DEPARTMENT OF DEFENCE
DEFENCE SCIENCE & TECHNOLOGY ORGANISATION
DSTO

A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1999 to March 2001

Colin Martin

DSTO-TN-0361

A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1999 to March 2001

Colin Martin

Airframes and Engines Division
Aeronautical and Maritime Research Laboratory

DSTO-TN-0361

ABSTRACT

This document has been prepared for presentation to the 27th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Toulouse France, 25th and 26th June 2001. 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 1999 to March 2001. The review covers fatigue–related research programs as well as fatigue investigations on specific military and civil aircraft.

RELEASE LIMITATION

Approved for public release

DEPARTMENT OF DEFENCE
DEFENCE SCIENCE & TECHNOLOGY ORGANISATION
DSTO

Published by

DSTO Aeronautical and Maritime Research Laboratory
506 Lorimer St
Fishermans Bend, Victoria 3207 Australia

Telephone: (03) 9626 7000
Fax: (03) 9626 7999

AR-011-879
May 2001

APPROVED FOR PUBLIC RELEASE

A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1999 to March 2001

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 Aeronautical and Maritime Research Laboratory (AMRL) 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. In contrasts to previous years, there are no items in this review under section 8.3 Fatigue programs on Civil Aircraft. However, work is under way in this area and reports can be expected in later reviews.

Contents

Table of Figures

8.1 INTRODUCTION

This review of Australian and New Zealand work in fields relating to aeronautical fatigue in the period 1999 to 2001 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.

AMRL	Aeronautical and Maritime Research Laboratory, GPO Box 4331 Melbourne, Victoria 3001, 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
Cooperative Research Centre for Advances Composite Structures	Aeronautical and Maritime Research Laboratory, GPO Box 4331 Melbourne, Victoria 3001, Australia
Aerostructures Australia	Level 14, 222 Kingsway, South Melbourne Victoria 3205, Australia.
Structural Monitoring Systems Ltd.	2 Dyer Road, Bassendean, Western Australia, 6054

8.2 FATIGUE PROGRAMS ON MILITARY AIRCRAFT

8.2.1 F/A-18 Structural Fatigue Refurbishment Program Assessments: M. Stimson and L. Molent; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

An extensive fatigue review of the RAAF F/A-18 has identified the need for significant structural upgrades if the aircraft is to achieve its planned withdrawal date. The fatigue refurbishments to the F/A-18 structure will fall within two programs aligned with phases of the Hornet Upgrade Program (HUG). These programs have been named Structural Refurbishment Program (SRP) 1 slated for 2002/04 and SRP 2 in 2006/08. To assess which structural locations require attention in each program, all available fatigue defect data have been considered, as well as the proposed and available repair options.

An electronic database of all Notices of Structural Deficiency (NSD) from previous full-scale fatigue tests, including the ongoing IFOSTP tests, and available data from fleet inspections has been prepared. There were nearly 1700 entries. Data were then grouped by Damage Item Location (DIL) and the DIL assessed against various safety and economic criteria for inclusion in one of the programs. The assessments were in accordance with the F/A-18 RAAF Aircraft Structural Integrity Management Plan (ASIMP), which uses DEF STAN 00-970, Amendment 13 as guidance.

The results support the conclusion that the centre barrel section of the center fuselage will need replacement at SRP 2 to prevent fatigue maintenance costs rising dramatically. The center barrel contains the 3 wing-attachment bulkheads, the major Safety of Flight structural items in the centre fuselage. Due to the potential presence of widespread fatigue damage the centre barrel replacement option was preferred over a piecemeal repair process

In addition to the centre barrel region which requires modifications, recent teardown results of the two inner wings from the IFOSTP FT55 centre fuselage article, indicate that alternative management strategies for the wings may be required. The wings present additional challenges due to the non-inspectable nature of many significant components. Work is proceeding in this area.

Although the SRP program will represent a substantial investment, it once again demonstrates the necessity of conducting high quality fatigue testing and analysis to support the through life management of a fleet.

8.2.2 Repair and Life Assessment of Fatigue Damaged F/A-18 Aluminium Alloy Structure: S. Barter, L. Molent, K. Sharp & G. Clark; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Research at the Aeronautical and Maritime Research Laboratory (AMRL) into the material factors likely to affect F/A-18 fatigue life highlighted the critical role played by surface condition in determining the service fatigue life of aircraft structure of the type used in RAAF F/A-18 aircraft. Further research capitalised on this understanding by developing a method for localised life extension [1]; the aim was to be able to overcome any threat to fleet airworthiness by extending the life of regions which displayed unexpectedly rapid or early fatigue cracking in service. This method involves removing a specified amount of "fatigue damaged" material followed by a controlled shot peening operation.

Much of the AMRL research has focussed on the highly stressed 7050 aluminium alloy wing carry-through bulkheads, although the rework/repair method should be applicable to most of the critical airframe structure. It is notable that the critical areas of cracking found in full-scale fatigue tests on the F/A-18 airframe were mostly independent of fastener holes due to fatigue enhancements at these holes, and optimised design procedures which provide highly uniform stressing. Most problem areas have been found to be associated with low Kt structural details that are usually coated in the standard corrosion preventative system. This system includes Ion Vapour Deposited (IVD) aluminium. The application of IVD aluminium is carried out on the pre-etched surface of the 7050 aluminium alloy in the F/A-18 aircraft. This etching has introduced a source of fatigue crack initiation, which must be examined in addition to those considered traditionally.

In order to quantify the effect of the etching process a fatigue test program was carried out on coupons made from 7050-T7451 aluminium alloy and designed to be representative of F/A-18 aircraft structural details. During these tests new fractographic methods were explored for characterising the crack-like effect of etch pitting. The important initial results of this testing, along with the full-scale testing of the F/A-18 aircraft, are that cracking from flaws such as inclusions is almost non-existent. The etch pits appear to be considerably more effective in starting fatigue cracks than cracked inclusions. The fatigue cracks begin to grow virtually from the beginning of fatigue cycling.

One region of F/A-18 structure in particular was used [1] to highlight the proposed repair process. This is the highly stressed X19 location in the Y470 bulkhead where cracking is believed to be developing to the extent that the fleet could require unacceptably high levels of inspection and maintenance.

Since RAAF Hornet aircraft are fatigue monitored individually, a repair can be optimised for each aircraft. This can be achieved by specifying an amount of material to be removed based on the calculation from the fatigue accumulated by an individual aircraft. The proposed method for the repair of critical regions will allow continued safe operation of the F/A-18 fleet.

8.2.3 Managing F/A-18 Fatigue Damage Using Operational Parameters: L Molent , Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Stores configuration and the Point In The Sky (PITS), where manoeuvres are completed, can have a marked effect on fatigue damage accumulation of agile fighter aircraft. Strain and flight parameter data recorded by the Royal Australian Air Force's (RAAF) F/A-18 fleet on-board Maintenance Signal Data Recording System (MSDRS) was used to identify relationships between fatigue damage and stores configuration or the PITS where the aircraft completed its manoeuvres [2]. The results of these relationships may in future allow the RAAF to tailor its missions and flying practices, so that fatigue damage is minimised.

Relationships between fatigue damage and the PITS were identified by calculating the average strain recorded, at 7 altitude ranges and 10 speed ranges, for 8 normal acceleration ranges (5 were for strain peaks and 3 were for strain valleys). As fatigue damage is a function of strain range, the trends identified between the strain peaks and strain valleys at the different PITS, could be related to fatigue damage.

Relationships between fatigue damage and stores configuration were developed by calculating the average strain at different PITS for different aircraft configurations. The difference between the average strain for two different configurations allowed conclusions to be made with regards to the configuration and fatigue damage.

The results of this work has lead to numerous conclusions about the influence of PITS and stores configuration on fatigue damage. An increased altitude was shown to decrease fatigue damage, as was a reduced Mach number. Empty centre line fuel tanks were shown to have a negligible affect on fatigue life but wing stores have the potential to reduce fatigue damage (although analysis thus far is based on symmetric manoeuvring only).

This information will allow the RAAF to tailor missions and provide advice to pilots so that the life of its aircraft are optimised.

8.2.4 Bounding Structural Durability Due to Buffet Induced Loading: L Molent , Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Australia and Canada is conducting a collaborative full-scale structural testing program known as the F/A-18 International Follow-On Structural Test Project (IFOSTP) to determine the economic and safe life of the F/A-18 structure. Canada is testing the centre fuselage and wings and Australia is testing the aft fuselage and empennage.

A unique full-scale fatigue test system has been developed by AMRL which combines buffet induced dynamic loading with manoeuvre loading to simulate the flight loading conditions experienced by an F/A-18 aircraft under typical operations. The system uses a pneumatic loading system to apply the distributed aerodynamic and inertial loads induced by aircraft manoeuvres. High powered, high displacement electromagnetic shakers apply the severe buffet dynamic loading experienced in flight.

Although the primary purpose of IFOSTP was to conduct and interpret fatigue tests, it has enabled other research to be conducted, including the assessment of buffet load related variability. Reference [3] addresses some factors that impact on the buffet load variability and in turn, traditional safety factors applied during design or in-service life monitoring. These factors include:

- Differences in buffet loading at nominally identical Points-In-The-Sky (PITS) due to the non-linear and quasi-random nature of the vortices generated by the LEX;
- Load response measurement differences between vertical tails (VTs) due to gauge scaling effects and structural redundancy;
- Differences in natural frequencies between VTs;
- Large strain gradients at VT structural details making generic critical points difficult to identify; and
- Errors in measuring dynamic loading.

A preliminary assessment of these factors was made to bound the structural effect of the variability induced by buffet loading using a safety factor approach [3]. This assessment which is reported in reference [3] was achieved by redeveloping the spectrum applied to FT46 using two additional and independent data sets.

8.2.5 Review of Fatigue Monitoring of Agile Military Aircraft: L Molent; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The fatigue management of an aircraft starts in the design process with the application of a design philosophy, stress spectra, material data and a damage theory to estimate the fatigue life. This estimate is then certified through a structural fatigue test, following which (or sometimes before) the aircraft operator collects service load data and puts together a management policy. The process of collecting service load data is termed fatigue monitoring and airworthiness regulations require all fighter type aircraft to be fitted with an on-board usage monitoring system.

Reference [4] presents a summary of a literature review (approx. 120 papers) on fatigue monitoring philosophies, systems, fatigue models and practises.

Fatigue monitoring serves a number of purposes:

- to fulfil airworthiness requirements to ensure aircraft are not operated beyond an acceptable level of risk;

- to determine the fatigue life status of a fleet of aircraft throughout its life based on an operational spectrum;
- to determine the actual service load history (many operators have found that operational usage of an aircraft is significantly more severe than the design spectrum) to ensure that aircraft are not operated beyond the fatigue damage accumulation threshold for various components as demonstrated through full-scale testing;
- to improve or to optimise the structural integrity management of the fleet (when done in conjunction with a program based on tracking each aircraft in the fleet). The assertion here is that the utilisation of each aircraft is different and that using an average value is inaccurate when monitoring the whole fleet;
- to detect occurrences of structural overloads in a timely fashion, thus enhancing fleet safety; and
- to assist in the definition of a flight load spectra for new aircraft of the same type.

Current processes are presented and comprehensively examined and where appropriate the benefits and drawbacks of the respective methods were stated. The history of fatigue management is presented followed by an outline of usage monitoring programs currently used by operators. The paper examined the issues of strain gauge utilisation and calibration, collection of flight parameter data, data integrity, data handling, comparisons with fatigue test results and fatigue damage models. The paper also included a discussion on the problems that have arisen in the last decade due to high angle of attack capabilities and redundant structures of fighter aircraft.

8.2.6 Development and Validation of an Analytical Model to Predict Fatigue Crack Growth in Notch Plastic Fields: K. Walker Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

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 Figures 1 and 2 below) are particularly susceptible.

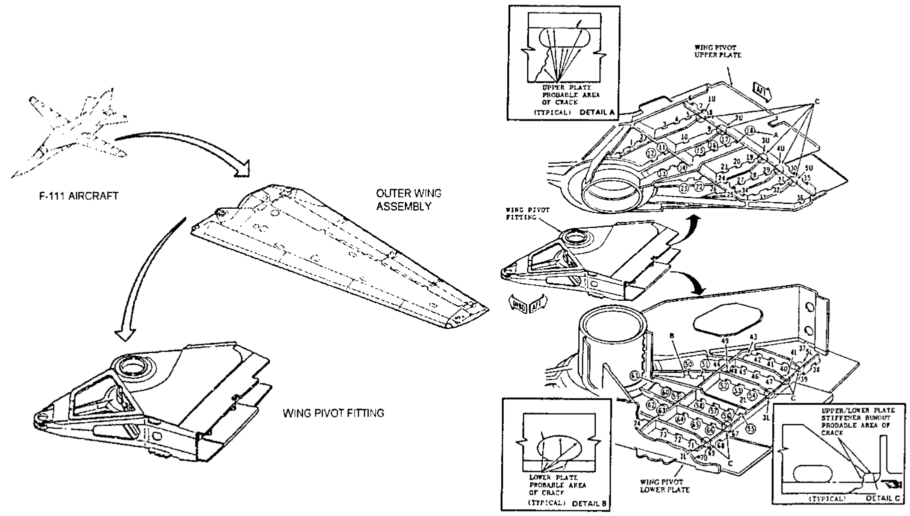


Figure 1: F-111 Wing Pivot Fitting Fuel Flow Vent Hole Details

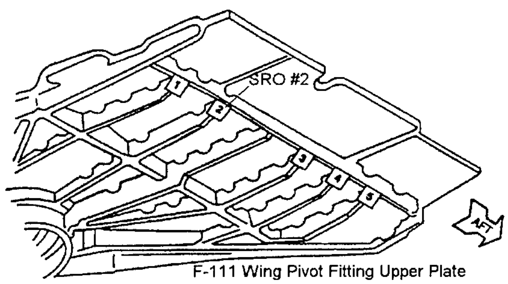


Figure 2: F-111 Wing Pivot Fitting SRO #2 location.

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 developed 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.

METLIFE analyses (References 5 and 6) have been conducted for the WPF upper plate FFVH #13 and SRO #2 locations. Data from testing programs involving specimens representative of the two locations were used to validate the METLIFE approach. In the case of SRO #2, two configurations were used; a baseline profile representative of the original design, and an optimised re-work profile designed to reduce peak stresses at the runout. Linear Elastic and Elastic-Plastic Finite Element Modelling, combined with strain gauge data, were used to determine representative load levels and load path offsets.

METLIFE analysis results were compared with an Elastic-Plastic Finite Element Analysis (ABAQUS) based strain response, and measured crack growth data. The analytical crack growth results correlate well with experimental data. 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.

Further development of METLIFE is still required, however the results thus far are encouraging. Areas for further improvement include the method of estimating the 3-D elastic constraint at the notch root, and refinements to the cyclic plasticity algorithm. 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.

8.2.7 Macchi Vertical Fin Crack Growth under Test and Service Spectra Nick Athiniotis and Graham Clark; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Damage Tolerance Analysis of the Macchi Vertical Fin forward attachment fitting (PN 3303-01-01) in December 1999 by Aerostructures Australia revealed an error in the original life analysis, leading to an incorrect estimate of the relationship between RAAF in-service loading and AerMacchi test loading. Consequently, FE crack growth analysis for a corner flaw (0.12 mm surface length) predicted a critical crack length of only 1.32 mm, and, based on an NDI inspection interval of 410 flight hours, the detectable crack size would have been 0.48 mm. With current NDI methods allowing a crack detectability of about 1 mm (based on RAAF assessment of detectable cracking in the component), the results of the analysis would have dramatically affected the airworthiness of the aircraft, possibly leading to indefinite grounding of the fleet. Validation of these results became the highest priority in the RAAF.

