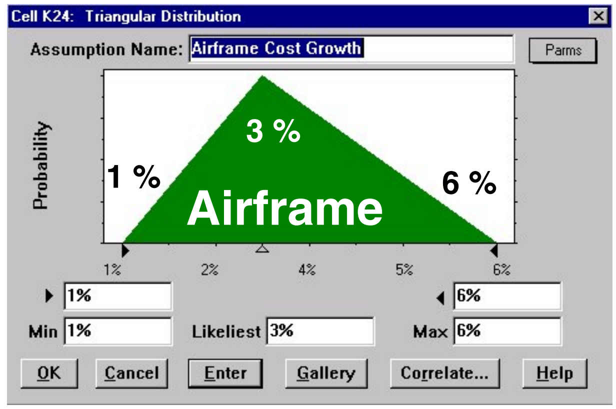
From the aforementioned discussions, it is apparent that an analyst has to deal with uncertainty. This is true in dealing with historical data as well as future projections. Anytime an analyst makes a projection, it is important to provide some idea of the variance of that projection. This can often be accomplished by providing a range of estimated costs. A very common way of dealing with this is to use probability distributions to model the uncertainty inherent in forecasting long-term cost estimates.



When estimating the economic service life of an aircraft fleet, the analyst typically must estimate many different costs. Because each of these estimates involves uncertainty, providing an overall range estimate of the total cost can be difficult. This task is made easier by spreadsheet tools that allow the analyst to take advantage of Monte Carlo techniques. Monte Carlo can be described as a method for estimating the answer to a problem by means of an experiment with random numbers[9]. The idea is to simultaneously vary several different inputs in a model to obtain the final output; this process is repeated many times to produce a distribution of final outputs. The average and variance of this distribution can be used to make a range estimate of the total cost.

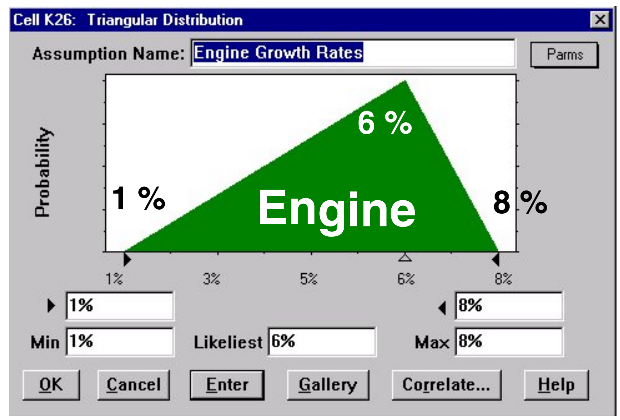
[9] E. S. Quade ed., An Appreciation of Analysis for Military Decisions, 1966

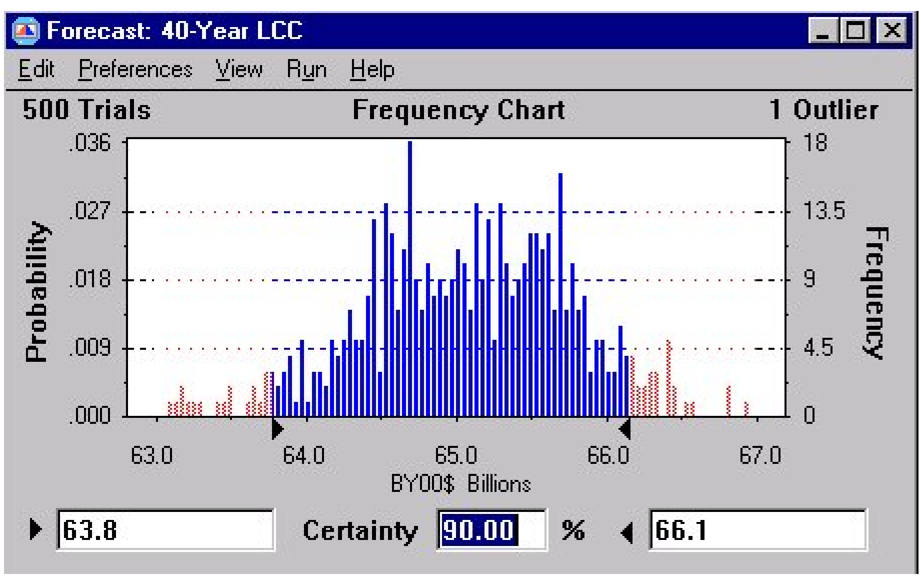
Sample input variables for Depot Level Repairables (DLRs), airframe maintenance, and engine maintenance are presented in terms of annual cost growth. Each input variable identifies the minimum, maximum and most likely values.



When all of the inputs are varied simultaneously, an overall cost estimate of the entire system is produced. Performing this operation numerous times produces a distribution of final life cycle cost estimates. This distribution allows the analyst to produce range estimates of the overall life cycle cost, allowing for the inherent uncertainty in the system.

A sample output from the ESL model is included below.

