A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 2001 to March 2003

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DSTO-TN-0489

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ABSTRACT

This document has been prepared for presentation to the 28th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Lucerne Switzerland, 5th and 6th May 2003. Brief summaries and references are provided on the aircraft fatigue research and associated activities of research laboratories, universities, and aerospace companies in Australia and New Zealand during the period April 2001 to March 2003. The review covers fatigue–related research programs as well as fatigue investigations on specific military and civil aircraft.

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Executive Summary

The Australasian delegate to the International Committee on Aeronautical Fatigue (ICAF) is responsible for preparing a review of aeronautical fatigue work in Australia and New Zealand for presentation at the biennial ICAF conference. The Defence Science and Technology Organisation (DSTO) has traditionally provided the Australasian delegate to ICAF and publishes the review as a DSTO document. This document later forms a chapter of the ICAF conference minutes published by the conference host nation. The format of the review reflects ICAF requirements.

Contents

8. Australasian Review

8.1 INTRODUCTION

This review of Australian and New Zealand work in fields relating to aeronautical fatigue in the period 2001 to 2003 comprises inputs from the organisations listed below. The author acknowledges these contributions with appreciation. Enquiries should be addressed to the person identified against the item of interest.

PSL	Platforms Sciences Laboratory, GPO Box 4331 Melbourne, Victoria 3001, Australia
ADFA	Australian Defence Force Academy, University of New South Wales, Canberra, Australia
CASA	Civil Aviation Safety Authority, Northbourne Ave, Civic, Canberra 2601, Australia.
DTA	Defence Technology Agency, Auckland, New Zealand
Monash University	Department of Mechanical Engineering, Wellington Road, Clayton, Victoria, Australia
University of Sydney	Department of Mechanical Engineering, Sydney University NSW 2006 Australia
CRCAS	Cooperative Research Centre for Aerospace Composite Structures. 506 Lorimer Street, Fishermans Bend, Victoria 3207, Australia.
Aerostructures Australia	Level 14, 222 Kingsway, South Melbourne, Victoria 3205, Australia.
Structural Monitoring Systems Ltd	Level 1, 5/15 Walters Drive, Osborne Park, Western Australia, 6017, Australia.
CSIRO	Commonwealth Scientific and Industrial Research Organisation, Division of Manufacturing, & Infrastructure Technology, Private bag 33, Clayton South MDC, VIC 3169, Australia.
BAESystems Australia	Edinburgh Parks, SA 5112, Australia.
UWS	IRC in Computer Aided Materials Engineering, University of Wales, Swansea, UK, SA2 8PP

8.2 FATIGUE INVESTIGATIONS ON MILITARY AIRCRAFT

8.2.1 International Follow On Structural Test Project (IFOSTP) (L. Molent, DSTO).

IFOSTP consists of three separate major full-scale fatigue tests supported by flight trials and load development programs. The Australian portion of IFOSTP consists of an unique full-scale fatigue test (known as FT46 [1,2]) which combines buffet induced dynamic loading with manoeuvre loading to reproduce the flight loading conditions experienced by an F/A-18 aircraft's aft fuselage and empennage under normal flight operations. The test was conducted at the Defence Science and Technology Organisation's (DSTO) laboratory in Melbourne, Australia.

The F/A-18 is a highly manoeuvrable, versatile, high performance fighter/attack aircraft. The inner wing leading edge extension (LEX) provides fuselage lift enabling it to achieve angles of attack (AOA) in excess of 60 degrees. The twin vertical tails canted slightly outward exploit the high energy vortices generated by each LEX to provide good directional stability at these high AOA conditions. Unfortunately, these vortices break down at high AOA, buffeting the empennage and exciting the vibration modes of the empennage structure. This buffet phenomenon results in severe empennage dynamic loading, as indicated by the high acceleration levels measured at the aft tip accelerometers of each vertical tail and horizontal stabilator. This empennage vibration also excites engine and other aft fuselage resonant dynamic response causing high stress levels in various structural components. The majority of the fatigue damage imparted to the aft fuselage and empennage is due to this severe buffet dynamic loading, which can occur simultaneously with high manoeuvre loading. There is a synergistic interaction between this quasi-static manoeuvre loading and the higher frequency buffet loading with respect to fatigue damage. This interaction, along with the large number of dynamic load cycles caused by the significant amount of time the aircraft spends above 10 degrees AOA, mandated that the aft fuselage and empennage fatigue test employ a test approach which would simulate as realistically as possible the combined manoeuvre and dynamic loading experienced in flight.

As the flight dynamic loads are fatigue-critical, the primary objective of the loading development process was to ensure that the test article was loaded in such a manner that its dynamic response matched as closely as possible that of an aircraft in flight. To accomplish this, a manoeuvre loading system was required that would not significantly affect the dynamic characteristics of the structure. Typical fatigue test loading systems use hydraulic actuators, loading beams and pads, and cables to load the structure. Such a system was unacceptable for FT46 testing, as it would add too much, mass, stiffness and damping to the structure and thus, alter the test article's dynamic characteristics too adversely.

The system adopted uses an Australian developed pneumatic loading system rather than the more conventional hydraulic loading system to apply the aircraft distributed aerodynamic and manoeuvre-induced inertial loads. The buffet-induced dynamic inertial loads experienced in flight were applied using a multi-channel vibration control system and high-powered, high displacement electromagnetic shakers.

FT46 has now completed 23,090.2 simulated flight hours (SFH) of testing using two spectra representative of distinct periods of RAAF and Canadian usage. The first test phase applied 1270.5 SFH of loading representative of fleet usage and dynamic/manoeuvre loading experienced prior to the installation of the LEX fences. The LEX fences were retrofitted by the OEM (Original Equipment Manufacturer) to alter the aerodynamics of the leading edge extension vortices to reduce the empennage buffet dynamic response. The second test phase (21819.7 SFH) represented the usage and loading experienced after the installation of the LEX fences to the aircraft. The third phase of the FT46 test program, residual strength testing (RST) has just been completed and the test article is now undergoing detailed teardown inspection.

References

1. Simpson, D.L., Landry, N., Roussel, J., Molent, L., Graham, A.D., and Schmidt, N., "The Canadian and Australian F/A-18 International Follow-On Structural Test Project", Proc. **ICAS 2002 Congress**, Toronto, Canada Sept. 2002.
2. Molent, L., Barter, S., White, P. and Conser, D., "Overview of the F/A-18 Aft Fuselage Combined Manoeuvre and Dynamic Buffet Fatigue Test", proc of **USAF Structural Integrity Program Conference,** Savannah, Georgia, 10-12 Dec 2002.

8.2.2 Risk-Based Life Assessment (L. Molent and P. White, DSTO)

The life of type of the RAAF Hornet has been assessed through the interpretation of fatigue tests results using guidance provided by the UK Defence Standard 970. In investigating alternatives to this traditional approach, improved methods of estimating the probabilistic risk of structural failure are being explored [3]. The improvements made by DSTO address a number of parameters additional to those normally used in probabilistic methods, and in the case study used, utilise the extensive crack growth measurements in representative coupons and the full-scale test articles and other experimentally derived fatigue-life-determining parameters available for the F/A-18 airframe.

The improvements were demonstrated by considering the safety-of-flight wing attachment bulkheads of a specific fighter aircraft. However the method is generic and applicable to all aircraft types. The risk analysis calculates the probability of failure for each flight over the life of the aircraft type. Because fatigue cracks grow over time (in the absence of inspections and removal of cracks) the risk of failure increases over time until an unacceptable level of risk is approached. Probabilistic fracture mechanics is used as the basis of the analysis and in this example, where possible, all relevant data were collected directly from the aircraft's fatigue tests supplemented by laboratory coupons. To address loading variability, the benefit and accuracy of the individual aircraft fatigue monitoring system is also considered. The analysis attempts to include all parameters that significantly influence the growth of cracks in metallic airframes of this vintage.

An important outcome of fatigue testing is the highlighting of areas in the structure prone to cracking. The measured sizes of cracks detected throughout the test life, to correlate with the crack growth rates and initial flaw sizes from fractography of coupon tests, allowed an estimate of the stress at these locations to be made. The fatigue test also provided a convenient spectrum from which a probability of exceeding a load level can be determined.

From the experimental coupon data, probabilistic distributions were determined for the initial equivalent crack sizes, crack growth rates and the fracture toughness of the material. It was concluded that the data available could not be extrapolated to the extent required for confident life assessment, and that this was a key factor which limited applicability of the probabilistic approach. The accuracy of the individual aircraft fatigue monitoring system was also assessed from another series of coupon tests covering a range of aircraft usage severities. By comparing predictions against the results from the coupon test program the distribution of relative errors in the crack growth predictions was determined. Without the monitoring system much more conservative estimates of error would be required.

By combining the probability distributions of the crack growth variability parameters with the probability of exceeding a given load level, whilst accounting for the accuracy of fleet wide crack growth estimation, a probability of failure can be calculated for each location prone to cracking. Combining the risk of failure for each of these locations could then allow estimation of the single flight probability of failure distribution for the components. By utilising data available from fatigue tests supplemented by tailored coupon tests for a specific aircraft type, rather than generic data, the risk analysis for a specific aircraft type may be optimised and the potential exists for appropriate levels of airworthiness to be maintained without undue conservatism. The preliminary process described in this paper is directed at this aim. The difficulties associated with this method, principally that of dealing with lack of data on initial defect size distributions, and a requirement for improved initial defect data, have also been identified.

3. White, P., Barter, S. and Molent, L., "Probabilistic Fracture Prediction Based On Aircraft Specific Fatigue Test Data", proc. 6th Joint FAA/DoD/NASA Aging Aircraft Conference, San Diego, Sept.16-19, 2002.

8.2.3 Fatigue Monitoring or HUMS (L. Molent, DSTO)

Significant research and development efforts directed towards the RAAF's F/A-18 fatigue monitoring system have been reported in previous ICAF reviews. Many lessons were learnt during this process and these have been documented for future acquisition projects [4]. The role of a Health and Usage Monitoring Systems (HUMS) for rotating and related components in helicopters airframes and engines is well defined and understood. In more recent times the term HUMS has been applied to fixed wing airframes. With this development it was timely to compare those lessons learnt, with the aims envisaged by HUMS.

Generally these fixed wing airframe HUMS consider the health (H) of an airframe through the fatigue monitoring or usage aspects (U) rather than a separate diagnostic capability monitoring health or structural integrity as found in most rotating component HUMS. However with the development of so-called smart structures and miniaturisation some researchers are arguing that the "U" may be replaced by more direct "H", in other terms HuMS. In general it is envisaged that the health of the structure will be ensured through some "advanced" non-destructive inspection (NDI) techniques (using embedded sensors or the so-called smart structures or material) to detect damage, degradation and flaws before they cause structural failure. Whilst this may seem attractive, there are many significant through life structural integrity issues to be addressed before this can be viable. These issues regarding the application of HuMS to the through-life management of fixed wing manned airframes were canvassed in a recent paper [5]. It specifically considers whether the role of prognostic usage monitoring should be replaced with direct health monitoring techniques.

4. Molent, L. and Aktepe, B., "Review of Fatigue Monitoring of Agile Military Aircraft", **J. Fatigue and Fracture of Engineering Materials and Structures**, **23**, p767-785, Sept. 2000.
5. Molent, L., "Considering the Role of Health Monitoring in Fixed Wing Airframe HUMS", **Proc. HUMS2003**, Melbourne, Australia, 17 –18 Feb 2003.

8.2.4 F/A-18 Aging Aircraft Audit (K. Jackson, M. Stoessiger and S. Bandara, Aerostructures).

The RAAF F/A-18 fleet has reached its anticipated mid-service life. In accordance with Australian Defence Force (ADF) airworthiness doctrine, an Ageing Aircraft Audit (AAA) is currently being conducted to provide an accurate assessment of the structural condition of the ageing fleet. A key aspect of the AAA is to provide fleet condition data to supplement and support test interpretation activities using the results from the F/A-18 International Follow-On Structural Testing Program (IFOSTP).

The AAA has been broken down into two main stages; the first stage is complete and consists of a desktop audit [6]. This involved the collection and review of all available F/A-18 documentation and the extraction of all essential defect and repair information. An SQL database with a web-based data retrieval system was developed to store the large amount of data generated. The condition data collected was analysed in a variety of ways to establish meaningful trends within the fleet, including assessments with respect to defect cause, location, high defect parts and distribution by material type. The distribution of defects by servicing type, tail number, model and production block was also considered.

The second stage of the AAA is the conduct of a physical audit of several representative aircraft. This stage is currently in planning [7] and involves deeper level inspection of targeted locations not currently considered during current RAAF maintenance. One of the key drivers for the selection of locations will be critical areas that have experienced fatigue cracking during IFOSTP. These inspections will seek to determine whether these locations may also be prone to various forms of environmental degradation. The results of depot level maintenance activity and physical audit programs carried out by foreign operators of the F/A-18 shall also be considered.

6. ER-F18-51-ASM251, "F/A-18 Hornet Structural Condition Audit", Aerostructures, 31 August 2001.
7. WP-F18-51-ASM100, "F/A-18 Aging Aircraft Audit - Physical Audit Planning", Aerostructures, 31 January 2003.

8.2.5 F-111 Aircraft Structural Integrity Sole Operator Program (K. Watters, DSTO)

Following the retirement of the US fleet, Australia is now the sole operator of the F-111, and plans to operate the aircraft to 2020. A large program of work, known as the F-111 Aircraft Structural Integrity Sole Operator Program (ASI SOP) [8], has been set up to develop the in-country capability and carry out applied research and engineering work to support the F-111 over its remaining 20 years service. The F-111 ASI SOP comprises a holistic approach to the management of an ageing aircraft by safety-by-inspection. A critical analysis of design and service data complemented by teardown inspections will identify the critical structural locations which have to be managed. The external loads, the internal loads and stresses and the crack growth at critical locations are subjects of initial capability development, research, capability extension and practical application. The influence of corrosion as a degradation mechanism and as a precipitator of fatigue is being assessed. Research is being conducted into better NDI techniques and their probability of detection (POD). The degradation of bonded honeycomb panels, their effect on structural integrity, their damage tolerance and repair and their replacement with stiffened graphite/epoxy panels are being researched. Ageing aircraft effects relevant to the F-111 are being assessed. A pictorial description of the F-111 ASI SOP is shown in Figure 1.

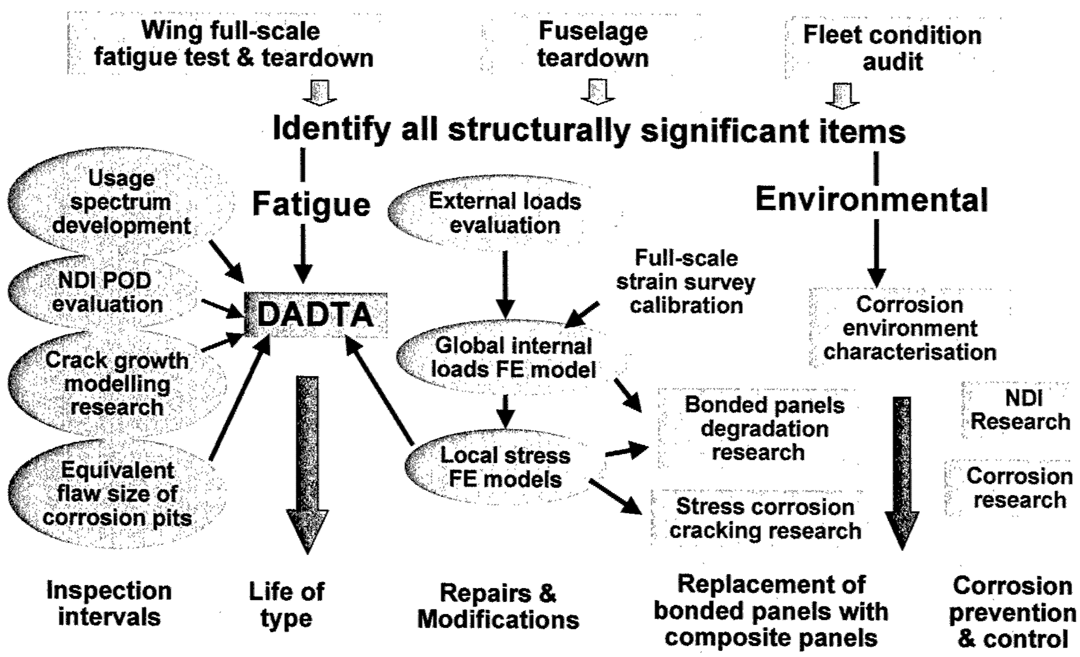


Figure 1: The F-111 Aircraft Structural Integrity Sole Operator program

Five significant tasks were carried out in conjunction with the F-111 OEM, Lockheed Martin Aircraft Company, LMAero at Fort Worth in the US. Attached Australian engineers and LMTAS engineers jointly staffed those tasks. The LMAero tasks are now complete and delivered, and comprised: a finite element internal loads model of the F-111 fuselage and wings; a comprehensive report on the durability and damage tolerance assessment (DADTA) of the Australian F-111; an extension of DADTA capability to model multi-site crack initiation; a report on the basis and status of external loads characterisation of the F-111; and a report documenting the set of Structurally Significant Items (SSI), which is the set of parts on the aircraft which may be prone to fatigue, environmental degradation or other in-service damage which compromises the structural integrity. Lockheed have also made their proprietary software for DADTA crack growth analysis available to Australia for use on the F-111. As a result, Australia now has a capability in DSTO and Industry to perform DADTAs of the F-111.

It was found that there was insufficient strain survey data available to correlate with the FE Internal Loads Model (ILM). A ground strain survey of an instrumented RAAF F-111 has been conducted and the data from 670 gauges is now being used to correlate the ILM. Fine-grid stress models have been developed from the ILM for parts of the F-111 structure with known problems. These are the D6ac steel 770 bulkhead where the horizontal and vertical tails attach to the fuselage, the D6ac steel 496 frame which forms the engine intake nacelle (Figure 2), the D6ac steel overwing longerons and the 7079 aluminium alloy forward fuselage fuel tanks. The latter of these is to investigate widespread stress corrosion cracking in the RAAF fleet. The fine-grid models await only the loads input from the correlated ILM.

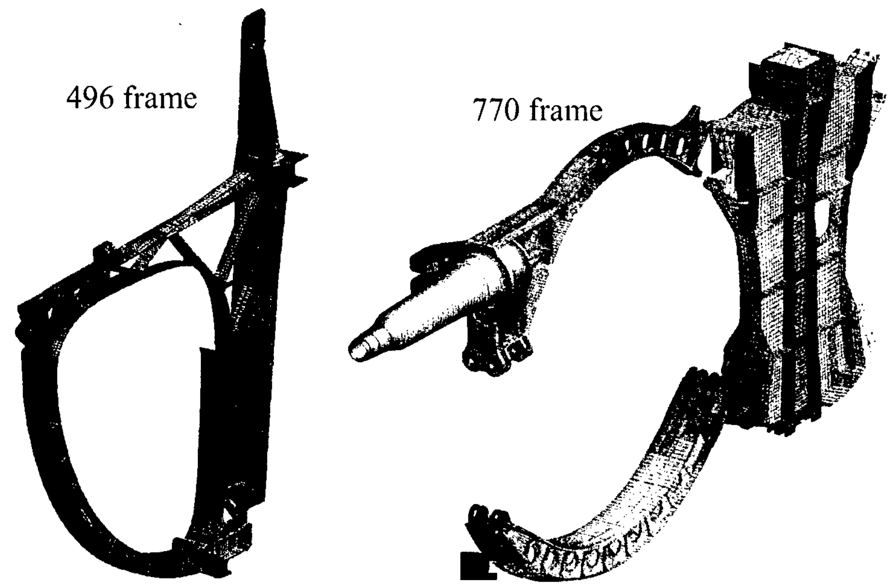


Figure 2: Fine-Grid FE Models of 496 and 770 frames

8. Agius, J., Connor, P., Madley, W. and Watters, K., *Royal Australian Air Force (RAAF) F-111 Aircraft Structural Integrity Sole Operator Program*, Proceedings of 1998 USAF ASIP Conference, San Antonio, Texas, USA, 1st to 3rd DEC 1998.

8.2.6 F-111 Wing test and teardown results (T. van Blaricum, DSTO)

Fuselage and wing teardown inspections are being conducted, the primary purpose being to reveal any new degradation sites in the F-111 structure which might appear in service over the next 20 years. The teardowns will also indicate the current state of degradation of the fleet, including any widespread degradation, providing a basis for projecting to the future state and making structural integrity management decisions.

An ex-RAAF F-111 wing with 5,500 hours and 25 years of service was further 'aged' by fatigue loading in a test rig before being subjected to a teardown inspection. Unfortunately, the wing failed prematurely and unexpectedly in the test and revealed a fatigue weakness (Figure 3). The failure occurred from a fatigue crack in the lower skin emanating from a Taper-lok fastener hole. The crack was triggered by poor surface finish in the hole bore (see section 8.2.7) and highlighted an emerging issue - the build quality of the F-111 wings. The wing failure occurred after 13,500 combined service and test hours, which was insufficient to demonstrate a safe life for the fleet wings to 2020. Indeed, many of the fleet wings had already passed the safe life derived from the test result and had to be grounded immediately.

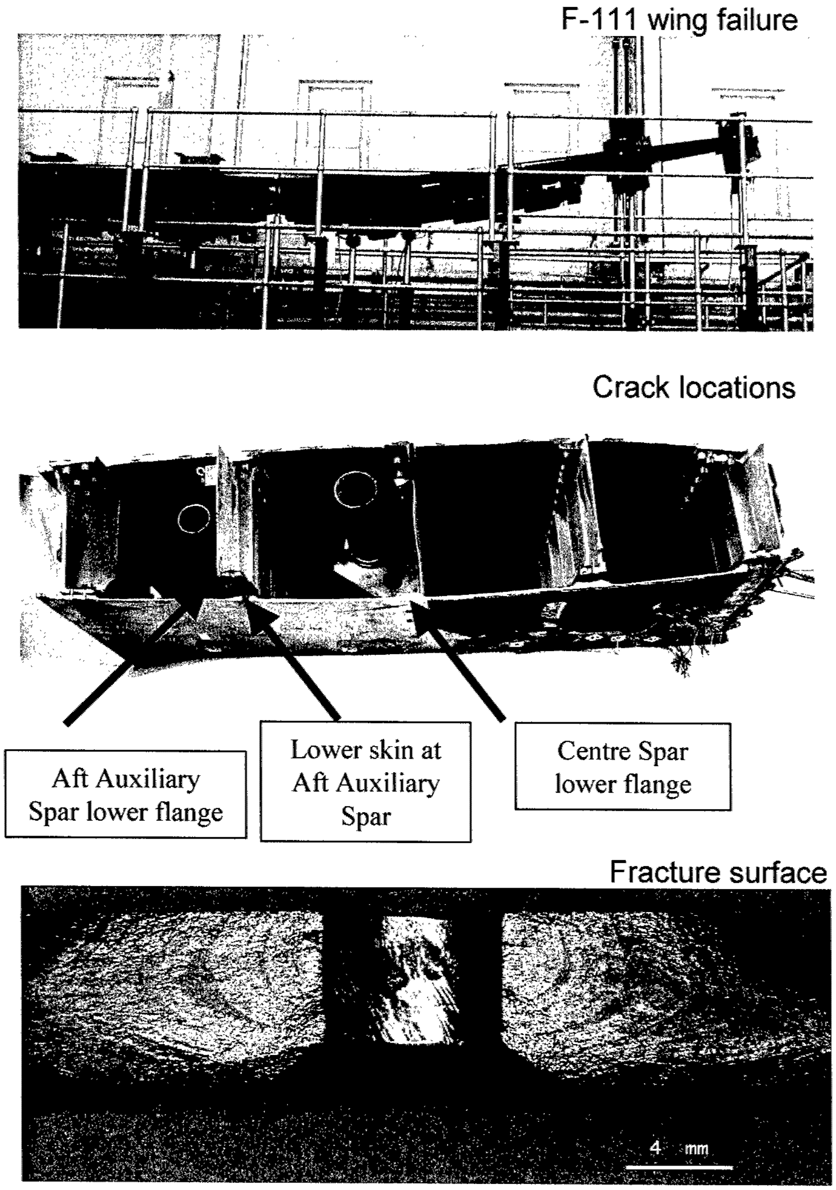


Figure 3: F-111 wing failure. Note the tunnelling of the crack which limits the surface-breaking indications.

The RAAF opted to replace all its fleet wings with ex-USAF F & D model wings, which had lesser accumulated fatigue damage and presumably better build quality (being later in the production cycle). The wing replacement program is progressing well and the RAAF has now restored adequate serviceable aircraft availability after a year of disruption.

The build quality of the replacement wings is being assessed by sample teardown inspections. One of the replacement wings will be fatigue tested to demonstrate the remaining life in RAAF service. The RAAF wish to bring the wings under safety-by-inspection management, as this is the certified management basis for the F-111, and efforts are underway to acquire and demonstrate the Probability of Detection (POD) available using an automated NDI system to deal with the rows of fastener holes attaching the lower wing skin to the five spars.

8.2.7 Hole condition and build quality issues with F-111 fatigue cracks (N. Athiniotis, DSTO)

During a full-scale fatigue test of an F-111 wing, cracking in the skin had initiated from a Taper-lok fastener hole. Cracking initiated from machining tears associated with poorly machined holes probably produced during the final reaming process.

Figure 4: Fracture surface of the cracking in the skin (shown macroscopically in Figure 3), showing the machining spiral marks from the reaming process, example arrowed.

These cracks originated at regions of extremely poor surface finish in the generally very poorly machined hole. Fractographic examination showed that the loads applied in the test and in service were clearly identifiable, the Cold Proof Load Test (CPLT) loads were clearly identified, and in addition the patterns of loads applied in the test were recognised

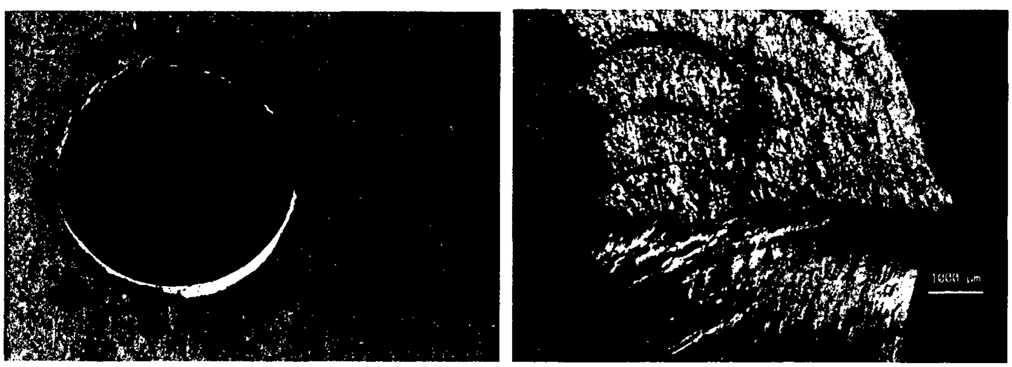


Figure 5: View of the top of the lower skin in the wing, showing the out-of-round hole and the large crack. The thin circular outline represents a fastener in the hole. The right image shows an Oblique view of the broken open hole. Note the well defined tear bands associated with the CPLT loads

8.2.8 Teardown Results on F-111 Fuselage (T. van Blaricum and N. Athiniotis, DSTO)

The teardown inspection of an ex-USAF F-111 fuselage with high service hours is scheduled for completion at the end of 2003. The fuselage was selected on the basis that was in similar state to the RAAF F-111 fuselages. No significant fatigue or environmental degradation instances (that were not known before the teardown) have emerged so far. A key result has been to confirm the extent of stress corrosion cracking in the 7079 aluminium forward fuselage structure (Figure 6).

