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A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 2009 to March 2011

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ABSTRACT

This document has been prepared for presentation to the 32nd Conference and 26th Symposium of the International Committee on Aeronautical Fatigue scheduled to be held in Montreal, Canada May 29th to June 3rd 2011. 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 2009 to March 2011. The review covers fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.

RELEASE LIMITATION

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1. INTRODUCTION

This document presents a review of Australian and New Zealand work in fields relating to aeronautical fatigue in the period 2009 to 2011, and is made up from inputs from the organisations listed below. The authors acknowledge these contributions with appreciation. Enquiries should be addressed to the person identified against the item of interest.

DSTO	Defence Science and Technology Organisation, 506 Lorimer Street, Fishermans Bend VIC 3207, Australia
UNSW	University of New South Wales, Kensington, NSW, Australia
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2. AUSTRALIA

2.1 RESEARCH ACTIVITIES

2.1.1 Stable tearing in fatigue crack growth (Mohd Fairuz, Ab Rahman, Reza D. Mohammed, Chun Wang, Graham Clark [RMIT] and Xiaobo Yu, Qianchu Liu [DSTO])

Introduction

Fatigue fracture surfaces in a range of structural metals sometimes exhibit bands, visible as dull crescent-shaped regions which contrast to the "bright" background fatigue surface. Such bands as in Figure 1 are known as stable tearing and believed to be created within a single load cycle under either constant-amplitude (CA) or variable-amplitude (VA) load conditions. Essentially, stable tearing is a recurring process in which fatigue crack growth is interrupted by significant jumps of crack growth, usually at the central portion of the component while the crack-front lags behind near the free surface. Microscopically, crack advance by tearing has similarity to the final unstable fracture of the component, with the difference that the tearing is *stable*: it arrests after a certain distance of crack front advance, which is then followed by further fatigue crack growth.

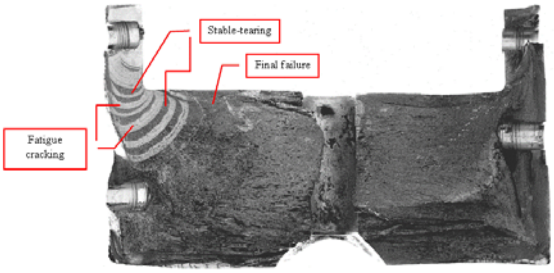

Figure 1: Fracture surface appearance of 7075 aluminium alloy lower wing spar from an Macchi MB326H, showing evidence of brighter regions of in-service fatigue crack growth, with progression markings interrupted by dull bands of stable tearing.

Post-failure analysis of fracture surfaces often relies on quantitative fractography to relate various crack-front progression markings to specific loads in the load history, in order to estimate the crack growth history in the component. Matching these data to predicted crack growth, however, is complicated by the fact that stable tearing is not included in fatigue predictive models, and so the presence of significant tearing can greatly influence the derivation of a crack growth history which can be matched to a crack growth model.

Methodology and Conclusions

This research involves a series of tests which produced tearing in 7075 aircraft aluminium alloy under CA and VA loading. The compact tension (CT) configuration was adopted and specimens were designed according to the relevant ASTM standard.

Specimens had an average width of and thickness of 40.00 ± 0.05 mm and 6.50 ± 0.01 mm respectively. Some of the results from these tests are shown in Figure 2. The test results have been used to examine the effectiveness of empirical models proposed for tearing analysis in failure investigation, leading to the conclusion that one of the models could be a useful engineering tool, relating the crack depth and the crack-front length. Modified slightly, it relates the stress intensity factor to the crack-front line length, and shows that this ratio could also be used as a simple engineering tool to relate the approximate levels of stress intensity factor applicable to any onset and arrest of tearing band observed.

The study reveals that the stress intensity factor is one of the key controlling parameters in tearing onset and arrest. The CA and VA tearing can be characterised by the first onset stress intensity factor, K_i^1, at which these two tearing conditions occur. The magnitude of K_i^1 for VA tearing was found to be equivalent to the material's plane strain fracture toughness, K_{Ic}, but considerably lower than the initiation K-value for CA tearing. The loading conditions also have been observed to impose different effects on the size of tearing. This study suggests that for similar K, the CA tearing at initiation has smaller tearing depth, Δa, than the VA tearing, but as the crack progresses, the size of VA tearing is markedly larger than that sustainable under CA loading. The CA condition seems to confer apparent resistance to tearing, which results in smaller tearing depth (crack jumps), than in VA loading conditions.

The other main conclusions relate to the notable differences between tearing under CA and VA loading. The CA and VA tearing are macroscopically different. The tearing after VA usually appears duller than the CA tearing, but both types of tearing are generally different to the fatigue crack growth region on fracture surface and both CA and VA tearing are also comparable to the face of the final unstable failure, as shown in Figure 2b.

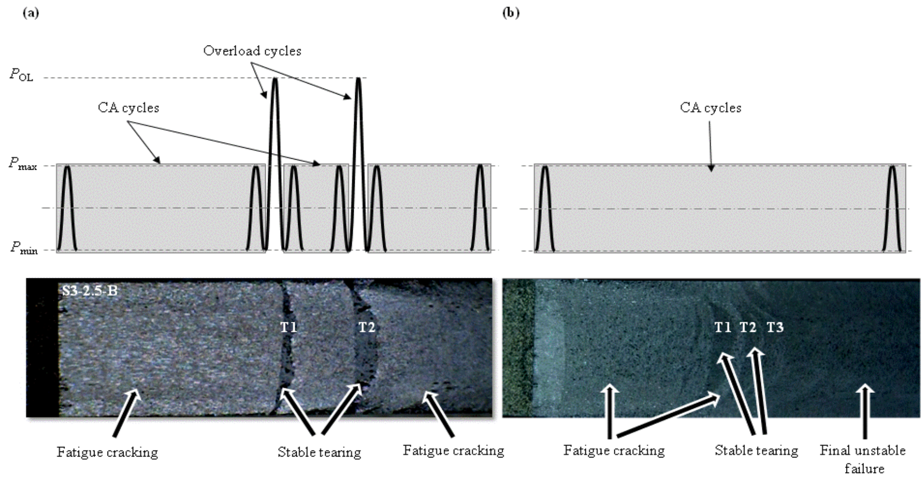

Figure 2: Stable tearing on fracture surface of a 6.5 mm thick 7075 aluminium alloy compact tension specimen. (a) The VA tear bands (marked T1 and T2) were produced during two high loads (overloads) in between background of low fatigue cycles; and (b) The three CA tear bands are marked as T1 to T3. The crack grows from left to right.

The static tearing curve is developed based on the standard K_R curve test method. The K_R (K-value with plastic zone correction) is plotted against the physical crack depth a_p as shown in Figure 3. The fatigue tearing data from previous tests is found to agree very well with the static tearing curve, with the CA and VA tearing scatter in the upper and lower limit of the curve respectively, as shown in Figure 3. The K-value at which static tearing commences is also approximately equivalent to the first onset stress intensity factor for VA tearing. This study suggests that the resistance curve method can be used to predict the first onset of tearing, and estimate the extent of tear under certain applied K.

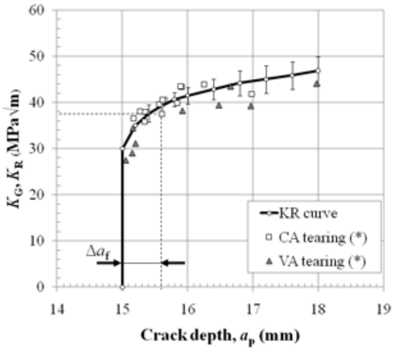

Figure 3: Distribution of tearing data against the static tearing curve.

The complex crack front curvatures observed at tearing arrest distort simple estimates of stress intensity factor and three-dimensional (3D) finite element (FE) analysis has been undertaken to estimate the through thickness stress intensity factor K_{3D} variation. Generally, the K-value at mid-thickness region reduces, while the K-values at the sides of the specimen increase as the crack-front becomes more curved. Figure 4 shows that the computed K_i and K_j at initiation and arrest of tearing show good agreement with the K-values estimated from the empirical models for analysing tearing. In all crack-front curvature at tearing arrest, the computed K values lies in between the K-values from the two models.

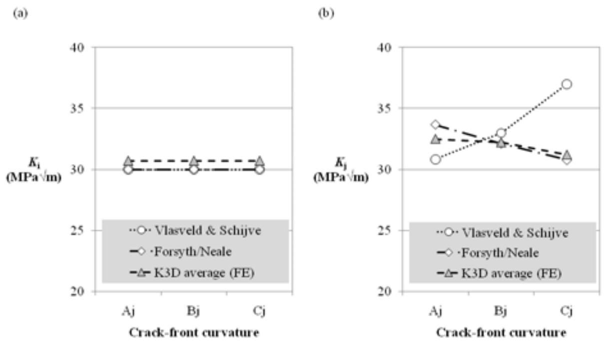

Figure 4: Comparison of K-values for three crack-front curvatures by using Forsyth and Schijve models, and computed through thickness K_{3D}. The K values at (a) onset of tearing; and (b) tearing arrest.

2.1.2 Critical strain performance of coating systems at aircraft joints (Ung Hing Tiong and Graham Clark [RMIT], in conjunction with Defence Materials Technology Centre)

Aircraft protective coatings (paint and sealant) protect the metallic alloy substrate from corrosive environments. Several locations in airframes are particularly susceptible to paint coating cracking, typically at the strain concentrations in joints. The cost of corrosion repair is high and so is the cost of repairing/replacing coatings. The objectives of this research are

(a) To determine the role of joint displacement in service as a factor contributing to degradation of corrosion-protective coating.

(b) Understanding this local strain effect (or loading history) is important if we are to predict or extend coating failures and lives.

Scanning electron microscope (SEM) and energy dispersive x-ray spectroscopy (EDS) analyses were performed on a retired P-3 wing panel, allowing a close examination of the paint condition at the fastener area. The panel is made of 7075-T6 aluminium alloy and consists of internal and external splices fastened via a column of countersunk Eddie bolts, see Figure 1.

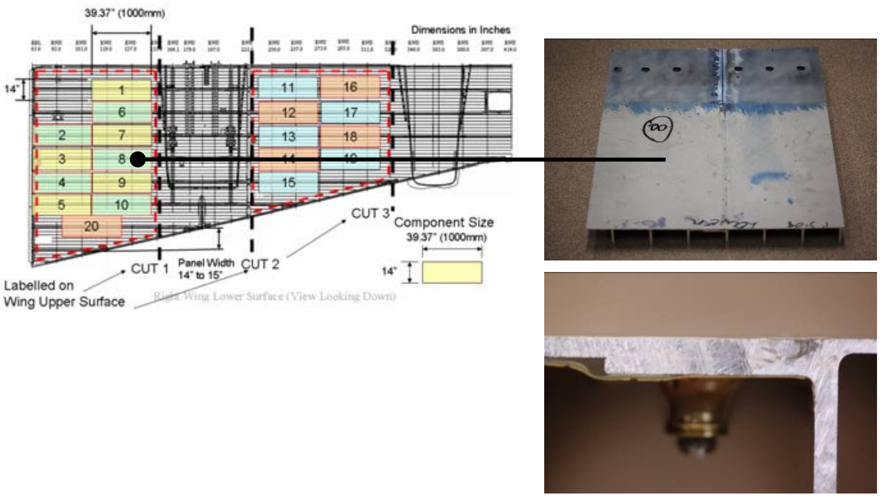

Figure 1: Schematic of P-3 lower wing panel cut-out

SEM-EDS results revealed excellent paint surface finish away from fastener head regions; but exhibited substantial paint defects, in the form of cracking and dislocation of layers (delamination); all of which were observed around fastener areas, see Figure 2. Numerous short, blunt cracks approximately normal to the paint surface had developed both at the external surface and interfacial layer. The cracks appeared to be consistent with movement associated with the fastener tilting or the stress concentration associated with joints as substantial displacement is expected at these locations. The examination supported a correlation between paint failure/cracking and mechanical displacement.

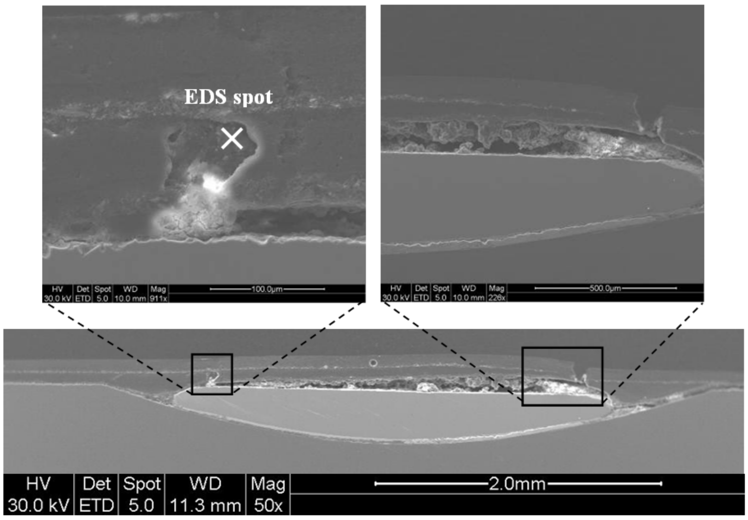

Figure 2: Scanning electron micrograph, showing paint defects and delamination underneath the paint film at fastener areas

Impact of joint end deflection on paint degradation

Finite element analysis was performed on two generic lap joints; one fastened with countersunk fastener, whist another one with dome-head. For a 45° bead of sealant and coating, length *l** (see Figure 3), at the exposed sheet ends, the predicted coating strain was given as 12% (countersunk) and 14% (dome-head). This level of strain could reduce coating integrity, especially for aged coatings as the intrinsic material strength of coating decays over time.