AMRL provided results of fracture toughness tests performed on the attachment fitting material and undertook fatigue tests on several components with known initial flaws to provide crack growth data. Fatigue testing revealed the existence of an internal manufacturing flaw from which several failures occurred during testing. In 1991, it is believed that the Italian Air Force lost an aircraft in-flight following failure of the fin, and it is thought that this was attributed to a manufacturing defect in the forging plane (flash line) at the 6 o'clock position. Inspections of the Italian fleet at that time revealed a 40 per cent incidence of defects in components manufactured prior to 1971. The RAAF fleet had most of the original fittings replaced during Life Of Type EXtension (LOTEX) in 1981, with a few – those having zero or low service life - carried over. The defects exposed at AMRL highlighted that RAAF aircraft had been flying with similar defects. Crack growth analysis suggested that the rate of cracking from these manufacturing defects was faster compared to cracking from the bore. An ultrasonic (U/T) method was developed at AMRL during testing to detect sub-surface flaws in the component.

8.2.8 Sbi Programme For The RAAF Caribou Fleet: Jon Kerr ; Aerostructures Technologies Pty Ltd, Melbourne, Australia

The Royal Australian Air Force (RAAF) currently operates 14 DHC-4 Caribou aircraft, based at Amberley Air Force Base Queensland. The RAAF DHC-4 fleet has seen active service in Vietnam and recently in East Timor. The aircraft were designed and built by De Havilland Aircraft Canada (DHC) and delivered to the RAAF between 1964 and 1971. The Defence White Paper [7] released in December 2000 advised that the Caribou would be withdrawn from service in 2010.

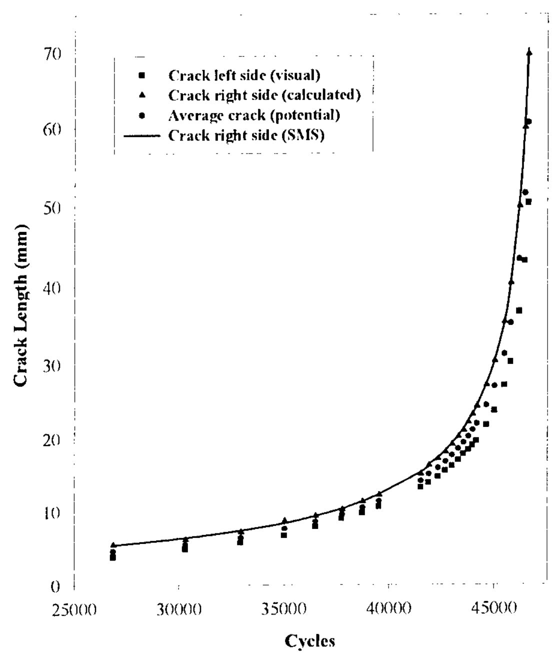


Figure 3 Crack Propagation Test, CCT Specimen

The RAAF Caribou fleet is now nearing the specified fatigue safe life limit for the wing. As the RAAF plans to continue operating the aircraft for another 10 years an alternative structural integrity management system is required. Aerostructures were tasked by the RAAF to develop a Safety By Inspection (SBI) programme, based on Damage Tolerance Analysis (DTA), to manage the wing structural integrity beyond the safe life limit.

Federal Aviation Regulation (FAR) 25.571 Amendment 25-96, Damage Tolerance and Fatigue Evaluation of Structure was selected as the design standard. This FAR standard is the successor to CAR4b.270, Fatigue Evaluation of Flight Structure, to which the Caribou was originally designed.

The development of the SBI programme is conducted in four stages as follows:

(a) Stage 1 – Identification of the DTA critical locations on the Caribou wing [8].

(b) Stage 2 – Development of stress spectra and DTA criteria for the critical locations [9]

(c) Stage 3 – Re-assessment of the task scope

(d) Stage 4 –DTA of the critical locations [10].

The RAAF Caribou SBI programme provides the means for managing the structural integrity of the Caribou wing beyond the original fatigue safe life limit until the planned withdrawal date.

8.2.9 Damage Tolerance Analysis Of The P-3c Orion: William Mullen, Michael Houston, Aerostructures Technologies Pty Ltd, Melbourne, Australia.

The Royal Australian Air Force Lockheed P-3 Orion is currently managed under a safe life philosophy. The RAAF is addressing P-3C structural integrity issues in the long term with its involvement in the United States Navy P-3 Service Life Assessment Program (SLAP). However, these issues are either effecting fleet readiness now or will do so well before SLAP results become available. Therefore, to investigate the possibility of short term relief and the potential for a durability and damage tolerance certification basis beyond the safe life limit, a damage tolerance analysis of five fatigue and logistically critical locations has been conducted [11].

The RAAF have not formally adopted or developed a design standard governing damage tolerance analyses for the P-3. Therefore, three standards, FAR 25, MIL-A-83444, and DEF STAN 00-970, were compared and conclusions drawn regarding the suitability of each. MIL-A-83444 was chosen for the purposes of the study based on the specific guidance provided by it and the associated USAF Damage Tolerant Design Handbook.

The damage tolerance analysis for each of these points included definition of configuration and geometrical detail, selection of appropriate material data, consideration of crack growth scenarios, residual strength and crack growth analysis. The recently completed P-3 Structural Life Monitoring Program (SLMP) [12] was used to produce stress spectra. The analyses also served to highlight several issues that would require resolution before the task of formally producing a DADTA baseline for the P-3C could advance. Such issues are common to many platforms for which fundamental changes in structural integrity management philosophy are being contemplated and include determination of appropriate rate data, development of realistic and conservative flaw development scenarios (including continuing damage) and selection of operationally suitable inspection techniques (and therefore detectable crack size).

8.2.10 P-3 Orion Structural Life Monitoring Developments: Michael Houston, Aerostructures Technologies Pty Ltd, Melbourne, Australia

RAAF P-3C Fatigue Life Indices (FLI) are calculated by the RAAF Service Life Evaluation Program II (SLEP II), a fatigue tracking program produced by Lockheed in 1989. SLEP II is based on pre-defined mission profiles and load spectra which are no longer representative of current usage [13], hence reducing the relevance of the FLI calculation. As the calculation becomes less relevant, the risk to airworthiness and the potential for economic loss and reductions in capability due to unnecessary inspections and lower aircraft availability increase. The RAAF does not possess a program that will produce certified FLI values for the TAP-3, P-3B(L) aircraft recently purchased from the USN to undertake pilot training and squadron support.

Both the P-3C and the TAP-3 fleets are close to their certified structural Life Of Type (LOT), and a large number of aircraft are already subject to the mandatory Safety-By-Inspection programme which is initiated at 75 FLI. However, the current Planned Withdrawal Date (PWD) for both types is 2015. At current Rates Of Effort (ROE), all aircraft will pass the current safe life limit well before this time. Hence, not only is the of the ability of the P-3 Structural Life Monitoring System (SLMS) to support pre-safe life limit structural integrity management under question, but consideration of post-100 FLI operations and consequent SLMS requirements is necessary.

Aerostructures, tasked by the RAAF, has recently undertaken two initiatives designed to rectify this situation. An enhanced Structural Life Monitoring Program (SLMP) has been developed for the P-3 [14, 15, 16]. Key improvements offered by the SLMP over SLEP II are:

1. the use of measured N_Z data to assemble load and stress spectra, compared to the fixed spectra in SLEP II;

2. the ability to calculate FLI for the TAP-3 aircraft;
3. the addition of some profile adjustments (eg. weight corrections) to improve the validity of fixed reference loads; and
4. the ability to generate stress spectra at each critical location for use in crack growth analysis.

The availability of the SLMP has already had a significant impact on plans for the through-life management of the TAP-3 by revealing that the risk of a structural LOT driven by fatigue is significantly lower than first thought.

The second initiative involves the introduction of an advanced Structural Data Recording Set (SDRS) to the P-3C and TAP-3 fleets [16,17]. This system is capable of recording mission profile parameters and local strain, as well as more traditional data (eg. N_{Z}, landings etc.). The SDRS will enable periodic review of actual mission profiles for comparison with profiles assumed by the P-3 SLMP and development of critical location stress spectra from locally measured strain for the purposes of crack growth analysis. The SDRS is yet to be fitted to the TAP-3, but data is being received from installations on several P-3C aircraft.

8.2.11 Australian Contribution to the International P-3 Service Life Assessment Program (SLAP): P. Jackson; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Australia has entered into a collaborative program with the USN, Canadian Forces and the Netherlands to conduct a series of full scale fatigue tests of the P-C aircraft. The program is titled the P-3 Service Life Assessment program (SLAP) and commenced in 1999. Australia's technical contribution to the SLAP program is being led by AMRL 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 that is in its original unmodified condition, and the conduct of a teardown on an ex RAAF and RNZAF P-3B(H) wing. Test preparations have been on-going at AMRL for the empennage test since 1999 and the test is due to start cycling in September 2001. This will be in line with the other fatigue tests on the wing/fuselage and undercarriage being conducted at Lockheed-Martin, Marietta GA on behalf of the USN. All tests are planned to cycle until the end of 2002. Test article 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.

8.2.12 B707 Structural Life of Type Studies: P. Jackson; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The RAAF fleet of B707 tanker/transport aircraft have all reached the original design life of the aircraft (20,000 flights or 60,000 flight hours) and are now subject to the Boeing developed Aging Aircraft related programs of structural inspections and mandatory modifications. The RAAF aircraft are of a similar age and commercial operator background to the aircraft being purchased and converted by the USAF for JSTARS operations. In 1996 the USAF conducted a lower wing teardown program on two high life aircraft and found evidence of widespread fatigue damage WFD) in the lower wing skin/stringer panels. An accompanying USAF structural risk assessment showed that the panels would need to be replaced to provide both airworthiness assurance and to limit future aircraft maintenance down time. A sampling inspection on RAAF aircraft revealed similar damage. The RAAF, QANTAS and AMRL all contributed to a study of the airworthiness and cost of ownership implications of continued operation of RAAF B707 aircraft. [18] This study included a risk analysis [19] of the wing lower skin panels at the location of the WFD. This study concluded that RAAF aircraft could be maintained for a short period under an increased inspection regime however the lower wing skin panels would require replacement in the medium term. This finding and other assessments of the condition of the structure of the RAAF fleet led to a decision by the RAAF to retire the B707 aircraft in the near future and seek a replacement AAR capability.

8.2.13 Australian Army, Sikorsky S-70A-9 Black Hawk, - Flight Loads Survey: D.C. Lombardo and G.F. Forsyth; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

A joint USAF/ADF flight loads measurement program was conducted on an Australian Army Black Hawk helicopter. The program aims to quantify airframe and dynamic component stresses that can be used in defining required structural enhancements. These enhancements will address several structural integrity issues that are common to the USAF's HH-60G and the Australian Army's S-70A-9 Black Hawks. The other participants in the program are the prime contractor for the USAF (Georgia Tech Research Institute - GTRI), the helicopter manufacturer (Sikorsky Aircraft, as a sub-contractor for flight test involvement and fatigue analysis), the Army Aircraft Logistics Management Squadron (AALMS), the Aircraft Research and Development Unit (ARDU) and DSTO (which provided scientific support to the ADF).

This is the most significant helicopter flight investigation yet performed in Australia, with over 65 flight hours of data being recorded from nearly 300 parameters during the period 21 July – 19 December 2000.

Analysis of the data is currently being performed at GTRI and at Sikorsky's facilities at West Palm Beach, Florida, and Stratford, Connecticut. DSTO has placed a senior officer on a long-term attachment at GTRI to gain an in-depth understanding of the methodologies used in analysing the data.

The program now enters its next phase, which will consist of a detailed data analysis to produce validated structural and aerodynamic models of the Australian and USAF versions of the Black Hawk helicopter. GTRI, in conjunction with Sikorsky, will undertake this work. The collaborative program will terminate once this phase is complete.

The availability of such a comprehensive flight loads data means that DSTO now has an invaluable resource for supporting the structural integrity of the Black Hawk and conducting research into helicopter structural fatigue in general.

8.2.14 Health and Usage Monitoring Systems (HUMS): D.C. Lombardo and G.F. Forsyth; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

DSTO International Conference on Health and Usage Monitoring (HUMS2001)

Following the HUMS conference held at AMRL in February 1999 (see 1999 ICAF report), over 100 people (almost half from outside Australia) attended the two-day HUMS2001 Conference organised by DSTO and held at the Duxton Hotel in Melbourne on 19-20 February 2001. Speakers from Australia, USA, Canada, France, UK and Israel presented twenty-eight papers and two panel sessions. Other countries represented included New Zealand and South Africa. The Australian Defence Organisation, other military services, civil aircraft operators, regulating authorities, OEMs and avionics suppliers were represented as well.

The conference provided an invaluable opportunity for DSTO and others in the Australian defence community to interface directly with lead international players in this important area. It also allowed attendees to gain an updated appreciation of the status of technologies now appearing, or anticipated, on ADF aircraft such as the Hawk, C-130J, Seasprite and the new armed reconnaissance helicopter. Technical topics included a description of the monitoring system in the RAN Super Seasprite, the requirements for the armed reconnaissance helicopter, the Dassault Rafale, HUMS projects for the UK military services and the US Navy, and background research being conducted by DSTO, US Navy, universities and industry, as well as presentations by suppliers of HUMS equipment.

Considerable interest was expressed in having a third conference in this series in February 2003.

8.2.15 Shape Optimisation of Critical Stiffener Runouts in the F-111 Wing Pivot Fitting: M. Heller, M. McDonald, M. Burchill and K.C. Watters; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Optimal rework shapes for the most fatigue critical stiffener runout (SRO) locations in the F-111 wing pivot fitting (WPF) upper plate region have been determined using a recently developed finite-element-based gradientless shape optimisation procedure **[20]**. The WPF region is shown in Figure 4. The resulting precise free-form shapes (which typically remove cracked material) render the local notch stress distributions near uniform and typically provide a 30–40% reduction in peak elastic stresses as compared to current rework shapes that exist for aircraft in service with the Royal Australian Air Force. They also typically represent a 50% improvement in stresses, as compared to the nominal blueprint shapes. The unique optimal shapes (of different sizes) have been determined for four of the most fatigue prone SRO locations in the F-111 WPF. The final peak stress level achieved at each of the optimised runouts was relatively consistent; therefore, a similar inspection interval can be expected for each location. A sample comparison of elastic stresses is given in Figure 5, where it should be noted that the key aim was to reduce the peak compressive stress (to minimse residual tensile stresses after cold proof load testing). The recommended rework configuration provides a manageable compromise between minimising the runout stresses and maintaining the WPF buckling strength to within acceptable limits. A number of important issues have been addressed in the present practical problem, including; use of higher-order elements for efficient robust stress prediction; accounting for the effect of size constraints on the optimal shapes; and assessing the robustness of the idealised optimal shapes to perturbations away from idealised conditions, such as those due to potential manufacturing errors.