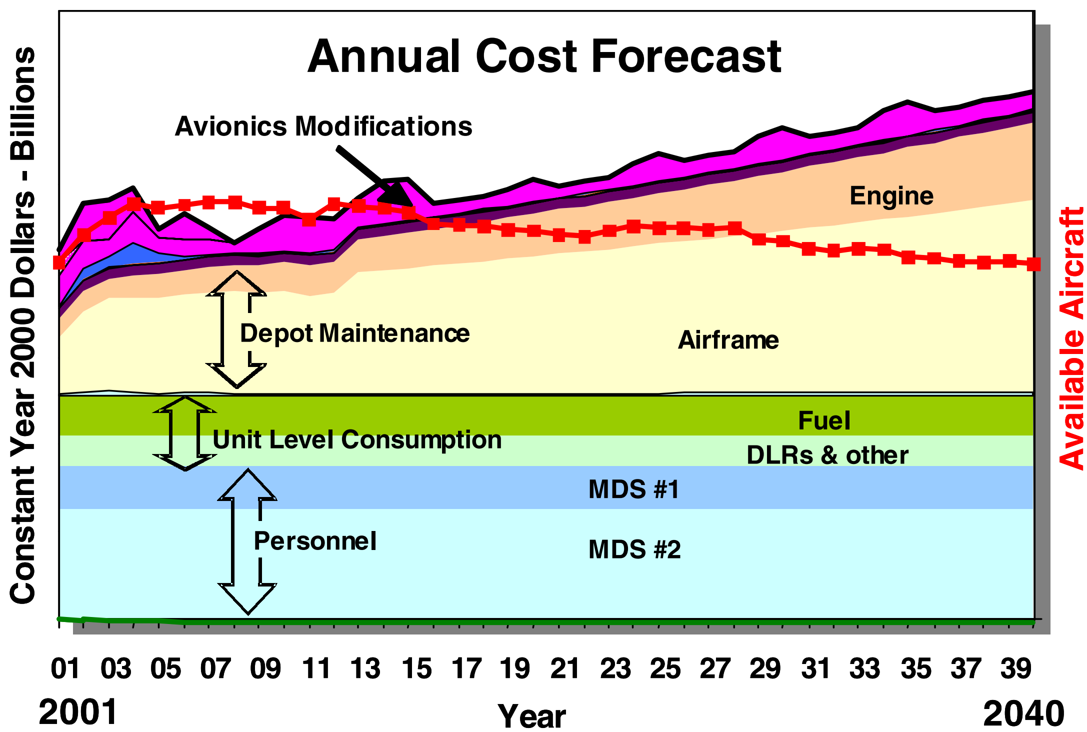




Economic Analysis

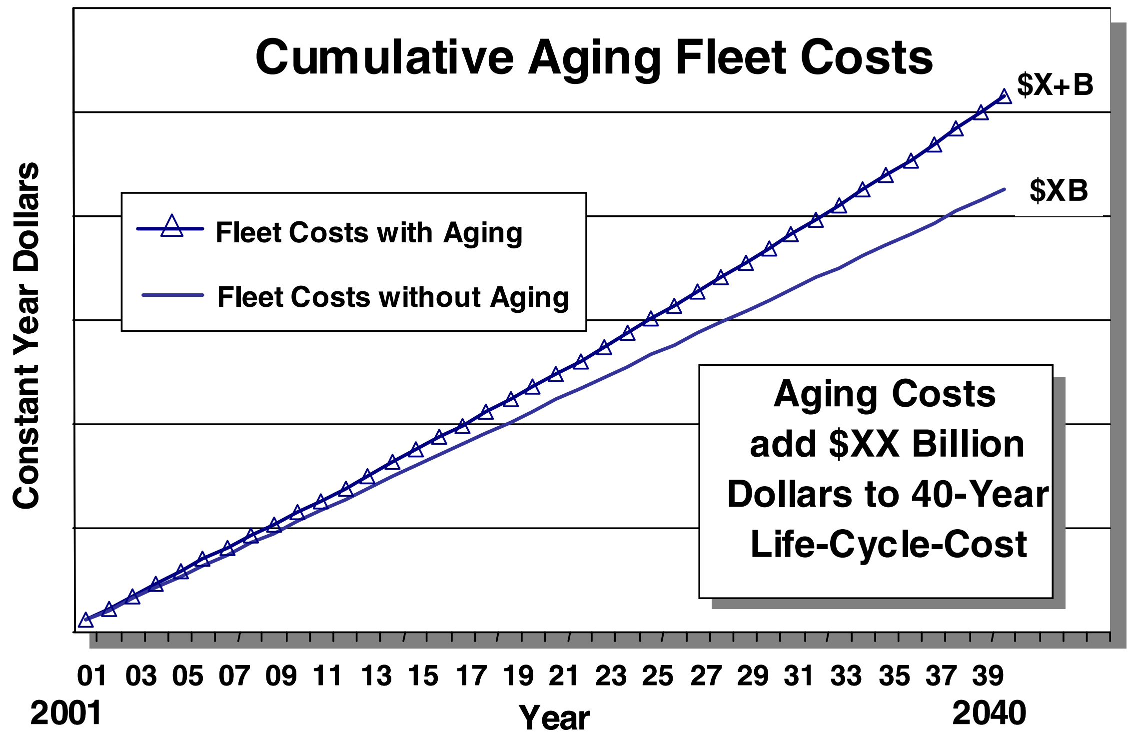
By definition an analysis of alternatives (AoA) is an analytical comparison of the operational effectiveness and cost of proposed materiel solutions to shortfalls in operational capability[10]. In the case of deciding whether or not to procure a new fleet of aircraft, an AoA requires comparing the costs of the new fleet to that of operating the current fleet. In the case of replacing a large fleet of expensive aircraft, this process will likely take place over many years. The result of this is that it would be helpful to have a tool that allows the analyst to perform several "what if" scenarios. These scenarios would naturally involve changing the number of aircraft and flying hours in the old and new fleets over time.

An AoA requires a cost model that includes the ability to model research and development costs, procurement costs, operations and support costs, and disposal costs. Traditional Air Force cost modeling (USAF AFI 65-503 CORE) can help in this respect. However, because of the need to compare costs over time, it is necessary to have a model that takes into account aging aircraft effects.



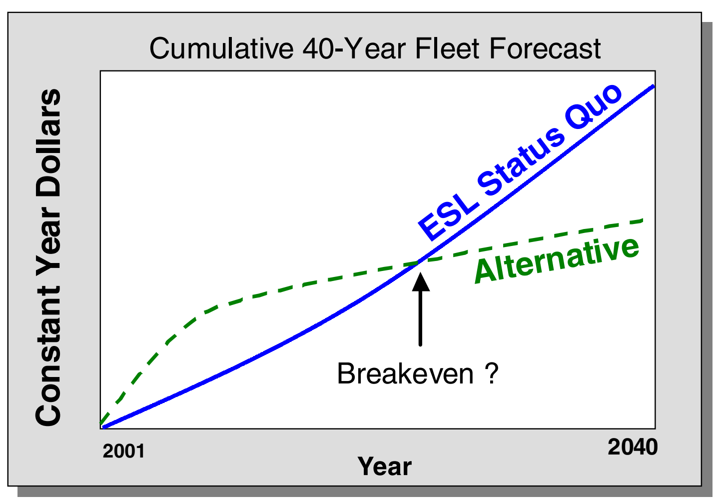
Taking all of the above observations into account, one of the goals of an AoA is to determine the economic service life of the current aircraft fleet. To determine the economic service life of an aging aircraft fleet, the model must be able to project the increasing costs of the current fleet and then compare that to the costs of a potential new fleet. The following two graphs depict what type of output the ESL model is capable of producing.