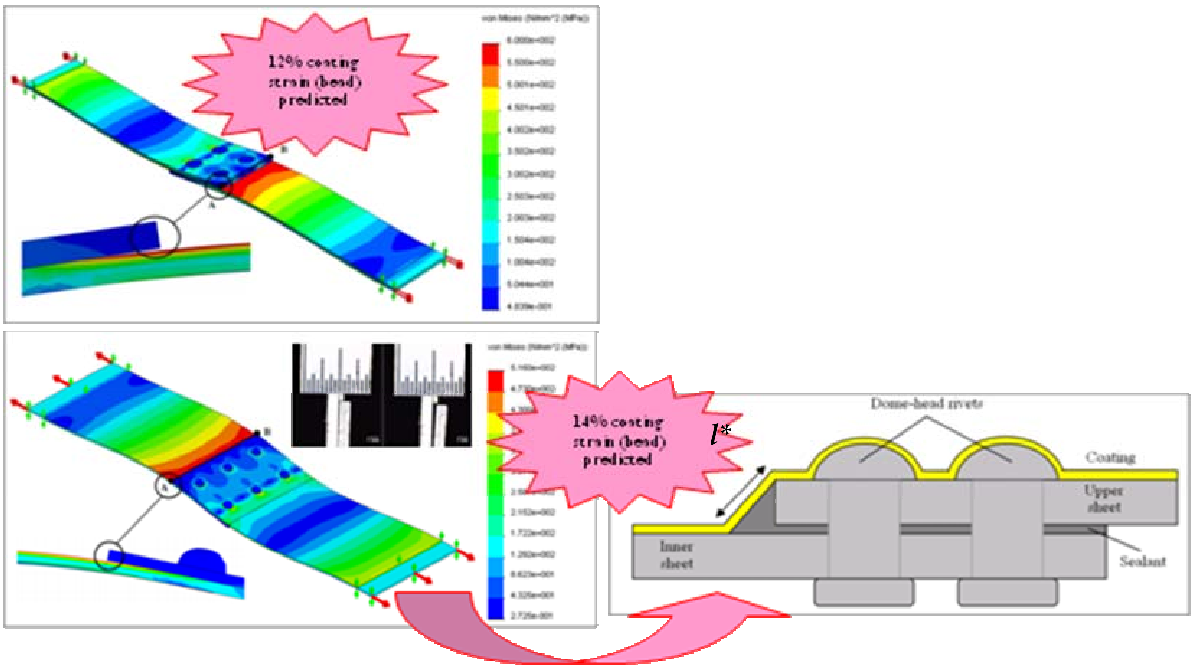

Figure 3: Finite element modelling, predicted strain along 45° sealant/coating bead

2.1.3 A review on the effect of corrosion inhibitors on the fatigue performance on riveted joints (Aditya Jaya, Ung Hing Tiong, Graham Clark [RMIT], Steve Swift [consultant] and Peter Trathen [DSTO])

Many General Aviation (GA) aircraft in Australia are operated well beyond their original planned service life, creating keen interest in the structural integrity and the overall safety of the aircraft, and the key factors such as corrosion, and fatigue which might degrade aircraft strength. Corrosion and fatigue commonly occur in riveted joints, and have significant effect on the overall maintenance cost of the aircraft as a result of the need for inspection and repair. A simple and effective way to protect aircraft against corrosion is to prevent the ingress of moisture, through the use of coatings and sealants, or by applying Corrosion-Inhibiting Chemicals (CICs).

The use of CICs has been extensive in the aircraft maintenance industry. Their use is emphasized in corrosion maintenance programs, such as Corrosion Prevention and Control Programs (CPCP). It has also become increasingly attractive for GA operators to use these corrosion inhibitors, even where such CPCPs are not available. While CICs can provide substantial benefits by preventing and retarding corrosion development, they are lubricating chemicals, and this provides a risk that structural fatigue life of riveted joints can be affected by their use. This could be particularly significant if the effect is substantial and limits the usefulness of CICs. Several investigations had been carried out in the past to investigate this effect, but the results were inconclusive; in many cases, the results appear to depend on many variables including, for example, the type of joint, type of fastener, stress level, and environmental conditions.

The objective of this research is to develop a more comprehensive picture of the fatigue behaviour of lubricated joints, to identify the underlying process, and to resolve the uncertainties from earlier research. In order to do so, fatigue tests were conducted using riveted joints typically used in small aircraft. Two types of riveted joints were tested, single lap joints made from 1 mm thick of 2024-T3 bare Aluminium sheets riveted using universal rivets with manual riveting technique, and single lap joints with force controlled riveting of countersunk rivets to join two 1.6 mm thick sheets of 2024-T3 ALCLAD Aluminium alloy together. Each type of joint was tested in two configurations, untreated and treated with CICs. A series of stress levels were tested to represent conditions of severely loaded joints. This was to allow identification of fatigue life trends, rather than a detailed assessment of fatigue life change at only one or two peak loads. The results were plotted in S-N plots, and were then compared.

In general, the application of CICs reduces the fatigue life of riveted joints at these relatively high loads, and a factor of 2 reduction was found to be the maximum. The reduction in fatigue life is mainly caused by the reduction in friction at the faying surface which promotes joint slippage. This causes more load to be transferred through the rivets, promoting earlier failure by rivet failure, or accelerate the crack growth rates in the materials around the holes. In addition, the application of CIC was observed to change the failure modes of the joints, especially for highly loaded joints. However, the severity of fatigue life reduction is highly dependent on the joint type as a function of stress level as shown in the figures below.

A potential reduction in fatigue life of riveted joints treated with CICs may seem initially to be a serious problem. But, we should note that corrosion, if untreated, can cause more significant problems to the structural integrity of the aircraft than the fatigue problems induced by the use of CICs. We have also seen the effectiveness of CICs in providing corrosion protection to real aircraft structures. This indicates that CICs would probably provide corrosion protection, which may well outweigh any structural problems. In addition, it needs to be noted that many of these "grandfathered" GA aircraft may not have a well-defined fatigue life anyway!

A potentially more serious issue for GA aircraft operators is the potential increase in rivet movement if CICs are used in joints. Any increase in the number of fasteners which exhibit "working" , or the severity of that working, could lead to very large additional maintenance costs, and these costs could be of far more significant to the industry than the increased risk of a fatigue life issue. In reality, of course, any repair of working joints would probably rectify any developing fatigue issues.

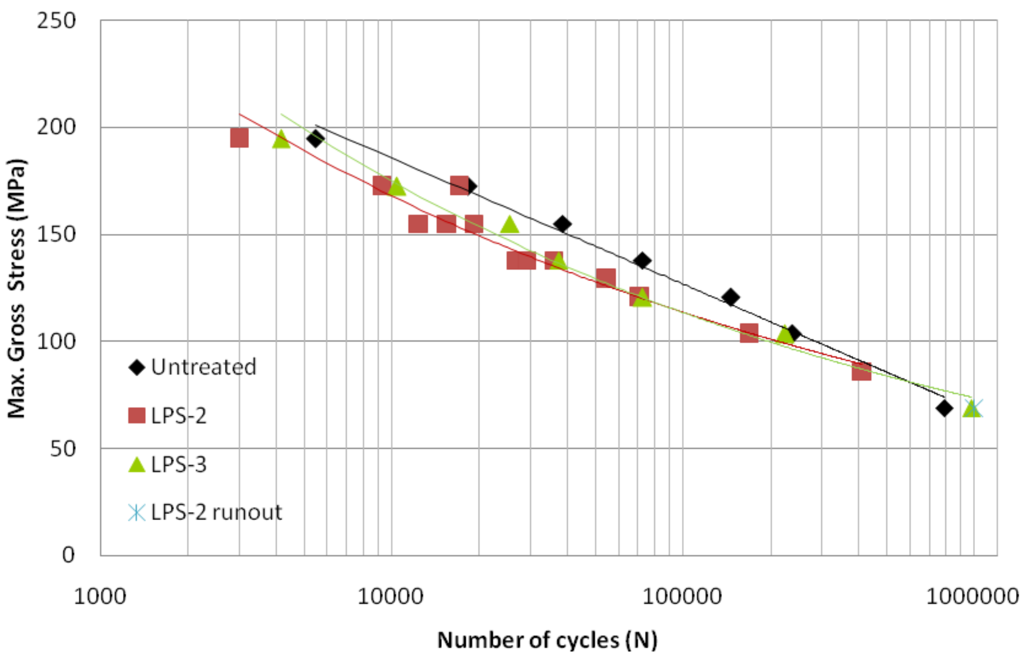

Figure 1: S-N plots of single lap joints with universal rivets and bare Aluminium sheets. The figure shows fatigue life comparison amongst untreated, LPS-2 treated, and LPS-3 treated specimens.

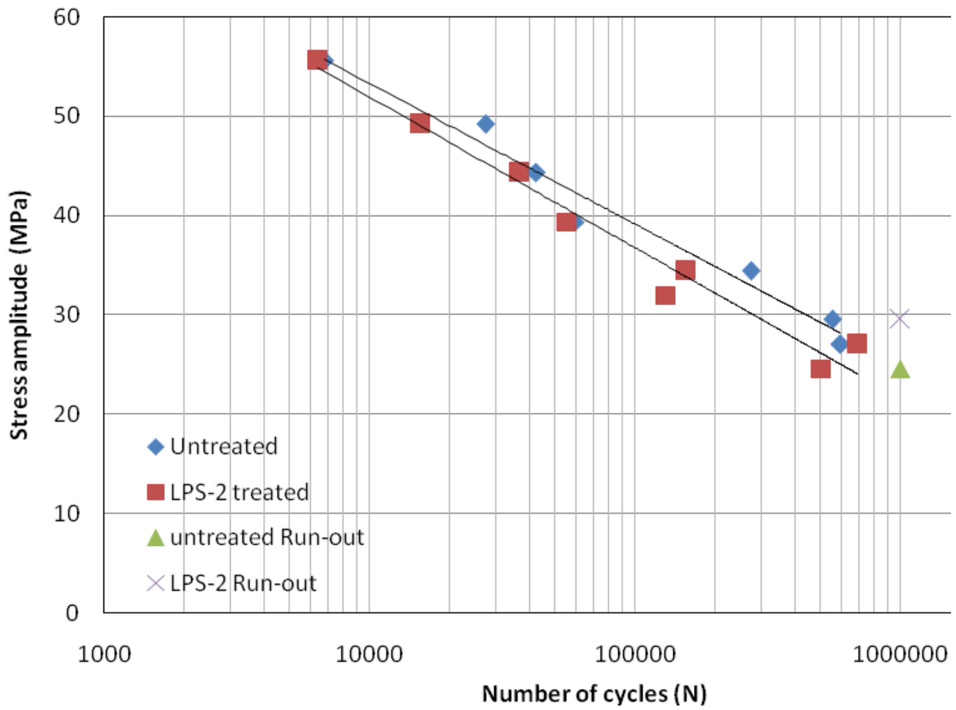

Figure 2: S-N plots of single lap joints with countersunk rivets and ALCLAD Aluminium sheets. The figure shows fatigue life comparison amongst untreated, LPS-2 treated, and LPS-3 treated specimens.

2.1.4 On the use of Supersonic Particle Deposition to restore the Structural Integrity of damaged aircraft structures (R Jones, C A Rodopoulos, K Cairns, S Pitt [DSTO Centre of Expertise in Structural Mechanics, Monash University] and N Matthews [Chief Design Engineer, Rosebank Engineering])

INTRODUCTION

The significance of this research is evident from the June 2007 Report to Congress by the Under Secretary of the Department of Defence (Acquisition, Technology and Logistics) [1]. This report estimated the cost of corrosion associated with US DoD systems to be between $10 billion and $20 billion annually. To address this problem Section 1067 of the Bob Stump National Defense Authorization Act, US Congress Public Law 107-3 14 (NDAA) requires the Secretary of Defense to designate an official or organization to be responsible for the prevention and mitigation of corrosion of military equipment and infrastructure. It also requires the development and implementation of a long-term strategy. As a result of a subsequent detailed study of the problem of corrosion the US DoD has focused its life-cycle corrosion research and development efforts on four primary areas [1]. One of these four areas is: “Repair processes that restore corroded materials to an acceptable level of structural integrity and functionality”. This topic is one of the focal points of the present paper.

To meet this challenge Monash and Rosebank Engineering, as part of a project sponsored by the Royal Australian navy Aircraft Systems project Office, have evaluated the potential for supersonic particle deposition (SPD) technology [2-7] to restore the structural integrity of damaged aluminium alloy structural components. The SPD process is currently mainly used to deposit metal, alloy, polymer, or composite powder material onto a substrate to provide a protective coating and in some instances to restore damaged/worn geometries, see Figure 1. The coating is formed by exposing the structure/component to high velocity (typically between 300 to 1200 m/s) solid-phase particles, which have been accelerated by a supersonic gas flow, usually either nitrogen or helium, at a temperature that can range between 400 to 900 ºC. When used to protect aluminium alloys from corrosion it usually involves the deposition of a pure aluminium surface layer/coating.

Figure 1: Rosebank Engineering Restoration of a Seahawk Helicopter Main Gear Box Module Aft Flight Control Pad, from [9].

It should be stressed that, to date, SPD has primarily been used to produce protective coatings [2-7]. However, more recent aerospace applications have seen SPD being used to restore damaged/worn geometry [6, 7]. Other than the work performed by Rosebank and the CoE-SM [8, 9] little (if any) attention has been given to using this technique to restore the damage tolerance of damaged aluminium alloy aerospace components, which is the focal point of this work. As a result of this study it would appear that this process has the potential to be a viable alternative to the use of externally bonded composite repairs [10, 11] for extending the fatigue life of damaged structural components.

EXPERIMENTAL STUDIES

To study the ability of an SPD doubler to reduce crack growth tests were performed on a single edge notch dogbone specimen, with a geometry as described above and an (initial) 1.4 mm long edge notch, see Figure 2. In the initial base line test there was no SPD and the specimen was tested under constant amplitude loading with a peak stress in the working section of σ_{max} = 93.36 MPa and R ($\sigma_{min}/\sigma_{max}$) = 0.1. The specimen lasted approximately 44,000 cycles and crack growth in the 2024-T3 plate was monitored using digital cameras. A second test was then performed whereby an SPD strip, see Figure 2, was applied to a notched specimen. In this case the test was stopped after approximately 345,000 cycles since there was no apparent crack growth at the notch (crack) or damage in the SPD.