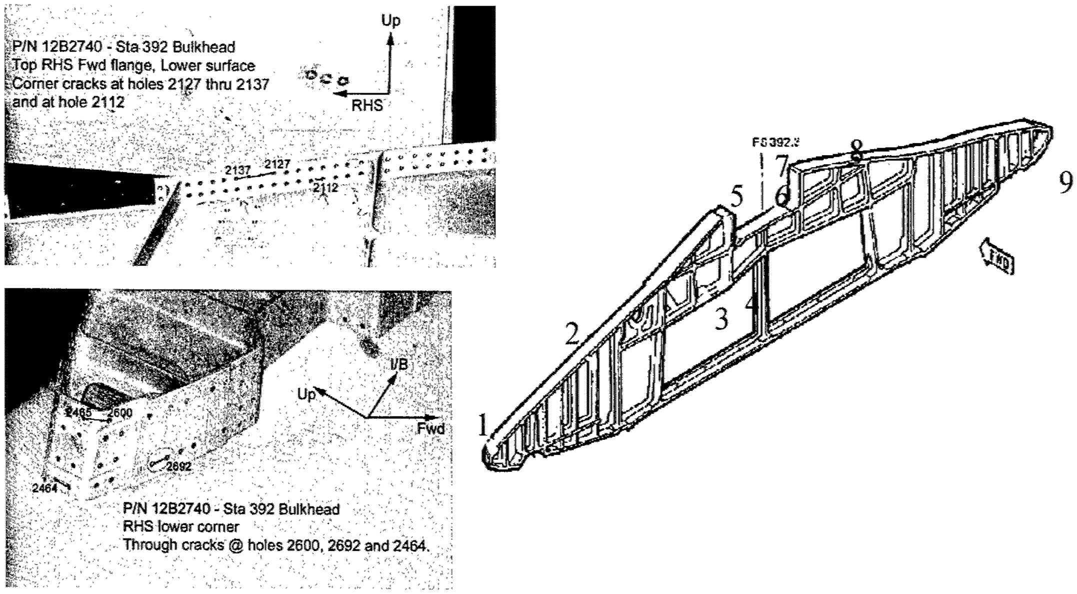


Figure 6: Typical Stress Corrosion Cracking in the F-111 forward fuselage frames

8.2.9 Development and Validation of an Analytical Model to Predict Fatigue Crack Growth in Notch Plastic Fields (K. Walker, S. Weller, J. Walker, DSTO, and D. Ball, Lockheed Martin USA)

Structural integrity for the RAAF F-111 fleet is assured under the Durability and Damage Tolerance Analysis approach. This involves the application of fracture mechanics techniques at critical locations to determine the crack growth characteristics and allow appropriate inspection intervals to be set. For locations on the F-111 which cannot be inspected by any other means, a Cold Proof Load Test (CPLT) is applied. This involves the application of Design Limit Load at a reduced temperature. The CPLT induces significant localised plasticity at some locations in the structure. The Fuel Flow Vent Holes (FFVHs) and Stiffener Run Outs (SROs) in the D6ac steel upper plate of the wing pivot fitting (see Figure 7 and Figure 8 below) are particularly susceptible.

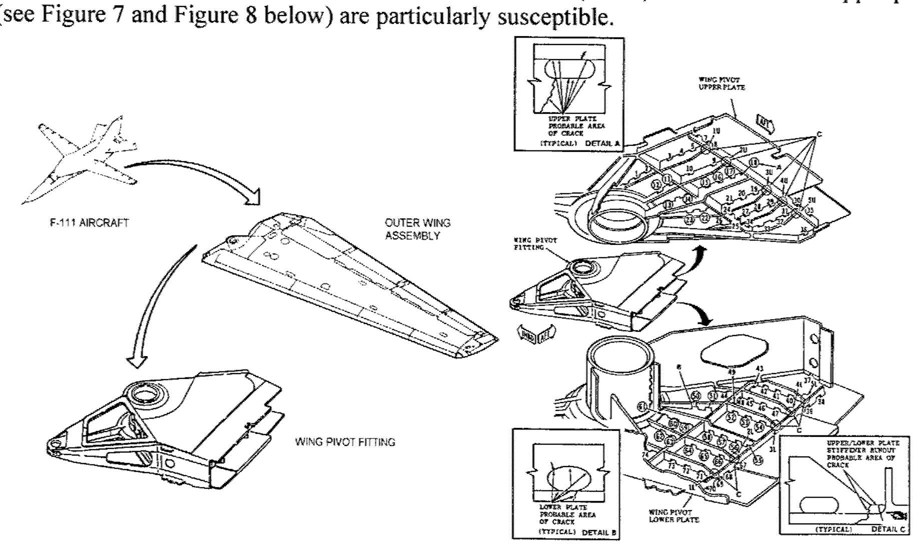


Figure 7: F-111 Wing Pivot Fitting Fuel Flow Vent Hole Details

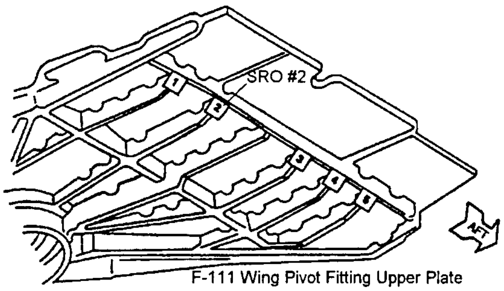


Figure 8: F-111 Wing Pivot Fitting SRO #2 location.

During the CPLT, the FFVHs and SROs in the upper plate experience localised compressive yielding which induces a residual tensile stress field. Although these locations are in a compression dominated loading environment, the tensile residual stress enables fatigue crack growth to occur. Analytical methods are required to quantify this type of cracking. The original manufacturers of the F-111, Lockheed Martin Aeronautics Company, have collaborated with DSTO researchers in the development of a computer program known as METLIFE for this purpose. METLIFE is based on a combination of the local notch strain approach (incorporating Neuber's Rule) and Linear Elastic Fracture Mechanics. The cyclic plasticity algorithm assumes non-linear kinematic hardening with Masings approach. Elements of this algorithm, including re-yielding in a given direction and reverse plasticity after yielding in the opposite direction have been significantly re-worked recently to rectify identified deficiencies.

METLIFE analysis has been carried out for a slanted oval hole specimen representative of FFVH # 13 (Reference 9). The model is detailed in Figure 9 below. The applied and response stress distributions from METLIFE are compared with elastic plastic FE results in Figure 10. The final crack growth is compared with experimental data in Figure 11.

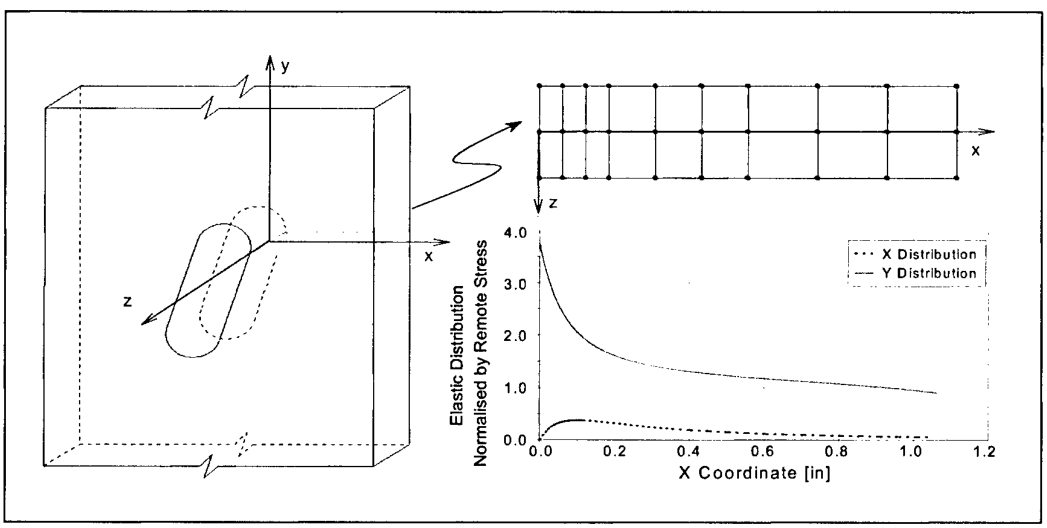


Figure 9: Model of Anticipated Crack Plane and Corresponding Elastic Stress Distributions.

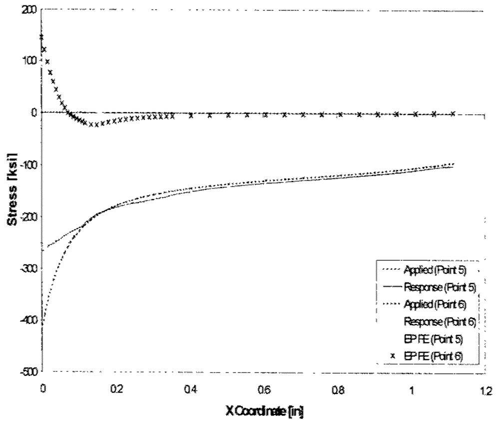


Figure 10: Comparison of FE and Notch Plasticity Stress Distributions at Load Point 5 (+7.33g load) and Load Point 6 (zero load) in the Proof Load Sequence.

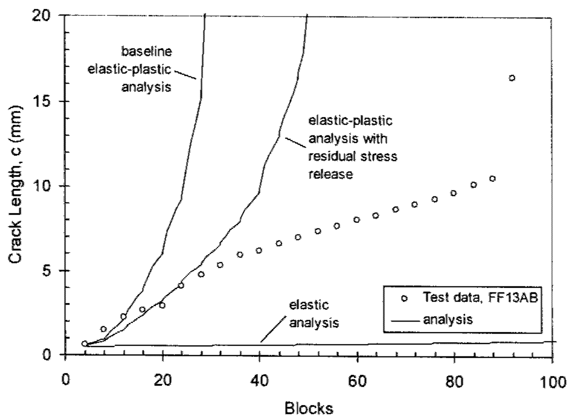


Figure 11: Comparison of Experimental and Analytical Crack Growth Results.

The elastic-plastic analysis using residual stress release provides a significant improvement in fit to the experimental data.

A similar exercise has been conducted for specimens representative of SRO # 2 (see Figure 12). The following METLIFE analyses were conducted for these specimens:

- A fully elastic analysis
- A fully elastic-plastic analysis
- A hybrid elastic analysis, where the residual stress distribution due to the CPLT loading is determined and imposed as an initial condition. Thereafter, a linear elastic analysis is conducted.

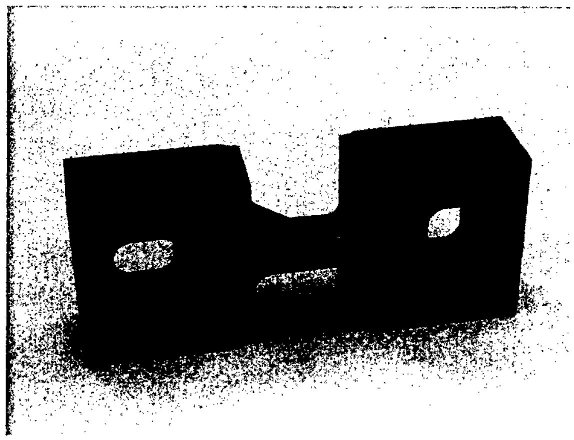


Figure 12: SRO #2 Representative Specimen

The results of the METLIFE analyses are detailed in Figure 13 below.

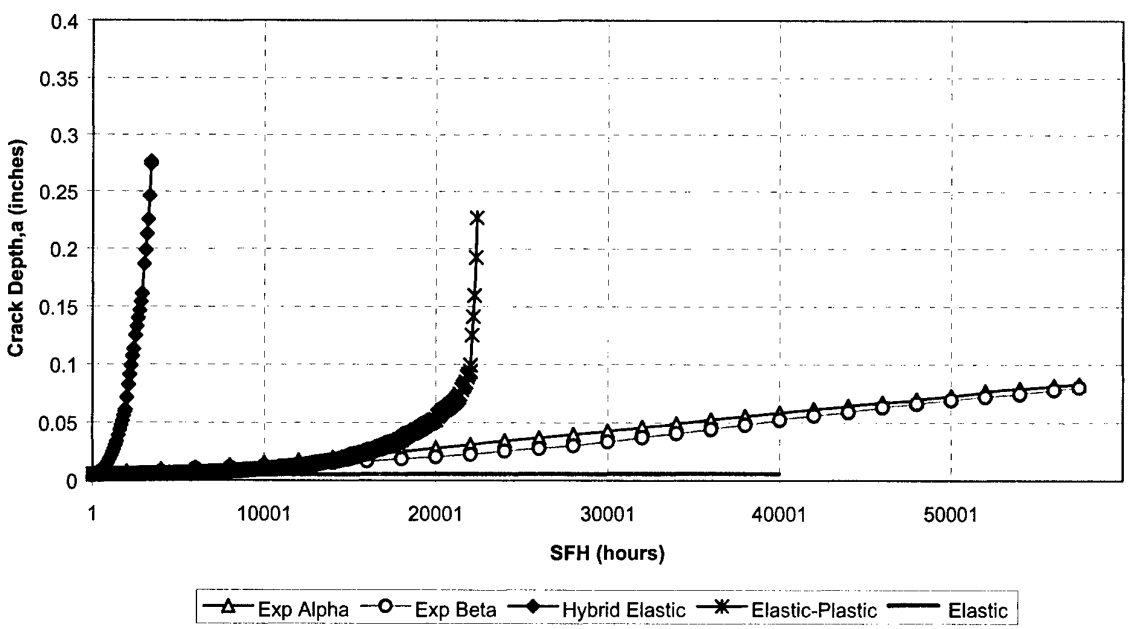


Figure 13: Experimental and Analytical Results for SRO # 2 Representative Specimen

The results being obtained from METLIFE are generally conservative and sufficient confidence has now been achieved to carry out predictions for fleet management. It is important to note however that fleet management decisions also consider other important factors including experimental data and service experience. Minimal crack growth is predicted for the optimised re-work profile cases, which are currently being implemented in the RAAF F-111 fleet as part of a long term management strategy. Linear elastic analysis for these locations gives no crack growth and an infinite life; i.e. clearly wrong and unconservative (see *Figure 10* and *Figure 13*). The METLIFE approach is therefore clearly a significant advance on the linear elastic analysis methods.

Further development of METLIFE is still required, however the results thus far are encouraging. Areas for further improvement include; refinements to the cyclic plasticity algorithm, improved methods to account for short crack effects, and methods to better account for load interaction and closure effects. It is envisaged that a fully validated version of METLIFE will be an essential tool in the structural integrity management of the RAAF F-111 fleet until the planned withdrawal date of 2020.

9. Ball, D.L., and Walker, K.F., "The Implementation and Preliminary Verification of an Algorithm for the Analysis of Fatigue Crack Growth at Notches", Presented at the Eighth International Fatigue Congress, FATIGUE 2002, Stockholm Sweden, 2-7 June 2002.

8.2.10 Durability and Damage Tolerance Assessment of RAAF F-111C Lower Wing Skin Critical Regions (K. Walker, S. Weller, DSTO)

A RAAF F-111C wing undergoing full scale fatigue testing at DSTO failed by fracture from a fatigue crack in February 2002. The failure originated at the surface of a taper lok fastener hole at Aft Auxiliary Spar Station 277.3 (see Figure 14 for the failure location). The fastener is one of a spanwise row connecting the skin and spar, and the failure occurred at approximately 13,500 hours. Subsequent investigations revealed numerous cracks at similar holes throughout the wing and also significant cracking in the spars. Failure by fatigue cracking at these holes was not anticipated, and prior to the failure there had been no known instances of this type of cracking in earlier fatigue tests or through service experience. Investigations revealed that the build quality of the wing was very poor, and many of the fasteners did not exhibit sufficient interference to be of any fatigue benefit. Also, the surface finish on many holes was extremely poor with significant gouges and out of round holes present. Examination of several other wings indicated that the test wing was a severe case, and probably the worst likely case.

The widespread nature of the problem (there are hundreds of fasteners, many with similar stress levels) meant that reliance on safety by inspection for a small number of discrete locations was not possible. However, it was possible to quantify the safe life for the wings and the durability limit for specific areas on the wing. A durability and damage tolerance analysis (Reference 10) was carried out for this purpose. Figure 3 details the fracture surface of the crack which caused the wing failure, and Figure 15 details the correlation of the crack growth model. Using this model, the durability of the wing was established and this was used to provide guidance on which regions were likely to require inspection. Figure 16 shows the areas (in red) of part of the wing estimated to require inspection at and subsequent to 8,000 hours. A key part of returning wings to a safety-by-inspection basis would be the development and validation of an automated NDT system for scanning large sections of wing. This is being pursued elsewhere in DSTO (see section 8.2.14); a potential system has been identified and is being trialled.

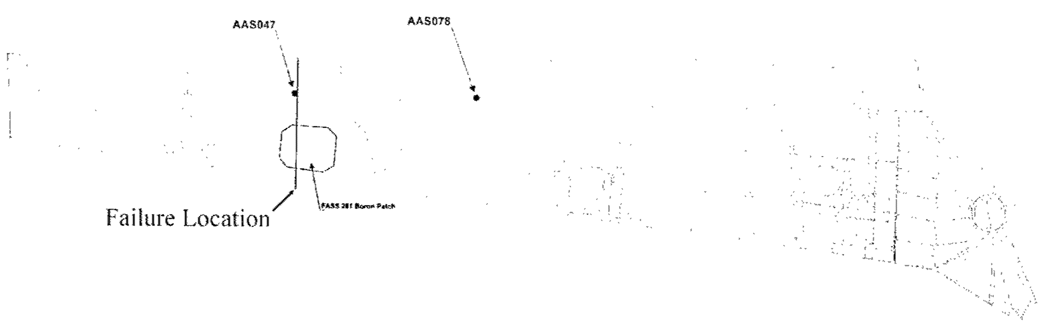


Figure 14: RAAF F-111C Fatigue Test Wing Failure Location

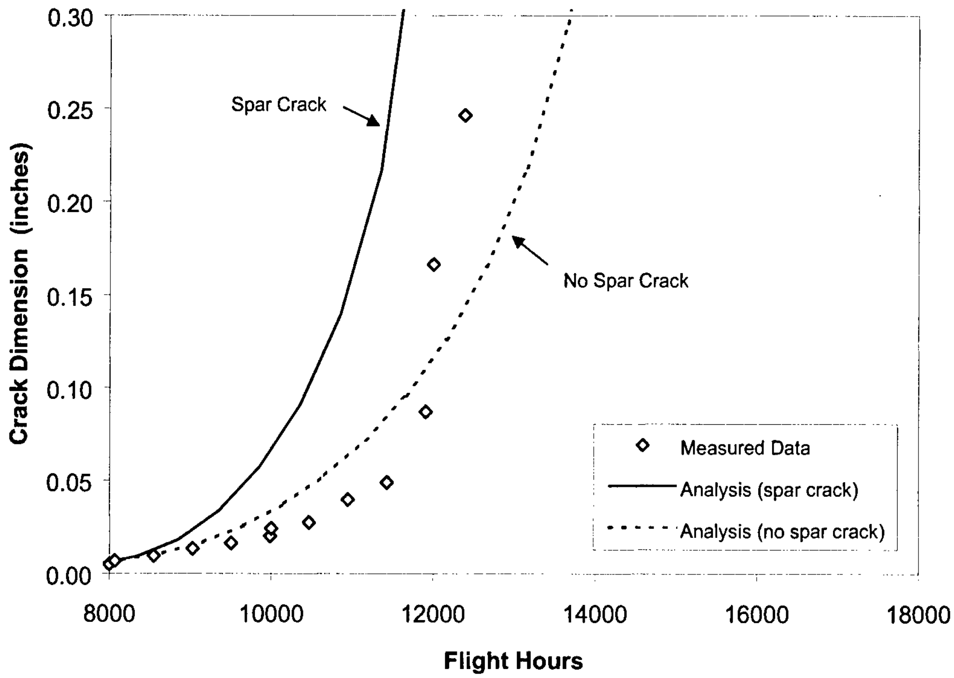


Figure 15: Crack Growth Correlation at Failure Location

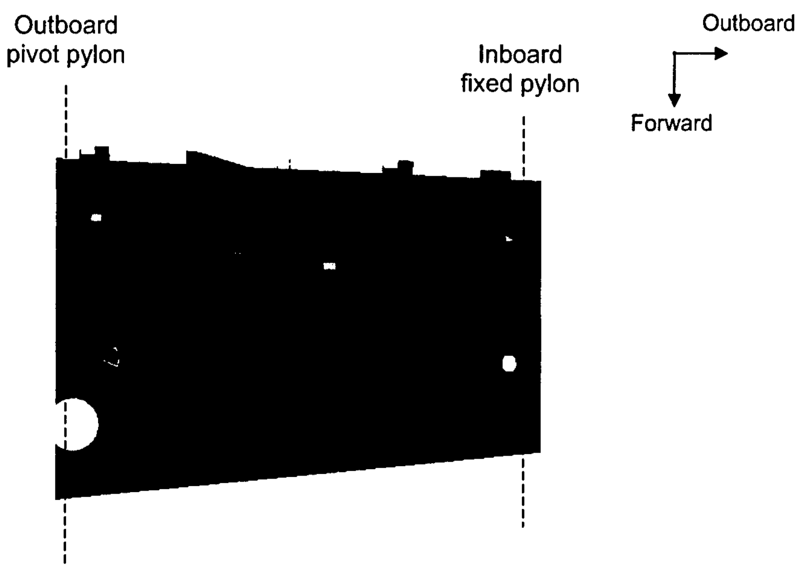


Figure 16: Approximate Areas of Wing Requiring Inspection at and after 8,000 hrs

10. Weller, S., and Walker, K., "Durability and Damage Tolerance Assessment of RAAF F-111C Lower Wing Skin Regions Around FASS 226 and 281 – Third Release", DSTO Letter Report, August 2002

8.2.11 F-111 Damage Tolerance Analysis (DTA) (S. Bandara, Aerostructures)

The structural integrity of the F-111 aircraft is managed primarily on a Safety By Inspection (SBI) basis. Fundamental to this is the knowledge of potential critical structural locations, the associated loading and usage spectra such that the appropriate inspection intervals and techniques can be implemented. Aerostructures conduct Durability and Damage Tolerance Analyses (DADTA) to obtain inspection intervals for critical F-111 structure so that they can be effectively managed.

Recently, work was carried out by Aerostructures in association with the Royal Australian Air Force (RAAF) and the Defence Science and Technology Organization (DSTO) to review the baseline inspection intervals as part of Aircraft Structural Integrity Management Plan (ASIMP) Review. The intention of this task was to ensure that there is a sound analytical basis to support the current baseline intervals for all DADTA locations. During the development of the SBI programme for the RAAF, Lockheed Martin Tactical Aircraft Systems (LMTAS) selected over 100 critical locations for DADT assessment. For each location, local stress spectra were derived, local geometry determined and a crack growth analysis carried out. However, over time, some of the fundamental assumptions, such as local load, spectrum or geometry, can change. Aerostructures was involved in reviewing the history and evolution of selected DADTA Items (DI). This task highlighted a number of locations that required further substantiation using DADTA analyses with representative loading and spectra.

Currently, work is underway to generate a new DADTA spectrum representative of current RAAF flying. D20 flight recorders will be installed on a number of F-111 variants to obtain flight data that will be used to generate this spectrum. Aerostructures have developed software to screen this data and will also be involved in the new DADTA spectrum generation process once sufficient data is obtained.

The F-111 Finite Element Internal Loads Model (ILM) (developed by LMTAS, Aerostructures and DSTO) has been correlated on a number of critical locations such as the Wing Carry Though Box (WCTB), FS 770 frame and overwing longerons and work is in progress to correlate regions such as the FS 496 former. Aerostructures has been involved in the correlation of the ILM that provides loads to be used for DADTA analyses. The DI locations highlighted at the ASIMP review will be reassessed using ILM loads.

Work is being carried out by Aerostructures in the spike island tab, FS 770 lower lug and critical wing regions using the ILM. Fine grid Finite Element (FE) models will be constructed of the critical DADTA locations at these regions. New stress equations will be developed for these locations using a pool of 30 design load cases that have been developed by DSTO for the F-111C ILM. The 30 load cases represent points in the sky flying that is typical of RAAF operations. The stress equations will be developed using regression analysis. DADTA will be performed at these locations using the revised load equations and the early DADTA spectra using LMTAS development DTA and spectrum generation software. Once the new DADTA spectrum is generated, these analyses will be re-run with the more representative spectrum.

Aerostructures also perform DADTA analyses for Strike Reconnaissance Systems Project Office (SRSPO) when DADTA locations are reworked due to repair or modification. Work is currently being carried out by Aerostructures on the FS496 nacelle former and overwing longerons with the intention of providing inspection intervals for the reworked structure at these locations.

8.2.12 Shape Optimisation of Critical Features in the F-111 Wing Pivot Fitting (M. Heller, M. McDonald, M. Burchill and K.C. Watters, DSTO)

Optimal reworks shapes for the most critical fatigue locations in the F-111 wing pivot fitting have been determined [11] using a recently developed finite-element-based gradientless shape optimisation procedure. The locations considered include stiffener runouts and fuel flow vent holes (Figure 17).