As part of an associated validation program, the precise shapes have been manufactured in two full-scale static test wings. Experimental strain measurements for the optimal shapes compare very well with predictions from finite element analysis. Further wing tests are currently underway in order to determine the damage tolerance and durability of these optimal shapes. Hence it is expected that the stress reductions predicted in the present work will be sufficient to provide a basis for extending inspection intervals by a least a factor of two, from 500 hours to 1000 hours. Implementation of such an extension to the F-111 fleet would provide a very significant maintenance cost saving. Based on the successful results to date, fleet-wide implementation of the optimal reworks is scheduled to commence this year, as part of the Australian F-111 Sole Operator Program.

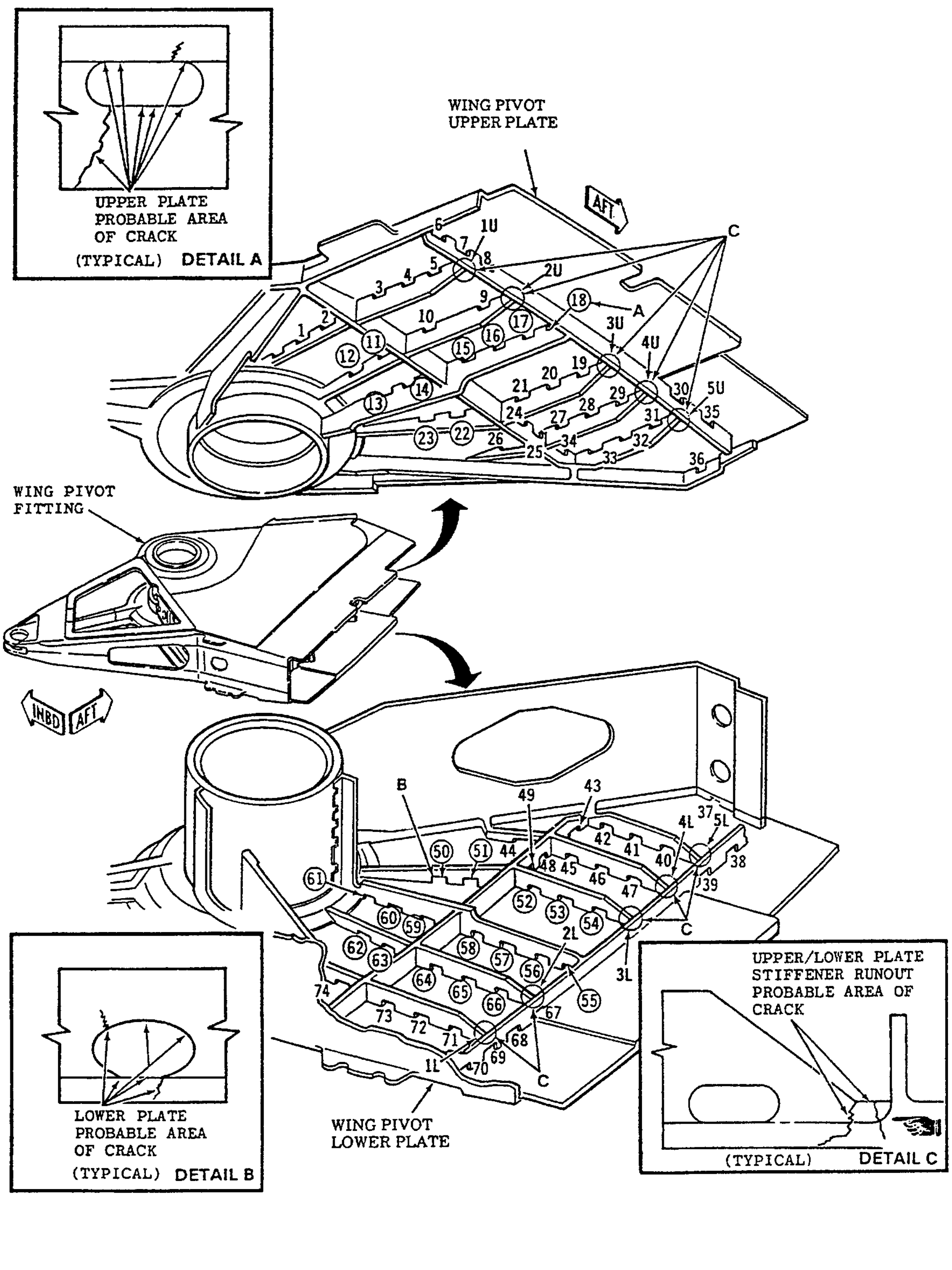


Figure 4: Detailed internal view of the wing pivot fitting structure showing fuel flow vent hole (FFVH) and stiffener runout (SRO) features.

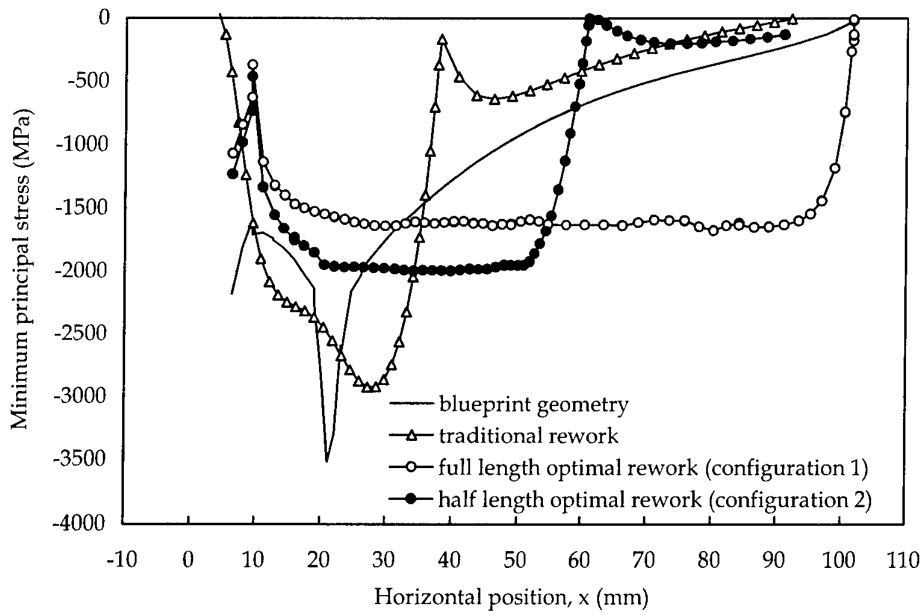


Figure 5: Elastic boundary stress distributions at SRO#2 for blueprint, traditional and optimised rework shapes for idealised Cold Proof Load Test loading of +7.33g.

8.2.16 Shape Optimisation of Critical Fuel Flow Vent Holes in the F-111 Wing Pivot Fitting: M. Heller, M. Burchill, M. McDonald and K.C. Watters; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Optimal rework shapes for the four most fatigue critical fuel flow vent hole (FFVH) locations in the F-111 WPF have been determined using a recently developed finite-element-based gradientless shape optimisation procedure **[21]**. The WPF region is shown in Figure 6. The resulting precise free-form shapes render the local notch stress distributions near uniform and typically provide a 35–40% reduction in peak elastic compressive stresses as compared to current rework shapes that exist for aircraft in service with the Royal Australian Air Force (RAAF). The unique optimal rework shapes have been determined for four of the most fatigue prone FFVH locations in the F-111 WPF, namely FFVHs 11, 12, 13 and 14. Here reworks of different sizes were considered to cover fleet requirements, taking into account difficult prevailing geometric constraints. Here reduction of multiple stress peaks around the hole boundaries (both tensile and compressive) was achieved. A sample comparison of elastic stresses is given in Figure 3, where it should be noted that the key aim was to reduce the peak compressive stresses (to minimise residual tensile stresses after cold proof load testing).

As for the runout optimisation discussed in the preceding section, good agreement with experimentally measured strains for the FFVHs was achieved. Also, further fatigue testing is currently underway. Hence it is expected that the stress reductions predicted in the present work will be sufficient to provide a basis for maximising inspection intervals, as well as justifying the existing FFVH inspection interval of 1000 hours. Implementation of the optimal shapes would be expected to provide significant cost savings and increased aircraft availability for the RAAF F-111 fleet. Again, fleet-wide implementation of the optimal reworks is scheduled to commence this year.

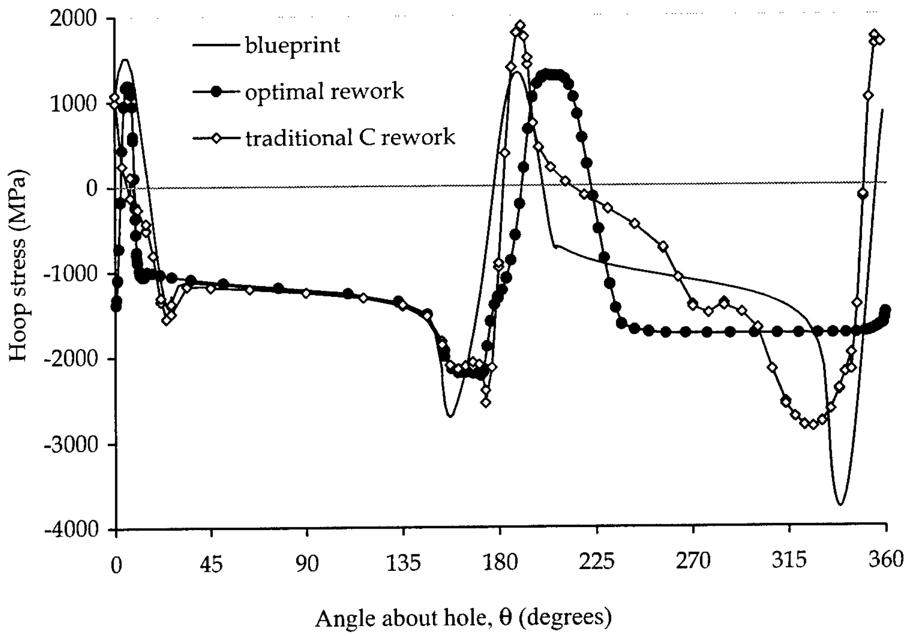


Figure 6: Elastic boundary stress distributions at FFVH 13 for blueprint shape, traditional C rework, and final optimal rework shapes, at idealised CPLT loading of +7.33g.

8.2.17 Robustness of the F-111 Wing Pivot Fitting Optimal Rework Shapes: M. McDonald, M. Heller, M. Goldstraw and A. Hew; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The robustness of the recently developed optimal rework shapes for critical locations in the F-111 wing pivot fitting has been determined by numerical sensitivity analyses. Here the effect on the peak stress for optimal reworks, due to variations in conditions such as loading, manufactured position and local shape accuracy, has been quantified using 2D and large-scale 3D finite element analyses **[22]**. It has previously been estimated that, under ideal conditions, the optimal reworks provide a 30–50% stress reduction compared to current shapes in the fleet. The numerical results from the present investigation show that, under the combined worst-case conditions, these stresses would increase again by only 10–20%. Hence the optimal shapes still provide a significant stress benefit, when manufactured to within the tolerances specified herein, and therefore are considered robust enough for practical use. It is anticipated that the implementation of the optimal rework shapes will be sufficient to extend inspection intervals, hence providing significant maintenance cost savings, increased aircraft availability, and allow achievement of the planned withdrawal date of 2020 for the F-111.

8.2.18 Optimal Free-Form Shapes for Shoulder Fillets in Flat Plates under Tension and Bending: W. Waldman, M. Heller and G.X. Chen; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

In engineering practice, section-transitioning fillets in plates have typically consisted of circular profiles. However, these profiles are not optimal and can result in significant stress concentrations. Also, there is a paucity of transferable data available in the literature relating to optimal fillets, which could offer an alternative option. 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 [23]. The geometry and notation for a typical loaded fillet is shown below in Figure 7. 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 given here offer the lowest possible stress concentration, subject to the bounding geometric constraints, and are presented in a tabular form that allows them to be easily used by designers. Here in all cases the hoop stresses have been rendered uniform along the fillet. Typical stress concentration results are given in Figure 8.

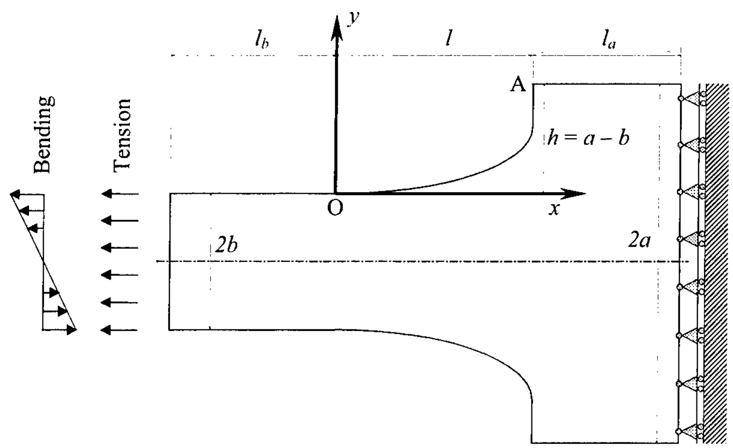


Figure 7: Geometry and notation for plate with fillet subjected to uniaxial tension or bending loading.

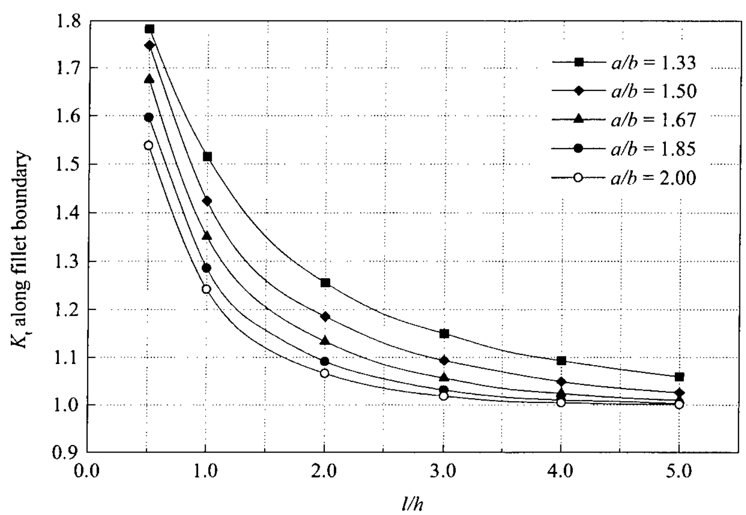


Figure 8. Variation of stress concentration factor K_t for optimal tension fillets.

8.2.19 Through-Thickness Shape Optimisation Of Bonded Repairs And Lap-Joints: R. H. Kaye And M. Heller; Aeronautical And Maritime Research Laboratory, Melbourne, Australia.

For realistic applications, the design of bonded repairs and lap-joints has often been undertaken through trial-and-error finite element analyses, or experiment. Recent experience indicates that, for more complex practical applications, unacceptably high adhesive stresses can occur in the adhesive layer. In the present work, an automated sensitivity-based shape optimisation procedure has been developed for the optimal design of free-form bonded repairs and lap joints, with the aim of achieving reduced adhesive stresses **[24]**. The approach has been demonstrated through application to a number of single- and double-sided configurations where both the shapes of the adhesive layer and the outer adherend are allowed to vary. The nominal initial lap-joint configuration under investigation for subsequent optimisation is shown in Figure 9. Here, assuming linear elastic conditions, the optimisation has provided significant improvements over conventional designs, as assessed by the reduction in peak adhesive stresses. (Figure 10) While these results indicate the scope for the use of numerical shape optimisation to improve standard designs, further work is to be undertaken to include temperature and plasticity effects.