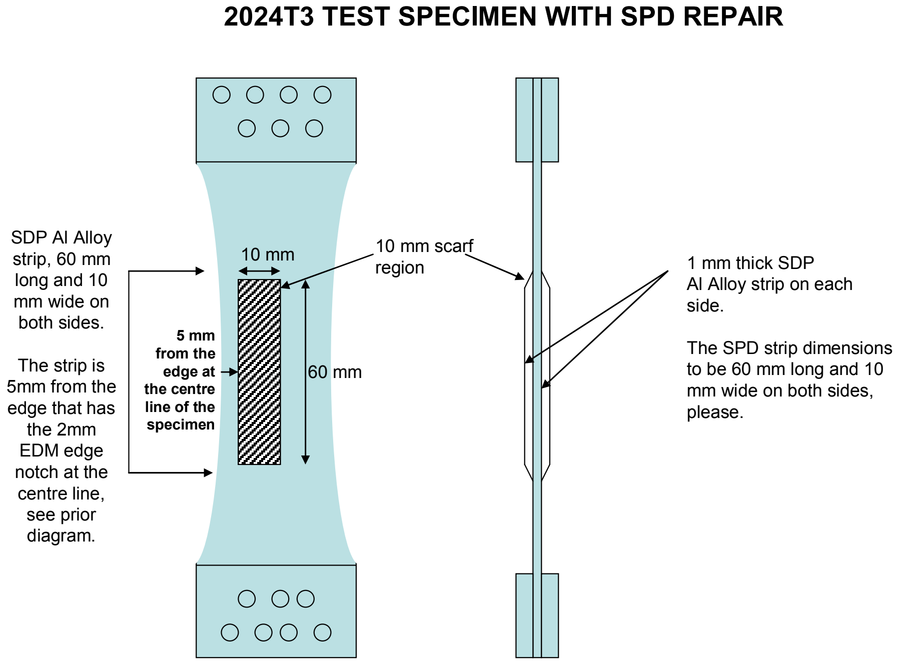

Figure 2: Schematic diagram showing the location of the SPD strip

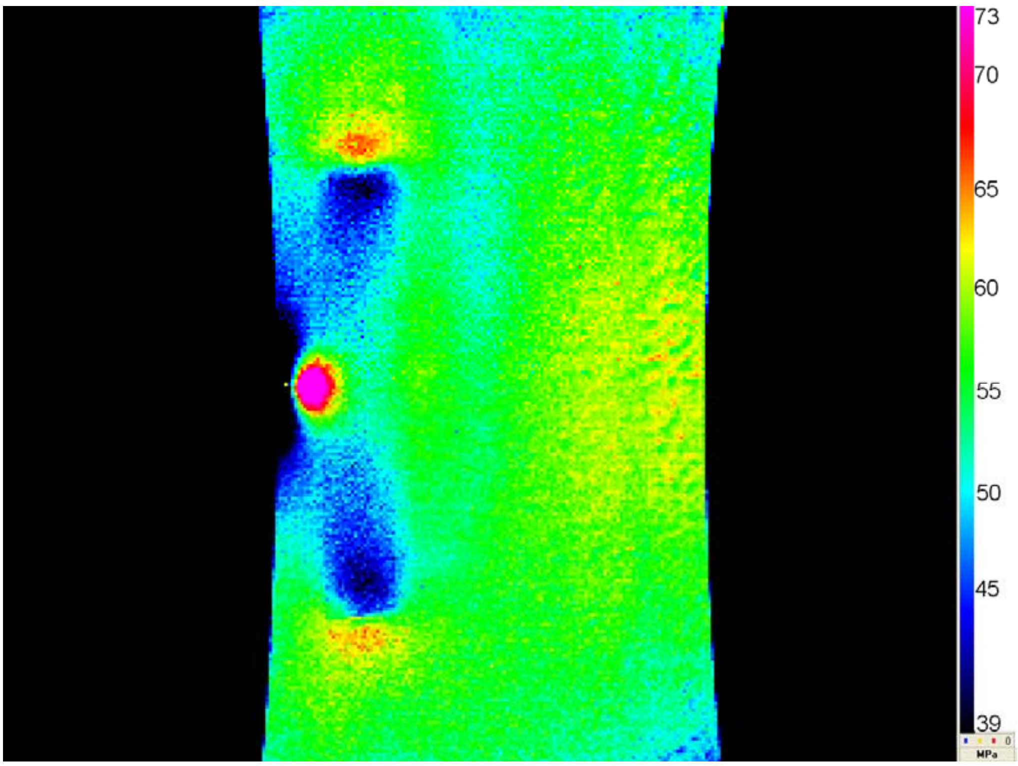

Figure 3: The stress field in the skin and the SPD strip

An infrared stress image captured shortly after the start of the test is shown in Figure 3. In this figure the picture was captured at a cyclic stress amplitude $\Delta\sigma$, remote from the centre line of the specimen, of approximately 53 MPa. This was done so as to not overly influence crack growth in the skin. Here we see how the stress field in the SPD ahead of the crack is contiguous with that in the plate, i.e. the SPD is taking load in the region ahead of the crack. We also see hot spots in the skin outboard of the ends of the SPD strip which establish that the SPD strip was indeed pulling load from the skin. This is essential if the process is to enhance the damage tolerance of the skin. The current work is to be reported in [12].

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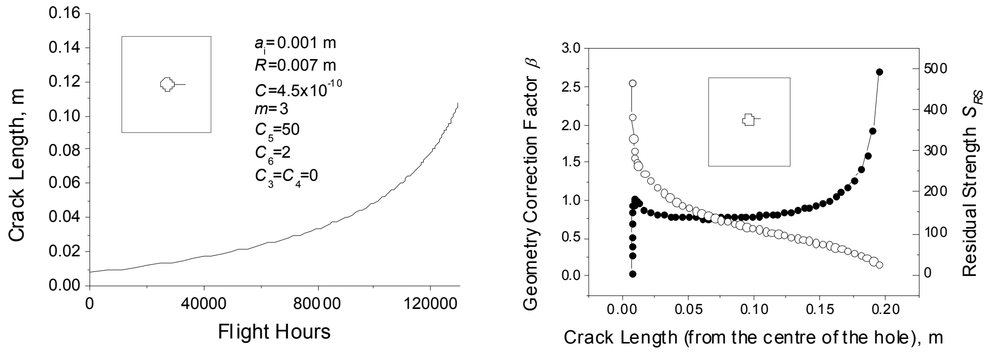
2.1.5 Integrated Probabilistic Analysis of Damage Tolerance and Risk for Airframe Structural Locations (W. Hu and R.F. Torregosa, [DSTO])

A new method for the integrated probabilistic analysis of damage tolerance and risk of failure that considers the effects of material property scatter and the load history effect on crack growth rates has been developed. The underlying damage tolerance analyses were conducted using a plasticity-induced crack closure model, with key inputs generated using the Monte Carlo method. The statistical behaviours of the following variables were modelled: the initial crack size, the peak stress of the load spectrum, the fracture toughness, the parameters defining the fatigue crack growth rate and the threshold stress intensity range. Each random variable was defined as either (i) a random variable with a normal, log-normal or Weibull distribution, or (ii) as a deterministic variable with a single specified value. The probabilistic crack growth model was implemented in the in-house CGAP and the risk analysis was implemented in another in-house computer code.

Risk analyses were conducted on two structural elements to demonstrate the use of the new method. Figure 1 (a) plots the master crack growth curve for a plate with a central hole made of aluminium 7075-T6, and Figure 1 (b) shows the geometry correction factor and the corresponding residual strength curve. The thin green curve in Figure 2(a) represents the probability of failure from the master curve approach and the thick black curve represent that from the probabilistic crack growth approach. The probabilistic crack growth approach produced lower risk of failure than the master curve approach. Figure 2(b) shows sample example distributions of crack length at different flight hours. Clearly, the master crack growth curve approach predicts larger crack sizes which lead to larger probability of failure. Similar observation was made for the centre crack tension specimen.

It is envisaged that by capturing the material property scatter and the load history effects the proposed approach would provide an improved probabilistic risk assessment for aircraft structures.



(a) (b)

Figure 1: (a) The master crack growth curve, and (b) the geometry correction factor and the residual strengths for a plate with a central hole.

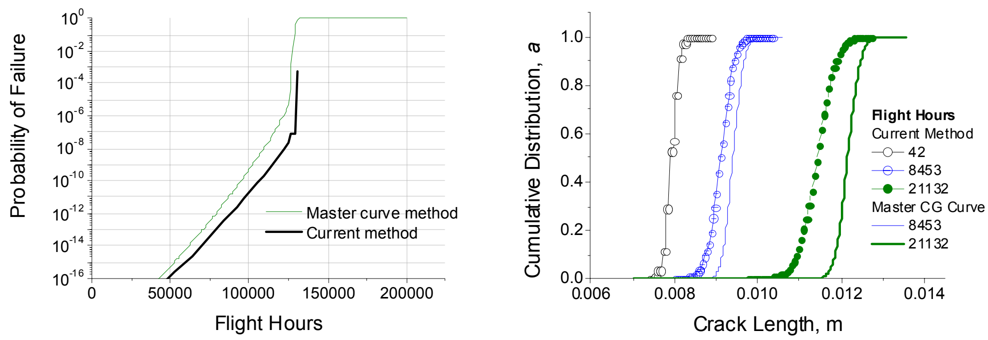


Figure 2: (a) The probability of failure curve (the 'current method' is the proposed integrated probabilistic method) and (b) the corresponding examples of distributions of crack length at different flight hours for a plate with a central hole.

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2.1.6 The effect of specimen thickness on fatigue crack growth rate and threshold behaviour in aluminium alloy 7075-T7351 (W. Zhuang, Q. Liu and W. Hu, [DSTO])

The effect of specimen thickness on fatigue crack growth behaviour in a common aircraft material was investigated experimentally. The objective of this investigation was two-fold; firstly to ascertain the crack growth rate data to be used for fatigue life analysis of aircraft structures involving varying thicknesses such as in F111 lower wing skin; secondly, to obtain accurate fatigue threshold values using the updated ASTM fatigue testing method.

Compact tension specimens were manufactured from thick aluminium alloy 7075-T7351 plates, and uniaxial loading was applied in the direction parallel to material process rolling orientation. The specimens with four different thicknesses (3.0, 8.0, 12.0 and 24.0 mm) were tested under constant amplitude loading at two stress ratios of R=0.1 and 0.5 in the ambient laboratory air. Fatigue crack growth data were obtained according to ASTM Standard E647 – 2008. The results from this experimental study indicated that, for the compact tension specimens, the thickness has negligible influence on fatigue crack growth rates, as shown in Figure 1. Subsequently, experiments were also carried out on the same specimens using constant-Kmax test method, to investigate threshold behaviour of fatigue crack growth. Compared with the conventional test method (the constant R load reduction test), the study found that constant-Kmax test method can generate more accurate fatigue threshold values, see Figure 2. It is envisaged that this study will lead to improved predictive capability for fatigue crack growth experienced by aircraft structures, and to enhance aircraft structural integrity management.

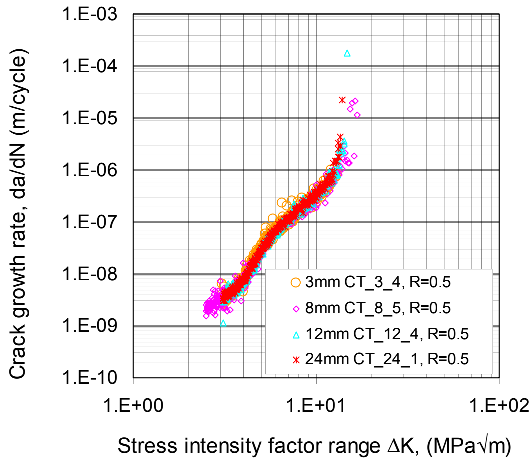

Figure 1: FCGR in AA7075-T7351 CT specimens with different thickness at R=0.5.

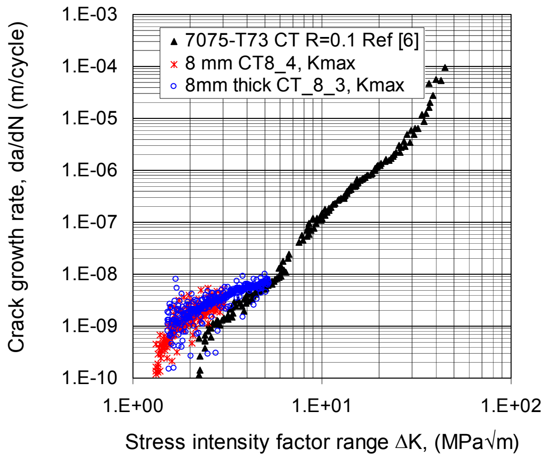

Figure 2: Threshold behaviour of AA7075-T7351 CT specimens for t=8.0 mm.

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2.1.7 A new method for evaluating the cyclic elastic-plastic stress distribution near an open hole under variable amplitude loading (Wallbrink C and Hu W [DSTO])

The assessment of fatigue crack growth under elastic-plastic conditions is computationally expensive due to the large number of load cycles involved. A new simplified elastic-plastic solution has been developed to address this issue. For a plate with a central open hole, a numerical procedure was developed to accurately and efficiently determine the distribution and evolution of elastic-plastic stresses in the plane of expected crack growth. The plane was discretized into a number of elements that extended from the edge of the hole to the edge of the plate, shown in Figure 1. At each load turning point, a generalised Neuber method was used to estimate the local elastic-plastic response in each element, and the load shed due to yielding was progressively redistributed to maintain the load bearing capacity of the plate. The redistribution was weighted by the updated distribution of the elastic stresses. Numerical accuracy was achieved by updating the distribution of the elastic stresses normal to and tangential to the plane. Globally, the problem was solved incrementally, but locally proportional loading was assumed in each element. The results showed that improved accuracy could be achieved by carefully considering the function of the redistributed load, the way the load was redistributed and the triaxiality of the stress state ahead of the notch. Results are presented in Figure 2 for both plane strain and plane stress conditions and are also compared to finite element solutions. The results in Figure 2 characterise material elastic-plastic behaviour using a Ramberg-Osgood constitutive relation. At 90% of the yield strength of the material the plastic zone size was approximately twice the hole radius and still the new method provided excellent agreement with finite element analysis. To further generalise the method for application to fatigue crack growth analysis the method was then extended for cyclic loading. Figure 3 shows a comparison of the results from finite element analyses and the present method for a variable amplitude test sequence. Here a bi-linear constitutive material relation was used to describe the material behaviour. Again the results from the new method and those from finite element analysis show excellent correlation. Further work is under way to extend the present method to more generic elliptical notches and it is envisaged that this will lead to better fatigue crack growth prediction in areas affected by local plasticity. For more information see [1].

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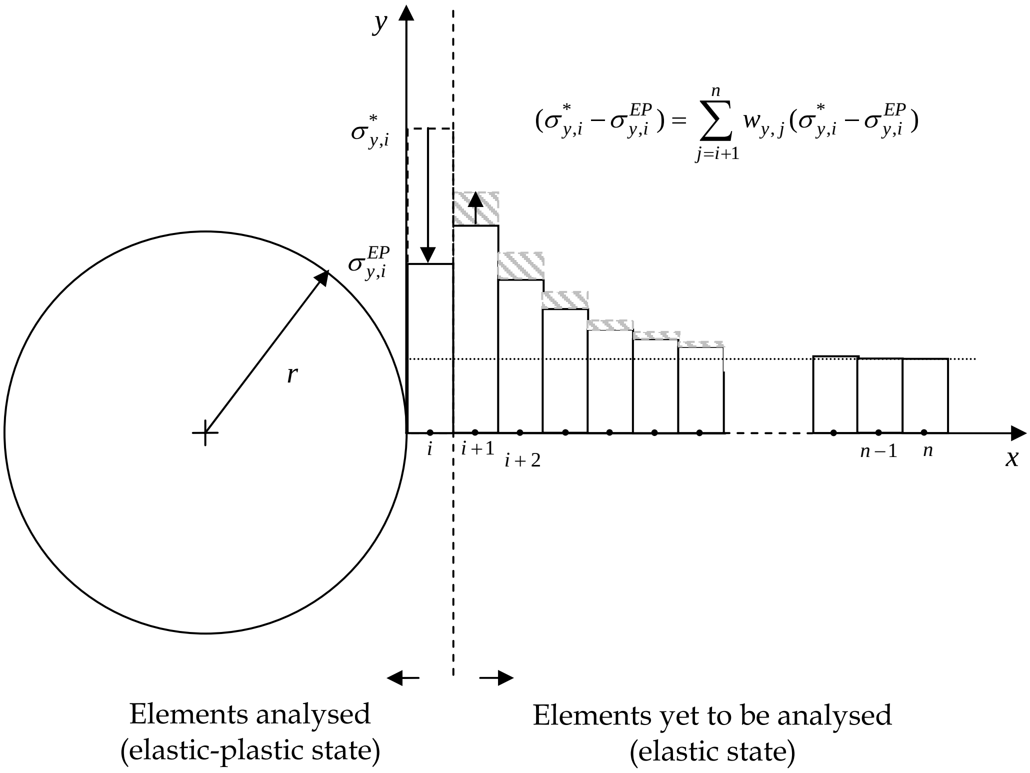

Figure 1: Redistributed stresses in the y-direction after the analysis of element i .