[10] Office of Aerospace Studies, AoA Handbook, June 2000



The ESL model combines the many elements that a cost analyst needs to help answer the complicated question of when an aircraft has reached its economic service life. This model will be extremely useful to Air Force analysts who are conducting AoAs on aging aircraft fleets.

One of the strengths of this ESL model outlined herein is the built-in flexibility. As new information becomes available over time, the analyst can update the cost estimates with the very latest information. One of the most important things to realize in estimating costs over a long period of time is that the estimates will certainly change with time. This model gives the analyst that flexibility to refine estimates as new information becomes available.



The bottom line is that this tool allows the analyst to provide the best information available to the decision maker in a useable format.

The Joint Strike Fighter (JSF) PHM and the Autonomic Logistic Concept: Potential Impact on Aging Aircraft Problems

Capt Simon Henley
UK Royal Navy
JSF Program Office

Mr. Andrew Hess, Mr. Leo Fila
Naval Air Warfare Center
Aircraft Division
22195 Elmer Road, Bldg. 106
Patuxent River, MD 20670
United States

ABSTRACT:

The JSF Autonomic Logistics (AL) system is a new supportability concept that will enable the aircraft to be better utilized throughout the life of the platform, and at a lower cost as compared with legacy aircraft. Autonomic Logistics is, simply put, the automation of the logistics environment such that little human intervention is needed to engage the logistics cycle. Actions that will be automated within the JSF supportability concept include maintenance scheduling, flight scheduling, ordering spare parts, and the like. The cornerstone of Autonomic Logistics is an advanced diagnostic and Prognostics and Health Management (PHM) system. The PHM provides the data, information, and knowledge for initiating the AutoLog chain of events. PHM is the ability of the aircraft to do fault detection (FD), fault isolation (FI), and accommodation real-time on-board the aircraft. Along with this FD/FI capability, some of the other facets of PHM include fault prediction on selected components, parts life usage tracking, fault filtering and reporting, and recommended action to the pilot only when action is necessary. It is intended that in most cases, maintenance actions and lifing decisions will be based on the actual material condition of the equipment components. The proposed architecture of this PHM concept includes a hierarchical approach where data begins at the sensor level and is transported up to area reasoners that turns this data into information about a particular subsystem. From the area reasoner, the information is then passed up to a top level Air Vehicle Reasoner where subsystem information is then fused to give knowledge about the health of the entire air vehicle. Additionally, many of the technologies that are being developed for the JSF PHM suite could be applied to legacy aircraft and would show significant benefits in respect to LCC and maintainability issues. In many cases, the capability of PHM sensors and prognostic technologies will enable the ability to "see" incipient faults in subsystem components very early prior to their progressing to final system failure. These capabilities will enable accurate useful life remaining predictions and more aggressive health management of assets. Through very accurate tracking of the life usage data for component parts, the JSF PHM and AL will be able to anticipate many problems that plague current legacy aircraft; giving a lead time to provide fixes before the actual problems become fleet wide and catastrophic. The PHM architecture will directly interface with the Joint Distributed Information System (JDIS), which is the information system that will enable the Autonomic Logistics functions. The JDIS could automatically forward to the Original Equipment Manufacturer (OEM) data on problems that arise within the fleet, thus alerting them to a developing situation sooner, and enabling them to provide faster, cheaper fixes to these problems. With these safeguards in place, the JSF will put itself in a position to not only quickly react to ageing aircraft problems, but to avoid many of them before they happen.

BACKGROUND:

In today's world, as defense budgets continue to shrink, it is becoming increasingly more difficult to maintain a constant level of force structure through new acquisitions. A direct result of these shrinking budgets and decreased purchases is a current fleet of increasingly aging aircraft. We are forced

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

to use the existing fleet assets much longer that originally projected and more often, in new and unanticipated roles. No longer will mass purchases of many models of new aircraft ensure that there are plenty of assets to meet all operational requirements. Much of today's aircraft fleet is well over 20 years old, which presents a new variety of problems in maintaining those aircraft that are nearing their operational life when there is no immediate sign of a replacement. All too frequently degraded equipment, assisted by inadequate maintenance practices, lead to preventable incidents in which people are injured or killed. This situation also ensures additional and very high operational and support costs; since older systems are less reliable, and harder and more costly to maintain.

A discussion of the Joint Strike Fighter's PHM system will be presented in this paper, with an emphasis on its capabilities and its aims to enhance aircraft safety; significant reduce operation and support costs; as well as minimize the impacts of the enviable aging fleet. The PHM system is, by design, envisioned to provide an information rich and highly intelligent aircraft. A major tenet of this discussion will be that robust information sources provided by the PHM system can and will be used to mitigate aging aircraft problems.

MAIN:

AUTONOMIC LOGISTICS

The JSF program is in a unique position to fully take advantage of modern technologies to significantly reduce operational and support cost as well as to be able to design the air vehicle with these aging aircraft problems in mind. In order to do this, the JSF program has devised a revolutionary new support concept called Autonomic Logistics. The aim and over-arching strategy of Autonomic Logistics is to provide a comprehensive logistic support environment for the JSF by including the following key features:

A highly reliable and maintainable (R+M) designed intelligent aircraft which encompasses a comprehensive Prognostics and Health Management (PHM) capability to enhance flight safety, improve efficiency of the logistics chain, and allow scheduling of logistic events to compliment operational planning.