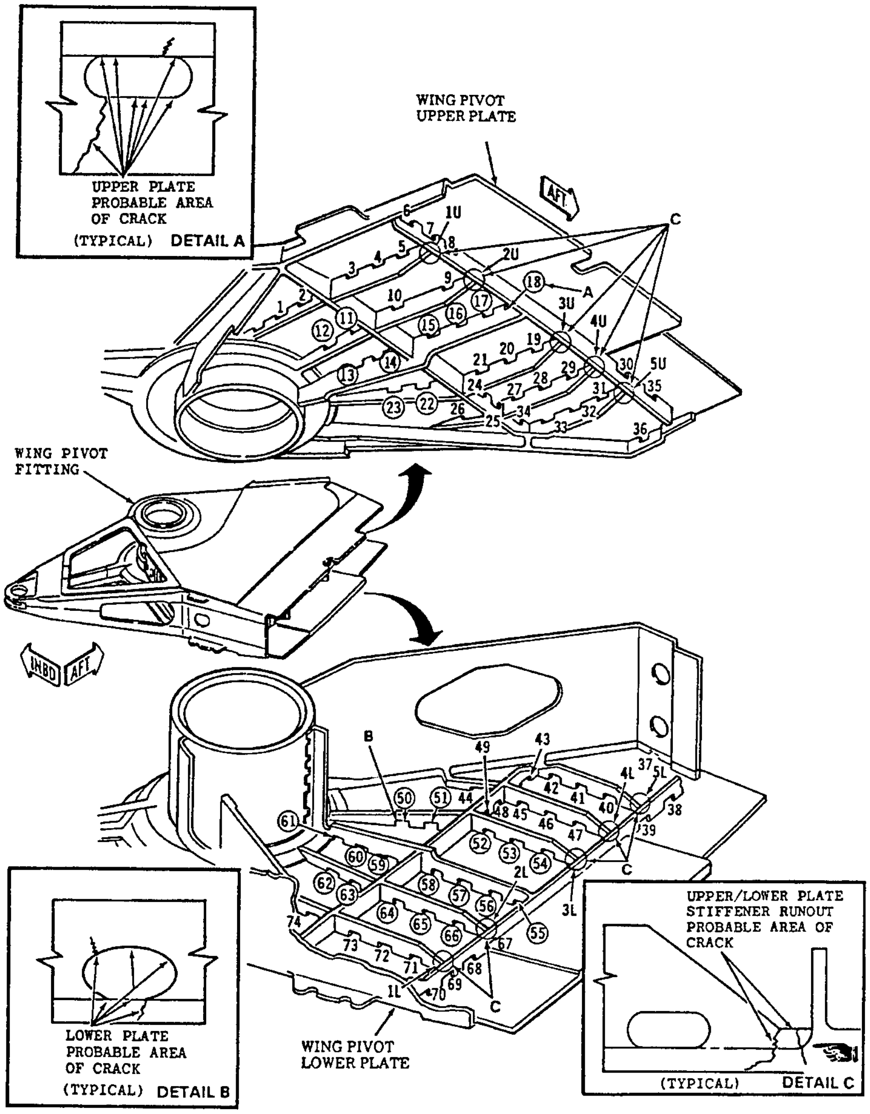
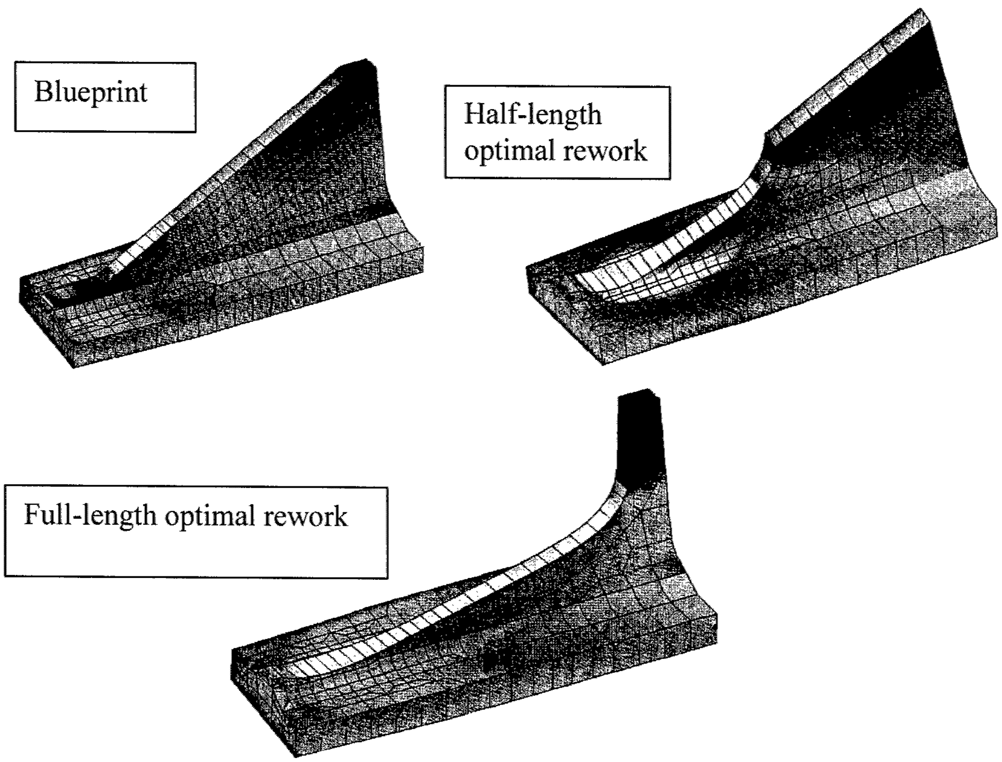


Figure 17: Detailed internal view of the F-111 wing pivot fitting structure showing typical fuel flow vent hole (FFVH) and stiffener runout (SRO) features.

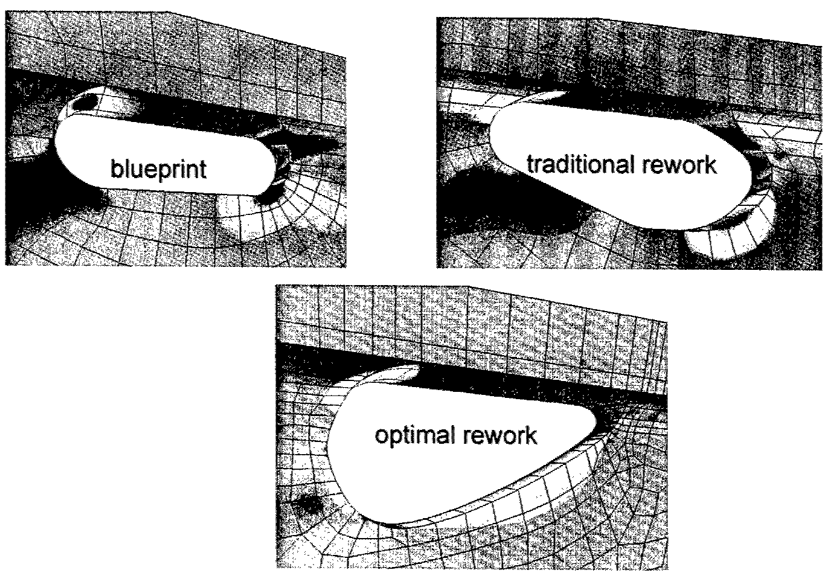
The resulting precise free-form shapes render the local notch stress distributions significantly more uniform and typically provide a 30-40% reduction in stresses as compared to the current rework that exists for aircraft in service with the RAAF. Typical rework shapes are shown in Figure 18, where they are compared to blueprint and/or traditional reworks.

Numerical predictions of peak strains for the optimal reworks have been validated by strain gauge results obtained by full-scale static wing tests. Full-scale damage tolerance and durability tests have been conducted to demonstrate experimentally the expected improvement in fatigue life [12]. A Durability And Damage Tolerance Analysis is in progress. It is anticipated that the stress reductions achieved in this work will be sufficient to provide a basis for extending inspection intervals by at least a factor of two at the critical locations. This will provide significant cost savings and increased aircraft availability for the RAAF.

The F-111 wing optimisation modification (WOM) has been accepted by the RAAF, and fleet-wide implementation commenced in 2002. The modification is effected by electro discharge machining the free-form optimal shapes in the D6ac steel wing pivot fitting. A number of F-111 wing sets (including replacement wing sets) have been modified.



(a)



(b)

Figure 18: Typical optimal rework shapes for F-111 WPF critical regions, (a) comparison of optimal to blueprint shapes for stiffener runout region, with 40-45% peak stress reduction, (b) comparison of various hole shapes where the optimal provides about 45% stress reduction compared to blueprint and 30% compared to the standard rework.

11. Heller, M., McDonald, M., Burchill, M. and Watters, K.C., 'F-111 airframe life extension through rework shape optimisation of critical features in the wing pivot fitting', *6th Joint FAA/DoD/NASA Aging Aircraft Conference,* San Francisco, USA, September 16-19, 2002.
12. Watters, K. C., Heller, M., McDonald, M. and Burchill, M., 'Fatigue Management of the F-111 Wing Pivot Fitting Upper Plate', DSTO-TR-1195, January 2002.

8.2.13 Shape Optimisation of Fillet Region for F-111 Wing Pivot Fitting Bushing (R. Evans and M. Heller)

A known cracking problem exists for the upper bushing located between the F-111 wing pivot pin and the upper wing lug, [13]. Figure 19 shows the location of the wing pivot fitting (WPF) bushing. Currently the upper bushing requires replacement every 1000 hours, mainly due to cracking or corrosion damage of the fillet radius, while the lower bushing does not need to be replaced as often. The potential frequency of replacement may also cause undesirable delays and considerable cost for the next F-111 wing test undertaken in Air Vehicles Division.

In the present work, subject to the simplifying assumptions made, an improved design of the precise profile of the fillet region of the bushing is determined with the aim of minimising the peak stresses in this region [14]. The design is developed through the application of iterative finite-element based shape optimisation as previously used by AVD staff. Here significant stress reductions of approximately 30% are predicted by incorporation of the specific profile provided, as shown in Figure 20. This would indicate a significant reduction in crack growth rates for these bushes as compared to the existing bush design. The revised design has relevance for possible implementation on fleet aircraft, as well as for wing test articles that may be fatigue tested in the future as part of the F-111 Sole Operator Program. Some optimised bushes have now been manufactured, and they are expected to be used on the upcoming F-111 wing fatigue test.

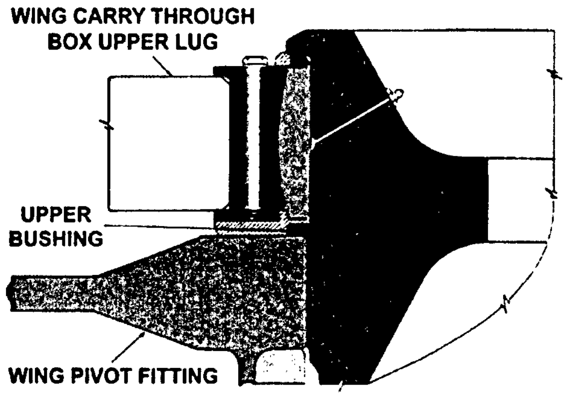


Figure 19: Cross section of F-111 upper pivot-pin region - the bushing is shown in red.

Figure 20: Local distribution of maximum principal stress for F-111 WPF bush: (a) nominal fillet, and (b) optimal non-circular shaped fillet (both rotated by 90 degrees clockwise, as compared to Figure 19).

13. Kaye, R. and Heller, M., 'Structural shape optimisation by iterative finite element solution', *Defence Science and Technology Organisation*, DSTO-RR-0105, June 1997.
14. Waldman, W., Heller, M., McDonald, M. and Chen, G., 'Developments in rework shape optimisation for life extension of aging airframes', *Third Australasian Congress on Applied Mechanics, ACAM 2002,* University of Sydney, Sydney, Australia, pp. 695–702, 20–22 February, 2002.
15. Evans, R. and Heller, M., 'Shape optimisation of fillet region for F-111 wing pivot fitting', *Defence Science and Technology Organisation,* File B2/129 Part 2, 10 February 2003.

8.2.14 Automated Nondestructive Inspection for Cracks in F-111 Wing Planks (G. Hugo, C. Harding, S. Bowles, P. Virtue, DSTO, G. Craven, D. Gannaway, D. Ward, RAAF NDTSL)

DSTO and RAAF Non-Destructive Testing Standards Laboratory are currently evaluating automated nondestructive inspection systems for detection of first and second layer cracks at Taper-lok fasteners in the lower wing skin of the F-111. Whilst such systems have been used by USAF to detect fatigue cracks in C-141 wings, they have not been deployed by RAAF previously.

Two systems have been selected for detailed evaluation: the Boeing MAUS and the SAIC Ultra Image International UltraSpect-MP. The primary focus will be on pulse-echo ultrasonic inspections. However, low-frequency eddy-current techniques will also be examined.

The performance of the systems for detection of cracks will be assessed under field conditions using an F-111 wing containing 60 EDM notches of different sizes and geometries inserted in-situ inside the Taper-lok fastener holes. Coupon specimens containing fatigue cracks will also be examined. Susceptibility of the automated inspections to false calls will be an important consideration. This is dictated by access limitations to the F-111 lower wing skin fasteners, which require that the whole upper wing plank must be removed in order to remove any lower skin fasteners for supplementary inspections or rework. Following assessment of the capabilities of the candidate systems, an inspection procedure will be developed for field application. A probability of detection trial will be conducted to validate the performance of the fielded inspection.

8.2.15 Australian Contribution to the International P-3 Service Life Assessment Program (SLAP) (P. Jackson, DSTO).

Australia is participating in a collaborative program with the USN, Canadian Forces and the Netherlands to conduct a series of full scale fatigue tests of the P-3C aircraft. The program, titled the P-3 Service Life Assessment program (SLAP), commenced in 1999. Australia's technical contribution to the P-3 SLAP program is being led by DSTO and consists of the analysis and delivery of flight loads data from a RAAF P-3 flight loads test program, the conduct of a full scale fatigue test on a retired P-3C empennage structure, and the conduct of a teardown on an ex RAAF and RNZAF P-3B(H) wing.

The flight loads test program was completed in 1999 and is the first time that P-3 flight loads have been collected since the early 1960's. The data has been used in the SLAP program not only to verify the accuracy of the analytically developed fatigue test loads, but also to correct areas of error and, in one case, to identify and characterise a previously unknown loading action. The teardown of a retired in-service wing has been completed by a contractor, Australian Aerospace (the maintenance contractor for the RAAF P-3 aircraft and a subsidiary of EADS). Findings included limited fatigue cracking but corrosion damage typical of the RAAF fleet.

The empennage full-scale fatigue test is currently underway at DSTO Melbourne, Australia. After a two year test build up and commissioning phase, test cycling commenced in January of 2002. The initial contractual period of 30000 test hours was completed by November that year and a period of extended testing has now commenced. This phase of testing is novel in that significant load augmentation has been applied to the test sequence in order to speed up the rate of damage accrual. Damage tolerance testing with the insertion of cracks in critical areas and a Residual Strength test are planned. Test teardown and interpretation activities will follow and the data will provide structural clearance information and on-going structural integrity management data for the RAAF P-3C fleet.

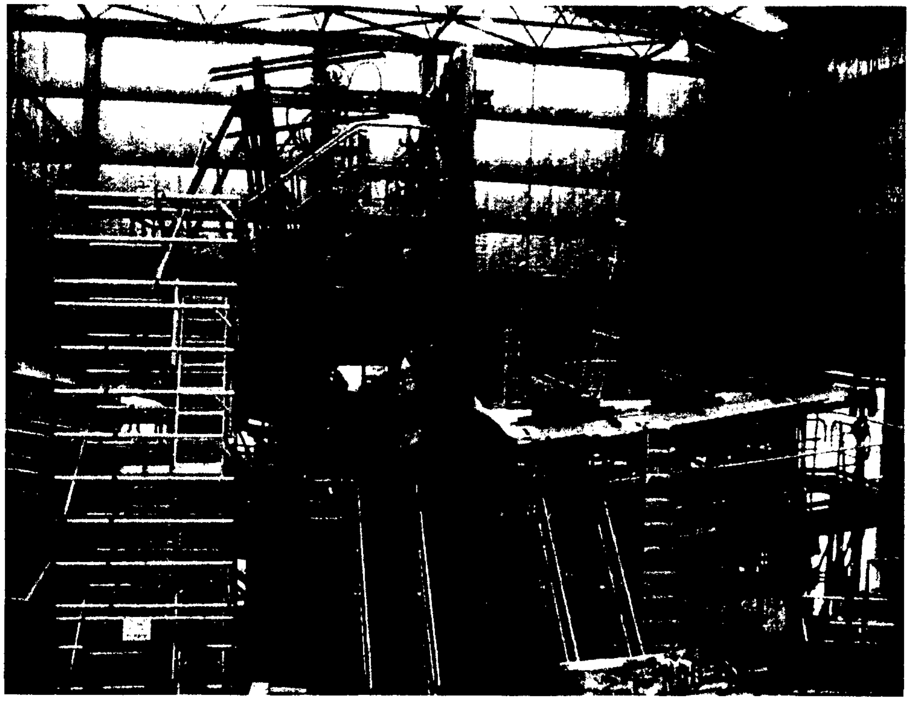


Figure 21: P-3C tail test at DSTO.

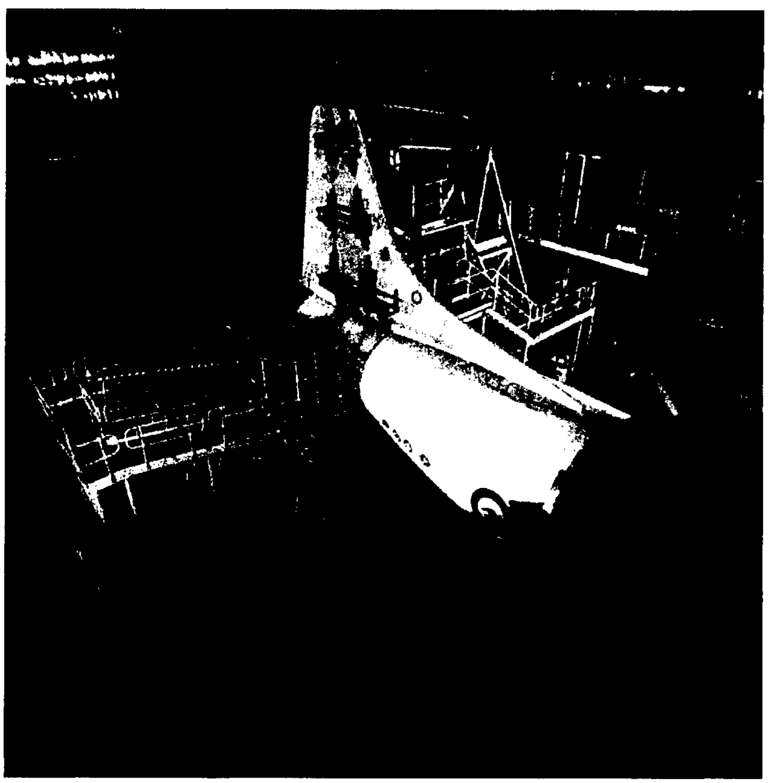


Figure 22: P-3C Tail test at DSTO.

16. Jackson P. & Cardrick A. W., "The challenge of Conducting a Meaningful Full Scale Fatigue Test on a Transport Aircraft Empennage", to be presented at ICAF 2003, Lucerne, Switzerland, May 2003.

8.2.16 Characterisation of P3 Missions Using the Structural Data Recording Set (R. Kashyap, Aerostructures)

RAAF P-3 aircraft are managed under a Safe Life philosophy, supplemented by limited inspection, where the life of the aircraft is monitored using a Fatigue Life Index (FLI). An index of 100 FLI represents the certified Life of Type (LOT) of the P-3 aircraft structure. FLI is calculated using the Service Life Evaluation Program II (SLEP II), a fatigue tracking program developed by Lockheed in 1989. SLEP II is based on pre-defined mission profiles and load spectra which may or may not be representative of actual usage. One way of establishing the difference between current usage and that assumed by SLEP II to calculate aircraft fatigue life is by using the Structural Data Recording Set (SDRS) data. Data from the SDRS can be used to plot actual flight profiles which can then be compared with that assumed by SLEP II to calculate FLI and assess validity of estimated damage.

The RAAF P-3 fleet are undergoing avionics upgrades as part of Project Air 5276 (PA 5276) which has the potential for changing the typical profiles of RAAF P-3C missions. To characterise pre-PA5276 missions, and allow better comparisons of past and future profiles, two aircraft, scheduled to be the last to receive the upgrade, were fitted with SDRS.

In October 2002 Aerostructures were tasked to determine if the data collected from these two aircraft was adequate to characterise pre-5276 missions. The adequacy of the data related to its quality and quantity.

It was determined that the SDRS was functional and delivering meaningful data that allowed confirmation of mission type in most cases. In most cases the collected samples were sufficient but for some mission types additional data was required to characterise those mission types. While data quantity was under question, the quality of the data provided by the SDRS gives confidence that the SDRS will be useful in its intended role.

The comparison of actual profiles plotted from SDRS with those assumed in SLEP II has potential to provide a means of revising calculated fatigue damage that will more accurately reflect actual flying. As a consequence, significant benefits, both economic and safety related, may be realised.

17. Aerostructures Letter Report, SDRS Project Management – Characterisation of pre-PA5276 Missions, 4-4-2-2-8.9GR2068, 6 November 2002.

8.2.17 Sensitivity Study Of DTA Inspection Intervals For RAAF C-130 Fleet (C. K. Rider, Aerostructures)

The RAAF C-130 fleet consists of 12 C-130H aircraft delivered in 1978 and 12 C-130J-30 aircraft which recently replaced the earlier C-130E model. Apart from the stretched fuselage, the two current models are structurally similar, with the fatigue assessment of the wing considering the same set of nominated locations.

Structural airworthiness is maintained by inspections based on a Damage Tolerance Assessment using slow crack growth analyses from 0.05 inch initial flaws to critical crack sizes for the limit load stresses at the chosen locations. Crack growth for wing lower surface structure, caused mainly by stress cycles caused by gust loads, is estimated by Forman's equation with retardation introduced by a method developed by Hsu.

Utilisation of the C-130H and the C-130J-30 aircraft in the fleet is likely to be significantly different, because of the varied capabilities of the two models. Consequently, in order to assess both predicted differences in required inspection intervals, and the relevance of any fatigue damage found in any aircraft for both models, the sensitivity of predicted crack growth to variations in usage parameters needs to be estimated.

Aerostructures has undertaken a program of work examining the effects of changes in usage related to variations in the set of missions, cargo and fuel weights and typical flight profiles.

8.2.18 Analysis of Widespread Fatigue Damage in B707 Wing Splice (P. White, DSTO)

Work is currently underway on assessing the widespread (interacting) fatigue cracking of the upper WS360 wing splice on the Boeing 707 aircraft. The orientation of cracking in this location allows for possible interaction of cracking at this location. Previous analyses have not considered interaction effects, but it is hoped that the probabilistic risk assessment of this location can account for the multiple interaction of cracks around the largest expected crack in the splice plate. It remains to be demonstrated that for a large number of cracks the largest crack will be involved in the most critical failure. Additional complications occur because the loading at each of the potential cracked fasteners is the same. This fact combined with the large number of fasteners in the joint, and being a transport aircraft where the probability of the largest load occurring is low, the probability of failure for these large number of locations cannot be simply combined. Thus the solution of this problem contains a number of interesting aspects to be resolved in using probabilistic methods in the analysis of widespread fatigue damage.

8.2.19 Implementation of Caribou Safety-By-Inspection Program (A. Bibby, Aerostructures)

The Royal Australian Air Force (RAAF) operates 14 DHC-4 Caribou aircraft in a short to medium range utility transport role. The expiration of the safe life of the wing and extension of planned withdrawal to 2010 necessitated an amendment to the original certification basis to provide an alternative fatigue management strategy past the safe life. The damage tolerance requirements of FAR 25.571 (amendment 25-96) were selected as the basis for ongoing airworthiness. A Safety-By-Inspection (SBI) program compliant with FAR 25.571 was therefore developed by Aerostructures.

Critical locations requiring recurring inspection under the SBI program were identified from fatigue test results, design documentation and in-service experience (Reference 18). Recurring inspection intervals for the critical locations were developed based on Damage Tolerance Analyses (DTA) (Reference 19).

Implementation of the SBI program is about to commence. Aerostructures has produced an implementation plan for the SBI program that details the initial inspection of each location under the SBI program and the recurring inspection requirements. The inspections will be incorporated into the current maintenance schedule to minimize impact to fleet availability. The implementation considers each aircraft individually and includes a risk assessment of the scheduling of each inspection.

Any previous inspections of the critical locations and the method used (visual, eddy current, etc) were required to be considered in the timing of the initial inspection. The initial inspections on individual aircraft were adjusted accordingly using the crack growth curves generated in the DTA.

18. Aerostructures, *Caribou Wing Safety By Inspection Program Stage 1 – Selection of Critical Locations*, ER-DHC4-51-ASM210, Revision 2, 8 July 2002
19. Aerostructures, *Caribou Wing Safety By Inspection Program Stage 4 – Damage Tolerance Analysis and Inspection Intervals*, ER-DHC4-51-ASM247, Revision 2, 8 July 2002

8.2.20 Australian Army Sikorsky S-70A-9 Black Hawk - Flight Loads Survey (R. P. Boykett, DSTO).

Analysis continues on the data acquired during the joint USAF/ADF flight loads measurement program, which was conducted in 2000 and reported in the 2001 ICAF Review. So far, analysis has concentrated on identification of manoeuvres that contribute fatigue damage to airframe components. In addition, algorithms for structural health monitoring have been refined by illustrating how airframe cracks can be identified from strain gauges located as far as 2 metres from the damage site. The extensive flight load data for dynamic components has also been used to predict revised retirement lives against the Australian-specific usage spectrum.

8.3 FATIGUE OF CIVIL AIRCRAFT

No specific fatigue related programs were offered for this report, but Steve Swift of the Civil Aviation and Safety Agency (CASA), highlighted two significant issues.

8.3.1 Significance of SIDs (S. Swift, CASA)

Cessna's SIDs (Supplemental Inspection Documents) for its 300 and 400 series twin-engined aircraft are a model of the retrospective application of damage tolerance analysis to old airframes that were never designed to be managed that way. Unlike the old big airliners, which have had SIDs for years, the old little airliners were not even designed to be fail-safe.

The major safety benefits that have come from Cessna's damage tolerance analysis have come from the simple fundamentals. For example, by inspecting *all* PSEs, not just those that are easily accessible, the SIDs have uncovered cracks in areas that the old inspection program had no hope of finding. Already, the SIDs have saved lives.

Also, by checking whether cracks are detectable before they become dangerous, Cessna found cases where they were not. One was the wing spar. Fifteen years ago, the Australian Civil Aviation Safety Authority (CASA) came to the same conclusion – that the spar is uninspectable. That is why, since then, CASA has required old Cessnas to have a reinforcing strap fitted to the spar. Cessna's new spar strap is very similar.

It would be disappointing if Cessna's efforts are wasted. As long as compliance with the SID is voluntary in the United States, it will probably be voluntary elsewhere. Few operators are expected to comply. Against this, CASA is trying to enforce the SID in Australia.

There is a message for us at ICAF. We not only need to convince *ourselves* that what we are doing is worthwhile. We also need to convince *others*.

8.3.2 Communication (S. Swift, CASA)

This story follows on from the last, because it relates to the way we communicate with end users. Sometimes, bad communication can undo good analysis.

Recently, communication was a contributory factor that led to an event that threatened the safety of Australian air travellers and triggered the collapse of a major national airline. After revising its damage tolerance analysis, an airliner manufacturer innocently and inadvertently presented the resulting changes to an inspection program in a way that could be confusing. One airline was confused and failed to schedule several structural inspections until they were well overdue. Passengers had been at risk. Affected airliners were grounded until inspected. Some were cracked.

There is a message for us at ICAF. We take care with our *analysis.* We also need to take care with its *presentation.* With damage tolerance, style can be just as important as substance. We allow for human error when we design new aircraft. We should leave no room for misinterpretation when we design their maintenance publications.

8.4 FATIGUE-RELATED RESEARCH PROGRAMS

8.4.1 Effect of Environmental Degradation on Structural Integrity (K. Sharp, DSTO)

DSTO has developed a number of models to predict the effect of corrosion has on fatigue life. The initial work was based on specific corrosion problems within the RAAF fleet, but the models have since been expanded to become increasing general in the type of alloy and the type of corrosion.

8.4.1.1 Pitting

A very good pitting model was developed for D6ac as part of the F-111 Recovery Program. The model predicted the location of the failure of a randomly pitted high k_t specimen simulating the fuel vent hole runout, and was shown to give good prediction of the fatigue lives of pitted steel specimens (Figure 23).

This model is now being used to assess the status of a number of durability parts on the F-111. By inputting the ECS data based on fleet pitting data a determination can be made as to the effect of corrosion pitting on inspection intervals.

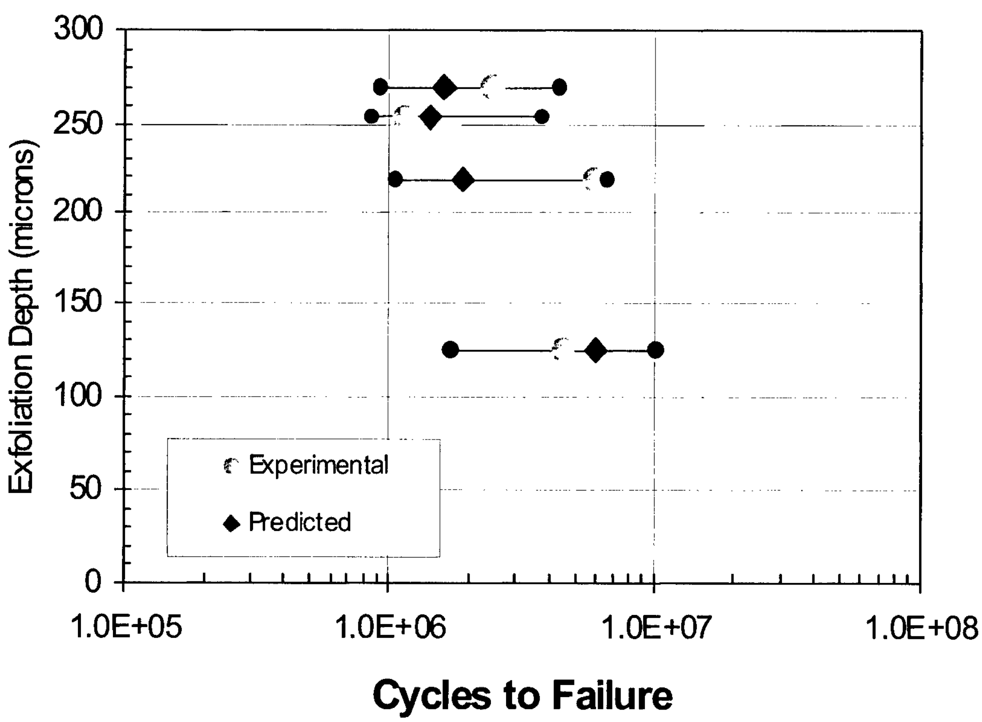


Figure 23: Prediction of remaining life for pitted steel specimens, and experimental results.