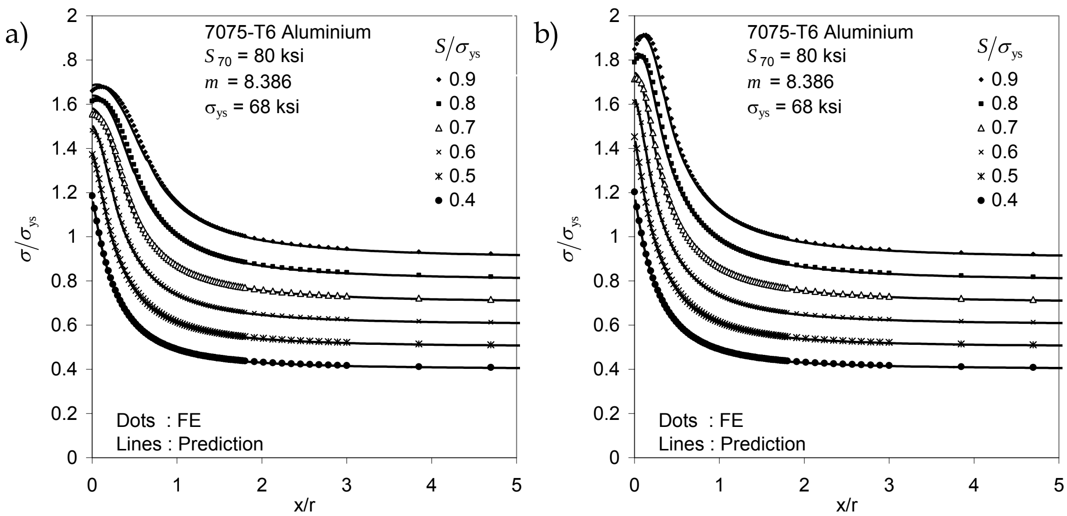

Figure 2: Comparison of the stress in the y-direction between the present methodology and FE analysis under a) plane stress and b) plane strain conditions for 7075-T6 Aluminium

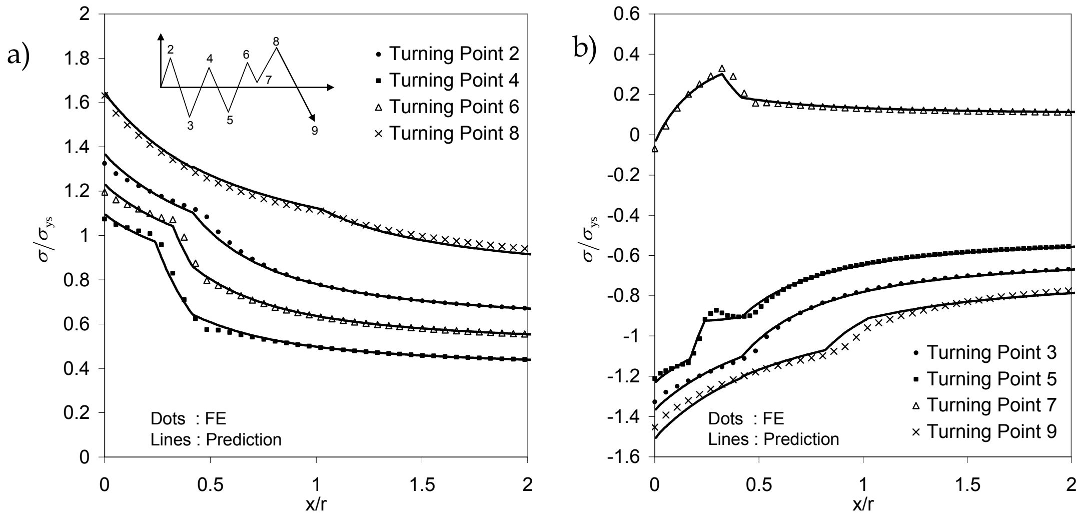

Figure 3: Comparison of the stress in the y-direction between the present methodology and FE analysis under plane stress conditions for 7050-T7451 Aluminium a) Peaks b) Valleys

2.1.8 Modelling fatigue crack growth near an open hole under spectrum loading using a new method for elastic-plastic stress calculation (Wallbrink C and Hu W [DSTO])

To date the analysis of fatigue crack growth from an open hole subjected to contained cyclic plastic deformation is problematic due to the prohibitive computational efforts involved. Thus, a new efficient method for calculating the elastic-plastic notch stress field under cyclic loading has been integrated into a crack growth analysis tool in an attempt to reduce computational effort and improve predictive capability. The anticipated crack path is defined along a line emanating from an open hole, and the line is discretized into a series of elements on which the elastic-plastic stress distribution is solved for the uncracked body. This solution is then used in conjunction with a weight function approach to calculate the stress intensity factors used in the crack growth analysis. This technique has been incorporated into a Defence Science and Technology Organisation proprietary crack growth analysis tool CGAP based on plasticity-induced crack closure. Results are presented in Figure 1 for cases involving service load spectra from the F/A-18 aircraft and are compared to those obtained from a commonly-used crack growth code that does not account for the effect of contained notch plasticity. It is hoped that this generic approach will lead to improved predictions for crack growth in plasticity-affected zones. For more information see [1].

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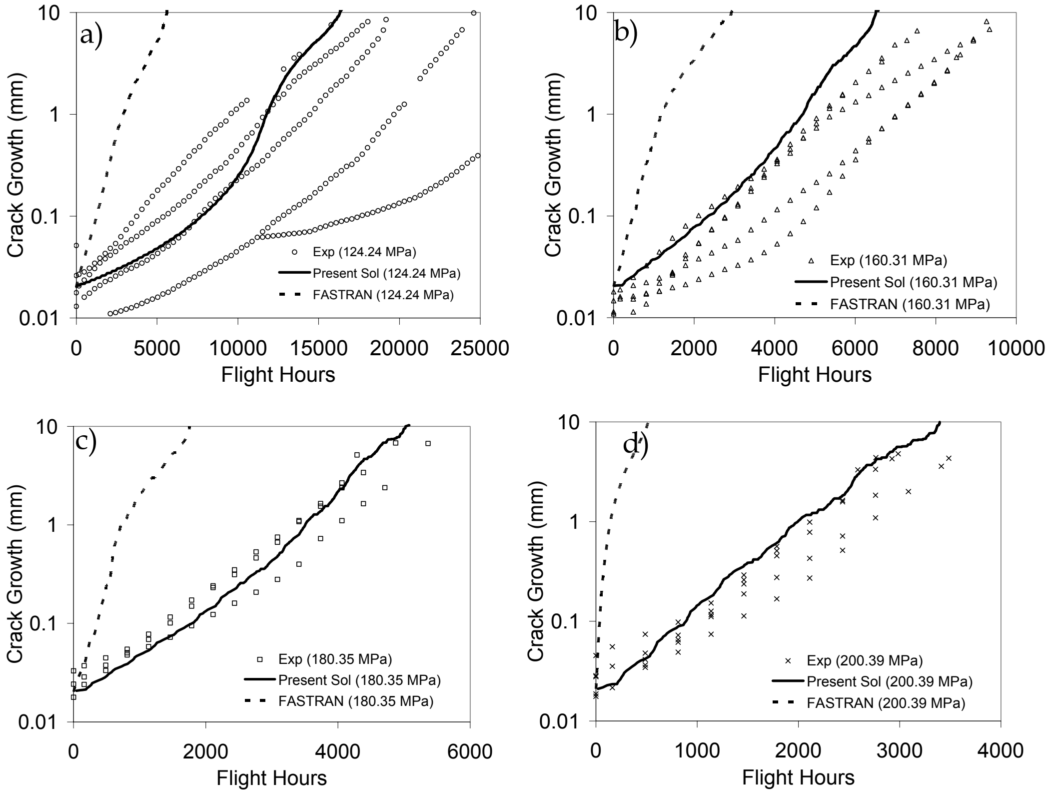

Figure 1: Crack growth curves for a single through- thickness crack emanating from the edge of a central hole in a plate subjected to an F/A-18 load sequence scaled by a) 124 MPa b) 160 MPa c) 180 MPa and d) 200 MPa.

2.1.9 A strain life module for CGAP: Theory, User Guide and Examples (Wallbrink C and Hu W [DSTO])

The assessment of fatigue damage in structures is an essential and an integral part of the management of air vehicles within the Australian Defence Force (ADF). As such the aircraft certification process requires that fatigue damage be assessed and critical areas susceptible to fatigue damage identified. This highlights the importance of methods for evaluating fatigue damage to the overall safe and economical management of aircraft. Hence the need for a robust and user-friendly software environment with which the engineer can use to assess the fatigue life of critical structural elements.

To this end the strain-life code FAMSH has been incorporated into an existing software tool CGAP. CGAP is a DSTO developed software tool that contains a set of algorithms commonly used to assess and analyse fatigue damage. It is envisaged that CGAP will gradually become a one-stop tool set with which the engineers can perform most of their required fatigue analyses. CGAP is a Microsoft Windows-based application with a graphical user interface (GUI) and an integrated database for the management of material properties, geometry sets and load cases. The GUI assists the user in entering computational parameters, inspecting spectra and checking for data consistency. It also provides a simple plotting capability for crack growth outputs. The inclusion of FAMSH to CGAP aims to provide improvements in prediction reliability which can be realised through reduced operational and maintenance costs, improved performance and combat readiness. For more information see [1].

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2.1.10 Crack growth rate curves: Which part dominates life prediction and when? (Wallbrink C, Jackson P and Hu W [DSTO])

Investigations into the importance of the crack growth rate data used in conjunction with FASTRAN [1] to make fatigue crack growth predictions have been undertaken. Selected fighter and transport aircraft load spectra have been used to investigate and determine the regions of the crack growth rate curve that dominate the crack growth predictions. The results of the analysis show that universally crack growth predictions under typical aircraft load spectra utilize a significant portion of the crack growth rate curve close to the threshold region as indicated in Figure 1. Figure 1 shows the probability that a cycle will have a certain ΔK_{eff}, where ΔK_{eff} is the stress intensity range for a cycle in the spectrum adjusted by the crack opening stress. It is observed that both fighter and transport aircraft display a similar behaviour throughout the entire crack growth prediction. As the threshold region dominates in terms of the number of cycles, this region will be particularly sensitive to errors in crack growth rates. Any error will be accumulated over a large number of cycles to ultimately affect the total life predictions. This finding demonstrates the need for accurate crack growth rate data for the threshold region in order to improve total fatigue life predictions. For more information see [2]

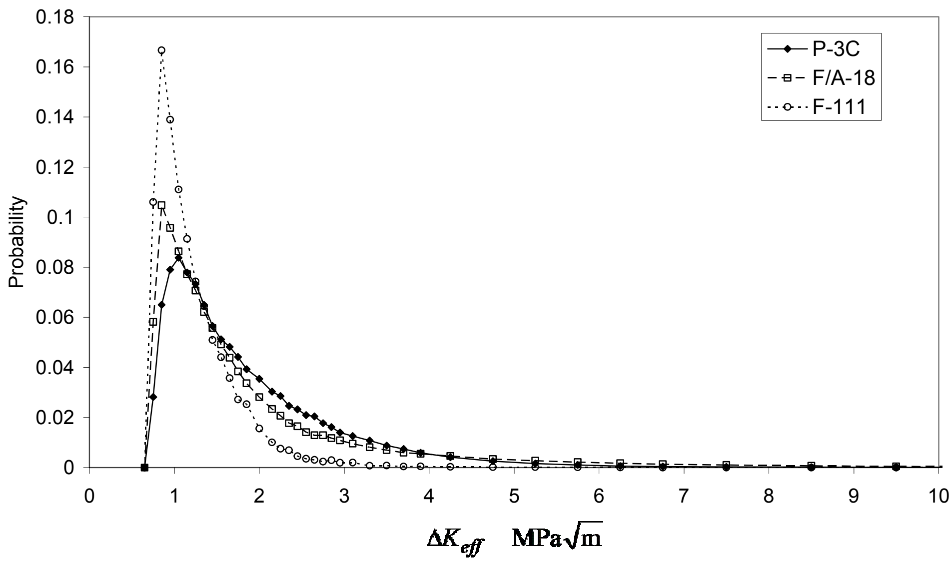

Figure 1: Probability density of ΔK_{eff} for crack growth under the F-111, F/A-18 and P-3C spectrums

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2.1.11 Probabilistic Risk Analysis for a C-130 CW-1 Location (R.F. Torregosa and W. Hu [DSTO])

As an exercise to build up the in-country capability in probabilistic risk analysis of aircraft structures, an analysis was conducted for a panel on the lower surface of the centre wing of C130-Hercules, made of aluminium alloy 7075-T73. The analysis was based on a scenario of a single crack propagating phase-by-phase in an analytically defined order through the structure, see Figure 2. An in-house probabilistic risk analysis program was developed to calculate the single flight probability of failure (SFPoF). The in-house program was verified by comparing the results with PROF [1] for two cases. As shown in Figure 1a and Figure 1b the results from the two programs correlate well.