A technologically enabled maintainer who, through the use of innovative and automated tools and technical publications, will be capable of efficiently and effectively maintaining the JSF with less specialized training and more "on the spot" training. This will be accomplished by the use of Interactive Electronic Tech Manuals (IETM's) and will allow the use of fewer maintainers, cross trained over many sub-systems.

A fully capable Joint Distributed Information System (JDIS) that incorporates advanced information technology to provide decision support tools and an effective communication network linking the JSF with the logistics infrastructure to provide proactive support.

A logistics infrastructure that is sufficiently responsive to support requirements within a timeframe that allows the JSF weapon system to generate the required number of effective sorties at the least cost.

These four elements of Autonomic Logistics each have a vital role to play in establishing a new paradigm of affordability of the Joint Strike Fighter weapon system for the 21st century. This paper discusses these elements and how they will be used to mitigate the problems associated with an aging aircraft fleet. Figure 1 shows a notional for how these four elements come together to "hold up" the overall view of Autonomic Logistics.

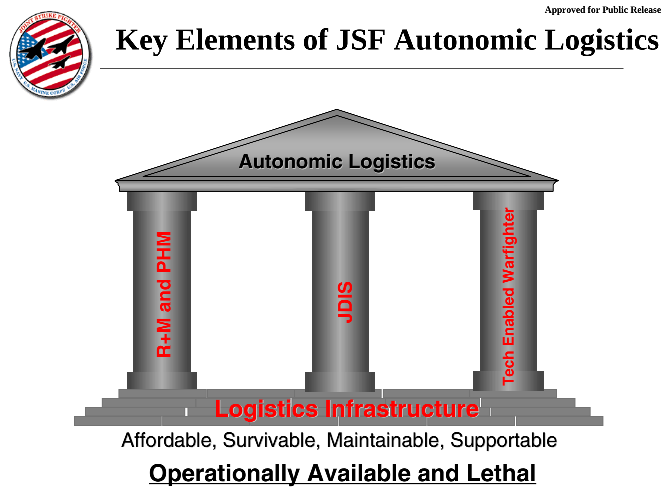


Figure 1 Autonomic Logistics Concept

PROGNOSTICS AND HEALTH MANAGEMENT

The Joint Strike Fighter PHM concept is the cornerstone of Autonomic Logistics. The system incorporates not only an integrated and comprehensive diagnostics system, but also prognostics, and an inclusive information system that utilizes decision support tools for the users. The key underlying theme of the JSF's PHM system is a minimal number of specialized sensors to be used in conjunction with an "area manager" architecture. These area managers contain software reasoners, or software modules in which data from various sources is fused together through means such as fuzzy logic, data fusion, neural nets, model based and case based reasoning and the like in search of anomalies or trends which may indicate defective or faulty parts. The idea is not to overload the aircraft with an abundant amount of sensors which tend to fail and induce an added ambiguity when attempting to isolate faults and failures, but to utilize as many already present aircraft parameters as possible with assistance coming from various strategically placed sensors.

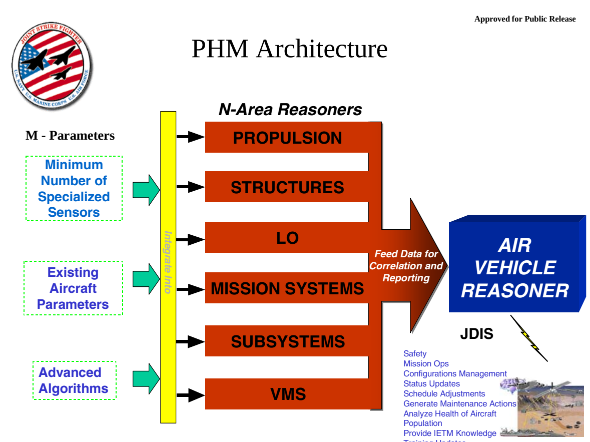


Figure 2: JSF PHM Architecture

Each area manager continually monitors a particular subsystem of the aircraft. Thus we have area managers for propulsion, structures, utilities & subsystems, vehicle management system, and mission systems. Having its own computing capability and software algorithms, the Area Manager automatically analyzes the signals from the sensors and other data sources to determine whether the devices or components of the given subsystem are behaving properly, or are exhibiting characteristics which may lead to failure. The JSF software architecture is constructed to minimize ambiguities. All of the subsystem area managers provide their output to a single air vehicle manager for further fusion of information across the air vehicle and elimination of ambiguities. This information is then filtered to send to the pilot only that information which he can use and act on, and the rest is conveyed to maintenance personnel for action (figure 2). The purpose of this software architecture is twofold: first, it allows the diagnostic system to perform more functions without the introduction of an abundant amount of specialized sensors. On today's digital aircraft, there is already an enormous amount of data present on flight data recorders and MUX busses. It is the job of the software architecture to turn this data into actual information that can then be used in the maintenance of the aircraft. Second, by fusing data from various sources, anomalies can be cross-checked with information from other subsystems in order to corroborate or dispute a potential alarm. With this concept, no longer will alarms be triggered based upon the input of a single sensor. All sensor data will be validated in the Area Managers and Air Vehicle Manager with other sensor data or aircraft parameters in order to verify the fault. Data filtering will be accomplished with new technology employing coherence analysis that detects component performance deviations from normal. Other non-critical information will be digitally stored for downloading upon landing.

Another aspect of PHM is to perform many of the prognostic calculations, remaining useful life calculations, cycle counting, and lifeing of components. This processed information, along with the rest of the information taken from the air vehicle is then passed along to the JDIS to inform the supply chain what it has to do to keep the airplane operationally effective. This is where many of the benefits will be realized. Prognostics, the way the JSF plans to implement it, has one overarching objective: to ensure that there are no surprises when maintaining the aircraft. Prognostics will occur across the air vehicle, just like the diagnostic system. From flight control systems to rotating machinery in the engine, work is ongoing to attempt to predict the life of all mission and flight critical parts of the aircraft, as well as the life of any parts where the technical risk and benefits can justify the undertaking. The metric to be used in this undertaking is probability of failure within a specified number of flight hours. For example, it is the goal that the JSF will be able to prognose that a certain hydraulic pump has a 90% chance of failing within the next 10 flight hours. This way, maintenance personnel will be able to make intelligent, informed decisions regarding the removal and replacement of parts. Another aim of the JSF prognostics system is to have an estimate of remaining useful life of a component at any given time that the component is on board an aircraft. With a new part, this number will most likely be the time set forth in the current RCM method; however, as the component approaches the removal point, prognostic algorithms will adjust the remaining life based upon how that part is being used. For instance, an aircraft structural member would exhibit a much longer useful life if the plane were primarily used for ferry missions as would the same member on an aircraft that was performing a lot of high G maneuvers. The prognostic algorithms will be able to account for this and adjust maintenance schedules accordingly.