During structural testing of the F-111 at DSTO, corrosion pits were introduced into the critical region. The modelling indicated that if any cracks form they would be very small - of the order of 100 μm. A recent teardown of the area indicated that a small crack had initiated at the corrosion pit as predicted by the model. Fractographic analysis is underway to determine the fatigue growth rate from the pit.

Further developments are being made on the ECS type approach to pitting corrosion for aluminium alloys. While the models works reasonably well, a greater degree of scatter is observed due to the more complex nature of pits in aluminium alloys compared to pits in steel.

8.4.1.2 Exfoliation:
A very detailed but impractical model was initially developed for exfoliation. After the excellent results with pitting corrosion an ECS approach was tested. Some recent results are shown in Figure 24.

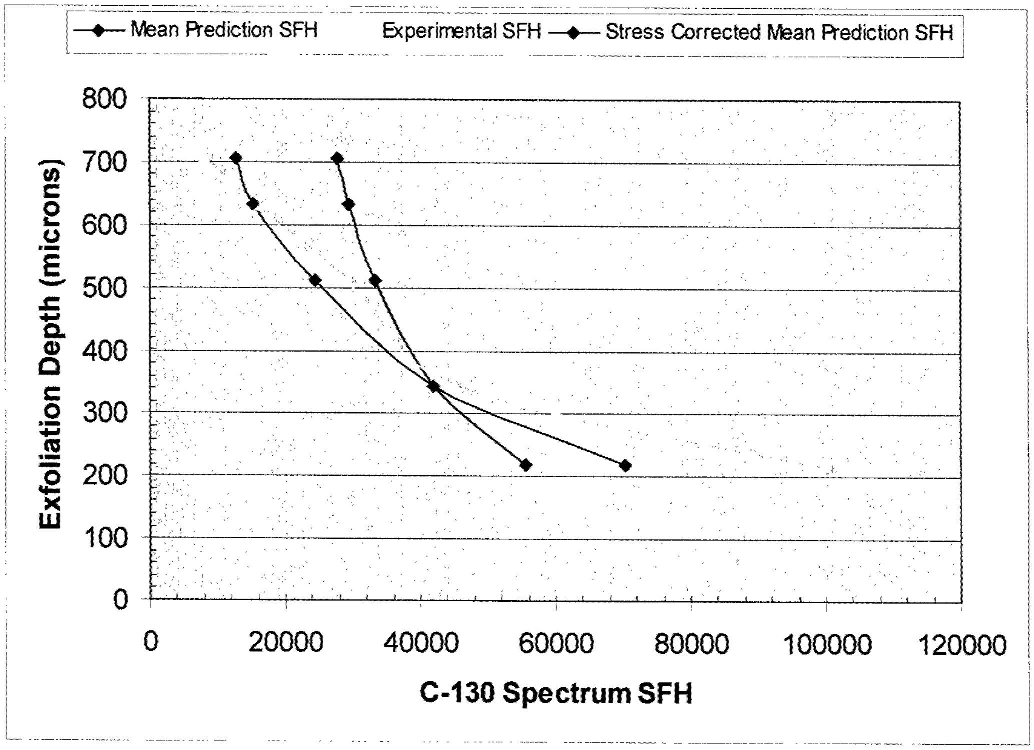


Figure 24: Exfoliation Corrosion modelling - a comparison of experimental fatigue life results vs mean prediction and mean prediction corrected for stress.

While the modelling looks good a number of problems have arisen; in real life exfoliation corrosion is a lot shallower than the extremes of this modelling. In fact it occurs near a knee in the ECS vs exfoliation depth curve, making this region very sensitive to exfoliation depth. Also when do we account for increased stress (through loss of thickness)? More work is therefore needed to increase the robustness of the model.

Research is also underway looking at better methods to determine the extent of exfoliation corrosion damage. This is extremely important, as it must form the basis of the corrosion metric used in the predictive model.

8.4.1.3 Total Life
The success of this type of crack growth modelling is now supporting the development of a holistic life model. This type of modelling is extremely complex but it is believed that due to the extensive data collected in Australia on both aircraft and base corrosion rates and the understanding of how pitting affects fatigue life, a total life model could be developed. The critical question is the extent to which some uncertainties might dominate the model.

At the same time work on corrosion sensors is being undertaken to assist in any total life model predictions. The major effort, however, is focussed on the options for the corrosion metric being measured and how this can be related back to the extent of corrosion.

8.4.2 Structural Integrity Assessment of Pitting Corrosion in Aircraft Structures (D. Hay, C. Urbani, CSIRO, A. J. Stonham, S. H. Spence, N. M. Williams, BAESystems, M. R. Bache, A. R. Ward, and W. J. Evans, UWS, and B. R. Crawford, C. Loader, K. Sharp and G. Clark, DSTO)

SICAS (Structural Integrity assessment of pitting Corrosion in Aircraft Structures) is a major international research project commissioned by BAE SYSTEMS addressing the effect of pitting corrosion on the fatigue behaviour of AA 7010-T7651. It is being undertaken by CSIRO Manufacturing and Infrastructure Technology and DSTO Air Vehicles Division in Australia and BAE SYSTEMS and the University of Wales Swansea in the UK. The project's aim is to develop a method of modelling corrosion pitting damage as equivalent initial flaws for airframe fatigue life calculations. If successful, this will mean pitting corrosion can be evaluated as part of an aircraft's Durability and Damage Tolerance Analyses (DADTA). This will greatly reduce maintenance costs.

A unique aspect of SICAS is the large number of repeat tests that have been conducted at various loading conditions. This has permitted the rigorous statistical analysis of the results. This rigour will allow the robustness and factors influencing the EIFS distribution to be investigated. Three surface conditions were investigated (as-machined, anodised-and-primed and anodised-and-corroded). In the case of the anodised-and-corroded condition, which was of principal interest, constant amplitude tests were conducted at twelve loading conditions (combining four peak stresses and three R-ratios), the aim being to set peak stress conditions to produce fatigue failures at target lives between approximately 10^4 and $5x10^5$ cycles. The R-ratios investigated were -0.3, 0.1 and 0.5.

Initial investigations, at R = 0.1, using a pit-area metric and fatigue crack growth data for 7010-T7651 from the NASGRO database have shown a reasonably good correlation between actual fatigue lives and those predicted using AFGROW, a fatigue life prediction code. These results are shown in Figure 1. However, the NASGRO fatigue tests were conducted in laboratory air while fatigue testing in the SICAS program was conducted at high (> 90% RH) humidities. Therefore, fatigue crack growth tests were conducted as part of the SICAS program. These were combined with crack growth measurements from marker bands to create a master fatigue crack growth curve for use in determining the EIFS distribution. The data obtained for R = 0.1 at 90% RH are shown in Figure 2. As can be seen the two data sets exhibit similar Paris Law exponents but do not overlap. Further experiments are being conducted to determine the cause of the observed differences in fatigue crack growth rates. Additionally, fatigue crack growth and marker band data are being collected for load ratios of R = 0.5 and R = -0.3. These will be reported subsequently.

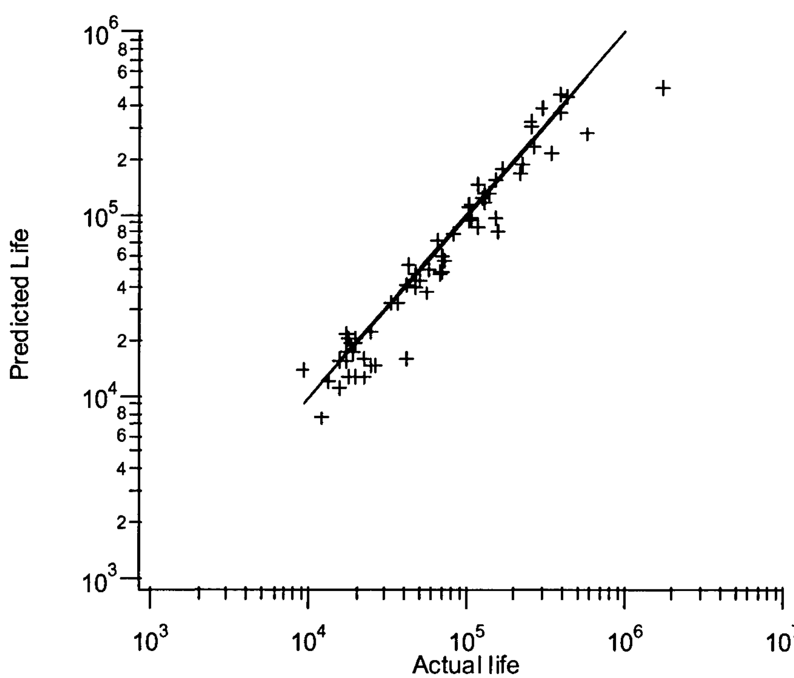


Figure 25: Comparison of actual and predicted lives for 7010-T7651 using NASGRO data included in the AFGROW fatigue life prediction code.

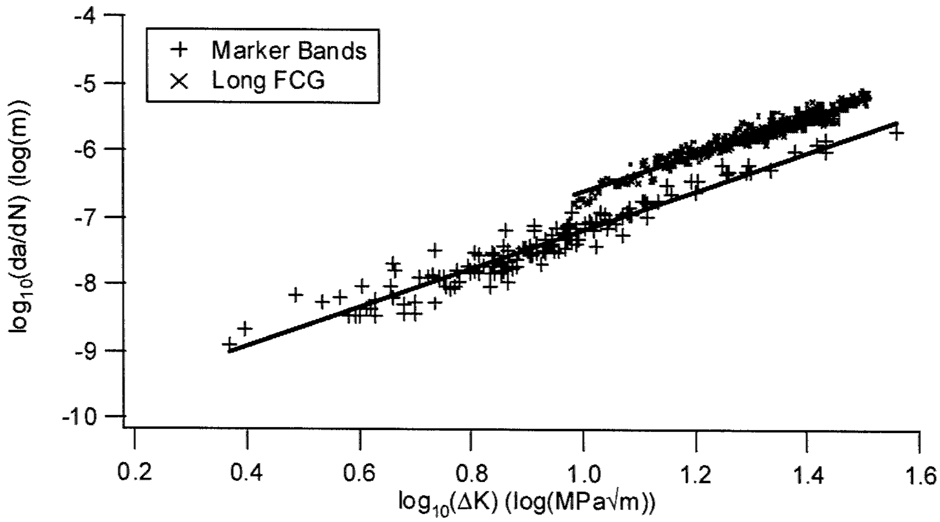


Figure 26: Comparison of long fatigue crack growth data with that obtained from marker band analysis

It is anticipated that the research outcomes will allow identification of the critical corrosion features producing a degradation in fatigue life. This will provide a basis for decision making concerning such corrosion and its management. This approach aims to minimise future testing by optimising the analytical design input, and is, therefore, expected to retain some flexibility, obviating the requirement for extensive test support if service conditions change significantly. While the programme focuses on military aerospace materials, the approaches developed are expected to be transferable to civil applications.

8.4.3 Effect of Corrosion and Corrosion Treatment on Fatigue Behaviour of Fuselage Lap Joints (B. Hinton, S. Russo, G. Clark, K. Sharp, DSTO, and K. Shankar, ADFA)

Previous experimental studies on the effect of Corrosion Preventative Compounds (CPCs) on mechanically fastened joints undertaken jointly by DSTO and ADFA have shown conclusively that application of CPCs significantly reduces fatigue life of one and half dog bone specimens especially at low stress levels. Similar results have also been obtained from bolted double lap joints tested by ADFA. The current project aims to extend this work to examine the influence of corrosion and corrosion treatment on riveted airframe lap splices, such as those of pressurised passenger aircraft. Fatigue tests are being conducted on riveted lap joints samples simulating those on the front fuselage of Boeing 737-200, in dry and in humid environment, with and without application of CPCs. The results will be compared to tests on samples with pre-corrosion to determine the relative effects of corrosion and corrosion treatment on the residual life of the joint.

8.4.4 Effect of Corrosion Prevention Compounds on Fatigue Crack Propagation Rates in Aluminium Alloys Compact Specimens (K. Shankar, ADFA)

The application of Corrosion Preventative Compounds (CPCs) to increase durability by controlling corrosion in airframe structural joints is widespread both in civilian and military environment. However fatigue tests on double lap joints, one and half dog bone specimens and riveted lap joints have shown that treatment with CPC significantly reduces the fatigue life of the joint, especially at low stress levels. While some of this is attributable to the reduction in friction due to the lubricative nature of CPCs, it appears that at least a part of the reduction in life is caused by an increase in crack propagation rates in the presence of CPCs. The current work involves measurement of fatigue crack growth rates in compact tension specimens made of aircraft grade aluminium alloys, immersed in CPC. Comparison tests are performed in laboratory air and in distilled water. Initial results, from tests performed on Alloy 2024-T351 indicate a statistically significant increase in crack propagation rate in the presence of CPC compared to those observed in dry air and distilled water. The study also includes fractography of the fracture surfaces with a view to identifying the mechanisms responsible for increase in crack propagation rates.

8.4.5 Development of Improved Structural Risk Assessment Methodology (P. White, DSTO).

In the past two years work has progressed on the structural risk assessment of aircraft. Following on from earlier work on the Boeing 707 risk assessment of widespread (non-interacting) fatigue damage of the lower wing skins, an analysis was undertaken on the F/A-18 centre fuselage bulkheads. This work was similar in approach to previous work using probabilistic models of the parameters involved in the fracture mechanics analysis of the structure. Variables considered were the equivalent flaw size, crack growth variability, fracture toughness, load variability and variation in the prediction of the fatigue damage consumed by the fleet tracking computer program. This program estimates the life consumed by the aircraft based on the recorded strain history of each aircraft. Scatter in all these variables were used to determine the single flight probability of failure and the total probability of failure. Fractographic examination of coupon test data was used to determine crack growth curves which were then back extrapolated to determine equivalent flaw sizes. Fatigue lives obtained from full-scale fatigue tests were used to estimate the apparent fatigue stress at each location and average crack growth rates. The probability of failure from the critical locations were combined to compare with the acceptable levels of failure for an aircraft from DEFSTAN and MIL standards. The probability of failure predictions were shown to be sensitive to the initial flaw size data, which indicated more effort is needed to more accurately determine the tail of the distribution used.

8.4.6 Shape Optimisation for Airframe Life Extension – DSTO Overview (M. Heller, DSTO)

Research commenced in 1996 to develop a capability in DSTOs Air Vehicles Division (formerly Airframes and Engines Division) for local shape optimisation in the context of life extension of ageing airframe components in service with the Royal Australian Air Force (RAAF), [20]. The essential aim is to reduce peak stresses at local stress concentrating features, and thereby delay or eliminate further fatigue damage growth. Implementation of such a strategy can lead to large savings in the cost of aircraft ownership, via reduced maintenance costs and/or extension of the aircraft withdrawal date. The implementation and effective use of iterative finite element (FE) procedures is a key feature of the work. The typical numerical algorithms for specifying the optimal shapes have been based on an analogy with biological growth. A key characteristic of the optimal shapes is that the stresses are minimised and rendered uniform along segments of the stress-concentrating boundary [20 and 21].

The investigations to date have focussed on the optimal design of precise rework profiles for metallic components, as well as on the local redesign of bonded repairs to metallic airframe components. Optimal reworking has the complication that only material removal is allowed in order to obtain the new shape, while the main complication for bonded repairs is that this approach involves a multi-material system where the key aim is to minimise peak stresses in the adhesive bond layer. Within the constraints of the available staff resources, three themes are continuing to be pursued as follows:

- (i) development and enhancement of generic approaches for achieving optimal shapes, including the use of in-house codes and commercial FE codes;
- (ii) solution of generic problems typical of stress concentrating features of airframe components, such as geometrically constrained holes and fillets (for which only limited good quality published solutions have been previously available); and
- (iii) shape optimisation for practical applications and demonstrators, for the life extension of RAAF airframe components. Here manufacturing constraints and robustness issues are of key importance. Currently, appropriate external interactions are being sought to complement the internal DSTO efforts.

Information relating to specific investigations, either completed or undertaken in the review period is given in the following sections.

20. Kaye, R. and Heller, M., 'Structural shape optimisation by iterative finite element solution', *Defence Science and Technology Organisation*, DSTO-RR-0105, June 1997.
21. Waldman, W., Heller, M., McDonald, M. and Chen, G., 'Developments in rework shape optimisation for life extension of aging airframes', *Third Australasian Congress on Applied Mechanics, ACAM 2002,* University of Sydney, Sydney, Australia, pp. 695–702, 20–22 February, 2002.

8.4.7 Structural Optimisation of Bonded Repairs (R. Kaye and M. Heller, DSTO)

Reports summarising the status of DSTO research on shape optimisation of bonded repairs have been published in the review period [22 and 23]. The motivation for work on this topic is that for certain practical applications, unacceptably high adhesive stresses can occur in the adhesive layer for bonded repairs. Hence it is desirable to develop and implement procedures to improve the design of bonded repairs. It is expected that such improvements would lead to further improved durability of bonded repairs.

In the present work three distinct issues are investigated in the context of improved through-thickness and in-plane shaping of bonded repairs.

Firstly, a finite element based design sensitivity approach is used for through-thickness shape optimisation of a double lap-joint configuration and an F/A-18 bulkhead reinforcement. Here both the shapes of the patch and the adhesive layer are allowed to vary either in the taper, crack or reinforcement regions. Significant improvements over conventional designs are obtained, as assessed by the reduction in peak adhesive stresses.

Secondly, discrete stepping of the taper region of a multilayer patch was investigated mathematically and via an iterative finite element approach for reducing the stress in the adhesive layer. Here the advantage of tailoring the length and stiffness of the first few layers in the patch to reduce peak stresses has been highlighted.

One example considered is the reinforcement to an F/A-18 aileron hinge. The numerical model of the aileron hinge is shown in Figure 27, while Figure 28 shows the use on non-uniform patch stepping to reduce adhesive shear stresses.

Finally exploratory numerical analysis of in-plane reinforcement shapes has also been undertaken using an inclusion analogy. The patch consists of one layer of boron/epoxy laminate (to represent a limiting case) bonded to an aluminium plate of 3 mm thickness, as shown in Figure 29(a). Here it has been found that the optimality of in-plane reinforcement shapes, in terms of the plate stress concentration near the patch ends, depends on the aspect ratio of the reinforcement, see Figure 29(b). For uniaxially loaded plates, rectangular patch shapes are essentially optimal for aspect ratios of 2:1 and higher. For lower aspect ratios, based on the non-uniformity of the local stresses (not shown), neither rectangular nor elliptical shapes are optimal, although elliptical patches are better. This indicates scope for further work.

Recent experimental work conducted by DSTO has confirmed the significant advantage in fatigue life, for bonded joints using optimised non-uniform patch stepping, as compared to standard uniform stepping [24]. Recent numerical work has investigated through-thickness shape optimisation of double lap-joints including effects of differential contraction during curing [25].

22. Heller, M. and Kaye, R., *Structural optimisation of bonded repairs – Chapter 10*, Advances in the bonded composite repair of metallic airframe structures, (Ed. Baker, Rose, and Jones), Elsevier Science limited, 2002.

23. Heller, M., and Kaye, R., 'Computational shape optimisation of bonded repairs', *6th Joint FAA/DoD/NASA Aging Aircraft Conference,* San Francisco, USA, September 16-19, 2002.

24. Baker, A. A., Personal communication, February 2003.

25. Kaye, R. and Heller, M., 'Through-thickness shape optimisation of double lap-joints including effects of differential contraction during curing', *(submitted to Journal of Adhesives and Adhesion, 2003)*.

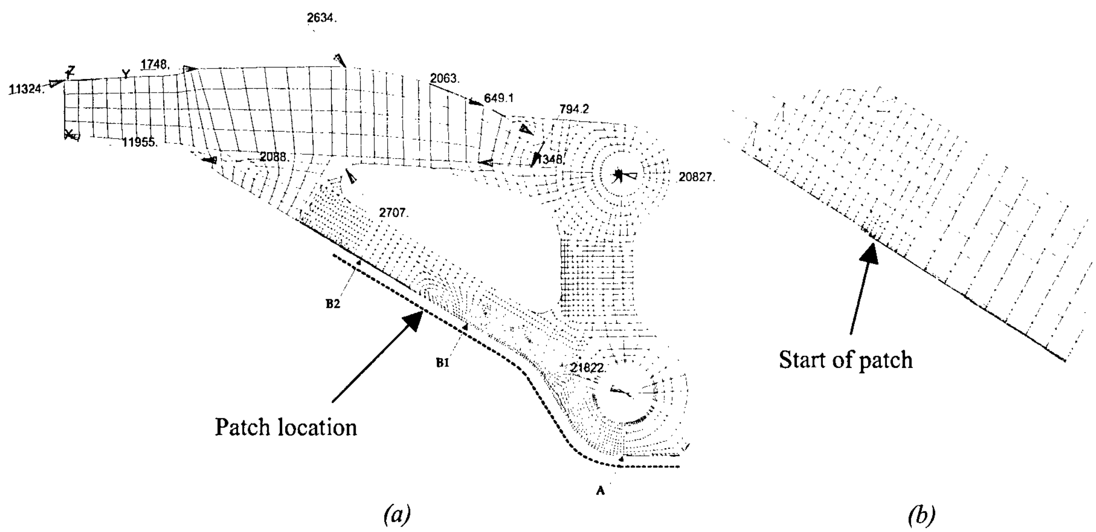


Figure 27: Analysis for optimised stepping of bonded patch for F/A-18 Aileron hinge: (a) overall model, with dominant loading, showing patch location, and (b) mesh refinement in the region at the beginning of the strut end of the F/A-18 aileron hinge patch.

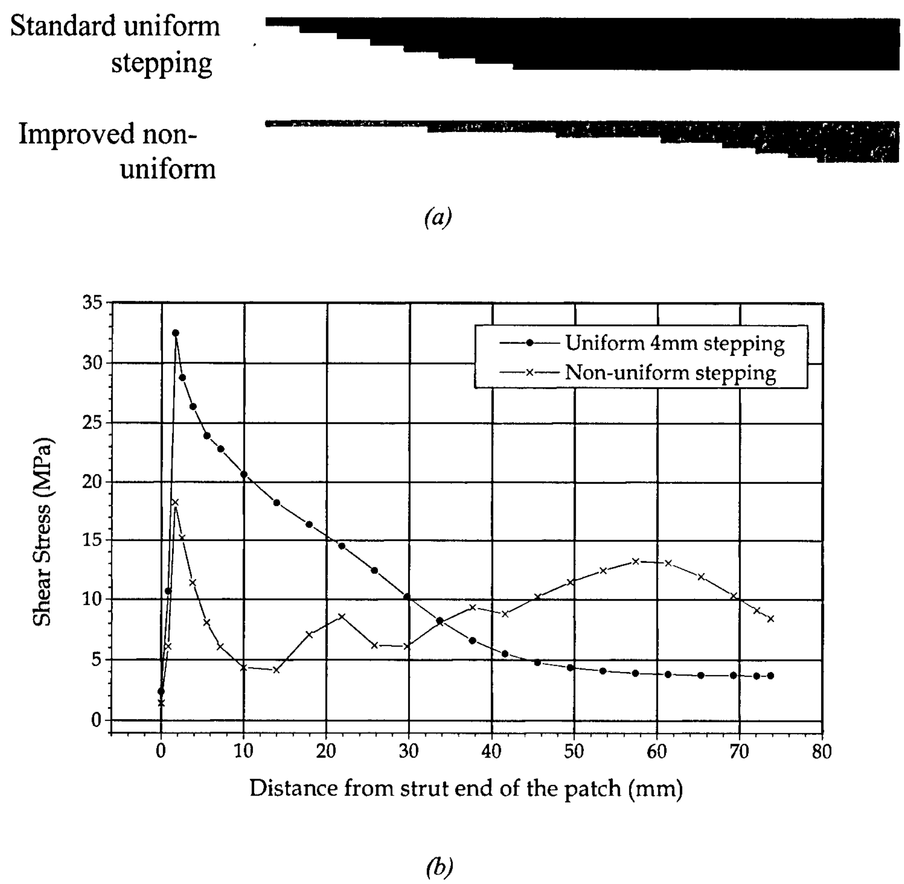


Figure 28: Results for taper stepping of bonded reinforcement to an F/A-18 aileron hinge, (a) stepping arrangements, and (b) adhesive shear stress distributions

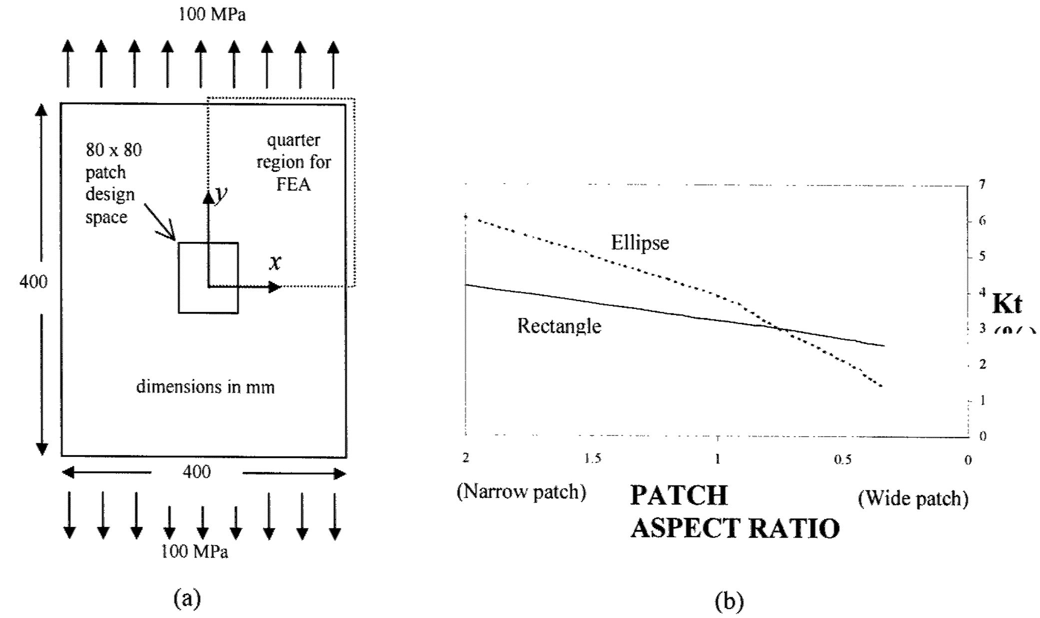


Figure 29: In-plane shape study of patch on plate, (a) geometry and loading arrangement for single layer of boron-epoxy patch on a 3 mm thick aluminium plate, and (b) percentage increase in plate peak stress concentration, as a function of patch aspect ratio.