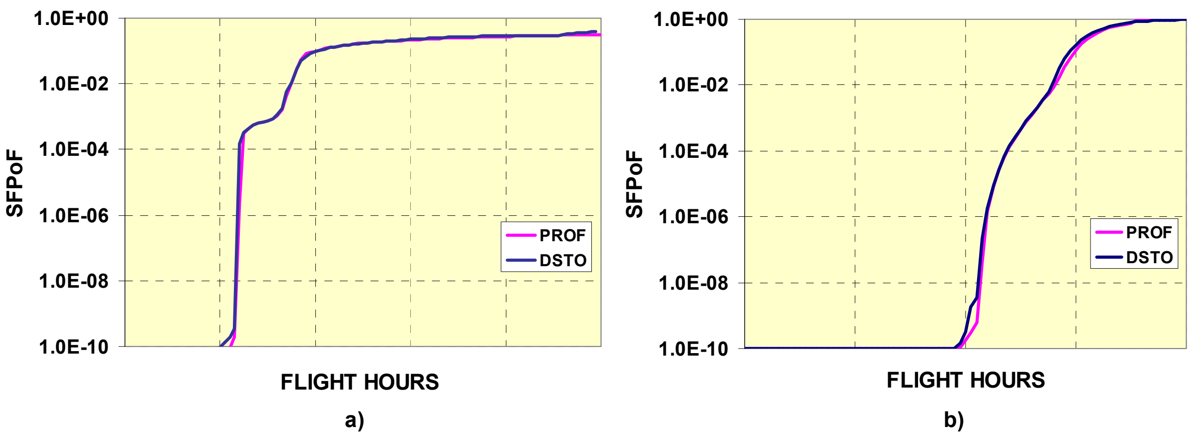

Figure 1: Comparison of PROF and DSTO in-house analysis programs a) SFPoF from PROF program sample problem; b) SFPoF for C130 CW-1

Probabilistic risk analysis was then conducted for the panel on the lower surface of the centre wing. It was assumed that cracking starts at the edge of the hole in Phase I and propagates to the edge of the panel. After reaching the edge of the panel the crack starts to grow at the other side of the hole, as shown in Figure 2.

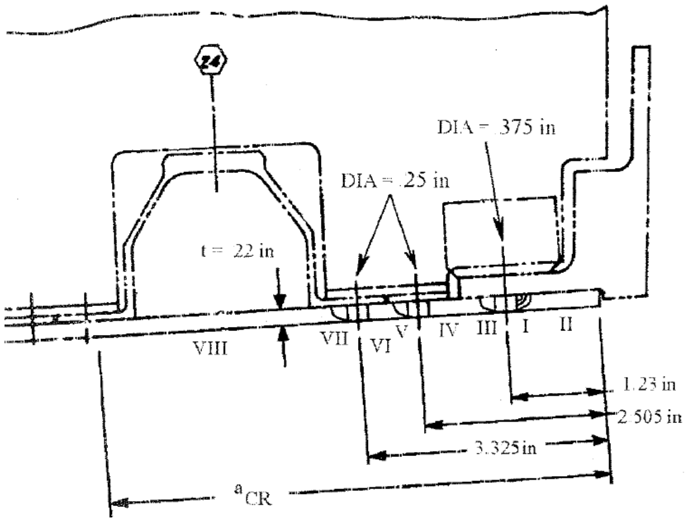

Figure 2: DTA analysis location showing crack growth phases

The probability of failure was calculated for the cases with and without inspection. It was observed that when an inspection was performed early, the reduction in the immediate probability of failure was very slight, as shown in Figure 3a. This is because when the inspection was performed the crack sizes are still very small. Hence the probability of detection is low. Furthermore, even when the mean of the crack size distribution was reduced slightly by the inspection, the probability of failure was not affected significantly as the residual strength of these small cracks changed little. However, the removal of detected cracks did have a noticeable effect on the probability of failure later in life, as demonstrated by the two curves in Figure 3a.

To illustrate more clearly the effect of the timing of the first inspection on SFPoF, a comparison was made between the probabilities of failure for inspections conducted at times t_a and t_b (Figure 3b). Clearly, the inspection at t_b had more impact on the immediate probability of failure.

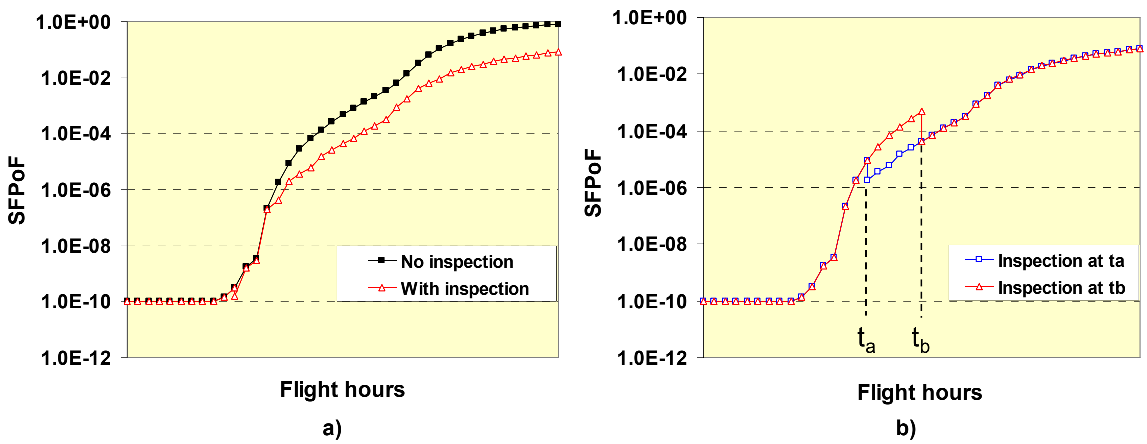

Figure 3: SFPoF curves comparison a) With and without inspection; b) Varying inspection times.

The outcome of this work demonstrates DSTO's capability in conducting probabilistic risk analysis and provides confidence to DGTA in evaluating results from such analyses.

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2.1.12 Research Investigations into the fatigue crack growth process (White P and Mongru D [DSTO])

Long range research into fatigue mechanisms in the Air Vehicles Division at DSTO is focused on developing greater understanding of the crack growth mechanisms due to variable amplitude fatigue. This is achieved by experimental work using fractography of fatigue cracks to compare the rates of crack growth and using atomic simulation of fatigue crack growth to elicit greater understanding of the mechanisms and provide a test bed for developing more effective crack growth equations.

Currently experimental work is under way to investigate the memory effect of aircraft materials. Because of the material memory, to predict the rate of crack growth due to variable amplitude loading sequence using constant amplitude crack growth data an equivalent cycle counting technique is required. The most common technique rainflow counting, ensures interrupted cycles keep a memory of the pre interruption state. By looking at the striation pattern on the fracture surface of small fatigue cracks under simple sequences we have been able to determine the significance of this memory effect, see Figure 1 for a typical optical microscope picture of a small sample of the fracture surface. The loading sequences applied to investigate this behaviour typically consist of approximately 10 to 20 cycles of a large cycle with an interrupted cycle to produce a recognisable band on the fracture surface. This is then followed in the same sequence by the same cycles re-ordered to their equivalent constant amplitude cycles as interpreted using rainflow counting. Initial results show that the re-ordered rainflow cycles produced the same degree of crack growth as the original variable amplitude sequence indicating rainflow counting is effective. However, it was discovered that other ordering of the sequence also produces equivalent rates of crack growth indicating the memory effect in Al7050-T7451 is small and that rainflow counting may not be required.

Molecular dynamic simulations have been developed to simulate the fatigue crack growth process at the tip of a crack subject to linear elastic fracture mechanisms boundary displacements. These displacement are applied to the cylinder of a material at the tip of a crack. The model is then cycled though a small stress intensity range and the crack tip is seen to advance in a cyclic fashion. Investigations are being conducted using pure aluminium model with the Embedded Atom Method potentials from a number of different sources in order that comparisons of the effect of the potentials can be made. The potentials enable a large range of phenomena to be simulated and produce reasonably accurate predications of material properties such as lattice spacing, various moduli, stacking fault and vacancy formation energies. Various structures such as stacking faults , dislocations and vacancies are seen to form at the crack tip. Also seen is a gradual tangling of glissile dislocations with sessile dislocations which gives rise to some degree of work hardening. Typically crack growth rates are seen to stabilise to a constant rate after the application of 5 to 7 cycles. Crack tip advance is seen to be continuous and much smaller than the lattice spacing since crack propagation is not merely cleavage of the atomic lattice but involves the development of a cyclic plastic zone at the crack tip. An example of the simulation is shown in Figure 2.

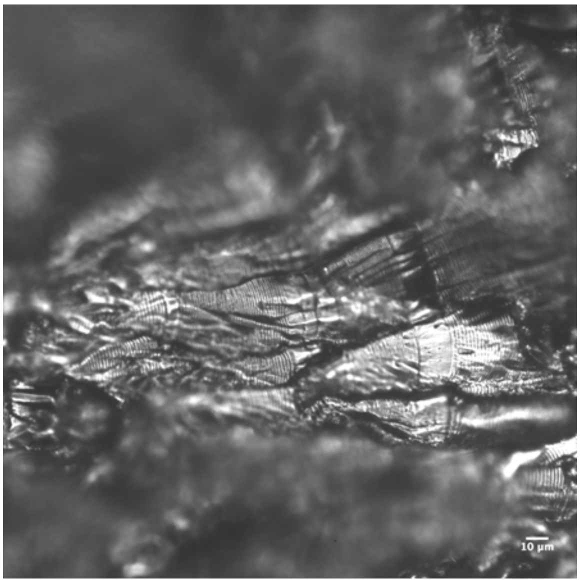


Figure 1: Shows the small segments of a particular variable amplitude or constant amplitude loading sequence. Comparisons of the crack growth for each band are made to determine equivalent crack growth cycles

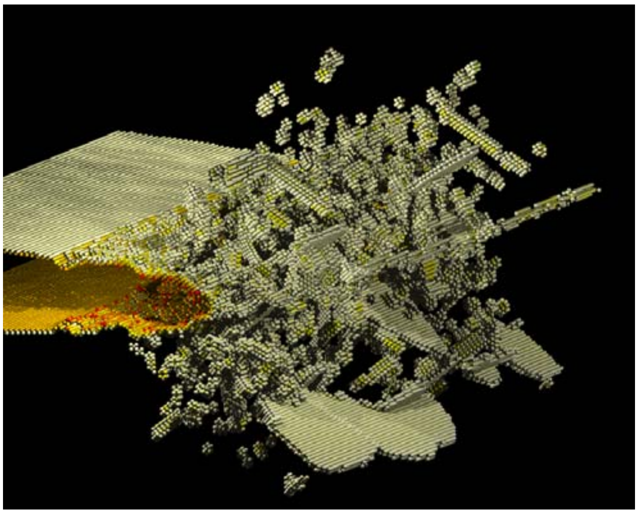


Figure 2: Molecular dynamic simulation showing the atomic arrangement at the tip of a fatigue crack . This figure shows only atoms displaced from the original fcc lattice. 15 loading cycles were applied to develop the complex structure at the tip

2.1.13 Investigation of Stress Intensity Factor for Overloaded Holes and Cold Expanded Holes (R.J. Callinan, R. Kaye and M. Heller [DSTO])

An understanding of how residual stresses due to overloads effect stress intensity factors for fatigue cracks at common stress concentrators, such as holes, is important. In this work a weight function approach is used to determine stress intensity factors for cracks in residual stress fields, due to three types of overloads for holes in metallic plates with and without subsequent remote loading.

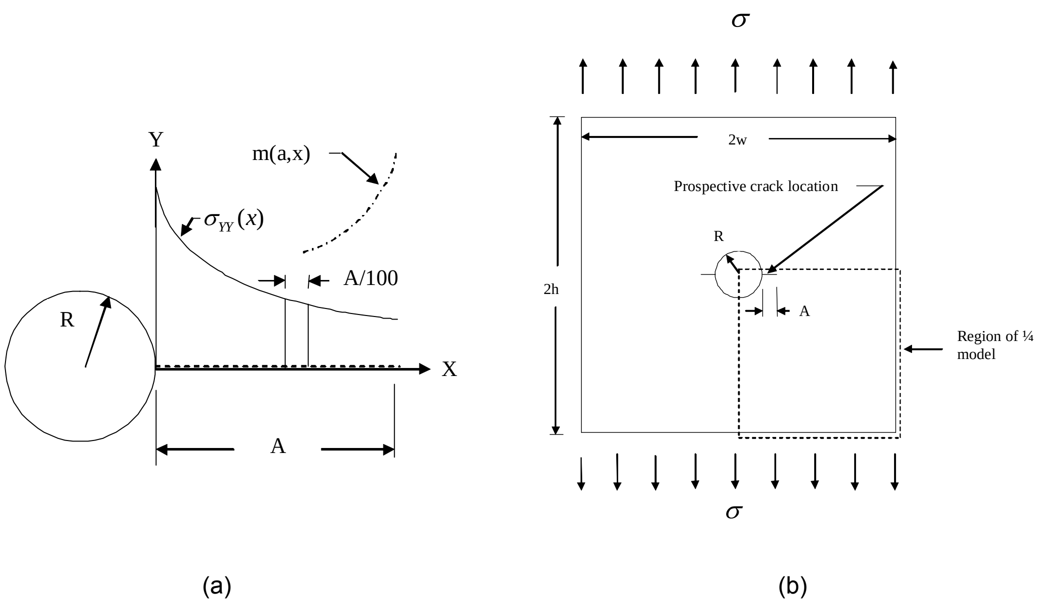


Figure 1: Notation and geometry for analysis of overloaded plate containing hole (a) local detail for calculation of stress intensity factor, (b) plate arrangement for FEA showing prospective crack location.

Consider an initial uncracked hole of radius R in a loaded plate as shown in Figure 1. For the corresponding hole with a crack of length X = A, the mode I stress intensity factor, K_I, can be calculated from the integral of the product of the weight function, *m(x,a),* and the transverse stress distribution, $\sigma_{YY}(x)$, in the uncracked plate perpendicular to the prospective crack plane. This is given by Bueckner [2] as:

$$K_I = \sqrt{W} \int_0^a \sigma_{YY}(x) m(x,a) dx \tag{1}$$

where W is a characteristic dimension, and for cracks emanating from a hole $W = R$. For the case of a symmetric crack emanating from a hole in a large plate it has been shown by Wu and Carlsson [3] that this weight function is given by:

$$m(x,a) = \frac{1}{\sqrt{2\pi a}} \sum_{i=1}^{3} \beta_i(a)(1 - \frac{x}{a})^{i-3/2} \tag{2}$$

Here a = A/R is the normalised crack length ratio, $\sigma_{YY}(x)$ is the stress in the uncracked body transverse to the crack path at the normalised distance x = X/R from hole edge, and $\beta_i(a)$ are coefficient values (i=1 to 3) taken from table 13.1 with N=2, Wu and Carlsson [3].

The cases resulting in compressive residual stresses are remote tension overload and hole cold expansion. Also considered is the case of a tensile residual stress field due to a compressive overload. Initially generic cases for a D6AC steel plate are considered. Stress intensity solutions are given for different crack sizes for different levels of overload which produce yield zones of different sizes. For both remote overload cases it is shown that once the crack length is the same or larger than the initial yield zone, the stress intensity factors are the same as for the case without the initial overload. However for the cold expanded hole case the beneficial reduction in stress intensity factor extends some distance outside the initial yield zone.