Prognostics will allow a lead time for the logistics pipeline to get parts and to train maintainers to change those parts. Also, it will allow for "opportunistic maintenance", or the act of grouping maintenance actions at a single time while an aircraft is already down. For instance, a hypothetical aircraft is down for a routine engine wash. While it is being attended to, the prognostics system informs maintainers that the primary auxiliary power unit (APU) has begun to degrade and will need to be replaced within the next 15 flight hours. It also informs the maintainers that the oil in the engine is beginning to show signs of coking and has an undesirably high content of fragments. Hence, all three maintenance actions can be taken care of with a single downing of the aircraft, vice three separate maintenance actions which would keep the plane out of commission for some time. As an additional benefit, these maintenance actions would keep the APU from failing at a later time, which may have been a very critical time in a mission being flown.

Another aspect of Autonomic Logistics, briefly mentioned above, is part tracking. With Autonomic Logistics, all component parts will be tracked by serial number across all aircraft tail numbers. This will assist in catching potential fleet-wide problems before they become problems. An example would be if the Navy were to switch bearing manufacturers for some reason (cheaper, original manufacturer went out of business, etc…). Suppose new bearings were not performing up to standard, at the first sign of a bad trend, the JDIS would be able to identify all bearings from the given shipment(s) and their locations and pull them before they started failing while in operation. This would avoid having to ground an entire fleet in order to check all bearings on all aircraft to identify the faulty ones.

JOINT DISTRIBUTED INFORMATION SYSTEM

The "backbone" of Autonomic Logistics is the Joint Distributed Information System (JDIS), and it works hand in hand with PHM. While the importance of fault prediction, detection, and isolation cannot be over-emphasized, without a means of properly communicating this information, the actual realized benefits can be much less than anticipated. JDIS provides the conduit that takes the prognostic and diagnostic information transmitted by the aircraft and determines from it the manpower numbers, capabilities and training requirements to complete the necessary tasks. Some tasks that will be automatically performed through and/or integrated with JDIS are: mission planning, maintenance action scheduling, ordering of spare parts, scheduling of flight and maintenance training, assignment of specific pilots to specific missions based upon experience and readiness, assigning specific aircraft to specific missions based upon aircraft availability and capability, and storing maintenance, training, spare part, and logistic information in the data warehouse. It is important to note here that while all of these tasks will be done automatically, at any given time a person with the necessary authority can access the system and make changes as required . By having this requirement, JSF personnel will not have to live with poor decisions that were made because of programming "glitches" or lack of foresight of the programmers.

The information fusion capability of the PHM system will allow JDIS to output and pass on actions and recommendations rather than just data. These decision support tools will include: maintenance information, supply chain management information, health and usage information, training management, and recommendations regarding best use of resources. JDIS will provide the "backbone" information distribution system to integrate PHM supplied data and information with all the other necessary maintenance management, logistics, supply, OEM, mission planning, etc., data bases required to ensure the most fully informed decision process possible. Figure 3 depicts an envisioned example of JDIS functionality.

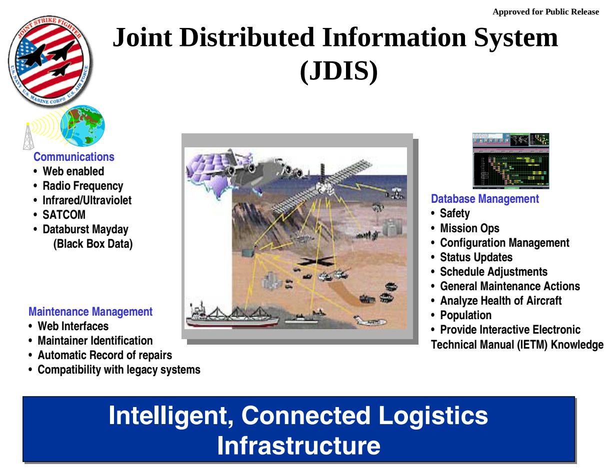


Figure 3 JDIS Functions

TECHNOLOGICALLY ENABLED MAINTAINER

The maintainer in the Autonomic Logistics concept has the full set of modern, technologically capable and appropriate tools at his immediate disposal to enable him to prepare the aircraft for its next and subsequent sorties in the most cost effective and timely manner. Coupled with a truly "World Class" and modern Training Environment, the maintenance technician's capabilities will be greatly enhanced by taking advance of new technology. Some of this new technology will enable truly Interactive Electronic Technical Manuals (IETM's), the Electronic Classroom, Training Continuum/Mission Rehearsal, Embedded Simulation, Virtual Reality feedback tools, On-line Support, etc. The tool sets include: comprehensive knowledge of the actual aircraft health before beginning work (PHM and JDIS), appropriate and timely training to conduct the task (Prognostics lead time, prior training methods), all the necessary material on hand before commencement of work (Prognostics lead time), and interactive guidance available in real time to provide supplementary information as required. The connectivity of the maintainer to the aircraft systems and maintenance management provides the response times necessary to conduct maintenance actions without compromising sortie generation requirements.