8.4.8 Rework Shape Optimisation for Two Closely Spaced Holes (W. Waldman, M. Heller, and L .R. F. Rose, DSTO)

Interacting closely spaced circular holes are a common feature in aircraft structures, and these can result in significant plate stress concentrations. In certain situations it is desirable to use hole shapes other than circles to achieve a lower stress concentration. However, there appear to be limited published solutions for optimal hole shapes that could offer an alternative option. In this work [26], the DSTO finite element gradientless shape optimisation procedure is employed to determine optimal hole shapes for two closely spaced holes in a large biaxially loaded plate, as shown in Figure 30. Two distinct biaxial loading cases are considered, namely tensile field (remote principal stresses have the same sign) or mixed field (remote principal stresses have opposite sign). The optimal solution shapes are non-circular (see Figure 31), and the stresses are either uniform around the hole boundary (tensile loading case), or the absolute magnitude of the tangential stresses has been rendered piecewise constant (mixed loading case). The resulting optimal shapes provide very large stress reductions as compared to the initial circular holes. In a key result, it is found that the optimal shapes completely eliminate stress interaction effects, with the stress concentration produced by the optimal holes for both loading cases being identical to the corresponding single hole optimal, as shown in Figure 32. The co-ordinates of the optimal shapes are presented in tabular form that allows them to be readily used by designers, [26].

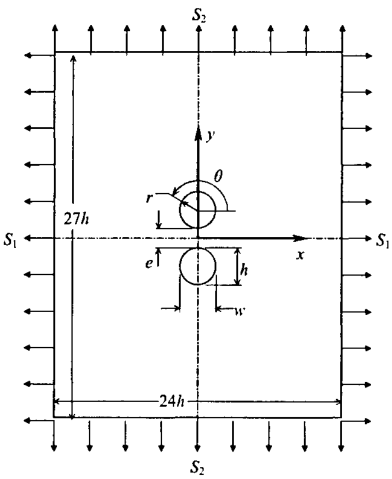


Figure 30: Shape optimisation of two closely spaced holes: plate loading and geometry showing initial circular hole,

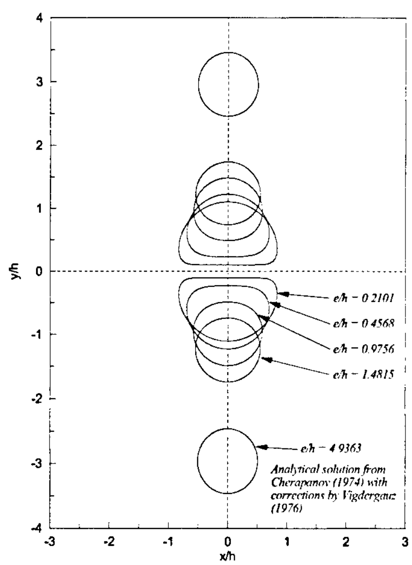


Figure 31: Shape optimisation of two closely spaced holes: shape of optimal holes as a function of interaction distance for loading $S_1=S_2$,

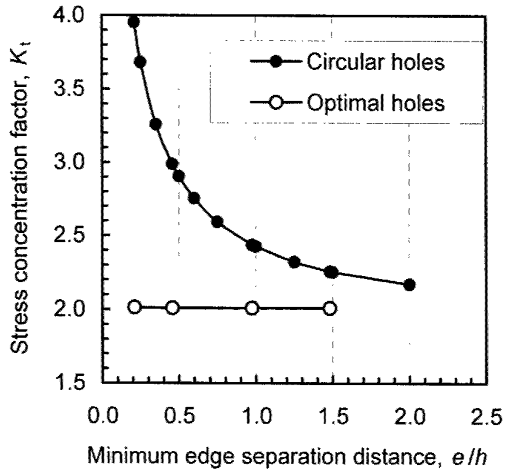


Figure 32: Shape optimisation of two closely spaced holes: comparison of stress concentration for circular and optimal holes for loading $S_1=S_2$.

26. Waldman, W., Heller, M., and Rose, L.R.F., 'Rework shape optimisation for two closely spaced holes', *Defence Science and Technology Organisation,* DSTO-RR-0166, 2003.

8.4.9 Robust Shape Optimisation of Notches for Fatigue Life Extension (M. McDonald and M. Heller, DSTO)

An iterative 2D finite-element-based shape optimisation procedure has been developed which incorporates robust design philosophies, [27]. This has been used to determine precise free-form shapes for a hole in a plate example, with the aim of maximising its fatigue life when exposed to varying load orientations. Past methods have typically considered only a single nominal load orientation, with empirical approaches to deal with orientation variability, thus resulting in sub-optimal solutions. Here a robust stress method is developed that produces a notch shape that minimises the peak stress and renders it constant for a range of load orientations.

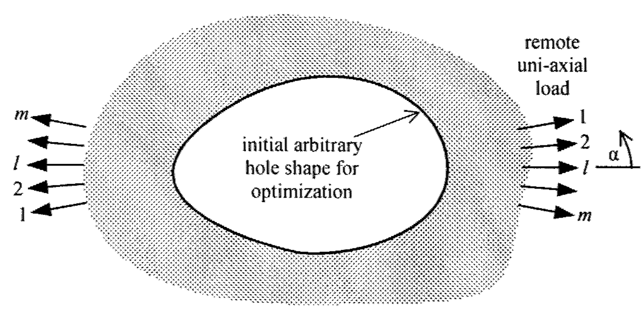


Figure 33: Arrangement for robust stress-based shape optimisation: arbitrary hole in a plate under a remote uni-axial load with varying orientation α, showing the discretised load cases representing the continuous variation,

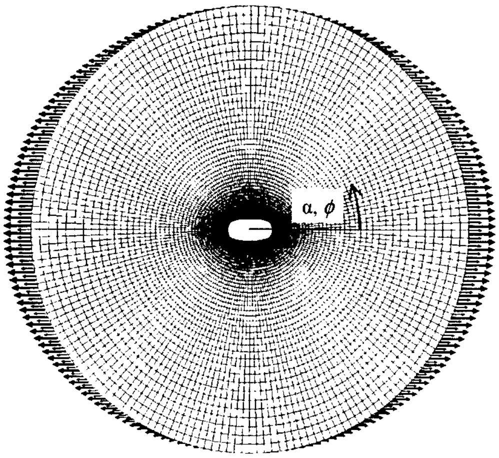


Figure 34: Arrangement for robust stress-based shape optimisation: FE mesh, showing the remote uni-axial stress applied at the nominal inclination angle of $\alpha = 0°$.

Figure 33 and Figure 34 show the arrangement for robust stress based shape optimisation for a sample problem, while the typical results obtained are given in Figure 35.

The key point is that for a prescribed load orientation range used in the optimisation, the peak stress magnitude on the hole boundary is minimised and is fixed in value, for any angle in that range. Furthermore, a more sophisticated robust fatigue-damage optimisation method is then developed to minimise the peak fatigue-damage for a given stochastic distribution of load orientations. Fatigue calculations, for an example problem with significant load orientation variation, show that the robust optimisation methods provide fatigue life extensions 2 to 8 times better than past methods. It is anticipated that the implementation of robust optimal shapes in airframe components would result in improved fatigue-life extension.

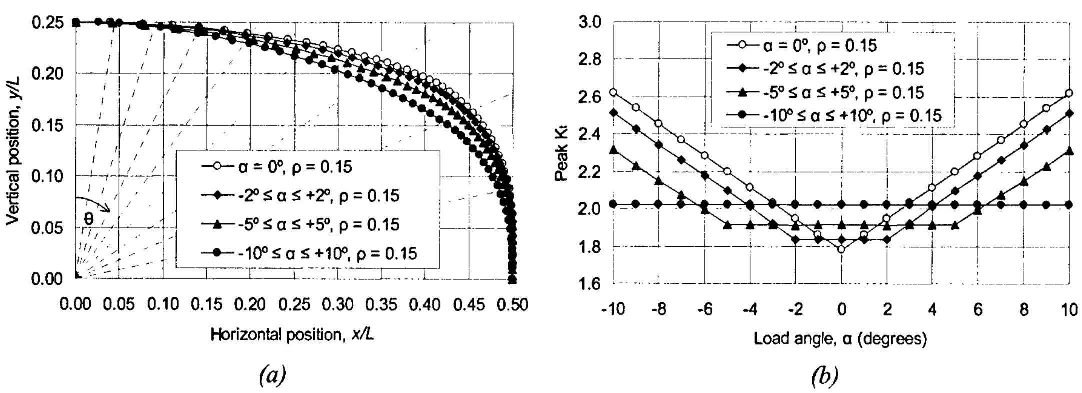


Figure 35: Robust stress minimisation results with a minimum radius constraint of $\rho = 0.15$: *(a) ¼ geometry of optimal hole shapes, for different remote load angle ranges, and (b)* K_t *sensitivity curves for the robust stress minimisation optimal hole shapes with a minimum radius constraint of* $\rho = 0.15$, *for various prescribed load variation range design cases.*

27. McDonald, M., and Heller, M., (2003) 'Robust shape optimisation of notches for fatigue life extension, (submitted to Journal of Structural and Multidisciplinary Optimisation).

8.4.10 Optimal Shapes for Notches and Holes in Flat Plates (M. Burchill and M. Heller, DSTO)

For the work on optimal notches, shapes have been obtained that offer the lowest possible stress concentration as a function of the notch aspect ratio and plate depth, see Figure 36, [28]. Here the DSTO finite element gradientless shape optimisation procedure was used. The non-dimensional coordinates of precise optimal shapes are given in a tabular form, and typical shapes are shown in Figure 37. All cases have been solved with a minimum radius of curvature constraint, which results in a 'robust' optimal shape that has no sharp corners. The optimal shapes can provide up to a 26% reduction in local peak stress as compared to circular shapes, and minimise the required length L. Typical stress distributions for the optimal notches are given in Figure 38. Optimal geometries are also given for notches that give no increase in local stress with the lowest possible stress concentration (i.e. $Kt_n = 1$). The results obtained are applicable to both the initial design of notched components and the reworking of existing components to remove damage, such as for corrosion grindouts.

In engineering practice, cutouts and holes in plates have typically consisted of circular or elliptical profiles. However, these profiles are typically not optimal and can result in significant stress concentrations. Hence in this work useful design data have been determined, defining optimal free-form hole shapes for use under both uniaxial and biaxial loading conditions for a key range of hole geometries, notably the hole aspect ratio [29]. These results have been obtained to a high fidelity using an iterative 2D finite element gradientless shape optimisation procedure. The optimal shapes that are given offer the lowest possible stress concentration, subject to the bounding geometric constraints.

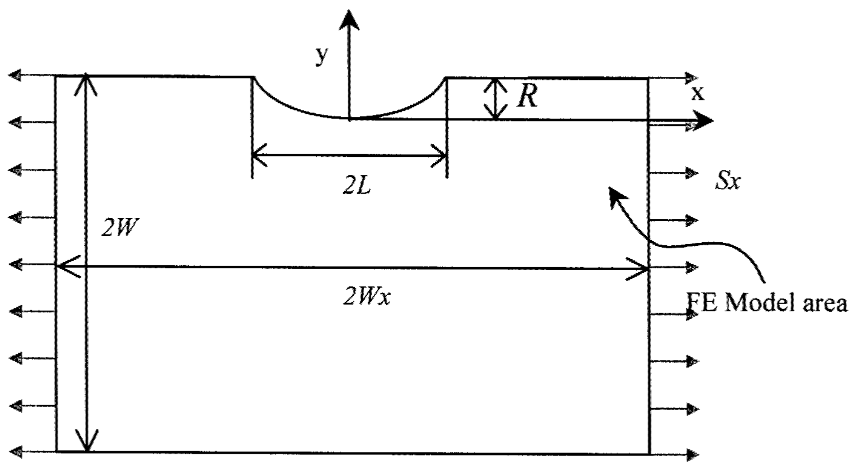


Figure 36: Shape optimisation of notched plate: plate loading and geometry,

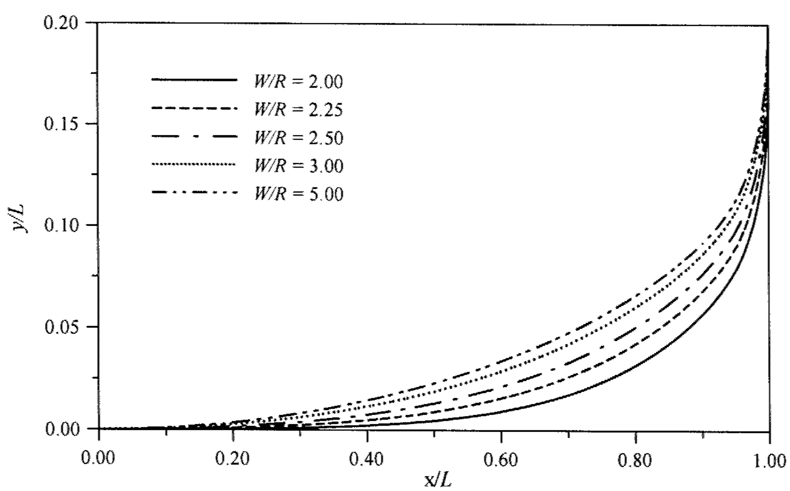


Figure 37: Shape optimisation of notched plate: typical optimal shapes for L/R = *5,* $\tilde{\rho}_{MIN}$ *= 0.10 for various ratios of* W/R *(x:y scale not equal),*

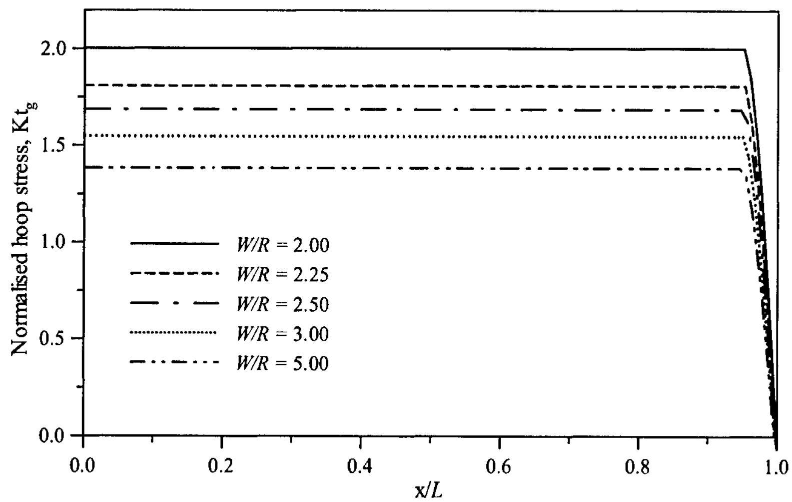


Figure 38: Shape optimisation of notched plate:) Typical Kt_g *values along optimal notches* L/R = *5,* $\tilde{\rho}_{MIN}$ *= 0.10 for various ratios of* W/R.

28. Burchill, M., and Heller, M., 'Optimal free-form shapes for notches in flat plates of varying thicknesses', (submitted to Journal of Fatigue, 2002).

29. Burchill, M., Heller, M., 'Optimal free-form shapes for holes in flat plates under uniaxial and biaxial loading', (Submitted to Journal of Fatigue, 2003).

8.4.11 Optimal Fillet Shapes Without Stress Concentration (S. Weller and M. Heller, DSTO)

This work provides useful design data defining optimal free-form fillet shapes for use under tension and bending loading conditions for a key range of fillet geometries, notably the relative fillet length, l/h, [30]. The key aspect of this work is that the limiting cases have been determined where the net section stress concentration is brought to unity, i.e. $K_{tnet} = 1$. The geometry and notation for a typical loaded fillet is shown below in Figure 39. The precise shape results have been obtained to a high fidelity using an iterative 2D finite element gradientless shape optimisation procedure. The precise optimal shapes that are determined are given as tabulated co-ordinates, which allow them to be easily used by designers. Typical optimal fillet shapes are given in Figure 40. In all cases here the hoop stresses have been rendered uniform all along the fillet (except for the region where $\tilde{\rho}_{min} = 0.15$), as shown in Figure 41.

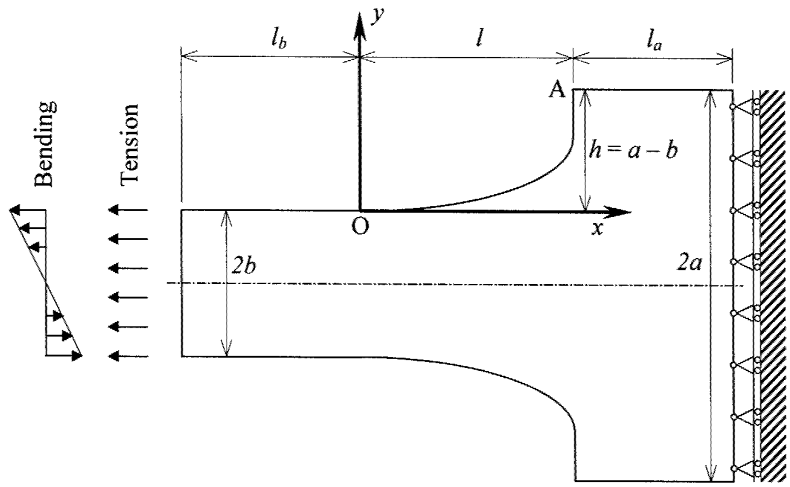


Figure 39: Fillet optimisation for $K_{tnet} = 1$: *geometry and notation for plate with fillet subjected to uniaxial tension or bending loading*

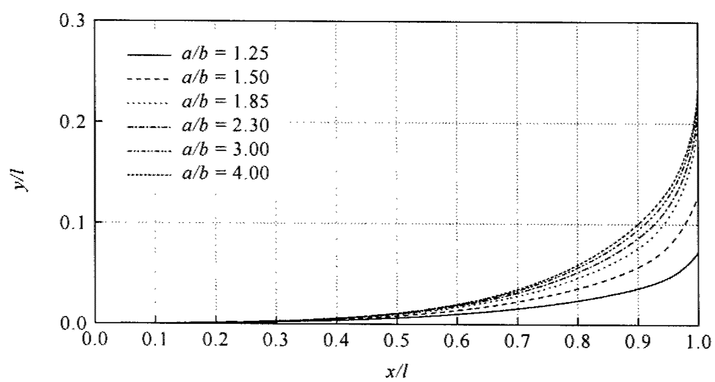


Figure 40: Fillet optimisation for $K_{tnet} = 1$: *a selection of typical non-dimensional optimal tension fillet shapes with* $K_t = 1$, *with minimum radius of curvature constraint,* $\tilde{\rho}_{min} = 0.15$.

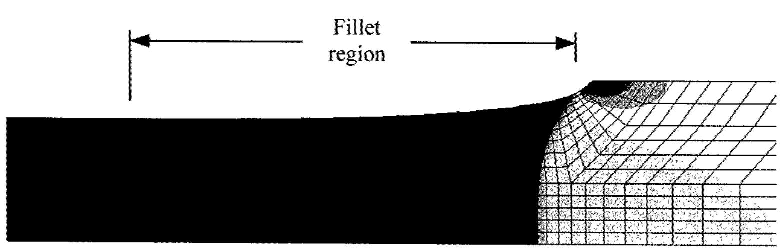


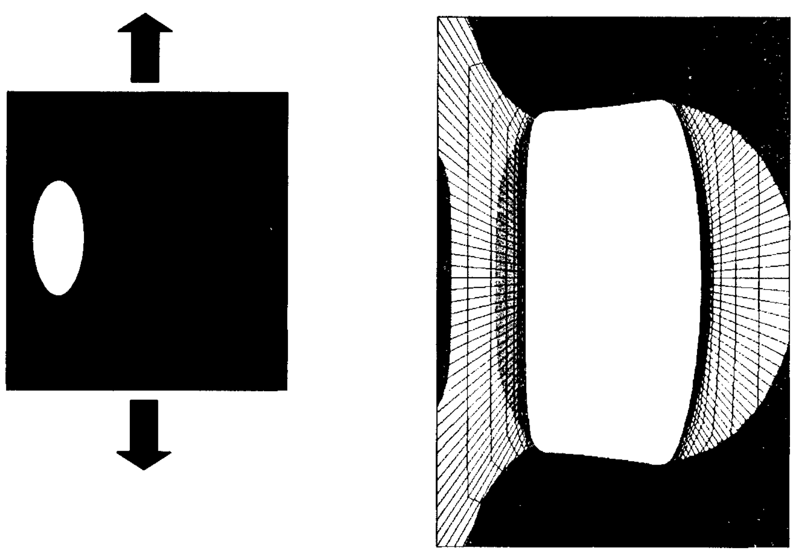
Figure 41: Fillet optimisation for $K_{tnet} = 1$: *contours of maximum principal stress for typical optimal tension fillet with* $K_t = 1$, *with* a/b = 1.333, l/h = 10.97, *and* $\tilde{\rho}_{min} = 0.15$.

30. Weller, S., and Heller, M., 'Optimal tension fillet shapes without stress concentration', (for publication in the 5th International Conference on Structural and Multidisciplinary Optimisation, Italy, June 2003).

8.4.12 Shape Optimisation of Holes for Multi Peak Stress Minimisation (W. Waldman and M. Heller, DSTO)

In DSTO work on shape optimisation to date, the fully automated approach has been typically implemented for a boundary segment consisting of a single stress state (either tensile or compressive). Good solutions have been obtained, where the stresses are rendered constant (and minimised) for the most critical segment of the hole boundary [31 and 32]. However, it is desirable to be able to minimise multiple segments around the hole boundary, which typically consist of tensile and compressive stress states. Here it can be noted that initially circular holes in loaded plates typically have two tensile and two compressive stress peaks, and for most practical loading and geometric constraint conditions, the stresses cannot be rendered uniform around the entire hole boundary. Hence in the present work, we generalise the method for the case of more complex constrained holes and simultaneously minimise hoop stresses on multiple segments around the hole boundary [33]. Here the multiple segments are defined by the individual stress states about the boundary. The aim is to simultaneously achieve a unique constant stress magnitude (and hence minimum peak stress) at each of the segments around the boundary.

The usefulness of the method is investigated by various numerical test cases. In all cases very good stress reductions were achieved. It is shown that the method yields optimal shapes matching those from known analytical solutions where they are available (for example an ellipse in a biaxially loaded plate, where the remote stresses are of the same sign, and a quasi-rectangular hole where the remote biaxial stresses are of opposite sign). Also this multi-stress-segment method typically gives solutions better than those obtained by the prior method based on a single-stress segment. A typical example is shown in Figure 42(a) that consists of a hole close to an edge in a uniaxially loaded plate. The optimal shape, where the hole aspect ratio is constrained to 2:1, is shown in Figure 42(b). Here there are four constant stress regions, with the peak stress significantly (27%) less than an equivalent ellipse.



(a) *(b)*

Figure 42: Multi peak shape optimisation of hole close to one edge in a loaded plate: (a) typical initial arrangement, and (b) detail of shape and stresses for optimal hole with constrained aspect ratio of 2:1, with four separate constant stress regions.

31. Kaye, R. and Heller, M., 'Structural shape optimisation by iterative finite element solution', Defence Science and Technology Organisation, DSTO-RR-0105, June 1997.

32. Waldman, W., Heller, M., McDonald, M. and Chen, G., 'Developments in rework shape optimisation for life extension of aging airframes', Third Australasian Congress on Applied Mechanics, ACAM 2002, University of Sydney, Sydney, Australia, pp. 695–702, 20–22 February, 2002.

33. Waldman, M., and Heller, M., ‘Shape optimisation of holes for multi peak stress minimisation’, (for publication in the 5th International Conference on Structural and Multidisciplinary Optimisation, Italy, June 2003.

8.4.13 Durability Based Evolutionary Structural Optimisation (ESO) Algorithm: (R. Das, M. Xie, and R. Jones, DSTO CoE-SM, CRC Rail)

It is known that stress optimised shapes are not necessarily optimal for fracture strength and fatigue life. To address this the ESO for shape optimisation has been extended for optimisation with fracture strength as design constraints. The cracks can be placed either in every element or certain specified number of elements on the design boundary.

The 2D crack configuration in each element is schematically shown in Figure 43. The crack runs from element boundary to its centre. Hence, the crack length is the normal distance from the element centre to the boundary. The present formulation uses stress field at the element centre for calculating stress intensity factor associated with each crack. The approximate expression proposed by Kujawski [34] was employed for this purpose. According to the Kujawski approximation the stress intensity factor at the deepest point of an edge crack for un-notched bodies can be estimated by:

$$K = Q\,\sigma\,\sqrt{(\pi\, l)}$$

Here K is the mode I stress intensity factor at the deepest point, Q is the shape factor, Q = 1.12 for a through the thickness crack, l is the length of the crack, and σ_t is the stress at the crack-tip normal to crack length. This stress can be approximated by the maximum principal stress at the element centre since the centres of the boundary elements lie very close to free edge with no normal stress and hence the stress parallel to the boundary (i.e. normal to crack orientation) will be the maximum principal stress.

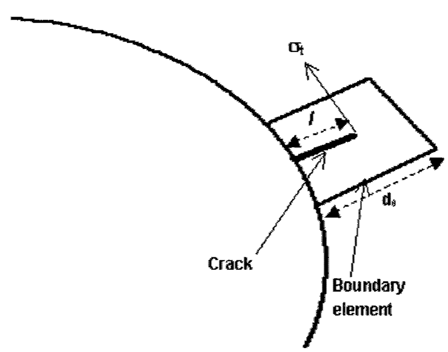


Figure 43: Crack configuration in each element

For implementation with the ESO algorithm the stress intensity factors for a crack in each element is considered as the ESO criteria and elements are removed if

$$\boldsymbol{K}_i \leq \boldsymbol{K}_{ref}$$

The objective functions are similar that presented in [36] with σ replaced by K and presented here for sake of completeness, viz:

(i) $$\boldsymbol{F} = \sqrt{\frac{\sum_{i=1}^{n}(K_i - K_{av})^2}{n-1}}$$

(ii) $$\boldsymbol{F} = \mathbf{Max}\,(K_1, K_2, \ldots\ldots, K_n)$$

Here $K_1, K_2, \ldots\ldots, K_n$ are the stress intensity factors associated with each crack. K_{av} and K_{ref} are the average and reference stress intensity factors respectively. K_{ref} is calculated as,

$$\mathbf{K_{ref} = K_{min} + F \times (K_{max} - K_{min})}$$

where F is a control factor.

The above equation determines material elimination level depending on current status of the structure. In the initial stages, usually there is a wide variation of stress level throughout the structure. Hence the stress range (difference between maximum and minimum stress) is high, which will result in higher volume removal per iteration. As the structure approaches close to optimum solution the ESO criteria, in present case K, tends to be uniform throughout the structure. In this situation, difference between the reference stress and elemental stresses will be less and number of elements removed per iteration will slowly decrease. This provides a better scope to record more number of closely spaced topologies near optimum region, thus furnishing the analyst with more options to choose from.

Example: To illustrate this technique let us consider the problem of a hole in a plate under biaxial loads where the initial shape is the stress optimised shape. For a biaxial stress of 50 MPa and 100 MPa the initial hole shape is a 2: 1 ellipse (20 mm x 40 mm) in a 600 mm square and 1 mm thick plate see Figure 44. The hole boundary was restricted within the design domain as shown in Figure 44. Two different initial crack sizes were considered, viz: 1 mm and 2.5 mm.