The results of a compression overload are given in Figure 2 and show that this leads to high tensile values of K_I ,and that for cracks longer than the initial plastic yielded zone the solution closely follows the curve for the case with no residual stress.

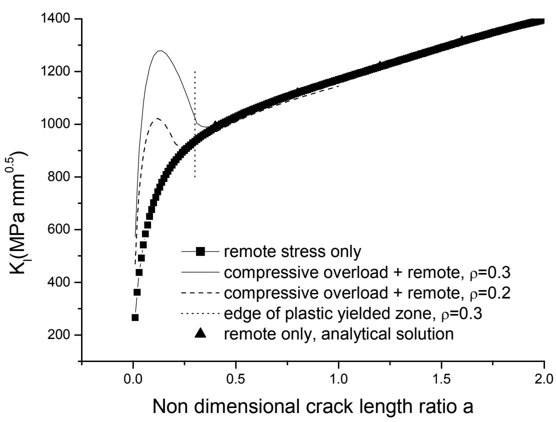
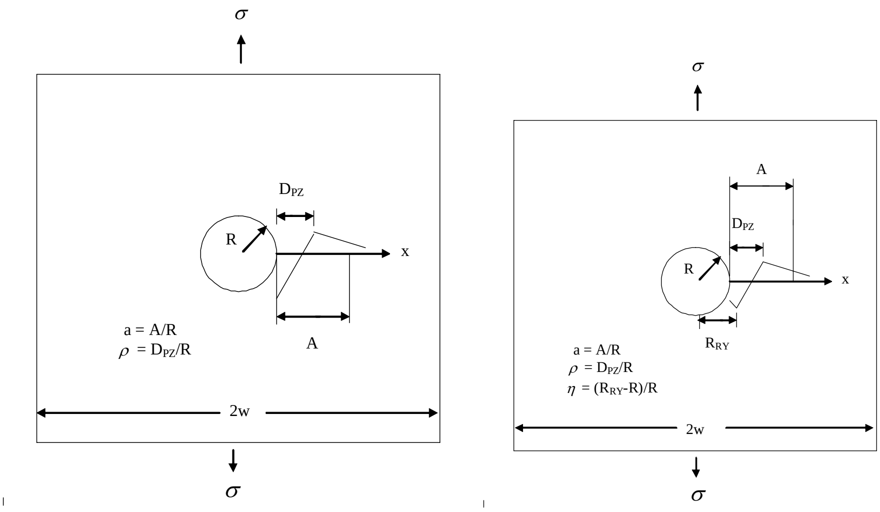


Figure 2: Stress intensity factor results due to remote compression overload for large D6AC plate, following overload, with subsequent remote stress of 200MPa.

The cold-expansion procedure involves pulling an oversize mandrel through the hole, where the radius of the mandrel is R_m. Shown in Figure 3 are the idealised yielding cases for the cold worked hole loaded in a finite width plate.



(a) (b)

Figure 3: Geometry and idealised yielding cases for cold worked hole in loaded finite width plate (a) no re-yielding following removal of mandrel, (b) with re-yielding following removal of mandrel

A comparison of K_I results between a tension overload case and a cold expanded hole case, both subjected to the same subsequent remote stress, is given in Figure 4. Both cases have an initial plastic yielded zone size of $\rho = 0.3$ and are for a hole in a large plate. It can be seen that the cold worked hole is more effective for the same plastic yielded zone size than the overload cases in minimising K_I. Notably, the residual stress due to cold expansion has a larger tail along a. Therefore, the effect of negative residual stress are not cancelled until a is large. In comparison, the residual stress due to an overload has a shorter tail. This is believed to be due to the fact that cold working affects the entire circumference of the hole while a tension overload effect is more localised. Hence the tension overload is only expected to have a beneficial effect for relatively small crack ratios.

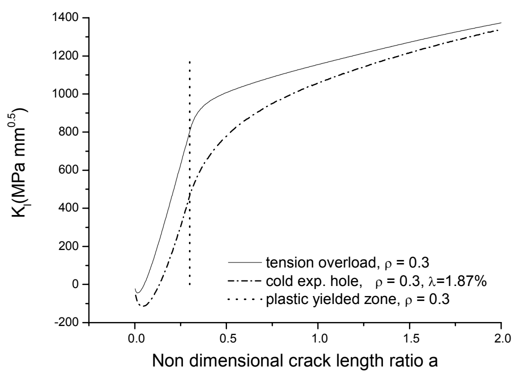


Figure 4: Comparison of stress intensity factors for large D6AC plate with remote tensile overload or hole cold working, with the same plastic zone size, and the same subsequent remote load.

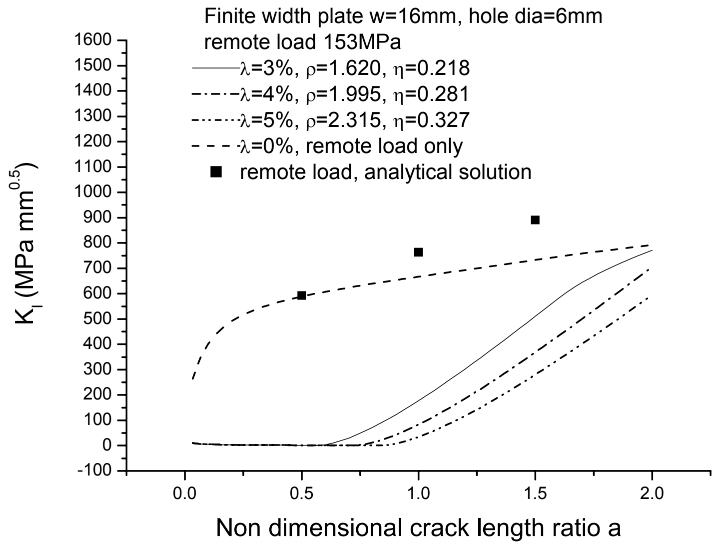


Figure 5: Stress intensity factor results for finite width Al. alloy C-130 test coupons with w/R=5.33 due to cold expansion and with subsequent remote stress of 153MPa.

The application of cold expansion to an Aluminium alloy fatigue test coupon representing a lower wing skin location in the C-130 is considered. It is shown that a much larger initial plastic zone size is achieved, Figure 5, by cold expansion resulting in an increased benefit in comparison to the D6AC steel example.

In summary, although not shown, if a fully opened crack is assumed then this may lead to potential inaccuracies in the case of a compressive residual stress field, which may lead to a partially closed crack. This effect reduces with increasing crack length. It is recommended that further work be undertaken to compare the weight function method to other numerical approaches where this effect can be accounted for, such as non-linear FEA or a more sophisticated weight function approach. For tensile residual stress and for remote tensile stresses, the weight function method is always valid. Also it has been shown that for the same plastic zone size, cold working is more effective than a tension overload in reducing *K*. Furthermore for cold expansion of the C-130 Aluminium wing skin it is shown that a much larger initial plastic zone size is achieved in comparison to D6AC steel. Finally it has also been shown for overloads that once the crack length is the same or larger than the initial yield zone size the stress intensity factors are the same as for the case without the initial overload.

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2.1.14 Contact Analysis Of Pin–Hole Interface Under Uniaxial Loading (W. Waldman, M. Opie and M. Heller [DSTO])

Finite element analysis work has been undertaken to study the effects of contact interaction between a neat-fit and a range of clearance-fit unloaded titanium alloy pins and a fastener hole in an aluminium alloy plate loaded in uniaxial tension (idealised as shown in Figure 1, with $S_y > 0$ and $S_x = 0$, and contact angle η). This work supports a fatigue coupon testing program that is being conducted for the validation of test interpretation activities associated with the BAE Systems Lead-In Fighter (LIF) Hawk full-scale fatigue test that is presently being undertaken by AVD. The aim is to gain a better understanding of the stress distributions that occur when fasteners are fitted in holes. Anecdotal evidence would seem to indicate that the crack initiation point at the hole boundary is offset by a small angle relative to the point of maximum stress when the hole is empty. Results have been obtained for linear-elastic pin–hole contact interactions, and these have been compared to the two-dimensional results obtained using a known analytical solution technique. Because it involves considerable numerical computations, this technique has been implemented in a Fortran 90 software program, and it permits the study of contact pressures and tangential stresses for pin–hole contact interactions where the elastic material properties of the pin and plate material are different, and where the applied loading can be biaxial or uniaxial in nature. Both 2D and 3D finite element analyses of the pin–hole contact problem have been conducted, for the case of an unloaded neat-fit titanium pin in a uniaxially-loaded aluminium plate. Excellent agreement has been obtained with the computed 2D analytical results (see Figure 2). Both the 2D and 3D linear elastic results indicate that the peak tangential stress around the hole occurs at an angle of approximately θ = 20° under uniaxial tension loading, rather than at θ = 0° as for an empty hole (see Figures 2 and 3). Further finite element analyses involving elasto–plastic material behaviour are now being conducted to study the effects of material plasticity and cyclic loading on the stress distribution associated with pin–hole contact problems.

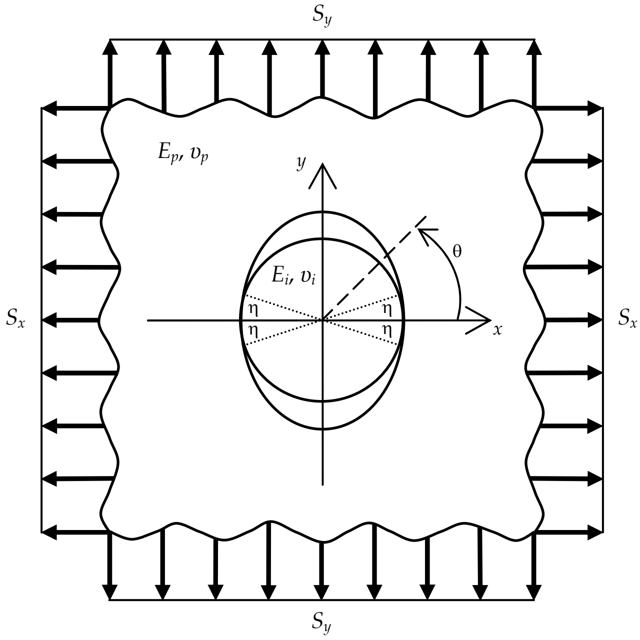

Figure 1: Geometrical configuration of an infinite 2D plate loaded by stresses S_x and S_y at infinity showing the contact angle η between the plate and the circular pin insert.

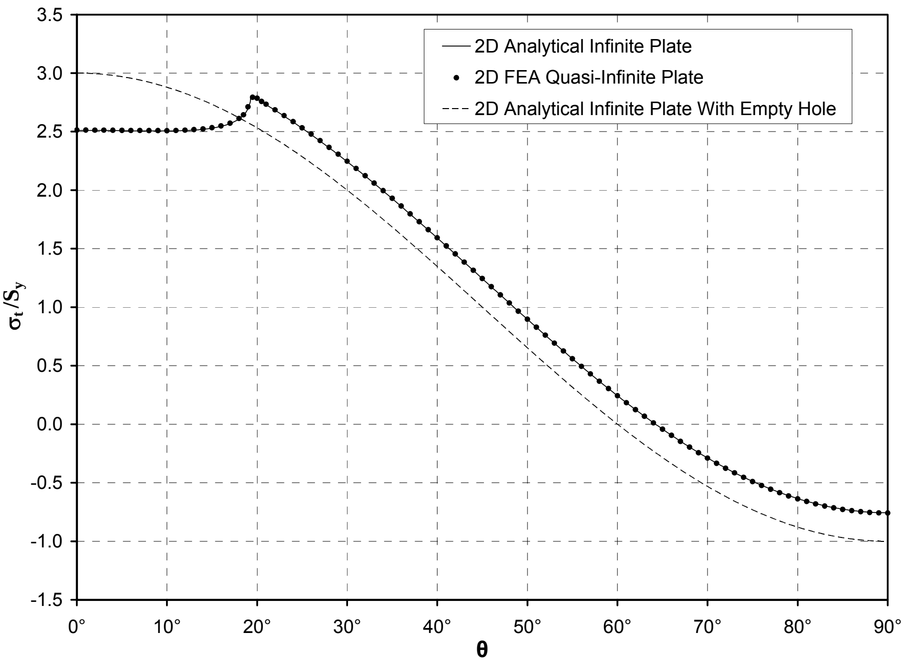

Figure 2: Variation of normalised tangential stress around a hole in a uniaxially-loaded square aluminium plate with a neat-fit titanium pin. Results obtained from 2D finite element analysis of a quasi-infinite plate and the corresponding analytical infinite-plate solution.

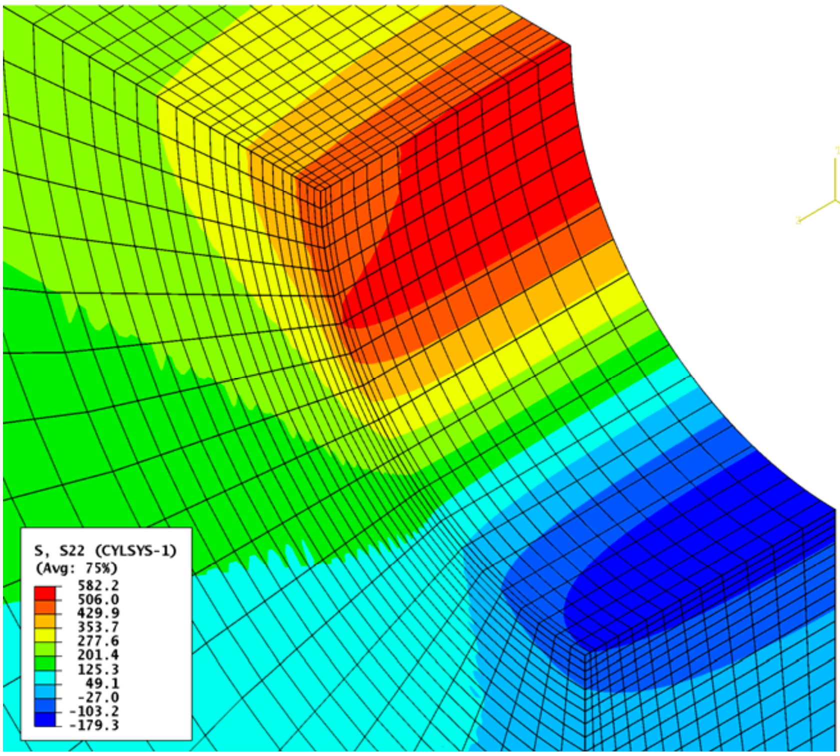

Figure 3: Contour plot of tangential stress in the vicinity of the hole for an aluminium plate loaded in uniaxial tension with a neat-fit titanium pin insert in the hole, as obtained from a 3D 1/8-symmetry linear-elastic finite element analysis.