It is important to note, however, that the logistics and PHM systems require minimal intervention from the maintainer. The ongoing activities within Autonomic Logistics and PHM should be as transparent as possible to the maintainer due to its automatic nature. Tasks are presented to the maintainer in a manner that provides proper procedures, safety, detail appropriate to skill level, rehearsal/review of the task when requested, tools and parts required and quality assurance. The maintenance management environment must aim to monitor, schedule, and prioritize all maintenance activity (on-condition and scheduled) based upon the data provided by PHM and JDIS. There will be less emphasis on diagnostic skills as a result of the highly capable PHM system being incorporated, but the maintainer must be trained in the proper functional utilization of the Autonomic Logistics environment. A full and comprehensive understanding of the

Autonomic Logistics and PHM systems is critical for the maintainer to develop the confidence in the tools that will dictate maintenance tasks, procedures and the training required.

ADVANCED LOGISTICS INFRASTRUCTURE

The potential advantages embodied in a substantial PHM capability, a technologically enabled maintainer, and a highly capable JDIS, producing accurate and comprehensive logistics information are all of no avail if the logistics infrastructure is not flexible and responsive enough to generate the necessary support in the right place when required. Equally, the importance of making JSF affordable through its life cycle demands a logistics strategy that explores every means of meeting the support requirements in the most cost effective fashion. Some of the issues to be tackled to have the appropriate infrastructure are as follows:

Levels of Maintenance – In order to minimize the logistics footprint and infrastructure costs, the minimum number of levels of maintenance to successfully meet mission requirements are envisioned.

Inventory Policies – Large quantities of spares carry substantial storage overheads, tie up capital and decrease flexibility within a supply chain management system. Just in time inventory becomes more realistic with this type of data sharing and prognostic capability.

Supply Chain Management – Greater contractor responsibility for logistics system effectiveness is likely to require less government intervention in supply chain management. The JSF supply chain will have to interface seamlessly with existing logistics chains for common items at the Front Line. Embarked, forward deployed, and on-base retail supply activities will most likely continue to be conducted by the Services.

These factors, along with a host of other factors will help to realize the potential benefits of Autonomic Logistics. While not directly attributable to reducing traditional aging aircraft problems, an advanced logistics infrastructure plays a role in realizing all of the benefits mentioned above, and thus provides an integral part of Autonomic Logistics.

APPLICABILITY TO AGING AIRCRAFT PROBLEMS :

Aircraft begin to age immediately after they are manufactured and introduced into fleet operations. The JSF PHM system and Autonomic Logistics concept, by vision and design, will provide levels of safety, maintainability, supportability, and affordability never achieved before. Many of the capabilities and benefits provided by this concept and these systems will significantly mitigate and/or avoid many typical aging aircraft problems as the JSF ages. While the PHM system and the Autonomic Logistics concept are being designed and integrated from the ground up with the "brand new' " JSF aircraft program; many their specific capabilities could also benefit older legacy platforms on a case-by-case basis.

With today's troubleshooting methods of aircraft failures, there are many cases in which aircraft parts are replaced and sent to the depot level where they are found to "retest ok" (RTOK). Many times, these parts that have been removed and tested ok find their way back to the flight line only to be re-installed into another aircraft. In these cases, the part may or may not be faulty; however, this adds a fairly large measure of uncertainty as to the ability of that part to perform its function. As aircraft platforms age, the number of these "uncertain" parts being used increases due to the need to re-use them because of the inability to order more (supplier may have gone out of business, too costly to order in small quantities, lead time to order too long, etc…). To further complicate this matter, over periods of time, these parts naturally tend to accumulate in the supply chain, thus making a clear distinction between good and bad parts somewhat uncertain. Implementing the PHM system as described above would lead to a much greater fault isolation rate and, consequently, fewer removals of good parts. This would lead to fewer (ideally zero) RTOK cases and, hence, zero bad parts in the supply chain.

The Prognostics capability of Autonomic Logistics will play a large part to improving the sortie generation rate of an aircraft and promote safety at the same time for aging aircraft. The additional lead time for maintenance actions will prove invaluable, not just in the latter stages of the aircraft's life cycle, but in all stages of the life cycle. As aircraft age, the number of parts reaching the end of their useful life will increase. This lead time will provide a valuable means of managing this natural turnover. By having this capability, the JSF will be kept at mission ready status over the life cycle with minimum amount of unscheduled grounded aircraft to reduce sortie generation capability.

Additionally, with the lead time notice to order more parts, ideally there would be no more cannibalization of aircraft. Many squadrons have what is known as a "hanger queen", or an aircraft that is in the hanger for the purpose of donating needed parts to operational aircraft. This presents a much faster solution than downing multiple aircraft while awaiting shipments, however it depletes the force structure of the squadron.

The Joint Distributed Information System also plays a key role in mitigating aging aircraft problems. By being able to automatically perform many of the JDIS functions, re-planning squadron and fleet operations will not be as difficult as previous, and what were once "severe" problems will be mitigated to minor inconveniences. For example, a hypothetical situation where a flight schedule has been planned out for a particular day during war time operations may turn disastrous when it is discovered that there is a defective load of control surface actuators that were shipped from the vendor. These actuators would effectively render useless all control surfaces that were being commanded by this shipment, thus would lead to loss of aircraft. In a legacy fleet, the entire wing would be grounded until the malfunctioning parts could be accounted for and replaced with good parts. Afterwards, a new flight schedule would have to be developed if time permitted to complete the missions. In the JSF world, a quick search of JDIS would lead to an immediate knowledge of the locations of the actuators and the aircraft they are housed on. The JDIS could then re-plan the flight operations with the new information of which aircraft were no longer functional, and operations would continue as normal if appropriate numbers of aircraft were unaffected, or on a limited basis as aircraft became available.

This situation could be easily modified to claim that the load of actuators were not defective, but had begun to reach the end of their useful life and were failing at an unusually high rate. In this case, the trend would have been noticed well in advance, and all actuators would have been effectively changed prior to the specific day in question. Again, a quick scan of JDIS would reveal the locations of the malfunctioning parts and those specific aircraft could be grounded at times that would convenience the flight schedule, thus achieving virtually no visible effects throughout the squadron.

Another way in which JDIS can mitigate many of the problems of aging aircraft is by keeping accurate maintenance records. If a particular part is succumbing to a particular failure mode, it may not be visible at the lowest wing or squadron level. But, as JDIS correlates this data across many wings and squadrons, it could possible "see" that the part is having trouble, thus prompting vendors to make design changes on the part to alleviate the problem.