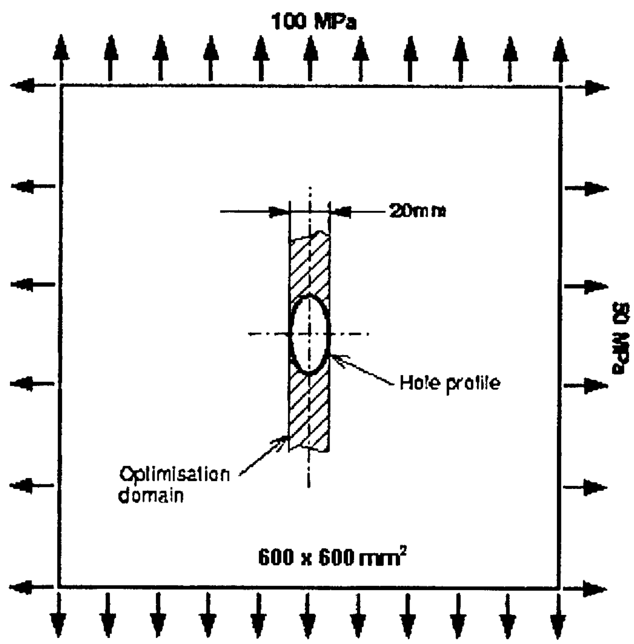


Figure 44: Schematic of elliptical hole in a square plate under biaxial stress

The first case considered of 1 mm initial flaws along the entire hole boundary. Due to symmetry, one quarter of the plate was meshed with 4-node quadrilateral elements. The mesh size in the optimisation domain was set approximately 2 mm in order to simulate the 1 mm initial cracks. The optimised hole shape was found to be an ellipse with an aspect ratio of 2.1: 1, i.e. 84 mm and 40 mm in present case, see Figure 45. This shape agrees reasonably well with that obtained using a 2D biological method [34]. In the initial elliptical profile, the stress intensity factors were 9.5 MPa $\sqrt{m}$ and 10.2 MPa $\sqrt{m}$ at the topmost and bottommost locations of the hole respectively. In the optimised shape, the corresponding values were 9 MPa $\sqrt{m}$ and 9.2 MPa $\sqrt{m}$ respectively. As usual, optimised boundary profile needs smoothing, as SIF is highly sensitive to sharp corners. The boundary was smoothened by fitting a spline through the boundary nodes.

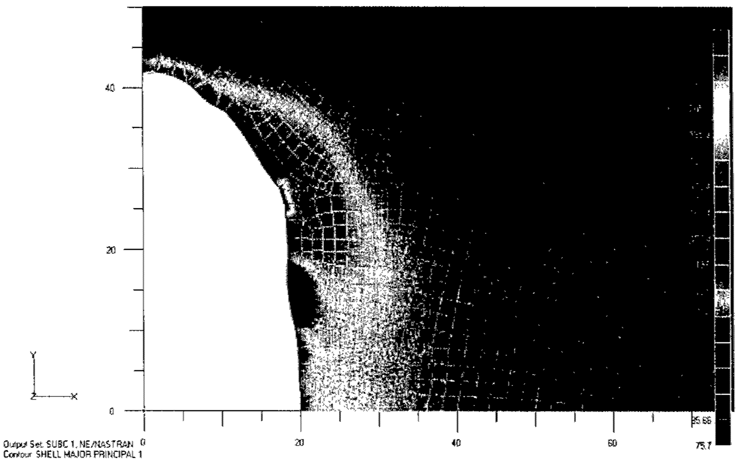


Figure 45: Optimised hole shape with smoothed boundary

The same problem was analysed for 2.5 mm initial flaws and the optimised shape was approximately elliptical with an aspect ratio of ~2.5:1 ellipse and agreed with results presented in [34]. Here also, the reduction of stress intensity factors were from 13.6 MPa √m to 12.9 MPa √m at the topmost point and from 15.9 MPa √m to 14.2 MPa √m at the bottommost point.

34. Kujawski, D. , "Estimation of stress intensity factors for small cracks at notches", Fatigue and Fracture Engineering Materials and Structures, 14(10), 953-965, 1991.

8.4.14 An Alternative Stop Drilling Method for Life Extension of Plates with Fatigue Cracks (G. X. Chen, M. Heller, and C. H. Wang, DSTO)

Conventional stop drilling techniques are useful for extending the fatigue lives of cracked components. However, due to the high level of stress concentration at the stop hole, only temporary delay in crack growth can achieved. The present work investigated an improved method involving drilling two circular holes above and below the conventional crack-tip stop hole (see Figure 46) with the view to reducing the level of stress concentration [35]. Analyses were carried out for a uniaxially loaded plate with a central crack using the finite element method. From the numerical results, parametric solutions were obtained for the associated stress concentration factors at the edge of each stop hole and the optimum hole spacing for various ratios of stop-hole radius. The optimum hole spacing is achieved when all the stop holes have the identical stress concentration factor. The results show that the stress concentration factor of the three-hole configuration at the optimum hole spacing is about 30% to 48% below that corresponding to a single stop hole. Results for the optimal spacing as a function of relative hole size are given in Figure 47(a), while the associated stress concentration factors at the optimum spacings are given in Figure 47(b). The significant reduction in the stress concentration factor implies that the alternative three-hole technique may prove a far more effective life extension method than the conventional single-hole stop-drilling technique.

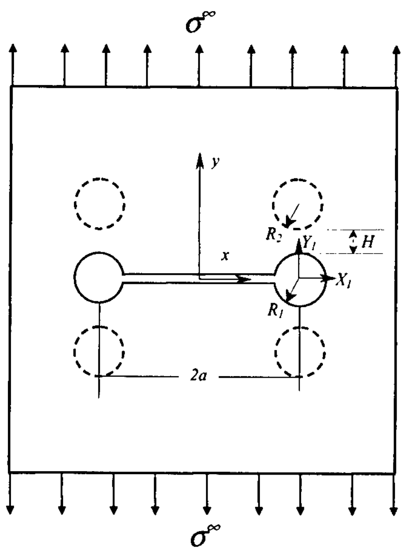


Figure 46: Geometry for a centre crack in an infinite plate subjected, with multiple holes subjected to a uniaxial stress field.

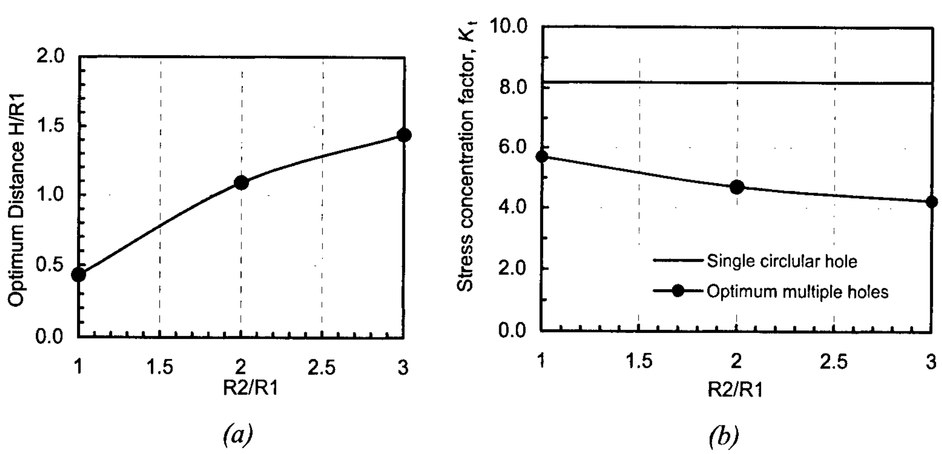


Figure 47: Results for study on optimum circular hole spacing: (a) optimum distance as a function of relative hole size, and (b) the associated stress concentration factor at optimum spacing.

35. Chen, G.X., Heller, M. and Wang, C.H., 'An alternative stop drilling method for life extension of fatigue cracks', *Australian Fracture Group Conference, 2002.*

8.4.15 Piezo-electric sensors: (W. K. Chiu, DSTO CoE-SM)

Several programs on piezo-electric sensors are under way at the DSTO CoE for Structural Mechanics at Monash University:

8.4.15.1 Detection of cracking:

The use of smart structures concept for structural health monitoring has attracted significant interest. In some applications, an actuator excites a point-location of the host structure and the response from another selected point location is received by a sensor. Any change in the condition of the host structure also results in a change in the response. However, successful detection relies on the method that is used to analyse the data received from the sensor. A reliable monitoring method will enable an engineer to make decisions to prevent total failure of the structure.

Recent work at the DSTO Monash Centre of Excellence in Structural Mechanics [36] has developed a transfer function (TF) method to detect and simultaneously locate the general area of damage i.e. a crack in a structure at an early stage.

This TF method works by acquiring the vibration signature in a certain range of broad band frequency from the piezoceramic (PZT) patches attached on the surface of the structure. In this method, PZT patches are distributed as arrays of actuators and sensors where they generate and measure the vibration signature. By taking the ratio of the vibration spectrum in a particular position of a sensor to an actuator one can produce the TF of this actuator-sensor pair. The arrays of the PZT patches can also be used to locate the general area of the damage in the structure and/or degradation of a particular sensor- actuator pair. If the TF of a particular sensor-actuator pair shows an anomaly, the region of the damage or degradation can be easily identified.

36. N. Wibowo and W. K. Chiu, "Crack Monitoring Of A Structural Component", Applied Mechanics: Progress and Applications, Proceedings ACAM2002, Sydney, 20 –22 February, 2002, Edited by L. Zhang, L. Tong and J. Gal, World Scientific Publishers, ISBN 981-02-4867-9, pp.439-444.

8.4.15.2 Fibre optic sensors: (S. Wade and R. Jones, DSTO CoE-SM)

An initial experimental program was conducted on cracked aluminium skins 3.14 mm thick, see [45 and 37] for more details. Each plate had an edge crack 10mm long, which was repaired with a semicircular unidirectional boron/epoxy patch. The aluminium skins were tested back to back separated by a honeycomb sandwich core, see Figure 48. The initial crack was then grown under constant amplitude fatigue loading. In this test program an array of surface mounted optical fibers, OF1 – OF4, was used, see Figure 48. Each fibre contained a number of Fibre Bragg Gratings (FBG's). To give a far field reading fibre OF1, containing a single sensor S1, was bonded to the aluminium skin using a cyano-acrylate adhesive. The remaining three optical fibers were bonded to the top surface of the boron repair also using a cyano-acrylate adhesive. These sensors were then completely buried beneath a thick layer of room temperature curing adhesive so as to protect the optical fibers and also provide a smooth surface for making the eddy current measurements of crack length. These gratings had high reflectivities (80-90%), FWHM of about 0.5nm, and a gauge length of around 5mm.

The crack was grown under a FALSTAFF fatigue load spectrum that attempts to simulate flight loads experienced by a military aircraft. As the crack progressed a series of static strain readings were periodically undertaken at load levels of 0 kN and 70 kN.

These results clearly illustrate the ability of an optical fiber sensor array to monitor crack growth under a bonded repair. This can be seen in the results from OF2 in Figure 49, located 25mm from the panel edge.

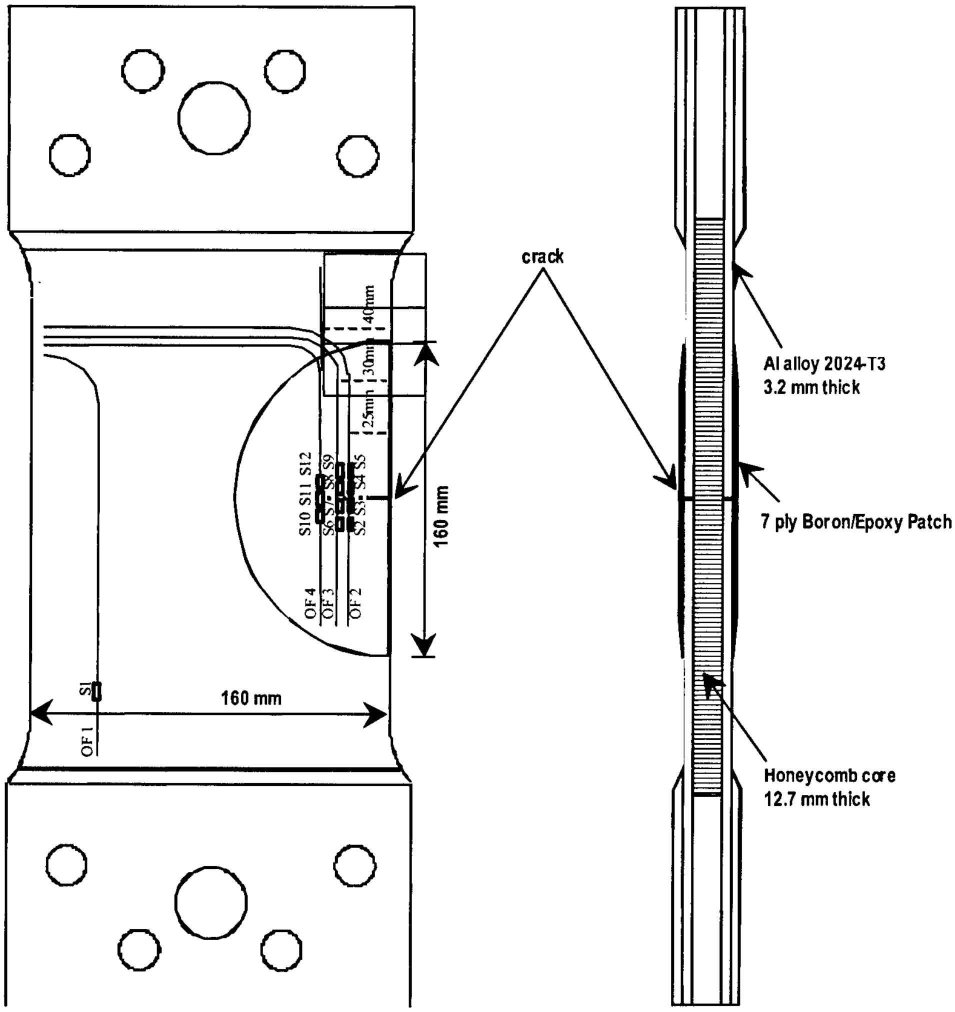


Figure 48: Schematic of the specimen used for this experimental program.

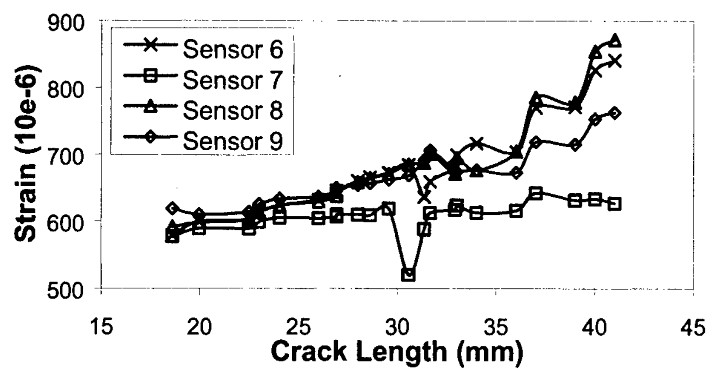


Figure 49: Change in Strain with Crack Length OF 2

Residual strains within the patch system can also be used as an in-situ structural health monitoring technique. This passive technique relies on the fact that when a composite repair is applied to a metallic substrate the difference in the coefficient of thermal expansion (CTE) between the composite and the metal results in significant residual thermal stresses in the patch system at room temperature. When disbonding or damage in the patching system occurs then redistribution in the residual strains occurs. Therefore monitoring the residual strains in the component gives an indication of the presence of damage. The magnitude of the change in residual strains depends on the type, severity and location of damage, the curing temperature, and the relative stiffness between the patch and the substrate.

In this study it was found that as the crack grew the sensors over the crack had a negative residual strains while those to the sides of the crack had a positive residual strain, see Figure 50 which shows the change in the residual strain as the crack grows under the sensors, see [45 and 37] for more details.

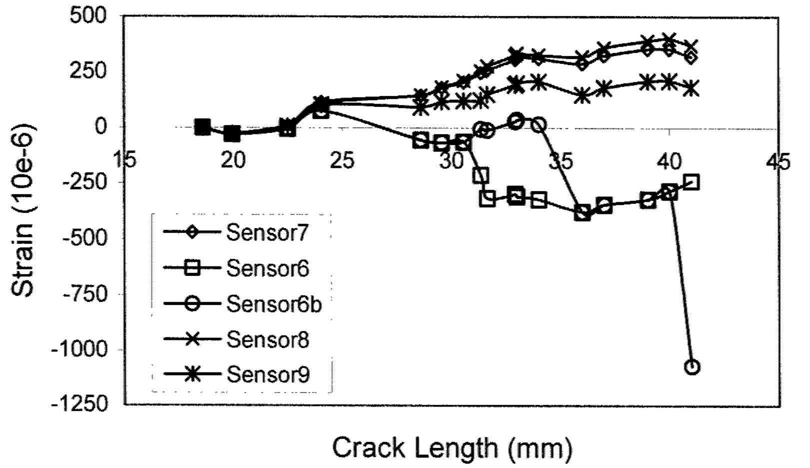


Figure 50: Residual strains for fibre OF3

37. R. Jones, S. Galea, and I. H. McKenzie, "Structural Health Monitoring Using Optical Fibres", Proceedings US Air Force Specialist International Workshop on Smart Composite Structures, Melbourne, 22nd November 2001, Edited by W. K. Chiu and F. G. Goretta, Journal of Composite Structures, Vol 58, 3, pp 397-404, 2002.

8.4.16 Statistical Analysis of Probability of Detection Hit/Miss Data for Small Data Sets (C. A. Harding and G. R. Hugo, DSTO)

Reliability of non-destructive inspections is generally characterised by probability of detection (POD) as a function of crack size. A recent paper [38] presented an improved analysis method for hit/miss POD data, which gives a lower 95% confidence limit POD curve which is valid for significantly smaller data sets than can be analysed using previously published methods.

The performance and robustness of the new analysis method was validated and compared to previously published methods using an extensive program of Monte Carlo simulations. Up to 2000 simulated POD trials were analysed using the different methods for each of a number of different hit/miss data set sizes. The new analysis algorithms give similar results to previously published algorithms for large hit/miss data sets. However, the simulations demonstrate that the new method gives robust lower 95% confidence limit POD curves for quite small data sets containing as few as 50 hit/miss observations, whereas previously published algorithms behave poorly for data sets containing fewer than 200 observations.

Collection of POD data through experimental trials is expensive and reducing the size of the data set required for analysis offers savings on the cost of trials and may make reliability studies viable for more applications.

38. Paper presented at the Review of Progress in Quantitative Nondestructive Evaluation, Bellingham, WA, July 14-19, 2002.

8.4.17 Health and Usage Monitoring Systems (HUMS) (G. F. Forsyth; DSTO)

The *DSTO International Conference on Health and Usage Monitoring (HUMS2003)* – the third in this series of conferences on HUMS was held in February 2003. Over 110 people, most from outside Australia, attended the two-day conference. The conference allowed researchers from all over the HUMS community to once more gather together to inform each another of the progress that had been made in their research. One interesting feature: the trend noticed in 2001 for HUMS technology to move from helicopters to fixed-wing aircraft has now extended to land and marine vehicles.

Considerable interest was expressed in having a fourth conference in this series in February 2005; a final decision will be made early in 2004.

SmartHUMS Initiative. - DSTO, in conjunction with Australian company GPS Online, has produced a design of a prototype miniature HUMS system. Testing of the prototypes is expected to commence in the second half of 2003.

8.4.18 Damage Tolerance Round Robin for a Helicopter Component (D. C. Lombardo; DSTO).

In 2002, DSTO staff participated in a Round Robin exercise to establish fatigue lives and inspection intervals for an aluminium helicopter airframe component. This Round Robin was initiated by Professor Phil Irving from Cranfield University, and a paper on the results will be presented at the next American Helicopter Society Forum [39]. In summary, various crack growth models were used to determine the crack growth of a defect. All participants were given geometry details, stress intensity solution, crack growth data and load spectrum data for the component. Twelve organisations participated and the predicted component lives ranged from 240 flight hours to over 12,500 flight hours. By way of comparison, two experimental validation tests were carried out under the specified same loading conditions as for the theoretical study and the resulting lives were 420 and 440 flight hours.

39. Damage Tolerance In Helicopters: Report On The Round Robin Challenge, P E Irving, J Lin, & J W Bristow, To be Presented at the American Helicopter Society 59th Annual Forum, Phoenix, Arizona, May 6 – 8, 2003.

8.4.19 Helicopter Structural Damage Detection (A. Wong, DSTO)

To support a proposed extension of service life of Black Hawk helicopters, the Australian Defence Force (ADF) and the United States Air Force (USAF) jointly conducted a comprehensive flight trial in 2000 to survey the structural distress areas. The test aircraft was heavily instrumented with 249 airframe strain gages, 18 accelerometers, 80 dynamic component strain gages and 28 flight state/control system parameters. The flight strain measurements at various flight conditions were conducted.

During the trial, an excessive level of stress was detected at one of the strain gauges. Follow-up inspection revealed a structural crack located just centimetres from the strain gauges. In this case, it was fortuitous that the propagation of a crack was detected because it was extremely close to a strain gauge. Questions arose as whether this crack could have been detected by gauges that are more remotely located, and whether it could be detected earlier or under less severe flight conditions if more sophisticated signal processing techniques were employed. As a result, a research program was initiated to find answers to these and related questions. So far, it has been shown that by applying similar techniques as those used in machine fault diagnostics (eg, synchronous re-sampling and time-frequency analysis), the presence of that structural fault could have been detected from remote gauges and under the most common and uniform manoeuvres (see Figure 51). More work has now been planned to develop and validate a general approach to this problem.

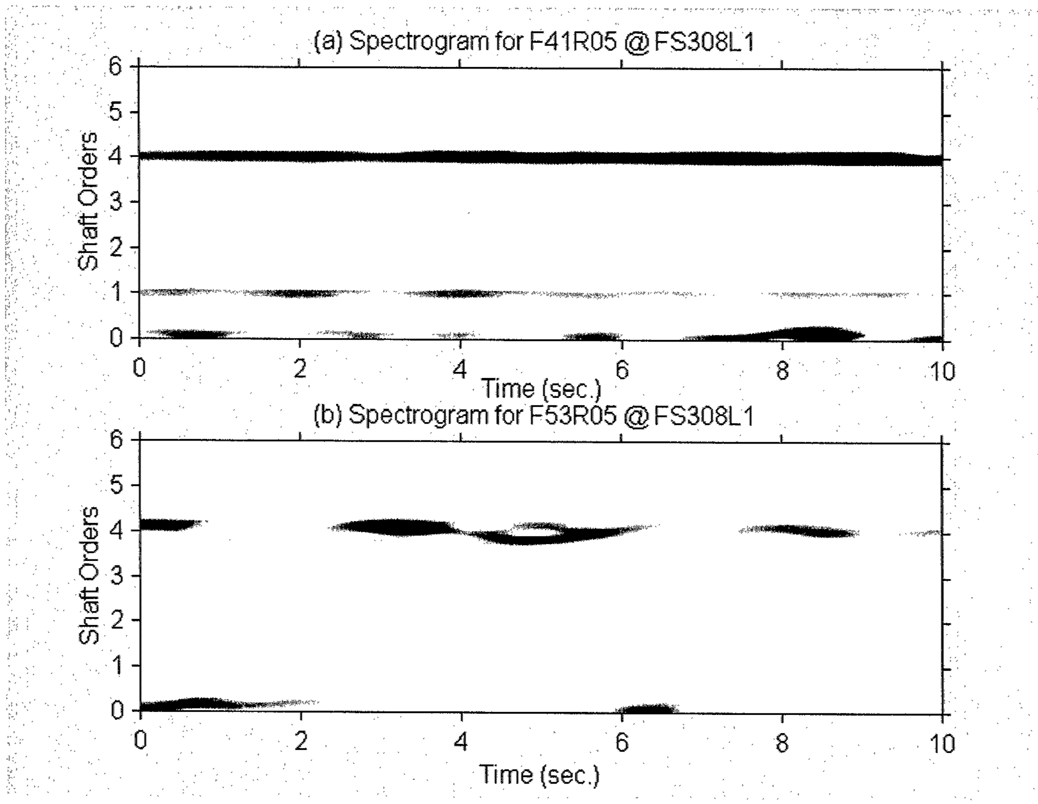
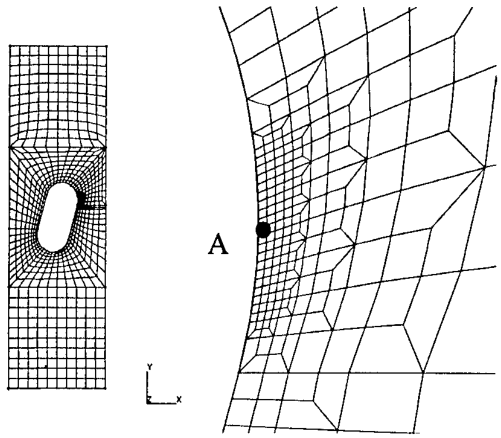


Figure 51: Spectrograms for synchronously re-sampled strain data of the furthest strain gauge from the crack location at **(a)** *the 41st (presumably healthy-state) and* **(b)** *the 53rd test flight hour (confirmed faulty-state). Both were taken during a common hovering flight condition. The frequency splitting/shifting as well the time-dependent features of the later spectrogram suggest the presence of the fault.*

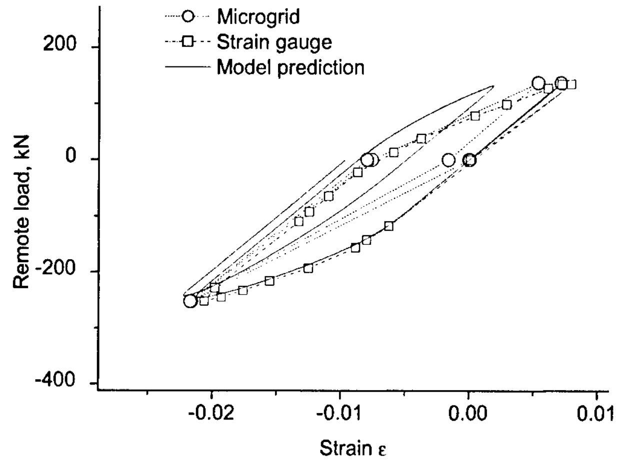
8.4.20 Improved Constitutive Model for Cyclic Plastic Deformation: (W. Hu and C. H. Wang, DSTO) (Update)

The nonlinear kinematic hardening constitutive model developed previously [40] has been successfully applied to analyse the stress-strain distribution near the root of a notch in a wing pivot fitting of a military aircraft [41]. The constitutive model uses multiple back stresses to accurately simulate the material behaviour over a fairly large strain range, which is characteristic of the loading condition during cold proof load tests periodically conducted on such structures. The numerical results for stress history have been compared to previously published experimental data at selected location, and good correlation has been observed, as demonstrated in Figure 52 below. The distribution and the evolution of the notch stress and strain are important input parameters for durability and damage tolerance analyses of the F-111 wing pivot fitting structure.

Work is in progress to implement the improved constitutive model in life assessment software such as FASTRAN.



Finite element model for a coupon representing a wing pivot fitting on a military aircraft, with an oval hole. (a) the whole model, (b) enlarged view of the critical location.



Comparison of experimental and numerical results for load-strain response for the coupon, under cold proof load test.

Figure 52: Finite Element Model, and its performance.

40. Wang, C. H., Hu, W. and Sawyer, J. (2000) Explicit numerical integration algorithm for a class of non-linear kinematic hardening model, *Computational Mechanics*, Vol.**26**, No.2, 140-147.
41. W. Hu and C. H. Wang, Notch Stress Analysis in a Wing Pivot Fitting using a Nonlinear Kinematic Hardening Model, Proceedings *SIF2002*, Perth, Australia.

8.4.21 The use of Comparative Vacuum Monitoring (CVM™) in Structural Health Monitoring (D P Barton, Structural Monitoring Systems Ltd, Australia)

Comparative Vacuum Monitoring or CVM™ is a relatively new technology for the detection of surface flaws in metals and composites. The technology has gained an acceptance within the testing laboratories of the major civilian OEM testing laboratories and the testing facilities of several military organisations.