2.1.15 The critical importance of correctly characterising fatigue crack growth rates in the threshold region (K. Walker and S. Barter, [DSTO])

The ability to accurately understand and characterise the behaviour of fatigue cracks in the near threshold regime is crucial for assuring structural integrity for high strength metallic aircraft structures. This is because the majority of the total fatigue life is often spent in this regime. In the past, attempts to model crack growth initiating from small surface breaking constituent particles present in 7050-T7451 aluminium alloy coupons tested under a fighter aircraft Variable Amplitude (VA) loading spectrum, using fracture mechanics based methods have produced poor results. See for an example Figure 11 re-produced from [1], which compares quantitative fractography (QF) results post test with calibrated (to the test lives) predictions. The shape of the growth curves is incorrect, and in order to get the life predictions close to those of the coupons, a relatively large initial crack size was required. The worst aspect is that the analytical predictions are non-conservative. This means that they under estimate the crack growth rate while the crack is small and as a result over estimate the life (if a realistic value of initial crack size were to be used). Conversely, in the Non-Destructive Inspection (NDI) region (> about 1 mm) they over estimate the crack growth forcing short inspection intervals (where required), which is expensive and wasteful.

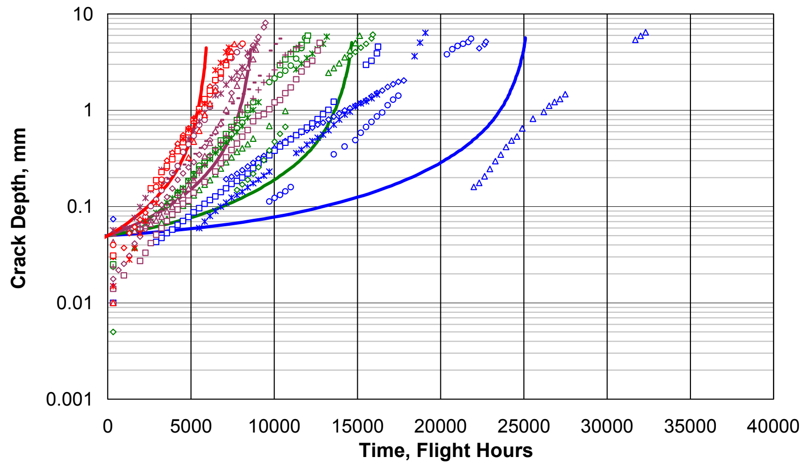

Figure 1: Example (from [1]) of poor analytical results compared with experiment. The experimental data are shown as symbols, and the analyses as solid curves. The four sets are for tests under the same load sequence at different stress scaling levels.

Recent work at DSTO has shown that the principal reason for the poor analytical results is that the baseline material crack growth rate characterisation data as collected by ASTM Standard E647 long crack tests when used in predictive programs on a cycle-by-cycle basis are inadequate. The generation of Constant Amplitude (CA) crack growth rate data in the threshold and near threshold regime is usually performed using the constant R load reduction method [2]. The load reduction method however has been demonstrated to produce higher thresholds and lower crack growth rates in the near-threshold regime than steady-state CA data on a number of materials (two aluminium alloys, two titanium alloys and a superalloy), especially those that develop rough and torturous crack surfaces [3-7].

The recent work has included a comparison of rate data determined by two separate, independent methods. The methods were applied to 7050-T7451 Alloy. Both methods avoid problems known to exist in methods such as the ASTM Standard E647 load reduction technique [2].

The first alternative method (Method 1) involves establishing the baseline CA crack growth rates by testing under compression-compression pre-cracking, which avoids the problems of remote closure which are known to compromise the results under conventional load reduction testing [8]. The second method (Method 2) involves growing natural cracks from typical discontinuities in un-notched specimens. The specimens are subjected to specially designed simple load sequences which consist of bands of CA loading with shorter bands of cycles with a changed R inserted periodically. It has been found [9] that the changes in R affect the crack path in such a way that individual CA bands can be identified on the fracture surface when it is examined under a high powered microscope (optical and/or Scanning Electron Microscope; SEM) and accurately measured. This process, known as QF, is carried out on the fractures surface produced by the testing. It produces CA growth rates for naturally occurring small cracks since these bands can be identified at very small crack depths.

Crack growth analyses were performed using the FASTRAN code [10] using rate data generated by Method s 1 and 2. The analyses were performed for cases including simple coupons and full scale fighter aircraft centre section bulkhead test. An example of the improved analyses compared with the experimental results for the bulkhead case is shown in Figure2. Further details of this work are to be presented at the 26th ICAF Symposium [11]. The results from this work offer an improved potential to maximise the life of aircraft structures while maintaining adequate levels of safety and therefore contribute to a reduction in the capability and environmental impact of premature fatigue related retirement and/or replacement.

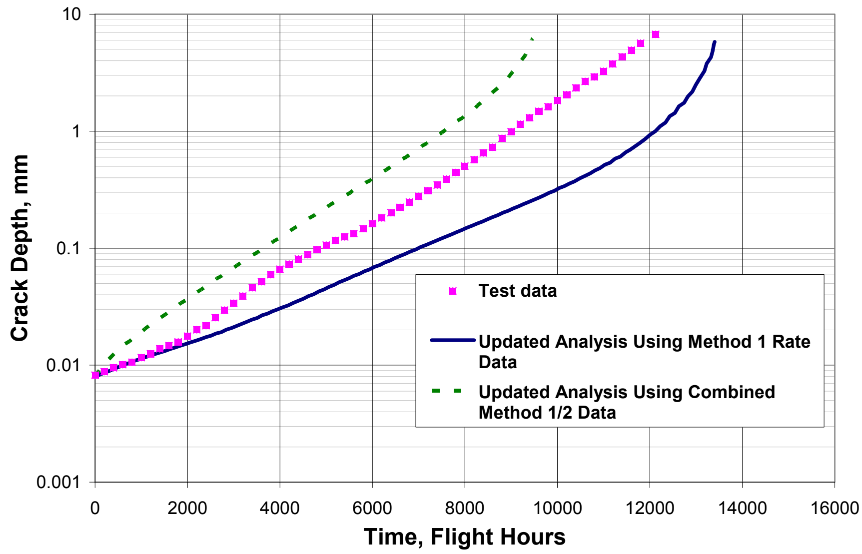

Figure 2: Comparison between test data and updated analysis showing greatly improved result

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2.1.16 Fatigue characterisation of surface preparation effects produced by bonded repair of aluminium alloy 7050-T7451 (S. Barter [DSTO])

A coupon fatigue test program has been completed to characterise the effect that the various stages of surface treatment have on the fatigue performance of aluminium alloy 7050-T7451 (6.35mm thick) when a bonded repair is applied. Two types of doubler were applied; aluminium and boron/epoxy composite, and two types of initial surface condition were used prior to surface preparation. The loading was by a representative fighter wing root bending moment spectrum at a peak load level that gave one equivalent lifetime for the lowest life coupon. In each case, the strain peak in the coupons remained the same regardless of the surface treatment or the application of a doubler; strains were measured by gauges attached to the sides of the hourglass style flat 6.35mm thick by 12mm wide (in the wasted section) coupons. In addition to total life measurement, Quantitative fractography (QF) of the block-by-block crack growth was measured to understand better the effects of the surface treatments.

It was found that the overriding fatigue life decider for these coupons and loading conditions was the nature, and the density of, the crack initiating discontinuities on the surface. As the density was reduced by the subsequent surface treatments and the bonded doublers reduced the effect of the environment under the patches, further reducing the number of discontinuities that developed cracking, the life improved. The reduced number of discontinuities resulted in those that still exist being in locations that are not as favourable for cracking as would be more likely if there were a higher density of discontinuities. The improvement was found to be almost exclusively associated with slower crack growth of the cracks and those that failed the coupons were small, typically less than 1mm in depth. The graphs, shown in Figure 1 of the crack growth for two conditions: etched and boron/epoxy doubler applied, indicate that the crack growth for the boron patched coupons is notably slower up to about 1mm in depth. The etching was carried out to mimic a similar surface condition found on the primary structure of a RAAF fighter aircraft.

The density effect was so strong with these coupons that even with an increase in the applied loading of 15% to account for the residual tension, produced by the hot bonding of the boron doublers to the aluminium (due to thermal miss-match) did not reduce the total lives of the coupons back to the level found in the etched coupons. The higher strain test results are shown in Figure 2.

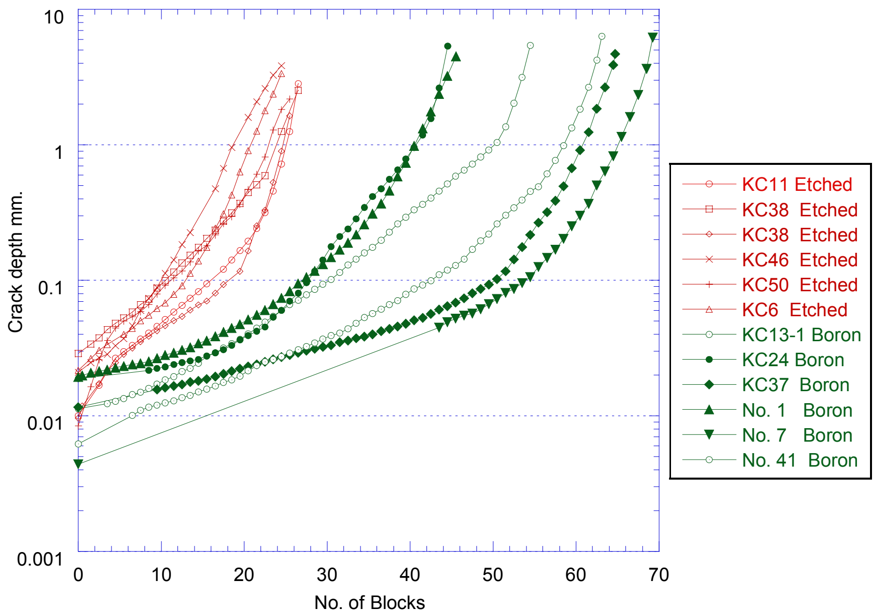

Figure 1: Crack growth curves for coupons tested with the same measured strains in the aluminium alloy. One set of graphs shows the results form coupons that were etched and the second for coupons that had boron doublers applied.

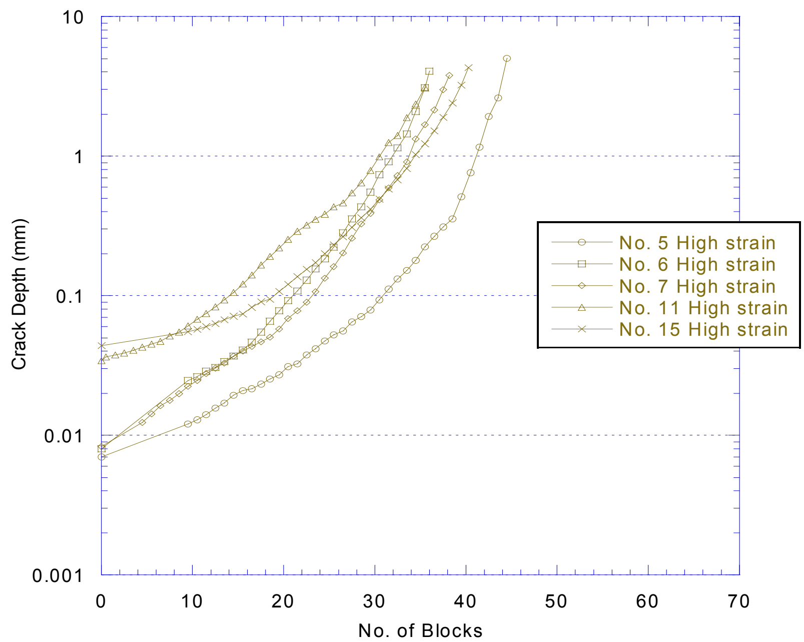

Figure 2: Crack growth curves for coupons tested with boron doublers applied loaded at strain levels about 15 % higher than those shown in Figure 1.

2.1.17 Investigations on Critical Load Cases for Robust and Efficient Shape Optimisations (X. Yu, [DSTO])

Rework shape optimisation has been developed in DSTO in the last decade as an effective approach to minimise peak stresses for life extension of critical airframe features [1]. An important issue here is the sensitivity of peak stresses to varying conditions including load variations. To address this issue, a robust shape optimisation was developed, incorporating multiple load cases which, for example, were defined by discretising continuous perturbation of stress orientations [1]. The present study is instigated by the desire to reduce the number of load cases while retaining the quality of optimal results [2]. Based on a linear elastic assumption, the engineering problem of selecting critical load cases is transposed into a mathematical problem of finding a convex hull that encloses all possible load cases. The critical load cases are found to be those located at the vertexes of the convex hull. All other load cases are redundant and can be omitted without affecting optimal results. Based on a geometrical interpretation of the convex hull, mathematical formulas are derived to estimate errors induced by omitting a critical load case. Two sample applications are presented. One application is on constructing load cases for a robust shape optimisation under biaxial tension with continuous variations of stress ratio and orientation, see Figure 1. As illustrated in Figure 1(c), the only load cases need to be considered are those relating to the two extreme stress ratios. Previously, intermediate stress ratios were considered without knowing their redundancy. The other application is on estimating errors induced by discretising continuous perturbation of stress orientations, see Figure 2. A comparison with FE results given in Figure 2(c) clearly demonstrates the upper-bound nature of the theoretical error estimations. With confidence in the upper limit of errors, the use of unnecessarily fine intervals of stress orientations can be avoided without adversely affecting optimal results.

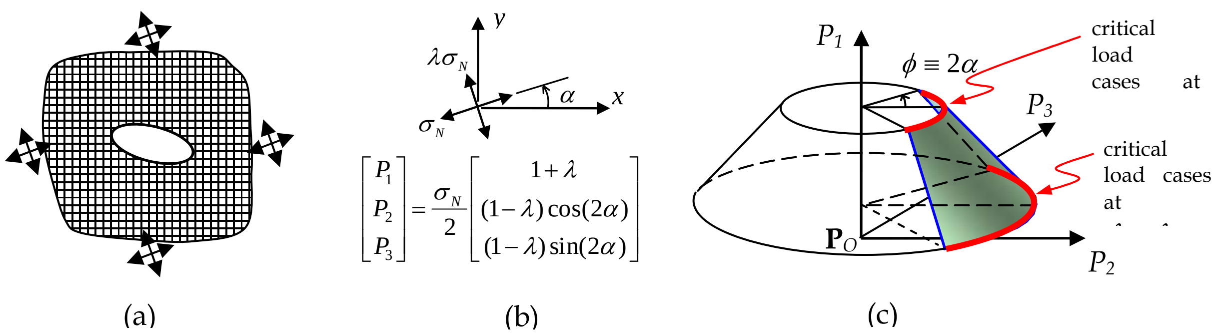

Figure 1: Constructing critical load cases for a centre-holed 2-D plate under remote biaxial stresses. (a) Illustration of the plate under remote stresses; (b) Continuous variations of stress ratio and orientation, represented by general load vector; (c) Critical load cases identified from collection of all load cases (the latter is shown as shaded area).