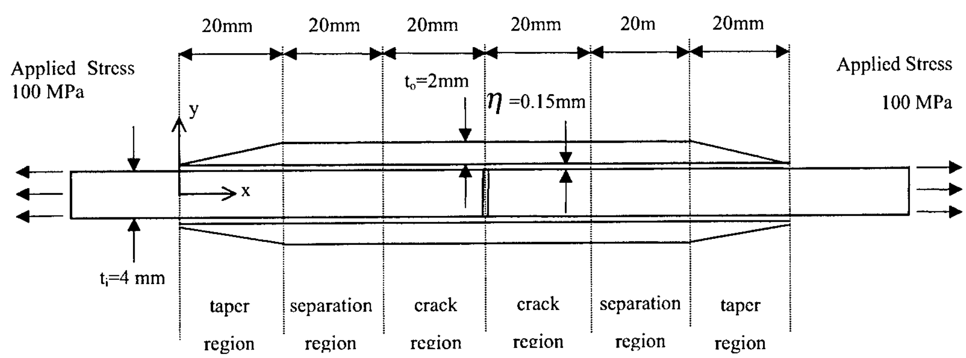


Figure 9: Dimensions and loading arrangement for double lap-joint.

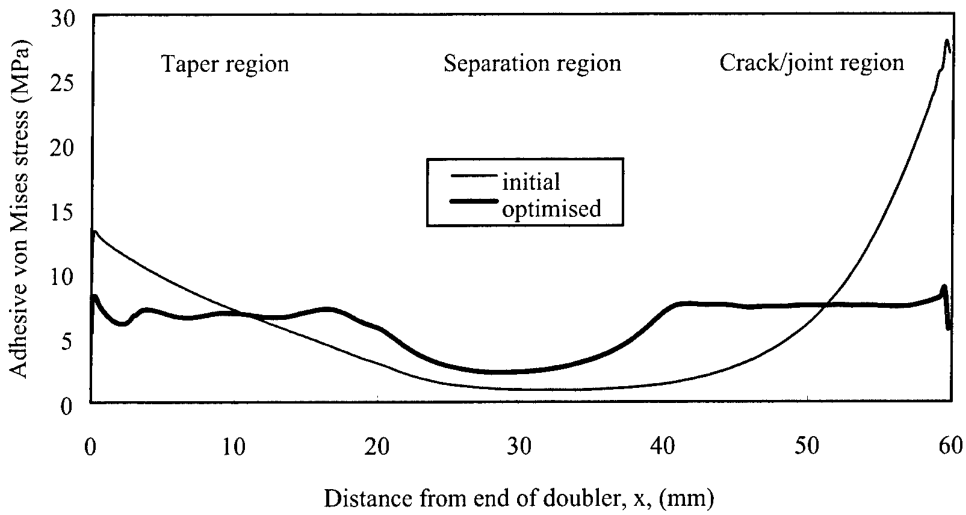


Figure 10: von Mises stresses in adhesive layer, for crack repair with boron/epoxy doublers, and aluminium inner adherend.

8.2.20 Investigation of Shape Optimisation for the Design of Life Extension Options for an F/A-18 Airframe FS 470 Bulkhead: R. Kaye and M. Heller; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

In the present work, an automated sensitivity-based shape optimisation procedure has been developed for the optimal design of free-form reworks and bonded reinforcements, and its usefulness demonstrated through application to a realistic practical problem, the F/A-18 FS 470 bulkhead [25]. Here, a least squares objective function written in terms of selected stress quantities is used, along with multiple shape basis vectors to specify allowable shape changes. For the rework option a unique optimal solution has been determined, which achieves a region of constant boundary hoop stress which is 25% less than for the nominal initial uncracked geometry, even though material removal at the critical location (of the crotch region) is accounted for. Subsequently, for the bonded reinforcement analyses, two distinct optimal designs were determined, corresponding to the case where either shear or peel stresses in the adhesive layer are minimised. Both the shape of the adhesive layer and the reinforcement are allowed to vary, and significant improvements over a conventional reinforcement design are obtained, as assessed by the reduction in peak stresses. The results for the minimum shear stress case are shown in Figure 11, where it is also seen there is no increase in the bulkhead stresses as compared to standard reinforcement.

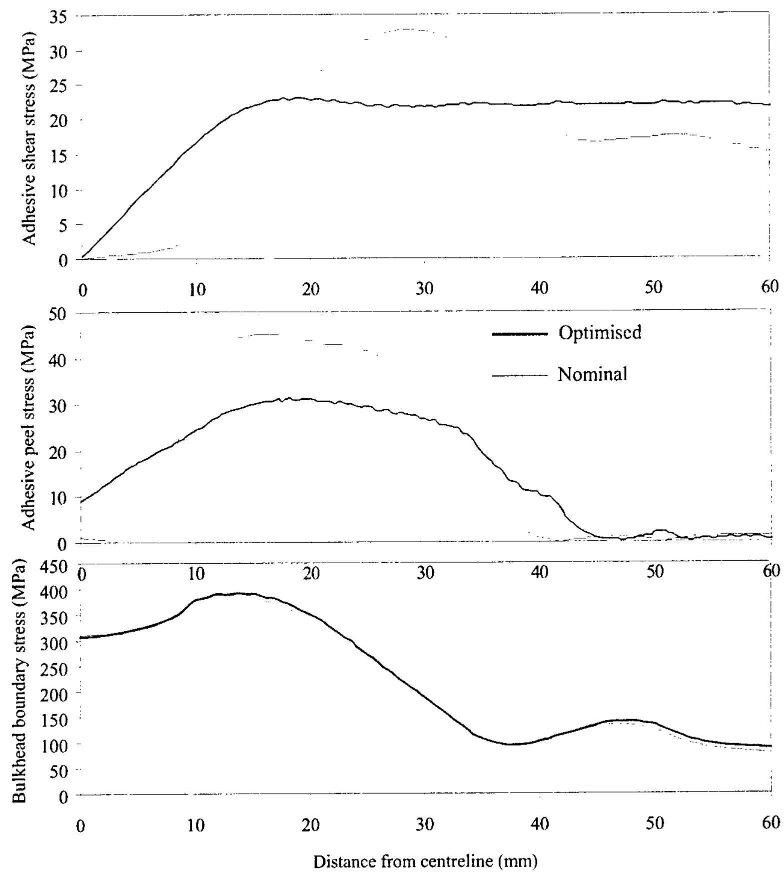
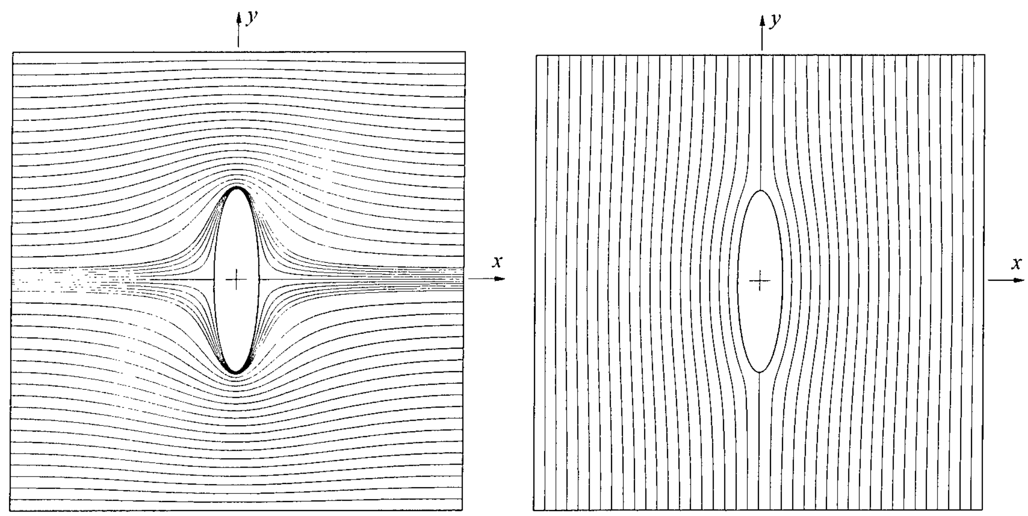


Figure 11 Stress solution for 470 bulkhead optimal reinforcement, with minimal adhesive shear stress.

8.2.21 Advances in Structural Loadflow Visualisation and Applications to Optimal Shapes: W. Waldman, M. Heller, R. Kaye and L.R.F. Rose Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Currently there is no generally accepted procedure for calculation of structural loadpaths, which would show how remote loads are equilibrated through a structure and could provide insight into how well a structure is performing its intended load-carrying functions. Recently a useful method has been proposed for computing loadflow orientations and loadpaths using finite element results, which is based on iterative solutions of non-linear equations **[26]**. In the present investigation the prior theoretical formulation and general procedure has been enhanced by deriving explicit expressions for computing loadflow orientations **[27]**. The new equations produce more accurate loadflow orientations and improves the fidelity of calculated loadpaths. In a series of benchmark problems, non-optimal and optimal holes in plates have been investigated using loadflow visualisation to identify their key features. It was found that significant recirculation is apparent for non-optimal hole shapes, whereas no recirculation zone is present for optimal shapes. Typical load paths for an optimal hole are shown below in Figure 12. The calculation of loadflow orientations using the new equations is simple, and could be used with any finite element analysis code, while a plotting package is required to display loadpaths. Loadflow visualisation is a powerful tool for use by structural designers to improve their understanding of structural performance, the application of which can potentially result in improvements in structural efficiency.



(a) Resolved x-direction loadpaths **(b) Resolved y-direction loadpaths**

Figure 12: Resolved x-direction and y-direction loadpaths obtained around an optimal 4:1 elliptical hole in a large square plate under y:x = 4:1 biaxial loading, showing no recirculation.

8.3 FATIGUE OF CIVIL AIRCRAFT

No Items

8.4 FATIGUE RELATED RESEARCH PROGRAMS

8.4.1 Initial Defects And Short Fatigue Crack Growth In 7050 T7451 Thick Section Plate : Simon Barter and Graham Clark; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

A number of studies have been commenced on short crack growth in 7050-T7451 aluminium alloys. These studies are being carried out on low Kt specimens loaded with a representative fighter aircraft wing root bending moment sequence. Quantitative fractography of regions around the origins of the cracking is being used to characterise the crack growth rates and study the interaction of adjacent cracks while the cracks remain very small. These studies have allowed a deeper understanding of the nature, type and distribution of flaws that initiate fatigue cracks representative of the cracking found in aircraft wing carry-through structure. Methods are being developed to use this data to aid in the understanding of F/A-18 component life and ways in which the life of such components can be enhanced by surface treatments or repair.

8.4.2 Corrosion Structural Impact Modeling: Khan Sharp and Graham Clark; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

Substantial progress has been made in developing a modeling process that would allow the incorporation of some types of corrosion into aircraft structural integrity management. The general effect of corrosion in terms of fatigue is greater at the lower stresses (near threshold) than the higher stresses (near yield) where fatigue cracks develop quickly. In terms of residual strength each case must be looked at on its own merits, though the calculations are relatively simple.

In particular, Equivalent Precrack Size (EPS) modeling has been explored for pitting corrosion and exfoliation corrosion, and has accurately predicted the mean remaining laboratory specimen life (within 10%). The scatter on these predictions varies with position along the "pit metric vs. EPS" curve, and the confidence limits imposed. At the smaller corrosion sizes (pit or exfoliation depth) the confidence bands are wider than at the higher corrosion sizes. The modeling process has also highlighted the criticality of the material fatigue crack growth data in obtaining EPS values (and hence in getting a good ECS vs Pit metric correlation); variability in material crack growth data can lead to large changes in the EPS value derived from crack growth life. Low kt constant amplitude laboratory specimen data has been used to predict the results of high kt CA laboratory experiments, and this data is now being used to predict crack growth lives for high kt spectrum loaded specimens. At the same time modeling is being performed on realistic coupons as well as on corroded components for a damage tolerance wing test on F-111.

Some long-term research is being conducted to determine if accelerated environmental testing is realistic. The test which will run for 1 year simulates the loading and environment of the P3C and will generate real-time crack growth data. Australia has had environmental sensors inside the P3 for at least 3 years and has collected substantial data on variation in corrosion activity throughout a mission. The data is superimposed on the load sequence generated for the P3 to represent service conditions. At the same time accelerated tests (high humidity and salt fog) are being run to compare with the long term test. Early results indicate that the fatigue and corrosion are activated at different times.

8.4.3 Reliability of Magnetic Rubber Nondestructive Inspection for Cracks in D6ac Steel: G. Hugo, C. Harding, C. Scala, H. Chin Quan; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

AMRL is conducting a program of work to determine the reliability of magnetic rubber inspections (MRI) performed on D6ac steel components on F-111. A relative unique feature of these inspections is the need to reliably detect, which a high degree of confidence, cracks smaller than 1mm in surface length and depth. This work has included an experimental program to determine statistically the probability of detection (POD) of fatigue cracks in D6ac as a function of crack size. A set of laboratory specimens containing fatigue cracks of difference sizes has been inspected by Royal Australian Air Force technicians. These inspections were conducted so as to simulate as closely as possible the conditions experienced by the technicians during in-service MRI of F-111 components. Statistical analysis of the experimental data will be completed by June 2001. The resulting POD information will be used as input to durability and damage tolerance analysis (DADTAs), which are used to determine the inspection intervals for in-service MRI of F-111 components.

8.4.4 Comparative Vacuum Monitoring (CVM); a Novel NDI Technique: D.P. Barton*, G. Wheatley and K.J. Davey - Structural Monitoring Systems Ltd. Bassenden, Western Australia.

Comparative Vacuum Monitoring (CVM) makes use of the principle that a steady-state vacuum, maintained within a small volume, is extremely sensitive to any leakage. The method requires three main components: 1) an extremely stable source of low vacuum, 2) a fluid flow meter and 3) a sensor covering, or integral within, the test object.

Sensors can either take the form of self-adhesive elastomeric "pads" with the sensing cavities moulded into the adhesive face, or the sensor cavities may form part of the structure from manufacture. The sensitivity of CVM is determined by the spacing of the cavities. SMS has also developed an application of CVM to measure surface crack propagation rates. (**Figures 13 and 14**)

CVM is gaining acceptance as a valuable new tool in fatigue testing laboratories. Earlier versions of the system were used by DSTO-AMRL as part of their F/A 18 FS488 bulkhead fatigue test program. The current laboratory system is being used by DSTO-AMRL as part of their testing program on F-111 wings, by a major aircraft manufacturer to test the integrity of lap-joints, to test crack initiation and crack growth propagation, by the US military to detect crack initiation in fatigue tests and by a German Institute to examine when the yield of punched rivet joints changes from elastic to plastic.

The system is now under development, including full environmental and field trials, to allow periodic monitoring of operational aircraft structures. Sensors would be fitted to target locations with vacuum tubing ducted to convenient inspection points. Portable equipment (comprising a vacuum source, fluid flow meter and data logger) would be used by ground-crew to determine any change in the integrity of the structure under the sensors, at a periodicity determined by the expected failure characteristics of the structures under surveillance. The value of the CVM approach is low cost, high accuracy monitoring while reducing the need for physical inspections that mostly require structural teardown for access. Field trials of CVM on operational military and civil aircraft are due to begin in the second half of 2001.