The most significant development since 2001 has been the development of thin load bearing (TLB) sensors to detect cracks within structure such as lap and butt joints. These sensors are installed within the structure and are able to detect cracks of 1.5 mm in length; specifically aimed at detecting multi-sight damage within airframe structure. The development of the TLB sensors has been driven by a major civilian OEM, and is also being used by the OEM to qualify new materials. Standard NDI techniques are not suitable for the certification programs of several new materials as they are not able to detect cracks sufficiently close to initiation in fastened joints.

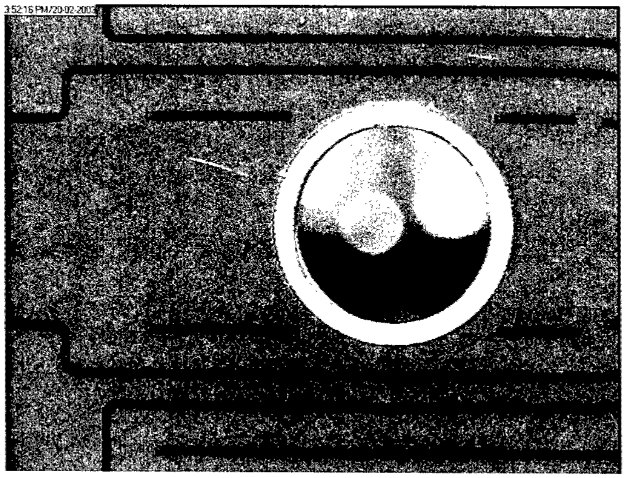


Figure 53: A close-up of a thin load bearing sensor installed over a rivet. This sensor is able to detect cracks 1.5 mm in length and is 125 μm thick.

CVM is also being investigated by the Australian Defence Science and Technology Organisation (DSTO) to enable them to detect the delamination of Boron epoxy repair patches. Initial tests have given positive results however a larger sample of measurements is required to more conclusively confirm the effectiveness of the technique. Installing sensors similar to the thin load bearing sensors below the epoxy bond is also being investigated to monitor the further growth of the crack over which the repair has been installed.

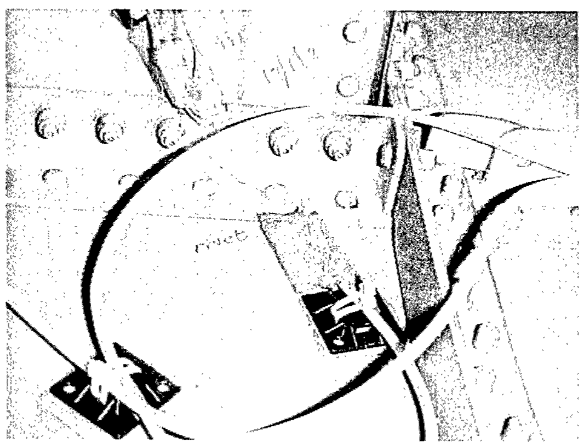


Figure 54: Sensors installed on US Navy CH-53 helicopter.

Sensors have been installed on a US Navy CH-53 helicopter based at Patuxent River Naval Air Base since January 2002. Two sensors were installed at each end of a 50 mm crack emanating from a fastener in the tail boom. The sensors have been monitored on a periodic basis equating to 25 flight hours (approximately every 6 weeks) using SMS' portable maintenance monitor (PM-3). The use of the SMS PM-3 has reduced the 4 hour strip down and visual inspection to a less than 5 minute test. The sensors and the hardware have worked efficiently, detecting the growth of the crack to a predetermined length in January 2003. The crack will continue to be monitored for growth to a second predetermined length until January 2004.

SMS currently has evaluation programs for in-service installations of CVM with the Royal Australian Air Force (in conjunction with DSTO), the Republic of Singapore Air Force, the US Navy, US Air Force, one civilian fleet operator and a manufacturer of armoured vehicles.

SMS is working closely with both major civilian OEMs to develop programs to fully certificate the use of CVM to monitor structure. CVM lends itself well to the developing field of Structural Health Monitoring or SHM. CVM sensors, either surface mounted or embedded within structure can be easily monitored on a periodic basis or in real time. The sensors and connecting tubing are lightweight and inert, with the same sensitivity achieved in the field as obtained in the laboratory. Since the method underlying the system relies on maintenance of a vacuum, the system is inherently fail-safe.

8.4.22 Laser Shock Peening (Q. Liu, DSTO)

Laser shock peening (LSP) technology is a new and promising surface treatment method. Substantial research work show that this technology can greatly improve the fatigue life of a material or a component. Laser shock peening or processing (LSP) has a number of advantages over conventional shot peening techniques. LSP can create a deeper residual compressive stress - up to 1.5 mm in the case of Al alloys whereas the conventional method only provides about 250 μm deep below the treated surface. Also, there is very little or no modification on the original surface profile. Even very thin sections can be laser processed without deformation of the part. In some component geometries which feature small holes and small slots, it is very difficult for conventional technology such as shot peening or cold expansion to be used due to limited accessibility, whereas LSP can easily reach these areas by using an optical fibre as a delivery system. In addition, the laser radiation is highly controllable and the process is readily amenable to automation. Therefore, LSP technology is potentially an important method for increasing fatigue life in developing modern aircraft, and repairing and maintaining "aging" aircraft. LSP has been successfully applied to increase performance of materials and structures by increasing fatigue life, reducing fretting fatigue, enhancing resistance to corrosion and increasing resistance to foreign object damage.

DSTO experiments designed to identify any technological risks associated with use of LSP in fatigue-critical aluminium aircraft structure revealed that there are some circumstances where problems might arise, preventing achievement of full life extension.

DSTO had conducted investigations on 7050-T7451 aluminium alloy, which was treated by laser shock peening (LSP). In this initial investigation, the usual large life extension was not observed; the averaged LSP fatigue life was lower than those from traditional glass bead peening at the same applied peak stress. An example is shown in Figure 55, where the applied peak stress was 390 MPa.

Metallurgical examination of fracture surfaces using optical microscopy revealed that significant cracking existed inside the specimen (Figure 56), leading to a reduction in fatigue life.

Based on the experimental results and FE analysis, the high laser power density and poor material were found to be the major factors causing the so-called internal cracking, which lead to low fatigue life of laser shock peened 7050 aluminium alloy. The sample geometry may also play a role, but this has not been fully explored.

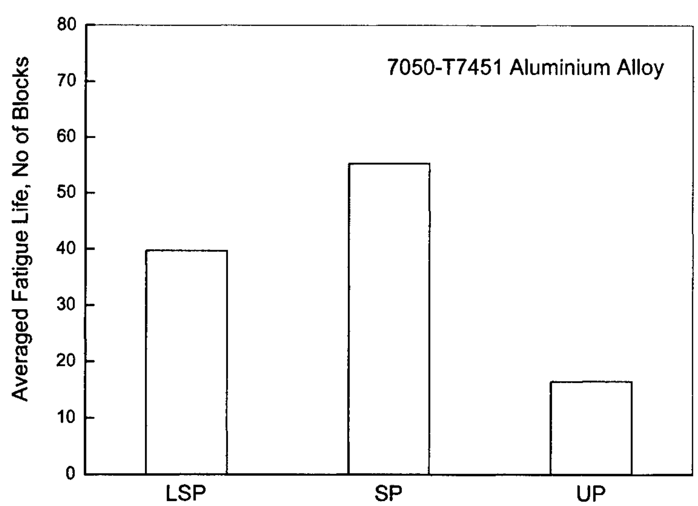


Figure 55: Comparison of fatigue life of LSP (laser shock peened), SP (glass bead peened) and UP (unpeened) 7050 specimens under spectrum loading at the applied stress of 390MPa

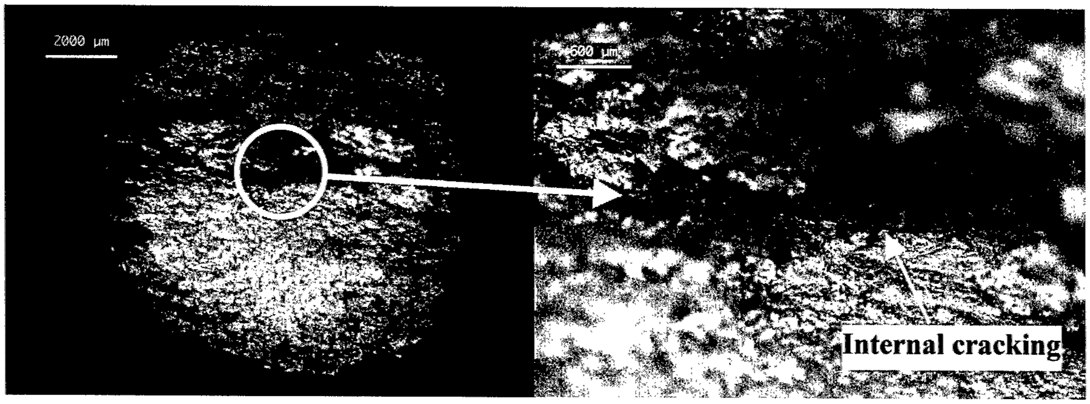


Figure 56: Macroscopic appearance due to LSP. The applied peak stress is 360MPa, where macroscopic cracks are clearly shown (7050-T74511 aluminium alloy).

It is important to emphasise that although the internal cracking has been found in the 7050-T7451 LSP specimens under the special conditions aimed at revealing any technological risk factors, LSP still has great potential for fatigue life enhancement of components or structures if the process can be well controlled.

Further DSTO research conducted an investigation on 7075-T7451 aluminium alloy, which was treated by laser shock peening (LSP). In this case the experimental results had demonstrated the effectiveness of LSP, as shown in Figure 57. Both SP and LSP produce a significant improvement in fatigue life for the studied material over the etched specimens (the reference surface condition). With decreasing applied stress the improvement of fatigue life is very obvious. Compared to glass bead peening, laser shock peening (LSP) further improves the fatigue life of a material although the experimental data for LSP specimens has more scatter at some applied stresses. The beneficial effect is believed to be associated with the deeper residual compressive stresses and better surface quality, both of which are expected to hinder crack development.

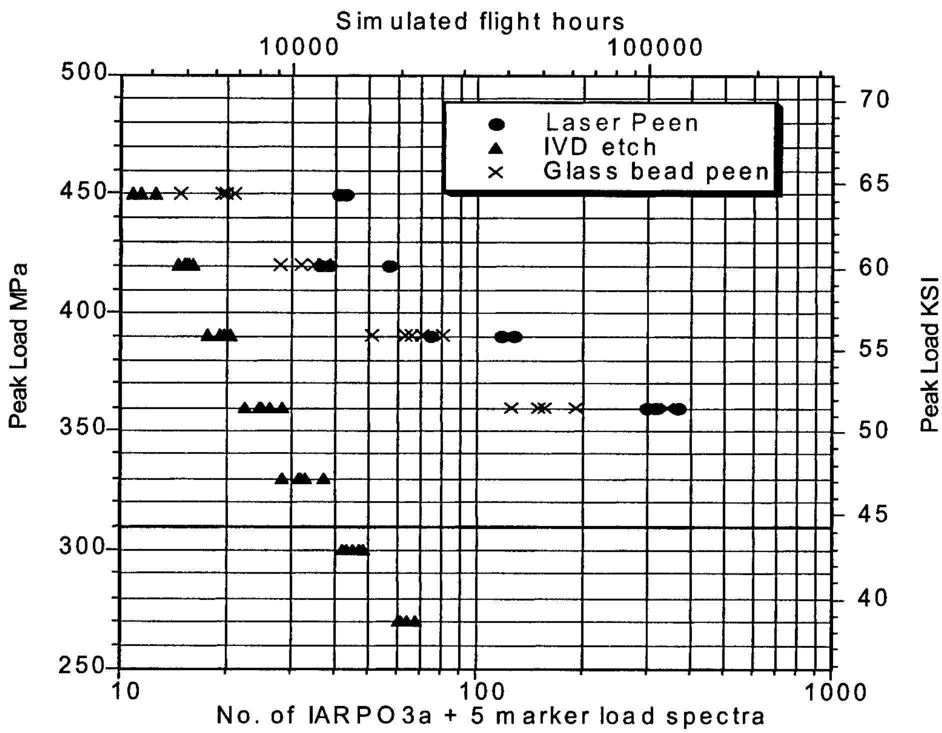


Figure 57: Fatigue tests on 7xxx series aluminium alloys for etched, glass bead peened and laser shock peened specimens.

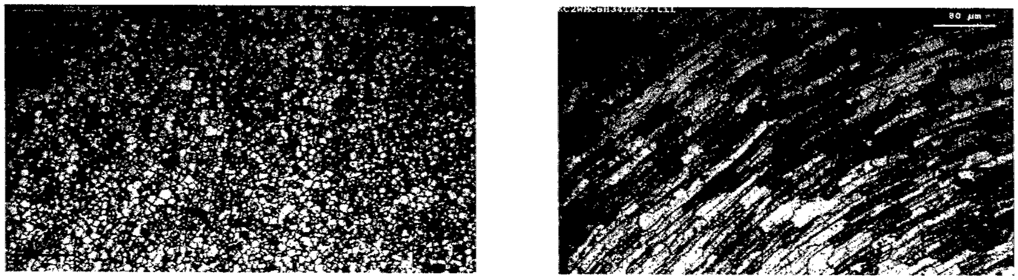
The research has shown that as is the case with conventional bead peening, peening conditions must be controlled. DSTO has since developed a model to predict the optimum LSP conditions for different materials. This model is being verified through a series of specimen fatigue tests.

8.4.23 Friction Stir Processing of Aluminium Alloys (P Baburamani, DSTO)

High strength aluminium alloys, eg 7000 series alloys, are used in the construction of civil, transport and military airframe components (wing, fuselage, etc.) in current generation aircraft, generally in wrought product forms. These alloys are considered to be un-weldable using conventional fusion welding processes. Over the last ten years, non-fusion, solid-state joining techniques such as friction stir welding (FSW), have emerged as a new joining process, especially for high strength aluminium alloys, paving the way for producing integral unitised airframe structure at reduced cost. The friction stir joining technology was developed and patented by The Welding Institute (TWI), UK, in 1991. The feasibility of using friction stir joining process for the repair of cracked or corrosion damaged aging aircraft components provides the scope for reduced through-life support costs for aircraft fleet operation. The purpose of the DSTO work was to evaluate the potential of the friction stir technology for repair of high strength aluminium alloy aircraft components as well as to document any technology risks in the application of friction stir process for airframe components.

Preliminary friction stir joining trials have been carried out at the Adelaide University CRC for Welded Structures' friction stir welding facility. The purpose of the trials was to: (a) develop the best combination of friction stir processing parameters for 7000 series aluminium alloys, (b) characterise the metallurgical, environmental and mechanical behaviours of the friction stir joints, and (c) to evaluate repair and rebuild capability of the technology, when applied to cracked and corroded surfaces, including technology demonstrators featuring RAAF aging aircraft components. The materials selected were high strength aluminium alloys, 7010 – 10 mm thick (LIF Hawk material) and 7050 – 6.35 mm thick (F/A-18 material). The alloys were heat treated by the supplier and were in T73651 (7050) and in T651 (7010) conditions. Bead-on-plate joints were produced in the preliminary trials. Some of the results obtained from these friction stir trials are presented below.

The microstructures developed during friction stir processing consist of several zones within the weld nugget. The joint microstructures were generally free from defects (cracks, porosity, etc.), normally found in fusion weldable alloys. The friction stirred (equi-axed grains) and thermo-mechanically affected (deformed grains) zones in the 7050 alloy are shown in Figure 58.



(a) *(b)*

Figure 58: Microstructures of (a) friction stirred and (b) thermo-mechanically affected zones in 7050 aluminium alloy

Fatigue test specimens were fabricated from as-friction stir processed plates and tested using F/A-18 spectrum (FT55+5M) at a maximum stress of 330 MPa and the results compared with base-line materials in Table 1.

Table 1. Fatigue test results (FT55+5M), σ_{max} = 330 MPa

Material	Condition	Average fatigue life (Number of blocks)
7010-T7651	FS welded	11.6
7010-T7651	Base	71.2
7050-T7451	FS welded	16.9
7050-T745	Base	75.8

The results clearly showed that the joint efficiencies in aircraft fatigue loading were 16% for 7010 and 22% for the 7050 alloys and that these would have to be improved significantly to be comparable with base-line materials or machined or forged components and to achieve the full benefit of the solid state joining process. However, the results should also be compared with typical riveted airframe structures available in the literature. Several options in the form of post-friction stir thermal and mechanical treatments to improve the fatigue lives of friction stirred joints are currently being explored by DSTO.

The fatigue fractures were examined in a scanning electron microscope (SEM) and a typical SEM images for the 7050 alloy are shown in Figure 59. Arrows in the figure mark the crack initiation site, where a crack appeared to initiate from laps produced on the surface of the weld nugget by the friction stir tool. Furthermore, cracks appeared to initiate from several such sources in the specimen, i.e. there was multiple crack initiation.

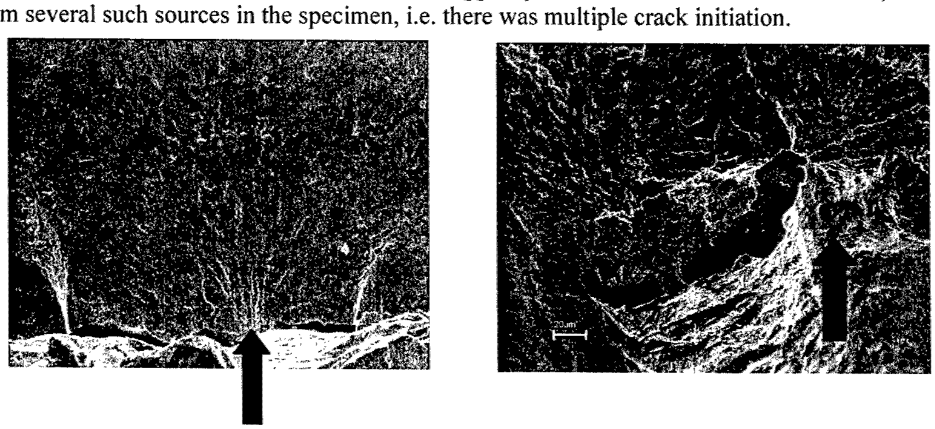


Figure 59: Scanning electron microscopic images of 7050 aluminium alloy fracture surface tested in fatigue (FT55+5M) loading, fracture origin (lap) marked by an arrow

8.4.24 Principal Component Thermography for Flaw Contrast Enhancement and Flaw Depth Characterisation (N. Rajic, DSTO)

Active thermal inspection technology, recently introduced into the RAAF, promises to provide a lower cost alternative to current nondestructive inspection methodologies for a wide range of structural integrity problems in F/A-18 and F-111 aircraft. The capacity of pulse thermography to furnish highly-informative broad-field inspection data for large structural components in a remarkably short time holds great practical appeal, and has served to accelerate its uptake amongst military and civilian fleet owners around the world. The technology is however deficient in (i) having a limited penetration depth compared to alternative active NDT (nondestructive testing) methods, and (ii) the considerable difficulty involved in obtaining robust and precise measures for various defect characteristics. These issues continue to stimulate vigorous research activity. Although a number of useful advances have been made on these matters, much scope for improvement and further development remains.

DSTO have developed a new methodology [42] that provides an integrated framework for both contrast enhancement and flaw depth estimation based on the singular value decomposition of an appropriately constructed matrix of observations. The methodology is shown to produce a highly compact and useful representation of the spatial and temporal variations relating to contrast information associated with underlying structural flaws. The method does not rely on the explicit formation of a contrast response, required by traditional approaches, and accordingly avoids the consequent risk of introducing bias in the flaw depth prediction. Analyses on synthetic and experimental data sets have underscored the practical efficacy of the methodology, its robustness to noise and bias, and a superior level of performance compared to pulse phase thermography. Figure 60 and Figure 61 illustrate the comparative performance of the technique on experimental data pertaining to a 3mm thick Al2024 plate with slots of varying depth and width milled into the back face.

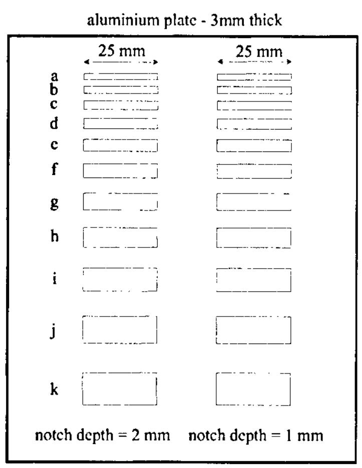


Figure 60: Schematic of spatial resolution test specimen

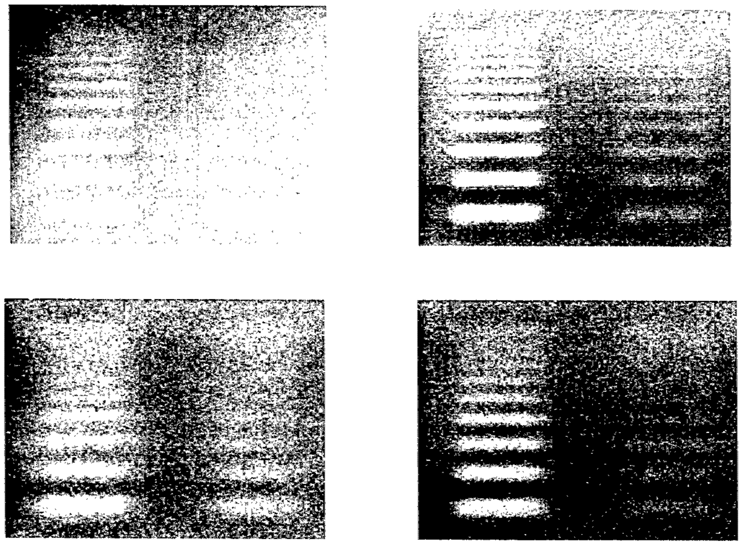


Figure 61: Clockwise from top left (i) Peak Contrast thermogram, (ii) Principal Component thermogram, (iii) Pulse Phase thermogram at twice frequency resolution and (iv) Pulse Phase thermogram at frequency resolution.

42. Principal Component Thermography for Flaw Contrast Enhancement and Flaw Depth Characterisation in Composite Structures (N Rajic, DSTO), Composite Structures, 58, pp 521-528

8.4.25 Inspections for Entrapped Water in F/A-18 Composite Skinned Honeycomb Components Using Active Infrared Thermography, (N. Rajic, S. Lamb and D. Rowlands, DSTO)

A thermographic survey of F/A-18 honeycomb components was undertaken at RAAF Base Williamtown to assess the efficacy of an active infrared thermographic system developed by DSTO, for the detection of entrapped water in rudders. In situ inspections of four rudders yielded no evidence of entrapped water, however internal structural features were easily resolved, which supports the expectation that entrapped water should be detectable. Supplementary laboratory work carried out at DSTO confirmed this, suggesting that the Advanced Thermal System developed by DSTO can reliably detect water entrapped in rudder structure at levels as low as 5% of total honeycomb cell volume. High sensitivity to moisture, along with the practical advantage of short inspection times, make active infrared thermography an attractive technique for the routine inspection of composite aircraft components for entrapped water.

Additional laboratory tests were conducted on representative rudder specimens using a low-cost portable system called RAAFTIS (RAAF Thermal Inspection System) (Figure 62) then under development at DSTO. This novel system is based on robust microbolometer imaging technology which provides a versatile platform for both active and passive thermographic inspection of aircraft components and systems. Despite the lower intrinsic sensitivity of microbolometer imaging technology compared to its photonic counterpart (as found in the Advanced Thermal System), tests revealed that relatively low levels of moisture could be detected in rudder structure using this portable system. This system has subsequently been delivered to the NDTSL (Nondestructive Testing Standards Laboratory) at RAAF base Amberley.

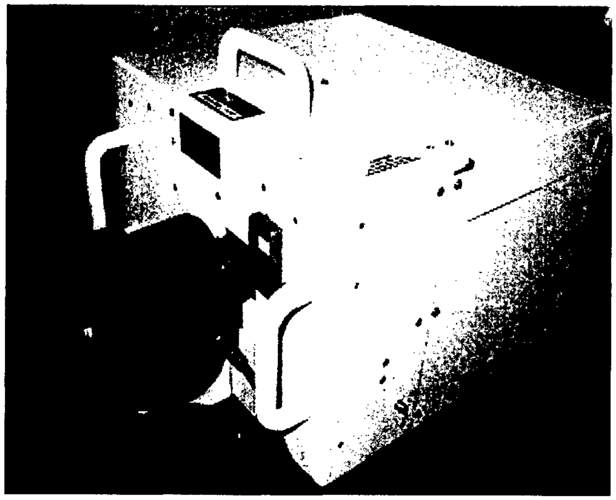


Figure 62: RAAF Thermal Inspection System, showing the SC2000 imager connected to a lightweight collapsible inspection head.

43. Inspections for Entrapped Water in F/A-18 Composite Skinned Honeycomb Components Using Active Infrared Thermography, DSTO-TN-0421, March 2002 (N. Rajic. S. Lamb and D. Rowlands, DSTO)

8.4.26 In-situ Health Monitoring of Composite Bonded Repairs (S. Galea, N. Rajic, DSTO, and W K Chiu, CoE-SM)

An experimental program of work aimed at developing and evaluating various, smart materials-based, in-situ health monitoring systems for composite bonded repairs have been conducted. These studies form part of an ongoing collaborative program, between the Australian Defence and Science Technology Organisation (DSTO) and Monash University (as part of the DSTO sponsored Centre of Expertise in Structural Mechanics), with the eventual goal of developing robust and reliable in-situ patch health monitoring systems. The techniques investigated include strain-based load transfer, residual strain, electro-mechanical impedance, transfer function and stress wave techniques utilizing embedded and/or surface mounted optical fibre sensors and/or piezotransducers. The experimental studies were undertaken on laboratory skin-doubler specimens subjected to cyclic uni-axial loading, and showed that disbond growth in the outer edges of the composite bonded repair could be successfully detected using any of these techniques.

However two novel techniques, residual strain monitoring and stress wave technique, showed significant potential for in-situ damage monitoring in composite bonded repairs. Figure 63 illustrates that the strains, at zero applied load, at the edge of the patch show a significant increase as damage progress under the patch with increasing number of cyclic loads. In the stress wave technique elastic stress waves were generated in the host by applying a short voltage pulse to a piezoelement, SP1, surface mounted to the metal substrate some distance from the patch edge as shown in Figure 64.

Part of the elastic energy is transmitted through the bond-line into the patch and received at the two piezosensor locations (SP2, located on step 3 of the patch, and SP3, located on the far-field region of the patch). The expectation was that continuous monitoring of the elastic energy at this sensor location would provide a robust basis for the assessment of disbond growth from the patch edge. Figure 64 shows the received signal power, measured by sensors SP2 and SP3, against the number of loading cycles. The results show that the measurements from sensor SP2, located directly above the region where disbond growth occurred, showed a significant level of sensitivity to disbond growth. These results show good correlation with the strain results measurements. One potentially concerning aspect of these measurements is the apparent complexity of the relationship between disbond growth and the evolution of signal power. In the case of SP3 for example this is demonstrated by the 20% increase in power immediately preceding the more substantial long-term decline in signal. Such complexities are likely to arise from wave diffraction behaviour. More work needs to be undertaken in order to understand such behaviour, with a view to exploiting it for the quantitative assessment of disbond growth.