The parts life tracking feature of the Autonomic Logistics system will also help to mitigate aging aircraft problems. By tracking parts and their actual life expended, fleet users will be aware of the actual condition of their assets and will gain an increased awareness as to their readiness. This awareness will translate to an increased confidence in the force structure and subsequently reduce the numbers of downed aircraft later in the life cycle that are undergoing preventative maintenance.

Additionally, the JSF is being designed with an open architecture system to make future unforeseen problems easier to cope with by facilitating the integration of technological fixes to the air vehicle. This is particularly true regarding future software changes and updates. This is being done by separating the Operational Maintenance Profile (OMP) from the Operational Flight Profile (OFP) software programs. This makes updating all PHM and JDIS related software not subject to the rigorous flight qualifications as the flight related software. This approach should enable new and unanticipated future problems to be addressed quickly and affordably. It is the goal of the JSF Autonomic Logistics concept to not only facilitate all maintenance actions and logistics functions, but to avoid the problems that traditionally occur in the aircraft regime, and the JSF Program Office feels that, with the proposed system, this will happen.

SUMMARY

All elements of Autonomic Logistics are part of a complete system that will manage JSF unit deployment and redeployment to ensure proper logistics support. A near-constant monitoring capability that ensures JSF units will be able to support the commander's intent will also become a reality with AL. This support system is designed to make the JSF air vehicle inherently safer to operate, much easier to maintain, more affordable to support, and much less problematic than legacy aircraft. Early in the JSF program much analysis was accomplished to provide an excellent understanding of legacy systems and the problems they are encountering. Many initiatives were undertaken to counter these problems in the design concept. The Autonomic Logistic concept and the PHM system embody the results of many of these initiatives, along with newly available technologies and supportability paradigm shifts.

As a direct result of these design studies, common problems that plague legacy systems will no longer be commonplace, and foresight by all people associated with the JSF program will help to facilitate the solving of any unforeseen problems that will inevitably arise as the fleet ages.

The AL system is, by design, envisioned to provide an information rich and highly intelligent aircraft. The robust information sources and increased knowledge provided by the PHM system can and will be used to mitigate many of the aging aircraft problems we experience today.

REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's References	3. Further Reference	4. Security Classification of Document
	RTO-MP-079(II) AC/323(AVT-085)TP/50	ISBN 92-837-1089-4	UNCLASSIFIED/ UNLIMITED

5. Originator Research and Technology Organisation
North Atlantic Treaty Organisation
BP 25, F-92201 Neuilly-sur-Seine Cedex, France

6. Title
Ageing Mechanisms and Control
Specialists' Meeting on Life Management Techniques for Ageing Air Vehicles

7. Presented at/sponsored by
the RTO Applied Vehicle Technology Panel (AVT) Specialists' Meeting
held in Manchester, United Kingdom, 8-11 October 2001.

8. Author(s)/Editor(s)	9. Date
Multiple	February 2003

10. Author's/Editor's Address	11. Pages
Multiple	416 (text) 46 (slides)

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

Aging tests (materials)
Aircraft engines
Aircraft maintenance
Airframes
Avionics
Corrosion
Failure analysis
Fatigue (materials)
Fleet management
Inspection
Life (durability)
Life cycle management
Life extension
Logistics management
Materials science
Mechanical properties
Methodology
Military aircraft
Nondestructive tests
Service life assessment
Structural integrity

14. Abstract

The costs of maintaining ageing aircraft are draining the existing budgets. The Specialist Meeting provided guidance on strategies for the development and implementation of technologies and logistic management processes to reduce this economic burden. The emphasis was on military aircraft, but many of the principles could be applied to other defence systems. The papers covered the entire range of ageing problems including structural integrity, corrosion, avionics, mechanical subsystems, structures and wiring as well as the role of information management.

Forty-two papers addressed the safety and economic implications such as fatigue cracking, corrosion, wear and material degradation. Key technologies were discussed, including non-destructive inspection, repair, modifications, prevention analysis, and health management. The shortcomings of current were highlighted and the investment required was identified.

The need for research and development was clearly identified.

This page has been deliberately left blank

Page intentionnellement blanche

NORTH ATLANTIC TREATY ORGANISATION

RESEARCH AND TECHNOLOGY ORGANISATION
BP 25 • 7 RUE ANCELLE
F-92201 NEUILLY-SUR-SEINE CEDEX • FRANCE
Télécopie 0(1)55.61.22.99 • E-mail mailbox@rta.nato.int

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
Streitkräfteamt / Abteilung III
Fachinformationszentrum der Bundeswehr, (FIZBw)
Friedrich-Ebert-Allee 34
D-53113 Bonn

BELGIQUE
Etat-Major de la Défense
Département d'Etat-Major Stratégie
ACOS-STRAT-STE – Coord. RTO
Quartier Reine Elisabeth
Rue d'Evère, B-1140 Bruxelles

CANADA
DSIGRD2
Bibliothécaire des ressources du savoir
R et D pour la défense Canada
Ministère de la Défense nationale
305, rue Rideau, 9e étage
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 21076-1320

FRANCE
O.N.E.R.A. (ISP)
29, Avenue de la Division Leclerc
BP 72, 92322 Châtillon Cedex

GRECE (Correspondant)
Defence Industry & Research General Directorate
Research Directorate
Fakinos Base Camp
S.T.G. 1020
Holargos, Athens

HONGRIE
Department for Scientific Analysis
Institute of Military Technology
Ministry of Defence
H-1525 Budapest P O Box 26

ISLANDE
Director of Aviation
c/o Flugrad
Reykjavik

ITALIE
Centro di Documentazione Tecnico-Scientifica della Difesa
Via XX Settembre 123a
00187 Roma

LUXEMBOURG
Voir Belgique

NORVEGE
Norwegian Defence Research Establishment
Attn: Biblioteket
P.O. Box 25, NO-2007 Kjeller