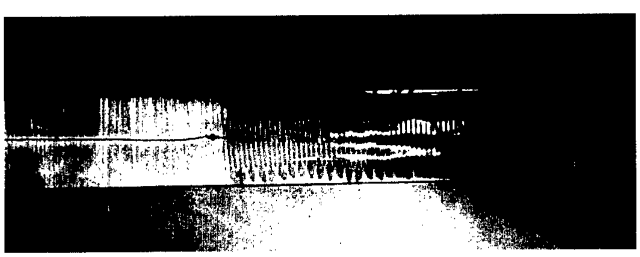


Figure 13 – Crack Propagation Sensor with an optical method

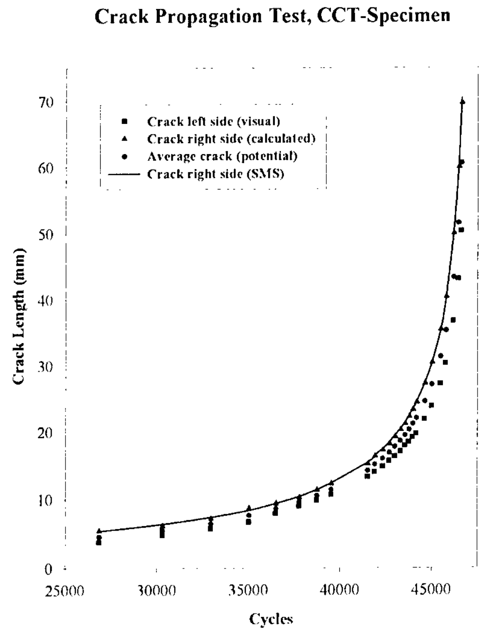


Figure 14 – Comparison of CVM (SMS)

8.4.5 Modelling The Effects Of F-111 Cold Proof Loads On Crack Growth: Simon Whitehead, Aerostructures Technologies Pty Ltd, Kevin Walker, Aeronautical and Maritime Research Laboratory, David Broek – Consultant to Aerostructures Technologies Pty Ltd.

Typically over loads applied during cold proof load testing of the RAAF F-111 fleet will introduce beneficial compressive residual stresses, although in some instances non-beneficial tension residual stresses may also be introduced. Current techniques employed in software provided by the original equipment manufacturer do not adequately model the beneficial effects. This results in shorter inspection intervals and subsequently higher maintenance costs.

The aim of this study was to develop an alternative approach to modelling the beneficial effect of over loads. Several F-111 high priority control points were used as case studies. Observations made during this study indicated that similitude concepts, in particular the use of plastic strain range as a similitude parameter [28], could be used to develop a retardation model that could adequately cope with over loads.

The retardation model (shown below) developed takes the form of multiplicative beta factors that can be used in conjunction with software provided by original equipment manufacturer. Results from this study indicated that use of this retardation model produces an increase in minimum inspection intervals while still retaining several conservative elements present in the current techniques [29].

$$\beta_p \; \frac{1-\dfrac{2}{\pi}\sqrt{\dfrac{2}{6}}\sqrt{\dfrac{r_{p_{proof}}}{r_{p_{max}}}}}{1-\dfrac{2}{\pi}\sqrt{\dfrac{2}{6}}} \quad \text{where} \quad \frac{r_{p_{proof}}}{r_{p_{max}}}=\left(\frac{\sigma_{proof}}{\sigma_{max}}\right)^2 \frac{a_{proof}}{a}$$

8.4.6 Multi-Site Fatigue Cracking In Plastically Loaded Notched 2024-T851 Aluminium Alloy: Mladen Ignjatovic, Aerostructures Technologies Pty Ltd, Kevin Walker, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The Aeronautical and Maritime Research Laboratory (AMRL) and Aerostructures provide F-111 Durability and Damage Tolerance Analysis (DADTA) support to the Royal Australian Air Force (RAAF) to ensure the continued safe operation of their F-111 fleet. Inspection intervals recommended by the original manufacturer of the F-111, Lockheed Martin Aeronautics Company (LMAC), for several fatigue critical locations are undergoing reassessment as they were found to be short and unacceptable [30, 31].

A multi-site crack initiation program in collaboration with LMAC is currently under way to develop and validate the analytical fatigue crack growth modelling capabilities for multi site cracking in 2024-T851 Aluminium Alloy material under conditions of notch plasticity. Cracking of this type can occur on the F-111 aircraft wing structure. Test data has been generated for a range of different loading conditions, all involving notch plasticity.

A Lockheed computer program called METLIFE was used to analyse this type of cracking [32, 33]. METLIFE is based on a combination of local notch strain approach (Neuber's Rule) and Linear Elastic Fracture Mechanics (LEFM). METLIFE uses the notch strain analysis to determine the effect that the plasticity will have on the stress distribution on the anticipated plane of crack growth. A Greens function method is then applied to determine the stress intensity factor and the crack growth is calculated using conventional LEFM techniques. Multiple crack initiation is being incorporated into the code, however at this stage the analysis is restricted to a single crack.

The analytical results compare very well with the test data [34]. This includes cases such as a compression dominated spectrum for which a conventional analysis would not predict any crack growth, where the testing demonstrated that crack growth would indeed occur. It was also discovered that the single crack analysis produced a conservative result, even in cases where multi site initiation behaviour was observed in the tests.

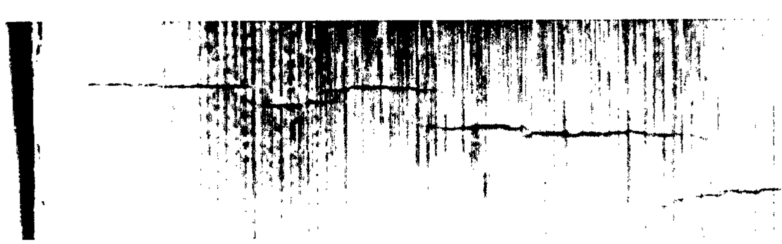


Figure 15. Crack Camera Image of Multi-Site Cracking at the Edge Notch

8.4.7 Green's Function Development for a Surface Crack in a Plate for use in the Elastic-Plastic Fatigue Crack Growth Code *METLIFE:* K. Walker and G. Swanton; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The original fleet of the Royal Australian Air Force's F-111 aircraft has been in service for some 27 years, with a planned withdrawal date of 2020. The on-going structural integrity of these aircraft is managed using the Durability and Damage Tolerance Analysis (DADTA) approach, whereby crack growth analysis studies are used to determine safe inspection intervals. Traditionally, only Linear Elastic Fracture Mechanics (LEFM) programs were available to perform the analyses. However, certain locations on the F-111 airframe have been known to experience plastic behaviour under severe loading conditions, in particular the Cold Proof Load Test, which renders LEFM inaccurate. As a result, more complex analysis tools have evolved to account for cyclic notch plasticity. One such tool is METLIFE, which was developed by Dr. Dale Ball of Lockheed Martin. The METLIFE software provides several options to calculate the Stress Intensity Factor (SIF), a parameter required to calculate crack growth rates.

The Green's Function is the one of the most accurate solution methods for calculating the SIF. Other techniques, such as handbook solutions rely upon a linear elastic applied stress, which is invalid for cases when the response is plastic. A more accurate technique in this case is the local stress/SIF solution, whereby the plastic response stress is used to generate SIF's and subsequent crack growth. However, the shortcoming here is that some accuracy is lost because the response stress, and hence SIF, is calculated at just one point on the crack tip. On the other hand, the Green's Function method uses the response stress distribution over the entire crack face to calculate the SIF at a point on the crack tip. The technique is also powerful because it can determine the SIF for arbitrary stress distributions.

Although there are many crack configurations that already have Green's Function solutions available to them, there are other scenarios that do not have this option. One such case is the surface crack in a plate, which is a flaw type relevant to some F-111 control points. Finite Element methods have been used to develop the Green's Function solution for the surface flaw in a plate, and the solution has been incorporated into the METLIFE code (see Reference 35).

8.4.8 No flaw-growth determination of impacted hat stiffened panels: S. Georgiadis and S. E. Dutton – Cooperative Research Centre –Advanced Composite Structures. Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The Cooperative Research Centre for Advanced Composite Structures (CRC-ACS) and AMRL are developing a generic approach for the substitution of ageing metallic honeycomb panels, which suffer degradation from corrosion and fatigue cracking, with stiffened composite panels. In support of the generic approach, an extensive test program is being undertaken, ranging from coupon specimens to full scale article.

A subset of the test program includes sub-component panels (250mm x 250mm) being subjected to compression and shear fatigue loading. The purpose of these tests is to determine what ratio of peak cyclic strain to static failure strain is required for no flaw growth after one million cycles. The data derived from this test will be used to design damage tolerant composite panels.

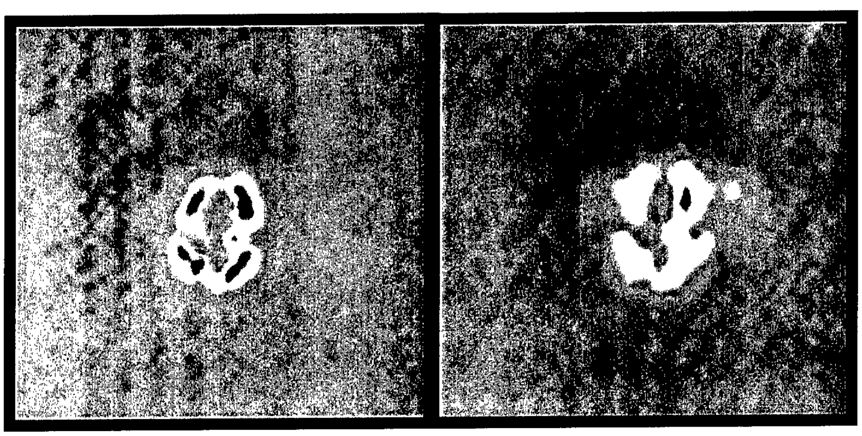


Figure 16: Thermography scans showing impact zone before and after compression-compression cyclic loading. (One million cycles, R=10, 5Hz constant amplitude loading)

8.4.9 Improved Constitutive Model for Cyclic Plastic Deformation:Weiping Hu and Chun H. Wang; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

To overcome a deficiency in the classical plasticity theory in dealing with relaxation of mean stress and strain ratchetting induced by overloads a new constitutive model has been developed for cyclic plasticity [36]. Incorporating isotropic and nonlinear kinematic hardening, the model is capable of quantifying the transient response of materials under asymmetric cyclic loading as well as the steady-state response. Multiple back stresses have been introduced to the original Armstrong-Fredrick type of model to extend the strain range of applicability [37]; a material constant-switching method has been devised to address the issue of transient cyclic softening exhibited by 7050 aluminium [38]; and a weight function has been introduced into the dynamic recovery term to improve the prediction of long term ratchetting rate [39, 40]. The model has been validated against experimental data for 7050 aluminium, as shown in the enclosed figure, and D6ac steel, with a very good agreement between the experimental and numerical results.

Numerical procedures have also been developed and implemented in a finite element code, ABAQUS, for the analysis of complex structures subjected to general multiaxial loading. This new capability has been applied to analyse the residual stress distribution near a fuel ventilation hole in the upper wing pivot fitting of F-111 aircraft, subjected to cold-proof load test sequence. Good correlation has been found between the experimental data and the numerical predictions. In addition, the proposed numerical methods have been applied to analyse notch plasticity in conjunction with the Neuber rule. Such an improvement has been implemented in a safe life prediction software (CI-89) developed by Boeing.

The development of the constitutive model and its various implementations add a new capability in the accurate prediction of cyclic stress-strain response in complex structures subjected to general loading sequences including occasional overloads.

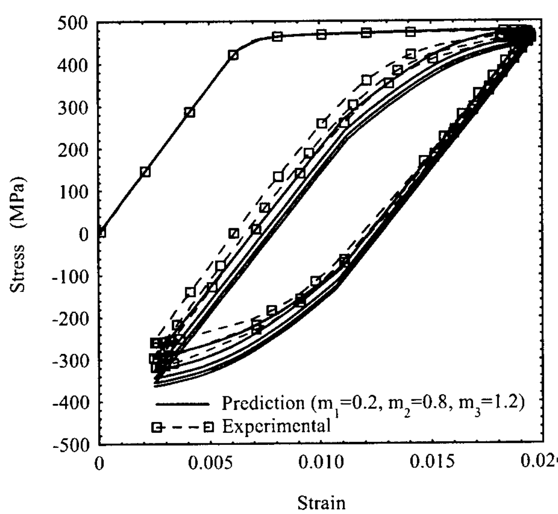
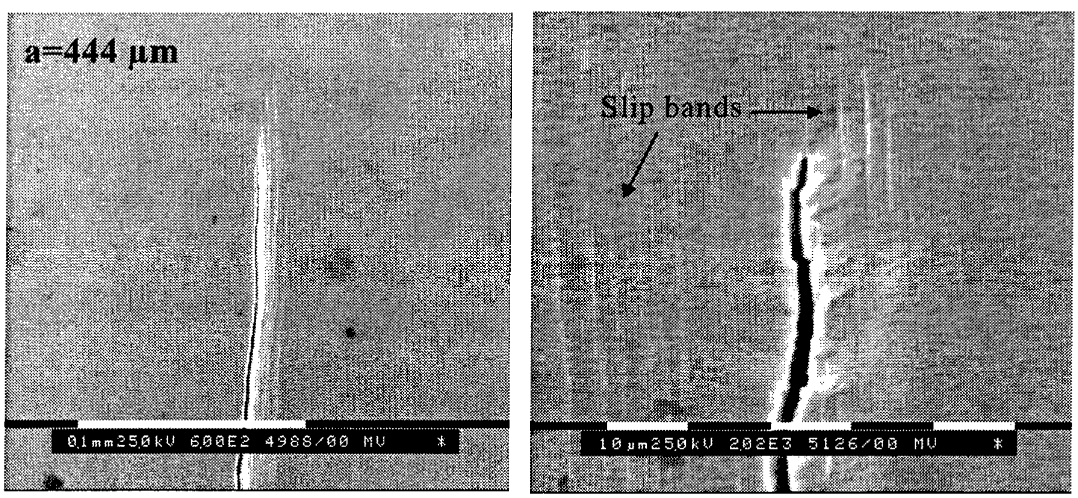


Figure 17 Comparison of experimental and numerical hysteresis loops for 7050 aluminium under asymmetric (strain-controlled) cyclic loading, illustrating mean stress relaxation.