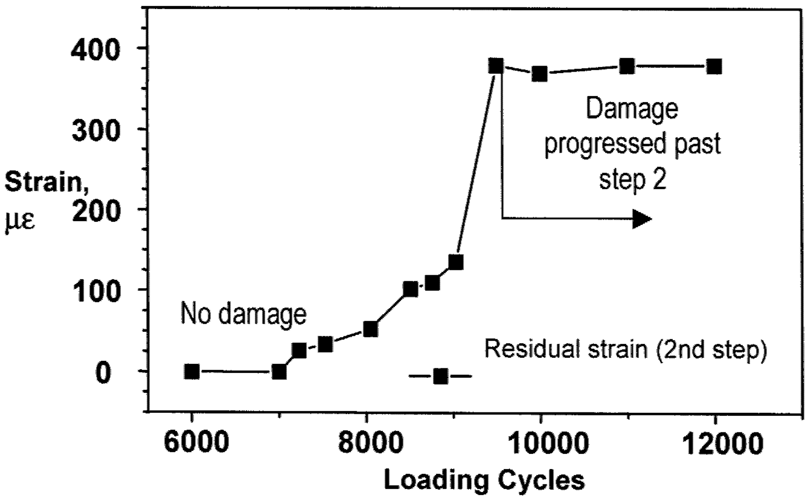


Figure 63: Variation of residual strains in tapered region of patch with increasing number of loading cycles for sensors on the 2nd step of skin-doubler specimen.

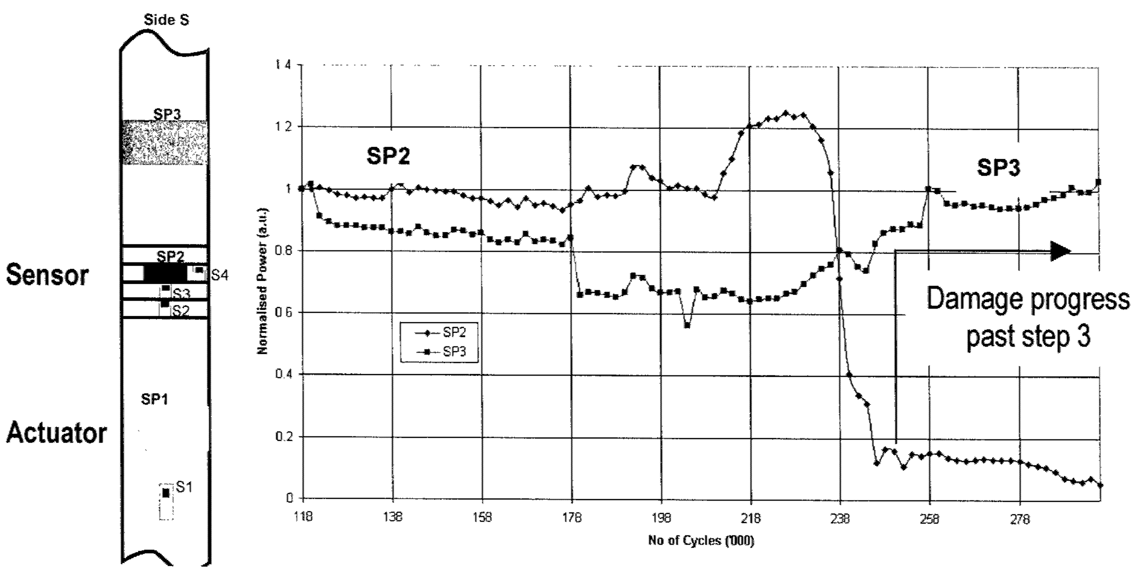


Figure 64: Signal power versus number of cycles for piezosensor SP2 and SP3

44. Y.L. Koh, W. K. Chiu, N. Rajic, and S.C. Galea, "The application of piezoceramic elements to the Detection of disbond growth in a Bonded Composite Repair patch", Applied Mechanics: Progress and Applications, Proceedings ACAM2002, Sydney, 20 –22 February, 2002, Edited by L. Zhang, L. Tong and J. Gal, World Scientific Publishers, ISBN 981-02-4867-9, pp.433-438.
45. Y.L. Koh, N. Rajic, W. K. Chiu, and S. C. Galea, "Smart Structure for Composite Repair", Composite Structures, 47(1), pp.745–752, 1999.
46. Galea, S., Rajic, N., Moss, S.D., McKenzie, I., Koh, Y.L. and Chiu, W.K. *Proceedings of the 3rd International Workshop on Structural Health Monitoring, Stanford University, 12-14 September 2001.* In-situ Health Monitoring of Composite Bonded Repairs.

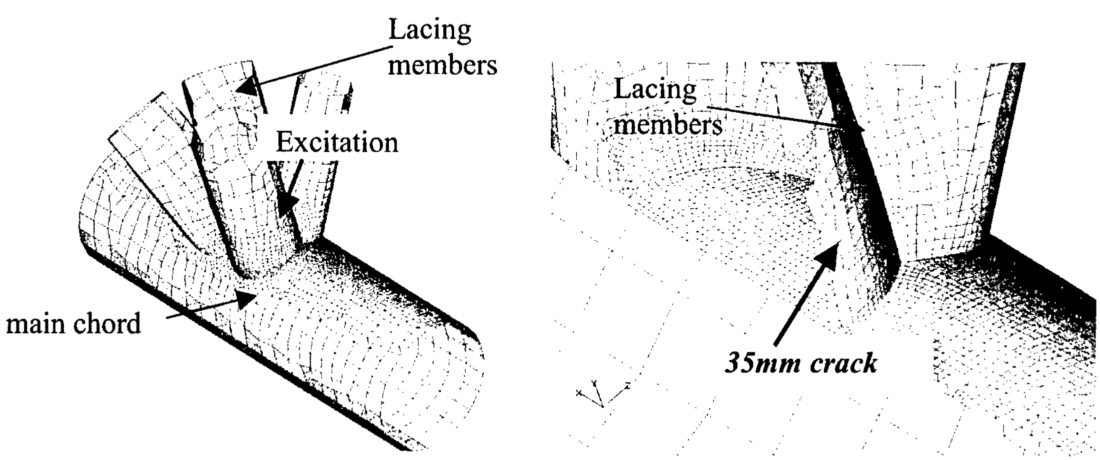
8.4.27 In Situ Health Monitoring of Bonded Composite Repairs using a Novel Fibre Bragg Grating Sensing Arrangement.(S. Galea, C. Davis, DSTO)

DSTO have been investigating the use of optical fibre Bragg gratings to measure the changes in thermal residual strain that occur when a composite patch starts to disbond from the parent structure [47]. Conventionally, the Bragg sensing mechanism relies on a shift in reflected wavelength, which requires the use of costly optical measurement tools. A modified sensing arrangement is proposed, which incorporates two Bragg gratings, and a fibre optic coupler. The reflection from the first Bragg grating acts as a reference source for an active Bragg grating on the patch. This modified arrangement allows a relative wavelength shift to be translated into a change in the optical power, which can be measured easily using a low cost interrogation system. The modified sensing arrangement also allows easier miniaturisation of the opto-electrical interrogation system, which would allow these systems to be more easily implemented on operational aircraft.

47. C. Davis, W. Baker, S.D. Moss, R. Jones and S.C. Galea, "In situ health monitoring of bonded composite repairs using a novel fibre Bragg grating sensing arrangement.", Proceedings of SPIE's International Symposium on Smart Materials, Nano-, and Micro-Smart Systems, Smart Materials II Conference, Vol 4934, Melbourne, Australia 16-18 December 2002. pp 140 – 149.

8.4.28 Interaction of Acoustic Waves and Cracks (R. Jones, DSTO CoE- SM)

To date most analytical and computational studies of ultrasonics have been conducted on simple structures rather than complex structures representative of large engineering components, which can have multiple load paths and have the potential for (complex) reflections of the acoustic wave. Recent work in the DSTO-CoE has studied the behaviour of ultrasonic waves in a complex structure, with multiple connecting members, containing a crack.



(a) (b)

Figure 65: (a) The finite element model. (b) Close up sectioned view of the model.

In this work a three dimensional model was created to approximate the geometry of a complex weld cluster, consisting of a main chord with several connecting members which are commonly referred to as lacings, in a large engineering structure, see Figure 65. For added complexity the cracks were assumed to lie in the footprint where the lacings intersect the main chord. This was done so as to create a problem in which the damage was hidden and could not be seen without dismantling the structure and where the structure would have multiple load paths and exhibit complex reflections of the wave form. Such problems are common in aerospace and ship structures.

The finite element mesh used in this analysis is shown in Figure 65. The material was assumed to be a steel with a modulus of elasticity **E** = 205GPa, a density ρ = 7850kg/m^3, and a Poisson's ratio υ = 0.32. The main chord was assumed to have a diameter of 406 mm and a thickness of 20 mm. Each of the lacing members was assumed to have a diameter of ~320 mm and a thickness of 20 mm.

The analysis was performed using the NE/NASTRAN finite element program and the element type used was the nine noded CQUADR shell element, which has an internal node at the centroid of the element and uses drilling degrees of freedom. It is important to refine the mesh in the area where the acoustic wave will travel. To overcome problems with reflections from adjacent elements a consistent uniform mesh was required. This problem will be discussed further later.

A triangular input wave with a duration of 0.1 μs was used to excite the input wave. The wave was then evolved over a period of 40 μs. A circular (input) wave was used in an attempt to produce the unique diffraction pattern associated with a circular wave front hitting a slit (crack), of length a. Three different crack lengths were considered, viz: 35 mm, 60 mm, and 100 mm. In each case we saw that even after several crack lengths the diffraction pattern was still significant. The evolution of the wave in the main chord under a lacing member, the reflection from the crack, and the resultant diffraction pattern is shown in Figure 66 for a 35 mm crack. This diffraction pattern closely resembles that associated with a circular wave front hitting a slit (crack), of length a. These views have been sectioned to view the propagation of the wave form. Similar, but more pronounced, images were seen for the 60 mm and the 100 mm flaws.

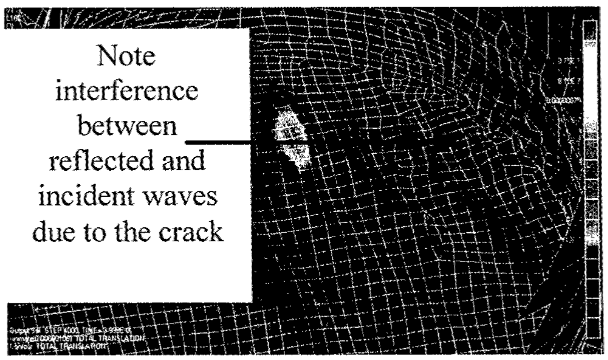


Figure 66: In plane wave propagation under footprint at 40μs

This study found that cracking and corrosion damage produce a diffraction pattern that resembles that associated with the traditional physics of wave motion. The extension of this hypothesis implies that it may be possible to use a simple ripple tanks to investigate how to best detect/sense and size a given damage state, e.g. cracking or corrosion. In this case we could use PMMA sheeting to alter the (local) depths of the water and thus simulate the corrosion damage. One advantage of this approach is the ability to rapidly change configurations and to obtain a full field view of the diffraction pattern. Furthermore, for the case of diffraction due to a slit (crack) it is well known that the diffraction is maximum when the wavelength λ of the incident wave is approximately equal to the length of the slit (crack). This implies that we may be able to optimize this approach by choosing the most appropriate wave form.

The present study also reveals that damage – a simulation of cracking or corrosion damage – can have a significant effect on both the period of the waveform and also on the local apparent refractive index of the material and that these effects have the potential to be used as damage indicators. At the same time recent advances in modern infra-red cameras mean that it may also be possible to use thermal emission to visualise this diffraction pattern for full scale structures.

8.4.29 Fatigue Life Extension of Aging Aluminium Alloy Structures Using Bonded Composite Reinforcements (A. A. Baker and S. A. Barter, DSTO)

The fatigue life of metallic aircraft components can be increased by the application of bonded reinforcements. This technology can be particularly beneficial in extending or recovering the life of ageing components that have for example, suffered damage by corrosion, or reached a significant proportion of their fatigue life. The fatigue life is extended in two ways by reinforcement: (a) by strain reduction and (b) by local reinforcement of the damaged region. Composites because of their formability, high strength and stiffness, low SG and outstanding fatigue and corrosion resistance are almost ideal reinforcements or doublers. However, suitable composites such as carbon/epoxy and boron/epoxy have a much lower thermal expansion coefficient than the metallic structure (even allowing for thermal expansion constraint), which leads to undesirable residual tensile stresses in the metal when reinforcements are bonded, particularly when bonded at elevated temperature.

Experiments were undertaken to evaluate the influence of these stresses in which the fatigue lives of 7075-T7451 dog-bone specimen were compared a) unpatched, b) reinforced with aluminium doublers and c) reinforced with boron/epoxy doublers of similar stiffness to the aluminium. Comparison between the boron and aluminium doubler results provided a direct measure of the influence of the residual stress.

Tests were performed under spectrum loading typical of that experienced by Australian F/A-18 aircraft. In order to isolate the influence of the doublers on the fatigue life all tests were performed at the same peak strain level in the aluminium coupon. Also a range of variables were studied, including heating simulating patch application and etching damage to represent the conditions that may be experienced in the F/A-18 components due to OEM surface treatments. Several important conclusions were reached concerning the use of reinforcing doublers in this program. The first being that; a very significant increase in life was obtained over the unreinforced specimen, when tested at the same nominal strain level. The other important conclusion was that, allowing for experimental scatter the composite doublers performed as well as the aluminium doublers – indicating, surprisingly that the residual stress had little effect on fatigue life. Possible reasons for these observations are discussed in the paper. Of course the life extension is very much greater when testing is performed at constant load.

However, the practical conclusion is that composite doublers are very efficient in extending the fatigue life of aging aircraft components and in view of the properties referred to previously should be the preferred material for this application. Indeed, composite reinforcement has been successfully applied in Australia to extend the life of F/A-18 test bulkheads. The OEM recommends the use of bonded metallic doublers.

8.4.30 Integrated Thermoplastic Surfaces for Composite Panels in Aerospace Applications (M. Scott, CRCAS)

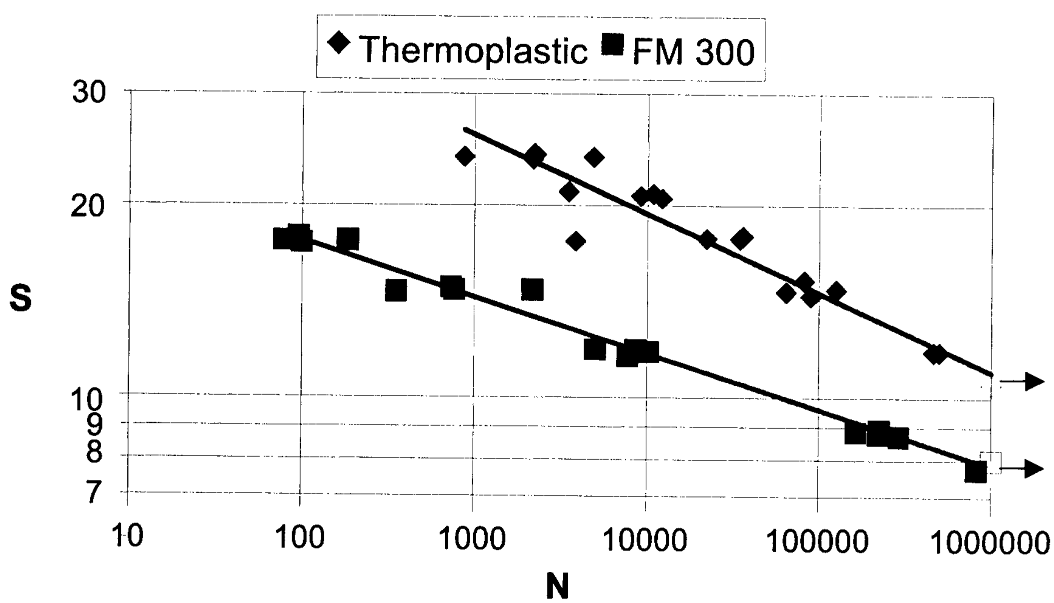


Figure 67: Fatigue of thermoplastic and adhesive surfaces for composite panels

The CRC-ACS has conducted development work on an integrated thermoplastic surface for epoxy-based composite panels. When welded they offer a cost-effective alternative to traditional adhesive systems for aerospace, such as Cytec FM-300M. A comparison of fatigue performance was made between the integrated thermoplastic and FM-300M for a Hexcel CF-F593-18 composite substrate. The integrated thermoplastic was found to substantially outperform the FM-300M in tension-tension fatigue testing of a single lap-shear joint. Failure of the FM-300M was generally through the adhesive. Failure of the integrated thermoplastic was largely through the composite substrate, indicating that the fatigue performance of the thermoplastic layer itself was higher than indicated in the test results. This work forms a part of ongoing testing and process development conducted by the CRC-ACS for the implementation of welded thermoset composites.

8.4.31 Fatigue research at the Centre for Advanced Materials Technology (L. Ye, CAMT, University of Sydney)

Several studies are addressing fatigue behaviour of advanced materials and the integrity/durability of advanced materials structures of various configurations. A significant part of this research aims to establish models to define the fatigue life and residual strength of advanced composites subject to cyclic loading and hostile environments. The outcomes add to understanding of durability of advanced composite structures for aerospace and maritime applications. The main research directions can be summarised briefly as follows.

8.4.31.1 Fatigue of piezoelectric materials
The study is focused on characterisation of fatigue crack initiation and propagation in poly-crystal and single crystal piezoelectric ceramics under combined cyclic mechanical and electrical loading for understanding fatigue behaviour and integrity of these materials for design and optimisation of smart materials systems.

8.4.31.2 Fatigue of shape memory alloys
This study is focused on characterisation of crack initiation and propagation in shape memory alloys under cyclic loading for understanding the crack initiation behaviour and phase transfer in shape memory alloys for design and optimisation of smart materials systems.

8.4.31.3 Fatigue of adhesive joints
The study addresses the critical issues of adhesively bonded joints and patched repairs of dissimilar materials in aircraft structures, aiming to understand crack initiation and propagation behaviour in different modes in relation to residual stresses and structural configurations. The outcomes will add to the critical knowledge for repair and maintenance of aging aircraft.

8.4.31.4 Fatigue of fibre-matrix interface
This study is focused on fibre reinforced composites with optimised fibre-matrix interfaces (with improved adhesion and micro/meso-structures) for producing toughened fibre matrix composites with improved damage resistance.

8.4.31.5 Fatigue of electronic and photonic joints
This is a new area of research, addressing the issues in electronic and photonic packages where cyclic electrical, thermal and mechanical loads exists, which may have significant impact on the durability of adhesively bonded or soldered joints in electronic and photonic assemblies.

8.5 FATIGUE INVESTIGATIONS IN NEW ZEALAND

8.5.1 C130 Hercules Cockpit Windshield Strain Survey (A. D. James, S.. K. Campbell, DTA, New Zealand)

During depot level maintenance at Safe Air's Blenheim facility, significant corrosion was discovered in the fuselage skin of an RNZAF C130 Hercules aircraft. This corrosion was in the skin (alloy 2024) attached to the lower window sill (Figure 68), immediately below where the cockpit window post joins the lower sill. Aircraft in the RNZAF C130 fleet have previously been fitted with a large doubler that covers the entire cockpit window structure (Figure 69). The corrosion was thus obscured by both the windshield sill and the doubler.

The OEM stress report for the location appeared to be conservative, and does not consider the windshield doubler. Given the complex load paths, presence of the doubler and conservative nature of the OEM stress report, it was not possible to determine the fatigue life impact of performing a "grind out" repair to this location.

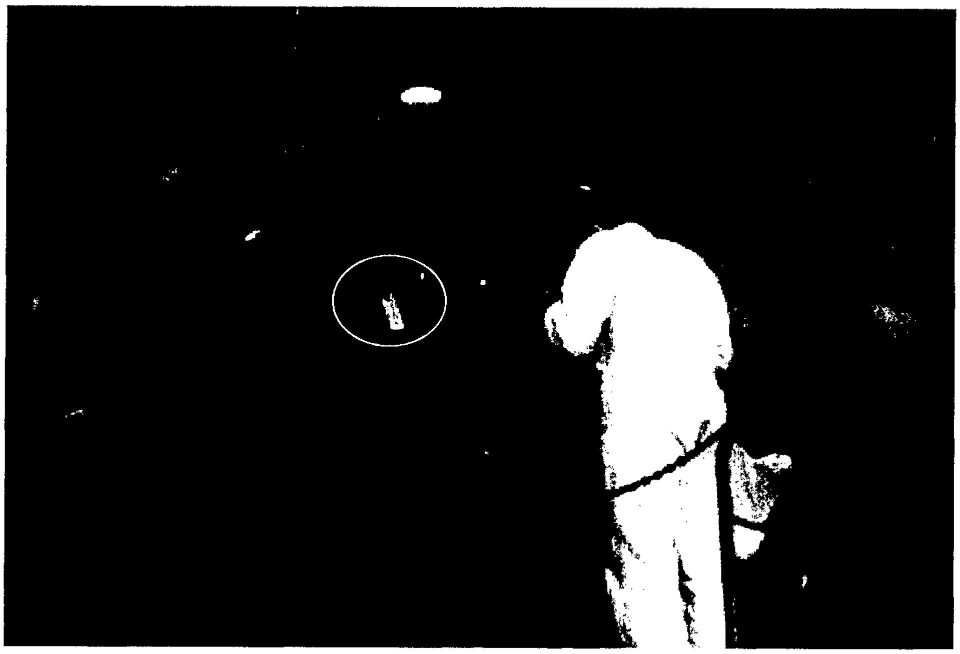


Figure 68: Corroded Skin Area

DTA and Safe Air undertook to instrument and measure the strains in this location under normal pressurisation loads. It was found that the strains in the lower sill were moderately high (approximately 1200 microstrain), and there were indications of secondary out of plane bending of the skin at the location. Current RNZAF inspection frequencies were deemed sufficient for the time being. The structure in this area will be re-evaluated during a planned structural life extension program for the RNZAF fleet.

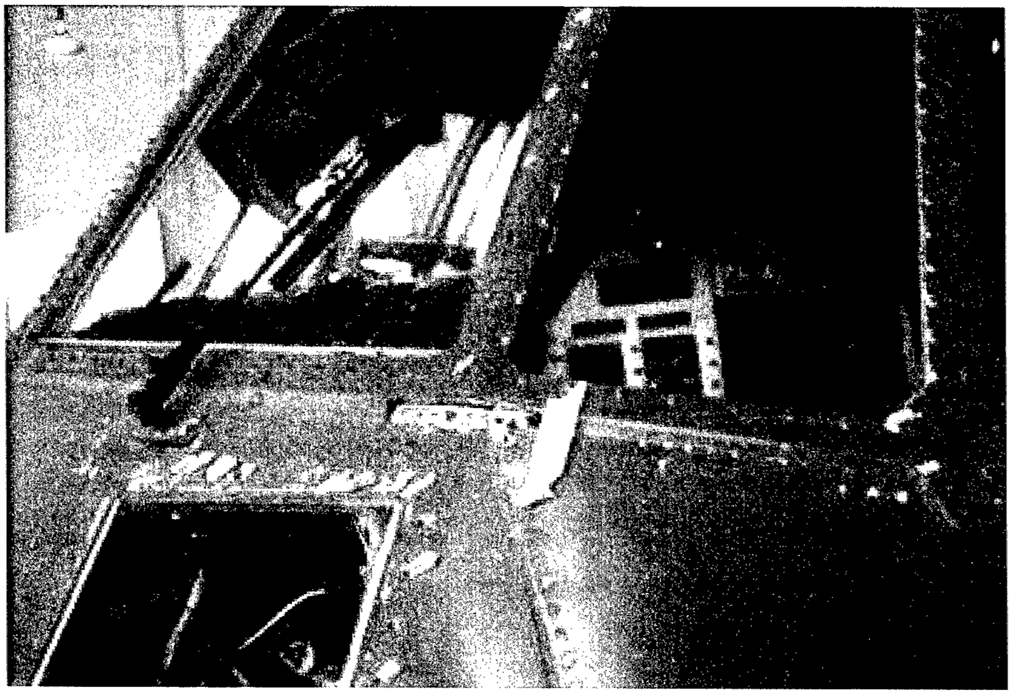


Figure 69: Close up of corroded area, showing the large doubler.

8.5.2 UH-1H (Iroquois) Main Rotor Blade – In-Service Fatigue Failure (A. D. James, P. C. Conor, M. J. Hollis, S. K. Campbell, I. P. Gatehouse, DTA, New Zealand)

In July 2001, while operating on UN Peacekeeping activities in East Timor, the RNZAF discovered a crack in a main rotor blade (MRB) during a pre-flight inspection of one of their UH-1H Iroquois helicopters. The crack was discovered behind and underneath the leading edge of the MRB, in line with the MRB / Rotor hub attachment bolthole. At this location, the MRB is of laminated aluminium construction. On the lower surface, a large steel grip pad is adhesively bonded to the aluminium structure (Figure 70). The steel pad surrounds the bolthole and is designed to transfer load out of the blade into the bolt and so reduce the fatigue and bearing stresses in the aluminium lower spar cap adjacent to the bolthole.

Subsequent failure investigations revealed a fatigue crack over a large portion of the cross section of the blade spar (Figure 71). The lower spar cap and spar webs had cracked through. Blade loads were evidently being supported by the upper spar cap and the blade nose, which is adhesively bonded to the front spar web. The crack is shown schematically in Figure 70. It was further found that the crack was initiated after an adhesive failure of the lower steel grip pad. The loss of adhesion resulted in a significant increase in the loads carried by the aluminium, accompanied by a corresponding reduction in time required to fatigue crack development in the spar.

Adhesive failure of the lower grip pad has been found to be widespread in blades in the RNZAF UH-1H fleet.

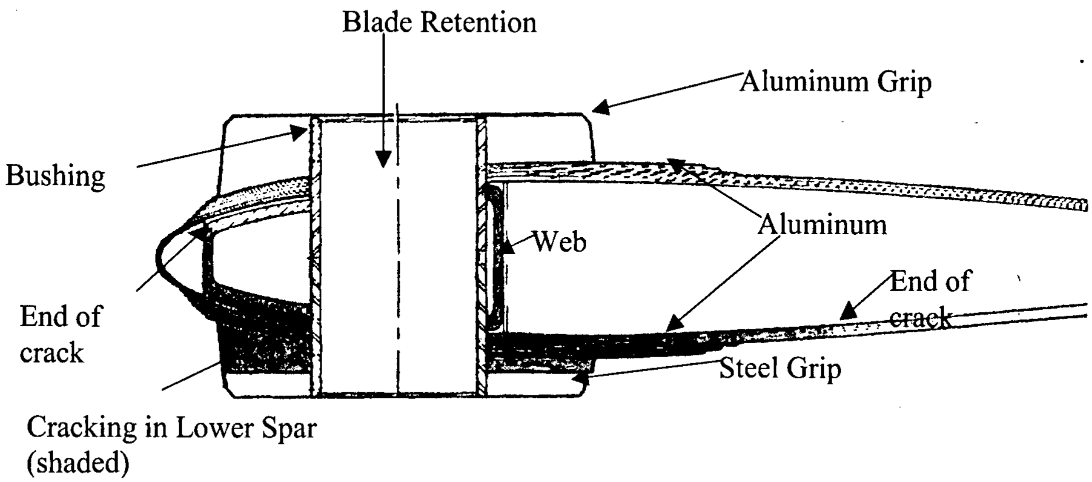


Figure 70: Blade Schematic. The cracked region is shaded

Figure 71: The Fatigue Crack

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19. ABSTRACT

This document has been prepared for presentation to the 28th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Lucerne Switzerland, 5th and 6th May 2003. Brief summaries and references are provided on the aircraft fatigue research and associated activities of research laboratories, universities, and aerospace companies in Australia and New Zealand during the period April 2001 to March 2003. The review covers fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.

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