PAYS-BAS
Royal Netherlands Military Academy Library
P.O. Box 90.002
4800 PA Breda

POLOGNE
Armament Policy Department
218 Niepodleglosci Av.
00-911 Warsaw

PORTUGAL
Estado Maior da Força Aérea
SDFA - Centro de Documentação
Alfragide
P-2720 Amadora

REPUBLIQUE TCHEQUE
DIC Czech Republic-NATO RTO
VTÚL a PVO Praha
Mladoboleslavská ul.
Praha 9, 197 06, Česká republika

ROYAUME-UNI
Dstl Knowledge Services
Kentigern House, Room 2246
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 21076-1320
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
Acquisitions
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 Groupe d'imprimerie St-Joseph inc.
(Membre de la Corporation St-Joseph)
1165, rue Kenaston, Ottawa (Ontario), Canada K1G 6S1

NORTH ATLANTIC TREATY ORGANISATION

RESEARCH AND TECHNOLOGY ORGANISATION
BP 25 • 7 RUE ANCELLE
F-92201 NEUILLY-SUR-SEINE CEDEX • FRANCE
Telefax 0(1)55.61.22.99 • E-mail mailbox@rta.nato.int

DISTRIBUTION OF UNCLASSIFIED RTO PUBLICATIONS

NATO's Research and Technology Organisation (RTO) holds limited quantities of some of its recent publications and those of the former AGARD (Advisory Group for Aerospace Research & Development of NATO), and these may be available for purchase in hard copy form. For more information, write or send a telefax to the address given above. **Please do not telephone**.

Further copies are sometimes available from the National Distribution Centres listed below. If you wish to receive all RTO publications, or just those relating to one or more specific RTO Panels, they may be willing to include you (or your organisation) in their distribution.

RTO and AGARD publications may be purchased from the Sales Agencies listed below, in photocopy or microfiche form. Original copies of some publications may be available from CASI.

NATIONAL DISTRIBUTION CENTRES

BELGIUM
Etat-Major de la Défense
Département d'Etat-Major Stratégie
ACOS-STRAT-STE – Coord. RTO
Quartier Reine Elisabeth
Rue d'Evère, B-1140 Bruxelles

CANADA
DRDKIM2
Knowledge Resources Librarian
Defence R&D Canada
Department of National Defence
305 Rideau Street, 9th Floor
Ottawa, Ontario K1A 0K2

CZECH REPUBLIC
DIC Czech Republic-NATO RTO
VTÚL a PVO Praha
Mladoboleslavská ul.
Praha 9, 197 06, Česká republika

DENMARK
Danish Defence Research Establishment
Ryvangs Allé 1, P.O. Box 2715
DK-2100 Copenhagen Ø

FRANCE
O.N.E.R.A. (ISP)
29 Avenue de la Division Leclerc
BP 72, 92322 Châtillon Cedex

GERMANY
Streitkräfteamt / Abteilung III
Fachinformationszentrum der Bundeswehr, (FIZBw)
Friedrich-Ebert-Allee 34
D-53113 Bonn

GREECE (Point of Contact)
Defence Industry & Research General Directorate
Research Directorate
Fakinos Base Camp
S.T.G. 1020
Holargos, Athens

HUNGARY
Department for Scientific Analysis
Institute of Military Technology
Ministry of Defence
H-1525 Budapest P O Box 26

ICELAND
Director of Aviation
c/o Flugrad
Reykjavik

ITALY
Centro di Documentazione Tecnico-Scientifica della Difesa
Via XX Settembre 123a
00187 Roma

LUXEMBOURG
See Belgium

NETHERLANDS
Royal Netherlands Military Academy Library
P.O. Box 90.002
4800 PA Breda

NORWAY
Norwegian Defence Research Establishment
Attn: Biblioteket
P.O. Box 25, NO-2007 Kjeller

POLAND
Armament Policy Department
218 Niepodleglosci Av.
00-911 Warsaw

PORTUGAL
Estado Maior da Força Aérea
SDFA - Centro de Documentação
Alfragide
P-2720 Amadora

SPAIN
INTA (RTO/AGARD Publications)
Carretera de Torrejón a Ajalvir, Pk.4
28850 Torrejón de Ardoz - Madrid

TURKEY
Millî Savunma Başkanliği (MSB)
ARGE Dairesi Başkanliği (MSB)
06650 Bakanliklar - Ankara

UNITED KINGDOM
Dstl Knowledge Services
Kentigern House, Room 2246
65 Brown Street
Glasgow G2 8EX

UNITED STATES
NASA Center for AeroSpace Information (CASI)
Parkway Center
7121 Standard Drive
Hanover, MD 21076-1320

SALES AGENCIES

NASA Center for AeroSpace Information (CASI)
Parkway Center
7121 Standard Drive
Hanover, MD 21076-1320
United States

The British Library Document Supply Centre
Boston Spa, Wetherby
West Yorkshire LS23 7BQ
United Kingdom

Canada Institute for Scientific and Technical Information (CISTI)
National Research Council
Acquisitions
Montreal Road, Building M-55
Ottawa K1A 0S2, Canada

Requests for RTO or AGARD documents should include the word 'RTO' or 'AGARD', as appropriate, followed by the serial number (for example AGARD-AG-315). Collateral information such as title and publication date is desirable. Full bibliographical references and abstracts of RTO and AGARD publications are given in the following journals:

Scientific and Technical Aerospace Reports (STAR)
STAR is available on-line at the following uniform resource locator:
http://www.sti.nasa.gov/Pubs/star/Star.html
STAR is published by CASI for the NASA Scientific and Technical Information (STI) Program
STI Program Office, MS 157A
NASA Langley Research Center
Hampton, Virginia 23681-0001
United States

Government Reports Announcements & Index (GRA&I)
published by the National Technical Information Service
Springfield
Virginia 22161
United States
(also available online in the NTIS Bibliographic Database or on CD-ROM)

Printed by St. Joseph Print Group Inc.
(A St. Joseph Corporation Company)
1165 Kenaston Street, Ottawa, Ontario, Canada K1G 6S1

ISBN 92-837-1089-4