8.4.10 Microstructurally Short Fatigue Crack Growth In Nickel-Based Single Crystal Superalloy SC16: X.P. Zhang[1], C.H. Wang[2], L. Ye[1] and Y.-W. Mai[1]

[1]Centre of Expertise in Damage Mechanics, School of Aerospace, Mechanical and Mechatronic Engineering, the University of Sydney, NSW 2006, Australia [2]Aeronautical and Maritime Research Laboratory, Melbourne, Australia

Recently, there is an acute demand to understand the growth characteristics of short cracks in single crystal alloys, since lifetime prediction and failure assessment based on *'retirement for cause'* (RFC) methodology for some important components or structures made of directionally solidified or single crystals, such as aero-engines and turbines, depend upon their fatigue performance in single crystal structures. The RFC methodology utilises techniques of damage tolerant design, taking account of crack initiation life and allowing for a certain crack growth within a critical scale. However, while there have been many studies on the crack growth of polycrystalline metallic materials, relatively fewer investigations have been carried out on the short fatigue crack growth behaviour in single crystal structural alloys. In addition, existing results are mainly for pure single crystal metals, such as copper and nickel, studying the short crack growth behaviour in a single crystal structural alloy would allow us to examine in detail the interaction between short crack and crystallographic orientation. The present work investigated the microstructurally short crack growth in a single crystal SC16 superalloy and identified three major mechanisms, when the single crystal alloy specimen is loaded in the axis nearly perpendicular to its main shear plane. Firstly, there does exist the so-called anomaly of short fatigue crack growth in the single crystal superalloy, similar to poly-crystal alloys. Secondly, fatigue crack growth the single crystal superalloy exhibited less deflected crack path and fairly flat fracture surfaces, compared to poly-crystal alloys. This implies that the dominant crack closure mechanism is the plasticity induced crack closure. Finally, short fatigue crack propagated dominantly along slip bands, which was nearly perpendicular to the globe mode I loading direction or the [001] crystallographic orientation.



(a) crack growing perpendicularly to [001] **(b) slip bands in both sides of crack**

Figure 18 Short crack propagation in single crystal superalloy SC16 from an initial length of 36 μm

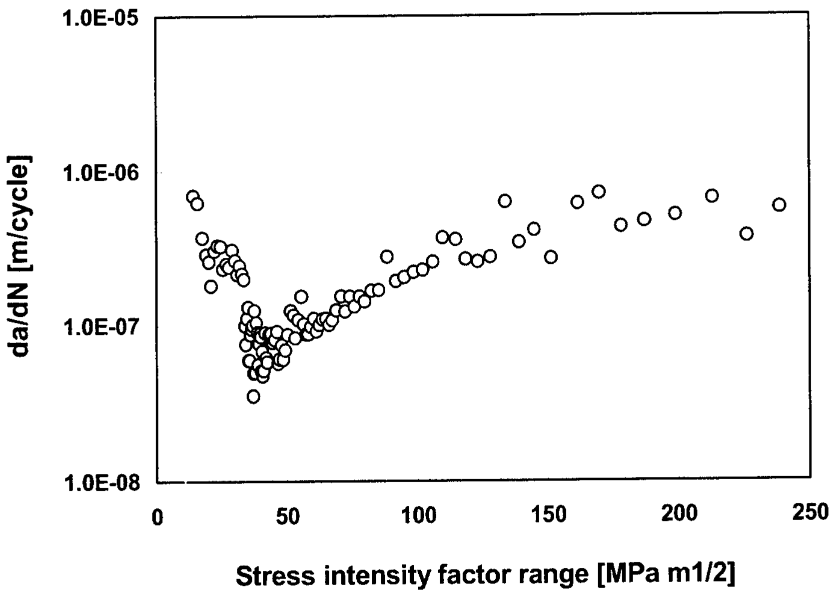


Figure 19: Crack growth rate versus stress intensity factor range, ΔK

8.4.11 Three-Dimensional Solutions for an Elastic Plate with a Deep Notch or Crack: A. Kotousov – COE, Monash University and C. H. Wang; Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

The cross-sectional thickness of plate-like structures has been found to have a strong influence on the propagation rates of fatigue cracks and hence the structure's fatigue life [41, 42]. This effect is especially prominent for fatigue cracks emanating from notch plastic zone where the crack growth driving force depends strongly on the plate thickness. However, due to the complexity of three-dimensional stress-state at notch root, conventional engineering practices are to treat the problem either as plane-stress or plane strain. Although such an approach would normally give satisfactory solutions of the in-plane stresses under elastic conditions, there is a significant difference between the plane-stress and the plane-strain solutions when plastic deformation occurs at the notch root. This difference arises due to the significant through-thickness stress under plane-strain conditions. In this case, it is important to account for the effect of the plate thickness on this through-thickness stress.

Analytical solutions of the three-dimensional stress constraint ahead of blunt V-notches and a crack in plates of arbitrary thickness have been obtained. By adopting the generalised plane-strain theory [43], which assumes that the through-the-thickness extensional strain is uniform in the thickness direction, closed-form solutions have been obtained for the stress constraint factor. It is shown that for deep notches the opening angle only a minor effect on the through-thickness stresses, which are tensile when the structure is subjected to tension. In the context of thick composite structures, these tensile stresses may have a significant implication for potential delaminations around cut-outs in thick composite structures. The obtained theoretical solutions compare very well with finite element results (see Figure 20).

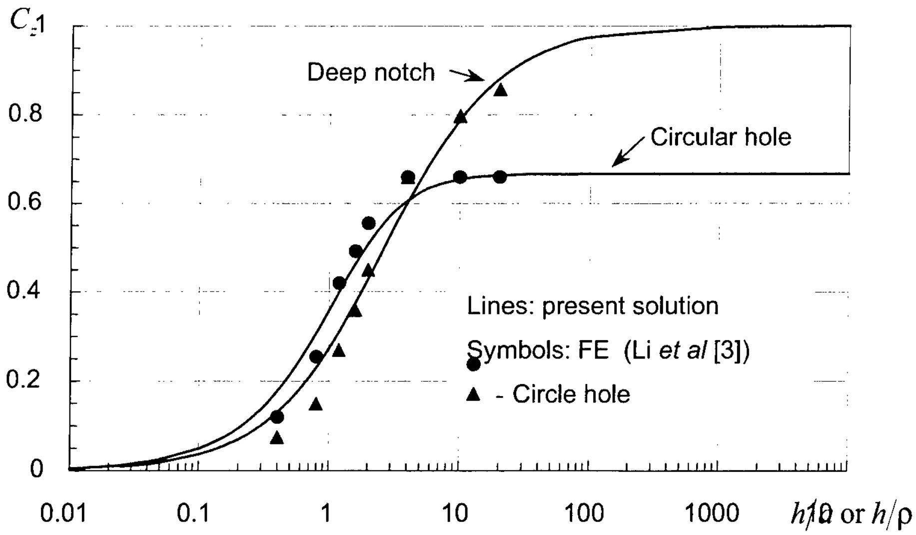


Figure 20: The maximum constraint factor for the deep notch with zero opening angle and circular hole in an infinite plate under uniaxial tension as a function of the thickness to radius ratios (Poisson's ratio $\nu = 0.3$)

8.4.12 Stress Intensity Factor for Small-to-Medium Cracks: Application to Shape Optimisation Problems; A. Kotousov and R. Jones – COE, Monash University, Melbourne, Australia.

A new and relatively simple engineering method for calculating the stress intensity factors for small-to-medium cracks emanating from a notch under arbitrary loading has been obtained. The formulation can be used in calculating the fatigue life of notched components as well as in the shape optimisation problems with durability constraints [44, 45].

As an example, let us to consider the problem of shape optimisation of a hole in an infinite plate subjected to arbitrary remote bi-axial loading. In this case a stress based optimisation, which minimises the maximum stress in the body, predicts the optimal shape to be an ellipse with the ratio of principle axis equal to the ratio of the principal remote stresses Cherapanov [46].

In this study we will limit the possible (hole) shapes to be elliptical and assume that all cracks are normal to the edge of the hole. Within this design space we will optimise the shape of the hole so as, for a range of given crack sizes, to have the maximum residual strength the resultant optimal shapes, as a function of the size l of the (pre-existing) edge crack, is shown in Figure 21 for various rations of principle remote stresses. Here ρ is the minimum radius curvature of the optimised hole.

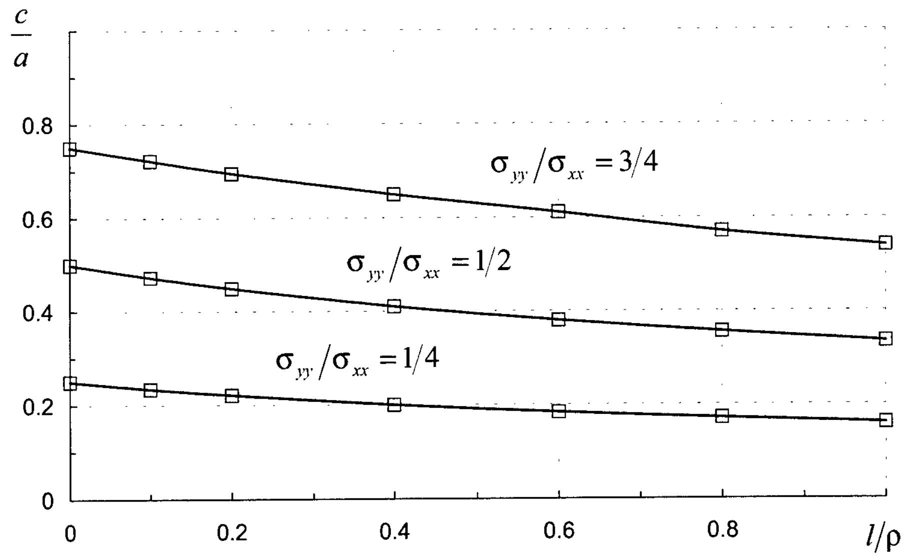


Figure 21: The optimal (fracture strength) shape as a function of the length of the pre-existing edge crack for various ratio's of the principle remote stresses. Here a/c is the ratio of half-axis of the ellipse and l/ρ is the non-dimensional length of the pre-existing crack.

8.4.13 A Combined Model of Short Crack Growth Accounting for Both Plasticity- and Roughness-Induced Crack Closures: X.P. Zhang[1], C.H. Wang[2], J.C. Li[3], L. Ye[1] and Y.-W. Mai[1]

[1]Centre of Expertise in Damage Mechanics, School of Aerospace, Mechanical and Mechatronic Engineering, the University of Sydney, Australia [2]Aeronautical and Maritime Research Laboratory, Melbourne, Australia [3] Department of Civil Engineering, University of Technology Sydney, Australia.

A short fatigue crack growth model should include both mechanisms of plasticity- and roughness-induced crack closures. Based on theoretical analysis and numerical modelling, a new crack growth model was developed to characterise short fatigue crack growth behaviour, accounting for both mechanisms of crack tip plasticity- and fracture surface roughness-induced crack closures. The results of crack closure and effective crack driving force simulated by the model agreed well with test data available in the literature and those analytic solutions obtained using Budiansky-Hutchinson's complex function analysis model. In addition, the results of short fatigue crack growth rate predicted using the model correlate well with the experimental data, and show reasonable improvement compared to Newman's plasticity-induced closure model.

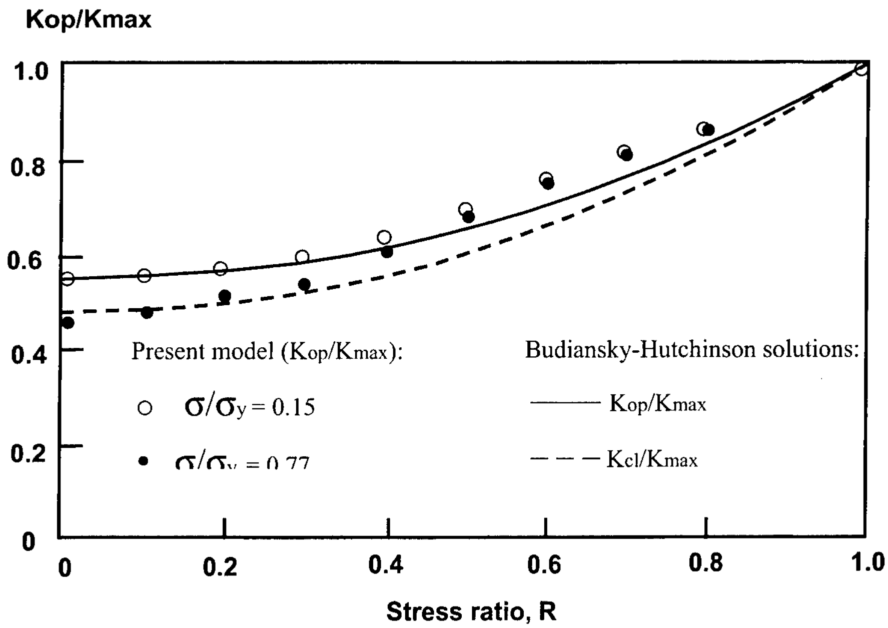


Figure 22: Comparisons of crack closure between the present numerical results and analytic solutions based on B-H analyses (both for plane stress condition)

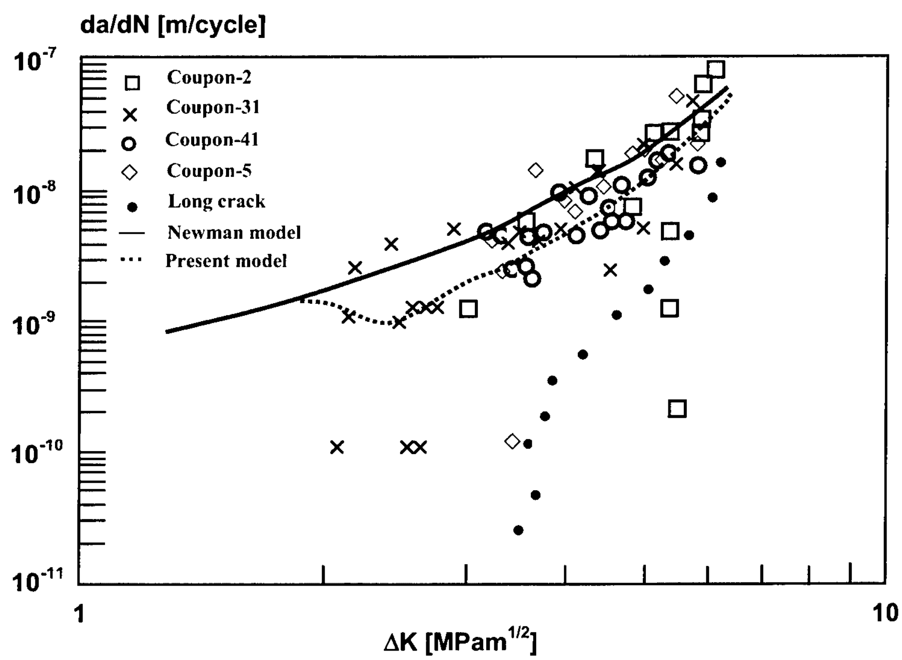
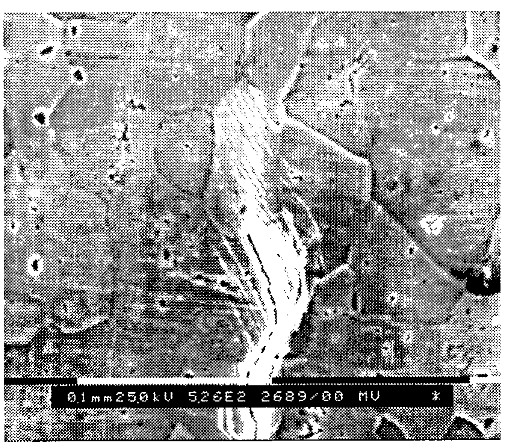


Figure 23: Short crack growth rate for 2024-T3 Al under constant amplitude (σ_m=78MPa), and comparisons with the present model and Newman model

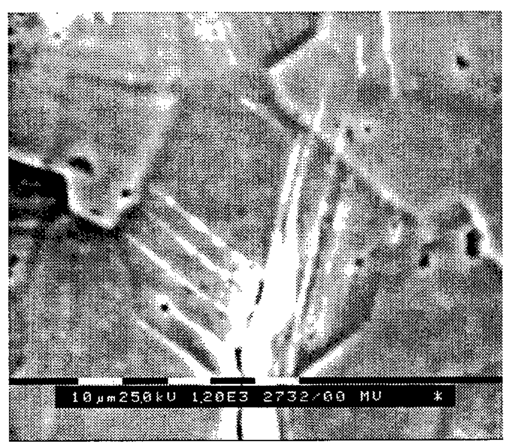
8.4.14 Growth Behaviour of Microstructurally Short Fatigue Cracks In 2024-T351Al By *In-Situ* SEM Fatigue Testing: X.P. Zhang[1], C.H. Wang[2], L. Ye[1] and Y.-W. Mai[1]

[1]Centre of Expertise in Damage Mechanics, School of Aerospace, Mechanical and Mechatronic Engineering, the University of Sydney, NSW 2006, Australia. [2]Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

For short fatigue crack propagation, it is especially important to characterise the microstructural influences on short crack growth behaviour, as this interaction strongly affects the plastic deformation at the crack tip. One most important aspect in material microstructures is the mismatch in crystallographical orientations across a grain boundary. The grain-boundary blocking effect has also been recognised as a major factor causing the anomaly of short fatigue crack. The present work studied the effects of grain boundary blocking and the micro-mechanisms of short fatigue crack growth in a poly-crystal 2024-T351 aluminium alloy (length varying from a few microns to tens of microns) by *in-situ* SEM fatigue testing. We are able to determine accurately the crack growth rates as the crack approaches and leaves the grain boundaries; and characterise the slip-band and boundary dependence of crack growth behaviour. *In-situ* SEM observations of short fatigue crack growth in the poly-crystalline aluminium alloy revealed that fatigue cracks may grow in a shear decohesion mode over a length of several times the grain size, far beyond the conventional stage I regime. Fatigue cracks were found to continue to grow along a single shear band even after two mutually perpendicular shear bands had formed at the crack tip (Figure 24). This type of fatigue crack growth poses a challenge to the current short fatigue crack growth models based on crack-tip opening displacement. In particular, it has been found that the cyclic crack-tip opening displacement, which accounts for both large-scale yielding and the plasticity-induced crack closure, is unable to unify the growth rates of short and long cracks in the 2024-T351 aluminium alloy (Figure 25).



(a) crack growing along a single shear band



(b) crack growing along shear band

Figure 24: Shear decohesion along shear band in 2024-T351Al with crack length 750 microns

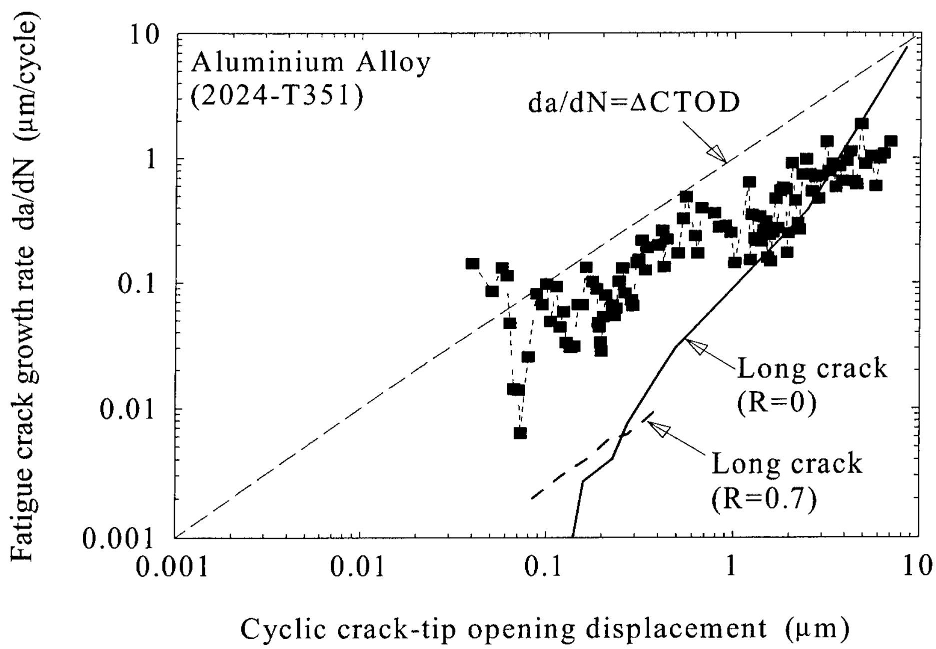


Figure 25: Fatigue crack growth rates against crack-tip opening displacement

8.4.15 Thermographic Inspection Methods: Steven Lamb, Aeronautical And Maritime Research Laboratory, Melbourne, Australia.

Acquisition of a high-speed Thermal Imaging System based on a state-of-the-art Raytheon IR camera, and a custom data acquisition and analysis interface, provides DSTO AMRL with one of the most advanced inspection systems currently available in Australia. The system has the potential to assist the ADF with difficult inspection problems such as the assessment of F111 metallic honeycomb components, and the detection/assessment of corrosion damage. The system has the particular advantage of providing for single-sided inspection at high speed in real time. Other potential applications for the system currently include the assessment of residual stresses and crack detection/assessment.

The new system has been successfully used to inspect a range of specimens including metallic and non-metallic honeycomb cored components, corrosion damaged components, and composite components. Successful trials of the new system have also been undertaken at RAAF Bases Amberley (Qld) and Williamtown (NSW) in support of the ADF acquisition of a thermographic inspection system.

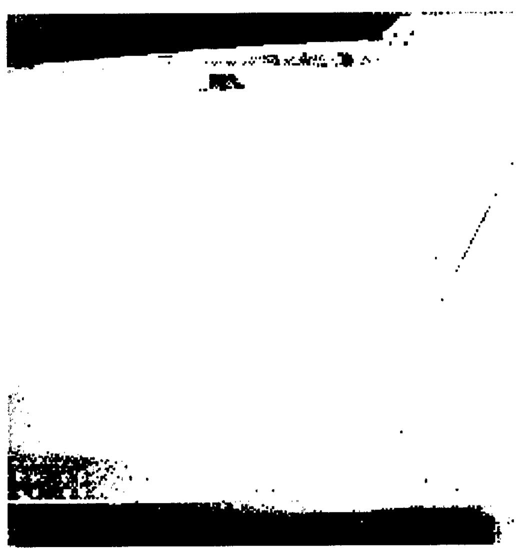


Figure 26: Thermogram of horizontal stabilator showing internal structural detail

8.5 REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN NEW ZEALAND

8.5.1 Parametric Sensitivities Affecting the Calculated Fatigue Life of a Fleet of Military Aircraft; Stephen Campbell, Defence Technology Agency (DTA), Auckland, New Zealand.

Recently the Royal New Zealand Air Force began a Life of Type Study (LOTS) for their fleet of Skyhawk aircraft. This study indicated that the most likely fatigue life limiting location of the Skyhawk wing was a 2-inch diameter hole in the intermediate wing spar. The critical hole encroaches the fillet radius between the spar shear web and cap. A detailed Finite Element stress analysis of this hole indicated a severe stress concentration (Kt 3.6).

RNZAF spars were a manufactured from Alloy 7075. Some were in T6 condition, others were in the T73 condition and some were thought to be have 7075-T6 spars that had been re-treated T73 conditions. Due to uncertainties associated with the different materials, RNZAF usage spectrum and other factors, a parametric sensitivity study of the fatigue critical location was conducted.

The sensitivity study involved numerical simulations to predict the fatigue life of the spar. The predictions were made using Neuber's rule, mean strain correction, rainflow counting and a linear damage summation model. The simulations investigated the sensitivity of calculated fatigue life to a number of input parameters including stress concentration, usage spectrum, cycle order and material properties. Significant variations in the predicted fatigue life were observed for relatively small changes in the input parameters.

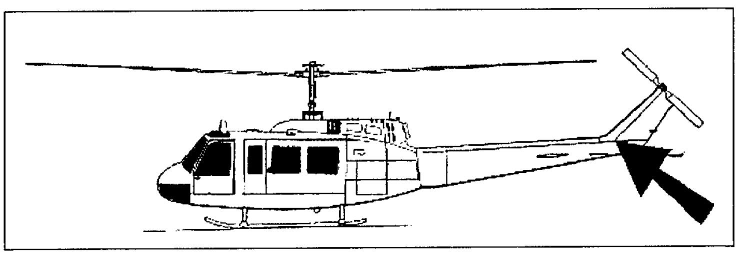
The observed sensitivity of the critical location to small changes in material properties, such as the stress-strain and strain-life curves, had a number of consequences. Firstly, it indicated that in some conditions the use of materials data derived from commonly used equations such as, Ramberg-Osgood and Coffin-Manson, can cause large variations in predicted fatigue life when compared predictions made using experimentally developed data. It also indicated that the statistics of the curve fit used to produce experimentally developed curves must be considered when predicting the fatigue life of the critical component using a strain life based method.

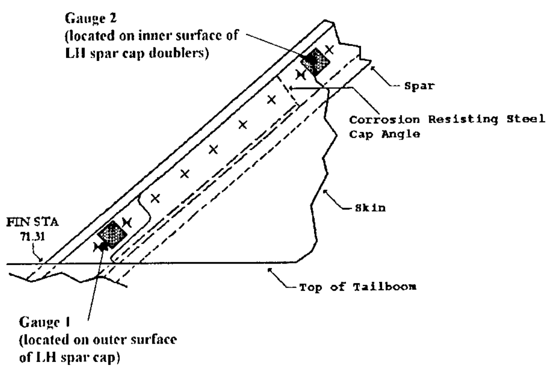
8.5.2 UH-1H (IROQUOIS) Fin Spar Wfd: Flight Strain Measurement and Fatigue Crack Growth Analysis Fraser J. McMaster, Alan D James, and Patrick C Conor Defence Technology Agency (DTA), Auckland, New Zealand.

Iroquois crashes in the USA, and subsequent Australian studies of fin spar fatigue damage rates raised concerns about the residual life of the vertical fin spar of the Royal New Zealand Air Force (RNZAF) UH-1H Iroquois fleet. Analysis in New Zealand indicated that the fin spars had exceeded their safe life and therefore flight safety would have to be maintained using a safety by inspection approach. This could have severe impact on availability and cost. There was also no published justification for the fin spar inspection intervals, and underlying assumptions about operational load spectra were under.

A flight load survey was recommended as a means of checking the validity of defined inspection intervals. An Iroquois fin spar was subsequently instrumented and in-flight strain data provided 11 flight hours of valid data. A simplified damage tolerance analysis (DTA) of the fatigue critical location (FCL) in the fin spar was performed to establish the most critical flight manoeuvre measured in the available flight strain data. The subsequent crack growth analysis performed using the flight data recorded from the instrumented Iroquois isolated a critical flight manoeuvre, which effectively decreased the crack growth life of the fin spar when compared to the crack growth life under a flight sequence that included all flight survey data. The most damaging condition involved flight at airspeeds greater than 110kts at full all up weight. Continuous operations involving this flight condition had the ability to increase the crack growth by a factor of 50 greater than the average crack growth rate using all flight spectrum data.

The initial flight load survey and damage tolerance assessment indicated that RNZAF Iroquois airframes that operate a well spread mission mix, will not suffer from rapid tail fin fatigue cracking and that existing inspection intervals are likely to be adequate. However, the flight load data also indicates that rapid fin spar crack growth can occur in helicopters exposed to extended periods of high speed flying. In this situation crack growth rates may be such that existing inspection procedures may not ensure that fatigue cracks are detected before they become critical. It was concluded that early replacement of the tail fins is highly desirable.





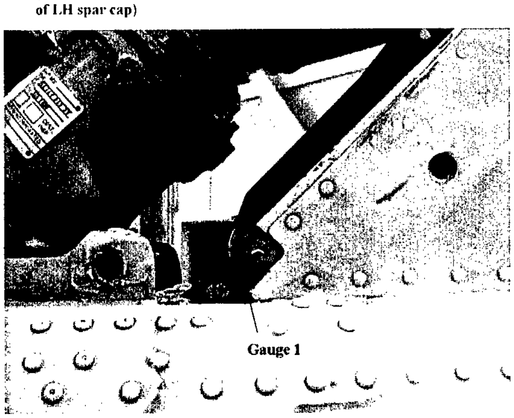


Figure 27 IROQUOIS Fin Spar Strain Measurements

References

1. S. Barter, L. Molent, K. Sharp & G. Clark, "Repair and Life Assessment of Critical Fatigue Damaged Aluminium Alloy Structure Using a Peening Rework Method", proc of USAF Structural Integrity Program Conference, San Antonio, TX, 5-7 Dec 2000.

2. Dickinson, T. and Molent, L., "Managing F/A-18 Fatigue Damage Using Operational Parameters", DSTO-TR-0985, May 2000.

3. Molent, L., White, P. and Harman, A., "Bounding Structural Durability Due to Buffet Induced Loading", proc. Of. 9th Aust. Intern. Aerospace Congress, Canberra, 4-8 March 2001.

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. Walker, K., Marsden, P., and Weller, S., "Development and Validation of an Analytical Model to Predict Fatigue Crack Growth in Notch Plastic Fields" presented at the 9th Australian International Aerospace Congress, Canberra, March 2001

6. Braemar, R., Walley, A. and Vanderkruk, D., "An Analytical and Test Program to Model Crack Growth Through a residual Tensile Stress Field in the F-111 Wing Pivot Fitting" presented at the 9th Australian International Aerospace Congress, Canberra, March 2001

7. White Paper, Defence 2000 – Our Future Defence Force, Commonwealth of Australia 2000 ISBN0642295441

8. Aerostructures Technologies Pty Ltd Report, ER-DHC4-51-ASM210, Caribou SBI Programme Stage 1 – Selection Of Critical Locations, 15 September 2000

9. Aerostructures Technologies Pty Ltd Report, ER-DHC4-51-ASM215, Caribou SBI Programme Stage 2 – Development Of Stress Spectra And DTA Criteria, 17 January 2001

10. Aerostructures Technologies Pty Ltd Report, Er-Dhc4-51-Asm236, Caribou Sbi Programme Stage 4 –Damage Tolerance Analysis And Verification, (Not Yet Issued)

11. Aerostructures Technologies Pty Ltd Report, ER-P3-57-ASM207, P-3 Preliminary Damage Tolerance Analysis, Revision 1, 24 July 2000

12. Aerostructures Technologies Pty Ltd Report, ER-P3-51-ASM188, P-3 Structural Life Monitoring Program Development and Usage Analysis, Revision 1, 27 June 2000

13. Aerostructures Technologies Pty Ltd Report, ER-P3C-51-142, RAAF P-3C Mission Profile Review and Spectrum Development – Part 1, Revision 3, 1 February 2000

14. Aerostructures Technologies Pty Ltd Report, ER-P3-51-167, An Improved Structural Life Monitoring Programme for RAAF P-3 Aircraft, Revision 2, 30 March 2000

15. Aerostructures Technologies Pty Ltd Report, ER-P3-51-ASM188, P-3 Structural Life Monitoring Program Development and Usage Analysis, Revision 1, 27 June 2000

16. Houston, M.I., The Future of Structural Life Monitoring for the RAAF P-3 Orion, Presented to Australian International Aerospace Congress 2001, and submitted to Aging Aircraft Conference 2001

17. Aerostructures Technologies Pty Ltd Report, ER-P3C-51-129, Installation of Structural data Recording Sets on P-3 Aircraft A09-756 and A09-662 – Design Report, Revision 2, 7 April 2000

18. Jackson P., Agius J. & Sexton D., "Structural Airworthiness and Cost of Ownership Implications of Continued Operation of RAAF B707 Aircraft Final Report August 1999, DSTO-TN-0267

19. Jackson P., "Structural Risk Assessment of Lower Wing Skin Panels of RAAF B707 Aircraft, to be presented at ICAF 2001, Toulouse, France, June 2001

20. Heller, M., McDonald, M., Burchill, M., and Watters, K.C., (2001) 'Shape Optimisation of Critical Stiffener Runouts in the F-111 Wing Pivot Fitting', DSTO-TR-1120, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

21. Heller, M., Burchill, M., McDonald, M. and Watters, K.C., (2001) 'Shape Optimisation of Critical Fuel Flow Vent Holes in the F-111 Wing Pivot' Fitting, DSTO-TR-1119, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

22. McDonald, M., Heller, M., Goldstraw, M. and Hew, A., (2001) 'Robustness of the F-111 Wing Pivot Fitting Optimal Rework Shapes', DSTO-TR-1121, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

23. Waldman, W., Heller, M. and Chen, G., (2000) 'Optimal free-form fillet shapes for tension and bending' (accepted for publication in the Journal of Fatigue).

24. Kaye, R., and Heller, M., (2001) 'Shape optimisation of bonded lap joints and crack repairs' (accepted for publication in International Journal of Adhesion and Adhesives).

25. Kaye, R., and Heller, M. (2000) 'Investigation of shape optimisation for the design of life extension options for an F/A-18 airframe FS 470 bulkhead', Journal of Strain Analysis, 35(6) 493–505.

26. D.W. Kelly and M. Elsley. (1995) A Procedure for Determining Load Paths in Elastic Continua. Engineering Computations, Vol. 12, pp. 415–424, 1995.

27. Waldman, W., Heller, M., Kaye, K. and Rose, L.R.F., (1999) 'Advances in structural loadflow visualisation and applications to optimal shapes', DSTO-RR-0166, Aeronautical and Maritime Research Laboratory, Melbourne, Australia.

28. Broek, D., A Similitude Criterion for Fatigue Crack Growth Modelling, Fracture Mechanics: Sixteenth Symposium, pp 347 – 360

29. Aerostructures Technologies Pty Ltd Report ER-F111-51-APM099, Whitehead S., Broek D., Modelling the Effects of F111 Cold Proof Loads, Revison 1, 31 May 2000

30. Anon., RAAF F-111C Durability and Damage Tolerance Analysis Results, FZS-12-5034, Lockheed Martin Corporation, USA, 15 September 1998

31. Anon., RAAF F-111C Durability and Damage Tolerance Analysis Details, FZS-12-5035, Lockheed Martin Corporation, USA, 15 September 1998

32. Ball D.L., Metallic Life Analysis Program: METLIFE V1.3 User's Guide, MR(FF)-1022, Revision C, Volume 1, Lockheed Martin Corporation, USA, 31 December 1999

33. Ball D.L., Metallic Life Analysis Program: METLIFE V1.3 Technical Approach, MR(FF)-1022, Revision C, Volume 2, Lockheed Martin Corporation, USA, 31 December 1999

34. Ignjatovic, M., Fatigue Testing and Analysis of 2024-T851 Edge Notch Specimens Under Conditions of Constant Amplitude and Spectrum Loading, Notch Plasticity and Multi-Site Crack Initiation, Aerostructures Technologies Engineering Report, in preparation

35. Swanton, G., and Ball, D., "Green's Function Development for a Surface Crack in a Plate for use in the Elastic-Plastic Fatigue Crack Growth Code METLIFE" presented at the 9th Australian International Aerospace Congress, Canberra, March 2001.

36. Hu, W., Wang, C. H. and Barter, S. (1999) Analysis of cyclic mean stress relaxation and strain ratchetting behaviour of aluminium 7050, *DSTO-RR-0153*.

37. Sawyer, J., Wang, C. H., Jones, R. (2000) An implicit algorithm using explicit correctors for the kinematic hardening model with multiple back stresses, *International Journal of Numerical Methods in Engineering*, Vol. **50**, 2093-2107.

38. Hu, W., Sawyer, J. and Wang, C. H. (2001) An integration scheme and its implementation for a class of constitutive models with weighted dynamic recovery, *Proceedings of Worley Australasian Finite Element Conference.*

39. 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.

40. Hu, W. and Wang, C. H. (2001) The implementation of a constitutive model with weighted dynamic recovery and its application, submitted to *International Journal of Numerical Methods in Engineering.*

41. Guo, W., Wang, C. H., and Rose, L. R. F. (1999) The influence of cross-sectional thickness on fatigue crack growth, *Fatigue and Fracture of Engineering Materials and Structures*, Vol. **22**, 437-444.

42. Kotousov, A. and Wang, C. H. (2001) Three-dimensional stress constraint in an elastic plate with a notch, *Int. J. Solids and Structures* (in press).

43. Li, Z. and Guo, W. (2001) Three dimensional elastic stress fields ahead of blunt V notches in finite thickness plate, *International Journal of Fracture*, Vol. **107**, 53-71

44. Kotousov, A. and Jones, R. (2001) Stress intensity factor for small-to-medium cracks: Application to shape optimisation problems, *Engineering Fracture Mechanics* (in press)

45. Chaperon, P., Jones, R., Heller, M., Pitt, S. and Rose, F. (2000) A methodology for structural optimisation with damage tolerance constraints. *Engineering Failure Analysis*, Vol. **7**, 281-300

46. Cherapanov, G. (1974) Inverse problems of the plane theory of elasticity. *PMM*, Vol. **38**, 963-979.

DISTRIBUTION LIST

A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1999 to March 2001

Colin Martin

AUSTRALIA

DEFENCE ORGANISATION

Task Sponsor
AED
S&T Program
Chief Defence Scientist
FAS Science Policy
AS Science Corporate Management
Director General Science Policy Development
} shared copy
Counsellor Defence Science, London (Doc Data Sheet)
Counsellor Defence Science, Washington (Doc Data Sheet)
Scientific Adviser to MRDC Thailand (Doc Data Sheet)
Scientific Adviser Policy and Command
Navy Scientific Adviser (Doc Data Sheet and distribution list only)
Scientific Adviser - Army (Doc Data Sheet and distribution list only)
Air Force Scientific Adviser
Director Trials

Aeronautical and Maritime Research Laboratory
Director
Chief of Airframes and Engines Division
Research Leader Structural Integrity: J. Sparrow

P.White
KC. Watters
G. Hugo
M. Stimson
I. Anderson
M. Heller
W. Waldman
LR.F. Rose
P. Jackson
C.M. Scala
G. Forsyth
K.F. Fraser
S. Barter
R.J. Callinan
R, Kaye
S. Burke
C.A. Martin (for ICAF distribution -212 copies)

M. Burchill
R.L. Evans
B.R.W. Hinton
S.C. Galea
L. Molent
D.C. Lombardo
A.A. Baker
G. Clark
K. Walker
G. Swanton
N.T. Goldsmith
P.K. Sharp
C.H. Wang
S.R. Lamb
N. Athiniotis

DSTO Library and Archives
Library Fishermans Bend (Doc Data Sheet)
Library Maribyrnong (Doc Data Sheet)
Library Salisbury
Australian Archives
Library, MOD, Pyrmont (Doc Data sheet only)
US Defense Technical Information Center, 2 copies
UK Defence Research Information Centre, 2 copies
Canada Defence Scientific Information Service, 1 copy
NZ Defence Information Centre, 1 copy

National Library of Australia, 1 copy

Capability Systems Staff

Director General Maritime Development (Doc Data Sheet only)
Director General Aerospace Development (Doc Data Sheet only)

Knowledge Staff

Director General Command, Control, Communications and Computers (DGC4) (Doc Data Sheet only)
Director General Intelligence, Surveillance, Reconnaissance, and Electronic Warfare (DGISREW)R1-3-A142 CANBERRA ACT 2600 (Doc Data Sheet only)
Director General Defence Knowledge Improvement Team (DGDKNIT) R1-5-A165, CANBERRA ACT 2600 (Doc Data Sheet only)

Army

Stuart Schnaars, ABCA Standardisation Officer, Tobruck Barracks, Puckapunyal, 3662(4 copies)
SO (Science), Deployable Joint Force Headquarters (DJFHQ) (L), MILPO Gallipoli Barracks, Enoggera QLD 4052 (Doc Data Sheet only)
NPOC QWG Engineer NBCD Combat Development Wing, Tobruk Barracks, Puckapunyal, 3662 (Doc Data Sheet relating to NBCD matters only)

Air Force

Aircraft Research and Development Unit
Tech Reports,CO Engineering Squadron,ARDU

DGTA
DAIRENG
ASI-DGTA
RAAF College, Point Cook
OIC ATF, ATS, RAAFSTT, WAGGA (2 copies)

Intelligence Program

DGSTA Defence Intelligence Organisation
Manager, Information Centre, Defence Intelligence Organisation

Corporate Support Program

Library Manager, DLS-Canberra

OTHER GOVERNMENT DEPARTMENTS AND AGENCIES

Civil Aviation Safety Authority
Library
S. Swift
M. Richmond
S. Emery
R. Fischer

UNIVERSITIES AND COLLEGES

Australian Defence Force Academy
Library
Head of Aerospace and Mechanical Engineering
Flinders
Library

Melbourne
Engineering Library
Dr J.P. Williams
D. Barter
Structural Monitoring Systems

Monash
Hargrave Library
Prof R. Jones
Dr Y.C. Lam
J. Price

Newcastle
Library
Prof R. Telfer, Institute of Aviation
Professor R. Melcher

New England
Library

Sydney
Engineering Library
X. Yu

NSW
Physical Sciences Library

Queensland
Library

Tasmania
Engineering Library

Western Australia
Library

RMIT
Library
A/Prof. M.L Scott

University College of the Northern Territory
Library

OTHER ORGANISATIONS

NASA (Canberra)
AusInfo
AeroSpace Technologies Australia
Manager
Library
Ansett Airlines of Australia
Library
J.H. Bibo
Aerostructures Australia
Mr R. Locket (3)
Mr B. Madley (3)
Qantas Airways Limited
Library
Hawker de Havilland Aust Pty Ltd, Victoria
Library
Hawker de Havilland Aust Pty Ltd, Bankstown
Library
Jayman Scinfo Services
J.Y. Mann
Cooperative Research Centre - Aerospace Structures
Dr I. Mair (2)

NEW ZEALAND

Defence Operational Technical Support Establishment
Library
P.C. Conor
W.L. Price
R.R. Aitken
S.K. Campbell
G. Rapley

RNZAF
Transport Ministry, Airworthiness Branch, Library

Universities

Canterbury
Library

Auckland
Library
Dr G.D. Mallinson
N F Ellis
A D James
D R W Horrigan

Wellington
Library

OUTSIDE AUSTRALIA

ABSTRACTING AND INFORMATION ORGANISATIONS
Library, Chemical Abstracts Reference Service
Engineering Societies Library, US
Materials Information, Cambridge Scientific Abstracts, US
Documents Librarian, The Center for Research Libraries, US

INFORMATION EXCHANGE AGREEMENT PARTNERS
Acquisitions Unit, Science Reference and Information Service, UK
Library - Exchange Desk, National Institute of Standards and Technology, US
National Aerospace Laboratory, Japan (
National Aerospace Laboratory, Netherlands

SPARES (5 copies)

Total number of copies: **352**

Page classification: UNCLASSIFIED

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION
DOCUMENT CONTROL DATA

1. PRIVACY MARKING/CAVEAT (OF DOCUMENT)

2. TITLE A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1999 to March 2001	3. SECURITY CLASSIFICATION (FOR UNCLASSIFIED REPORTS THAT ARE LIMITED RELEASE USE (L) NEXT TO DOCUMENT CLASSIFICATION) Document (U) Title (U) Abstract (U)
4. AUTHOR(S) Colin Martin	5. CORPORATE AUTHOR Aeronautical and Maritime Research Laboratory 506 Lorimer St Fishermans Bend Victoria 3207 Australia

6a. DSTO NUMBER	6b. AR NUMBER	6c. TYPE OF REPORT	7. DOCUMENT DATE
DSTO-TN-0361	AR-011-879	Technical Note	May 2001

8. FILE NUMBER	9. TASK NUMBER	10. TASK SPONSOR	11. NO. OF PAGES	12. NO. OF REFERENCES
M1/9/974	AE/MGT	AED	50	46

13. URL on the World Wide	14. RELEASE AUTHORITY
http://www.dsto.defence.gov.au/corporate/reports/DSTO-TN-0361.pdf	Chief, Airframes and Engines Division

15. SECONDARY RELEASE STATEMENT OF THIS DOCUMENT

Approved for public release

OVERSEAS ENQUIRIES OUTSIDE STATED LIMITATIONS SHOULD BE REFERRED THROUGH DOCUMENT EXCHANGE, PO BOX 1500, SALISBURY, SA 5108

16. DELIBERATE ANNOUNCEMENT

No Limitations

17. CASUAL ANNOUNCEMENT Yes

18. DEFTEST DESCRIPTORS

fatigue, fatigue tests, military aircraft, civil aircraft, research projects, Australia, New Zealand

19. ABSTRACT

Optimal rework shapes for the most fatigue critical stiffener runout (SRO) locations in the F-111 wing pivot fitting (WPF) upper plate region have been determined using a recently developed finite-element-based gradientless shape optimisation procedure **[20]**. The WPF region is shown in Figure 4. The resulting precise free-form shapes (which typically remove cracked material) render the local notch stress distributions near uniform and typically provide a 30–40% reduction in peak elastic stresses as compared to current rework shapes that exist for aircraft in service with the Royal Australian Air Force. They also typically represent a 50% improvement in stresses, as compared to the nominal blueprint shapes. The unique optimal shapes (of different sizes) have been determined for four of the most fatigue prone SRO locations in the F-111 WPF. The final peak stress level achieved at each of the optimised runouts was relatively consistent; therefore, a similar inspection interval can be expected for each location. A sample comparison of elastic stresses is given in Figure 5, where it should be noted that the key aim was to reduce the peak compressive stress (to minimse residual tensile stresses after cold proof load testing). The recommended rework configuration provides a manageable compromise between minimising the runout stresses and maintaining the WPF buckling strength to within acceptable limits. A number of important issues have been addressed in the present practical problem, including; use of higher-order elements for efficient robust stress prediction; accounting for the effect of size constraints on the optimal shapes; and assessing the robustness of the idealised optimal shapes to perturbations away from idealised conditions, such as those due to potential manufacturing errors.

As part of an associated validation program, the precise shapes have been manufactured in two full-scale static test wings. Experimental strain measurements for the optimal shapes compare very well with predictions from finite element analysis. Further wing tests are currently underway in order to determine the damage tolerance and durability of these optimal shapes. Hence it is expected that the stress reductions predicted in the present work will be sufficient to provide a basis for extending inspection intervals by a least a factor of two, from 500 hours to 1000 hours. Implementation of such an extension to the F-111 fleet would provide a very significant maintenance cost saving.

Page classification: UNCLASSIFIED