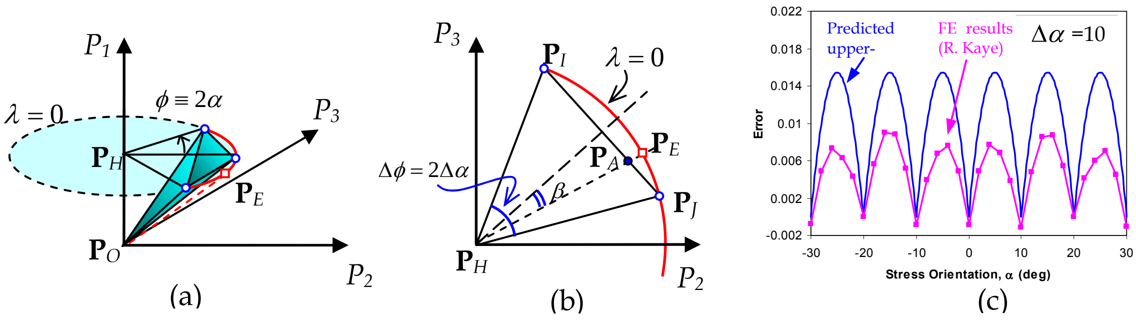

Figure 2: Errors induced by discretising continuous load cases. (a) Plot of load cases in a 3-D load space; (b) Load cases projected onto a 2-D sub-space; (c) Comparison with FE results showing upper-bound nature of the theoretical error estimation.

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2.1.18 Numerical & Experimental Evaluations of the Geometrical (Beta) Factor for F/A-18 High Kt and Low Kt coupon specimens (X. Yu, M. McDonald and W. Hu, [DSTO])

Usually the geometry (beta) factor, as used in LEFM-based crack growth rate models, is considered sufficient to characterise stress concentration factor (Kt) effects. However this was not observed on an F/A-18 coupon test comparison of Low and High-Kt results, particularly for small cracks (< 1 mm). A more detailed assessment of the beta factor solution was thus conducted, which revealed that the beta solution was significantly sensitive to: i) the assumed shape of the crack when it was small [see Figure 1(a)], and ii) higher stress cases where material yielding occurred, which created a significant residual compressive stress field [see Figure 1(b)], that affected the crack growth while it was small. These effects, when approximately accounted for, significantly, though not completely, improved the correlation between the Low and High-Kt specimen results [1]. This work forms a part of the effort toward fully understanding some short crack growth rate behaviours that traditional LEFM-based tools find difficult to account for.

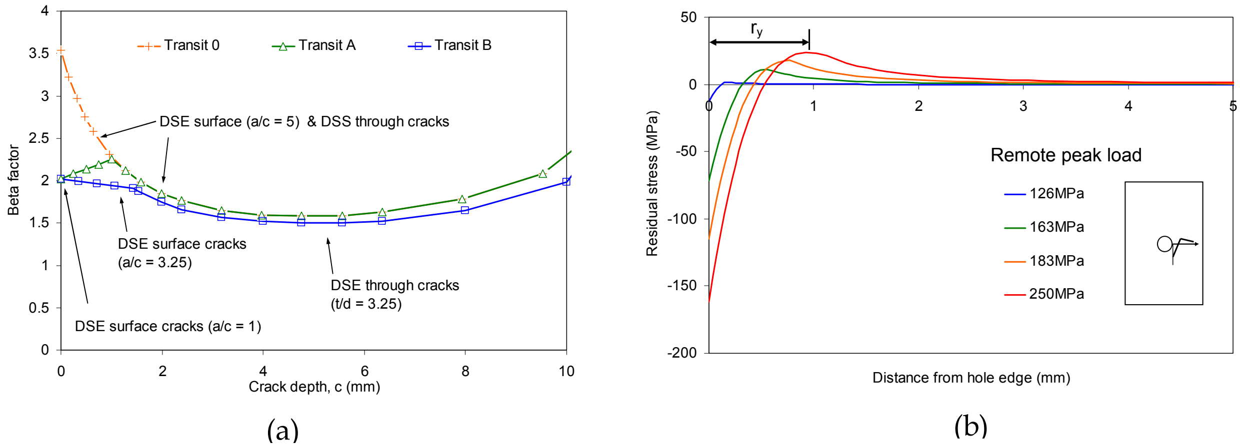

Figure 3: Plots showing sources that affect beta factor estimation. (a) Aspect ratio of DSE (double symmetry elliptical) surface crack in high Kt coupon (plate with a centre hole); (b) Residual stress induced by remote peak load in high Kt coupon.

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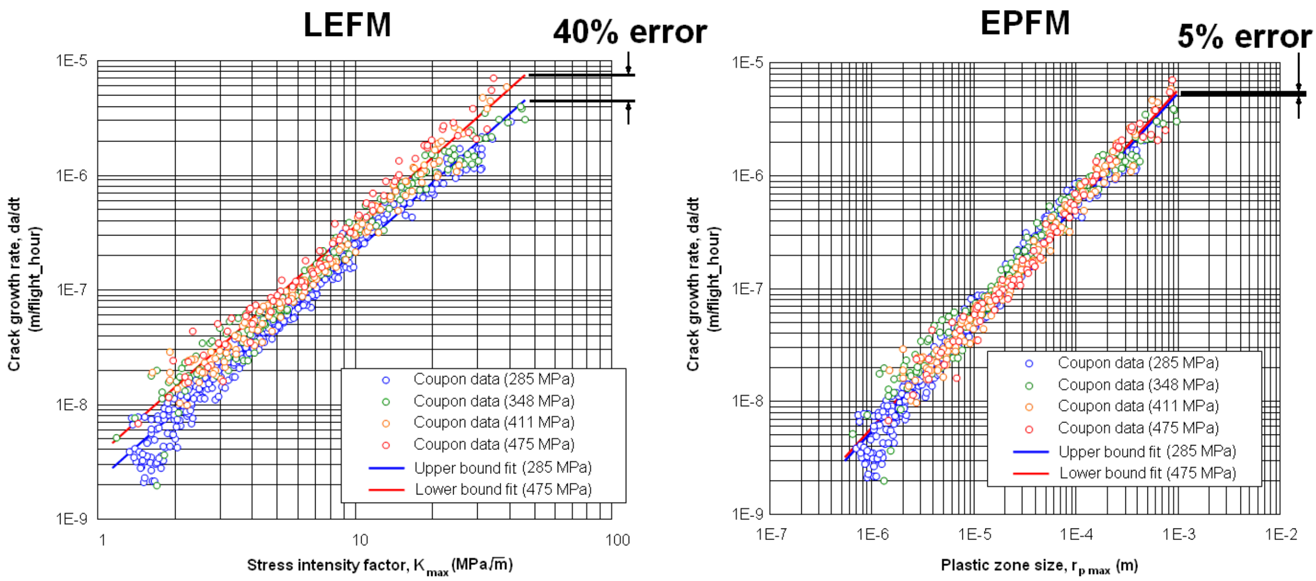
2.1.19 Elastic-Plastic Fracture Mechanics for improving the accuracy of metal fatigue life analyses (M. McDonald, [DSTO])

Combat aircraft design typically allows the airframe structure to be more highly stressed in order to achieve a low-weight/highly-manoeuvrable air vehicle. The effect that this has on the fatigue crack growth rates can be significant, yet it can be difficult to predict accurately. This is most likely because crack growth analysis methods are usually based on linear elastic fracture mechanics (LEFM), which are typically limited to low stress scenarios. The development of elastic-plastic fracture mechanics (EPFM) is anticipated to help provide a useful solution to this problem. This study used EPFM finite element modelling of AA7050-T7451 to determine if the plastic zone size, r_p, could provide better correlation of crack growth rates when considering stress levels close to the yield strength of the material (~60% to ~100% of yield in this case) compared to the traditional LEFM stress intensity factor, K. Crack growth rate da/dt data were obtained from variable-amplitude (fighter wing bending moment) spectrum short-crack fatigue tests [1]. An indication of the correlation ability was gained by simply plotting this da/dt data against the respective correlating parameters at the maximum spectrum value, as shown in Figure 1. Lines of best fit through the upper and lower stress level data give an estimate of the potential analysis error over this stress range, and hence indicate the potential accuracy improvement from moving from LEFM to EPFM models – in this case the error appeared to be reduced from 40% to 5%. This study, which was based on a low stress concentration factor test specimens (Kt ~1.05), is considered to be a first step toward the analysis of higher stress concentrating features, like fillets and fastener holes, which are often the cause for limiting the life of metallic airframes.



(a) *(b)*

Figure 1: AA7050-T7451 da/dt data for a variable-amplitude (fighter wing bending moment) spectrum plotted against: (a) an LEFM correlating parameter, stress intensity factor, K, and; (b) an EPFM correlating parameter, plastic zone size, r_p – indicating the potential analysis accuracy improvement of EPFM compared to LEFM.

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2.2 FULL SCALE TEST ACTIVITIES

2.2.1 Structural Integrity Support to the Royal Australian Air Force C-130J-30 Hercules (R. Ogden, L. Meadows & D. Hartley [DSTO]).

As presented in previous Australian National reviews, a significant program of work is currently underway to support structural integrity management of the Royal Australian Air Force (RAAF) C-130J-30 Hercules. Entering RAAF service in 1999, the fleet of twelve stretched versions of the C-130J aircraft (C-130J-30) encountered issues during the initial military structural airworthiness certification compliance finding activities. Of particular significance, the aircraft within RAAF service was found to be non compliant with respect to the Durability and Life Of Type (LOT) certification basis elements of the RAAF Certification Structural Design Standard (CSDSTD). As a result of this non compliance, the RAAF entered into a collaborative program with the Royal Air Force (RAF) who had similarly encountered certification issues. This collaborative test program encompassing a full-scale C-130J Wing Fatigue Test (WFT) and C-130J Operational Loads Monitoring (OLM) program was aimed at primarily addressing this certification non compliance, by providing robust and representative data to support both RAF and RAAF fleet management and structural airworthiness out to respective aircraft Planned Withdrawal Dates (PWD).

The remainder of this section provides an update on the major activities that are occurring in support of this wider program of work. In particular with the collaborative WFT program now up and running at the Marshall Aerospace facilities in Cambridge, UK, (see Figure 1) the focus of DSTO work has switched from support to test development to that of gearing up for interpretation of WFT results. In the context of this program, interpretation refers to the process of taking a specific test result (i.e. known crack on test) and reading this across to equivalent damage under RAAF representative spectrum conditions, (i.e. assuming that the test spectrum is not 100% representative of RAAF usage). Over the last two years, (with WFT testing commencing in March 2009), DSTO efforts have been specifically focused on developing the necessary tools and processes required for subsequent WFT interpretation. These are summarized as follows:

- Review applicability of the extant RAAF C-130J-30 CSDSTD.
- 7075-T7351 crack growth & total life coupon testing to support tool verification.
- Analytical crack growth (FASTRAN) and crack initiation (FAMS) tool verification and validation.
- Development of parametric 'Beta' solutions for C-130J-30 Damage Tolerance Analysis (DTA) locations.
- Development of RAAF unique loads spectrum (utilizing RAAF OLM aircraft)
- Structural Health Monitoring System (SHMS) verification and validation (utilizing RAAF OLM aircraft).
- Development of C-130J WFT Information Management System (IMS)

Several key elements of this work are singled out below for further detailed description. These include the extensive program of work to generate robust geometric stress intensity factor solutions (Beta Solutions) for use in subsequent Linear Elastic Fracture Mechanics (LEFM) interpretation of the respective test failures and the coupon test program designed to support the calibration/verification of adopted analytical crack initiation and crack growth tools.

The Beta solution development work has reached a mature position recently with activities in this area due to wrap up over the next six months. The program aim was to provide coverage of all extant C-130 DTA locations with the added ability to easily and efficiently tailor these solutions to cover other test arisings or subtle crack path variations. For this reason all solutions have been developed in a fully parametric fashion, (enabling geometry and associated loading assumptions to be tailored to the specific situation). As presented in Figure 2, solution development included three options. The first and highest fidelity involves the development of 3D Finite Element Models (FEM). Utilizing advanced computational fracture mechanics code (ESRD StressCheck), highly detailed models were developed for more complex geometries and to aid in assessment of lower fidelity solutions. Some key features of this approach included (i) adaptive mesh refinement via variable-order polynomial elements to achieve solution convergence, (ii) iterative nonlinear modelling of contact, and (iii) parametric geometry definitions including crack shape. The parametric nature of the modelling also allows multiple crack configurations to be considered at one location, to achieve solutions significantly more quickly than prior modelling approaches. Option 2 essentially involved the use of handbook solutions available in the literature, whilst Option 3 supplements the extant handbook solutions by developing 2D and 3D generic FEM to address solution aspects that were considered deficient, (based upon comparison with higher fidelity 3D FEM solutions). In particular the generic FEM developed by DSTO included those presented in Figure 3, (also including rationale for development).

The coupon test program (Aluminium 7075-T7351) aimed at supporting the test interpretation activities has also reached a mature position recently with sufficient data now gathered to address the issue of verifying or calibrating the adopted crack growth and crack initiation tools. Playing a significant role in reading across test to representative RAAF fleet damage, the ‘crack closure’ based CG tool (FASTRAN) in particular has undergone extensive review and correlation against the coupon test data. The process of effectively calibrating the tool relies upon “fine tuning” constraint parameters inherent within the FASTRAN model. This calibration process was carried out for a range of spectra locations (i.e. wing lower surface, wing upper surface, outer wing etc) and then additionally checked by performing a second series of coupon tests aimed at addressing the ability of the calibrated model to replicate coupon test results across a range of spectra severities. An example is shown in Figure 4 where it is shown that the calibrated model performed well against a range of spectra severities, (i.e. tactical, logistics, training and combined mission mix or WFT spectrum).

An additional activity highlighted in the bullet points above, relates to the use of the RAAF OLM aircraft to engage in an extensive evaluation of the RAAF C-130J-30 SHMS. Developed by Lockheed Martin, the SHMS is not currently utilized by the RAAF due to lack of validation evidence. As a result, a DSTO program of work is underway to utilize the fully calibrated OLM system and conduct SHMS validation activities. It is expected that this work will be completed within the next 12 months.

With the full-scale WFT now achieving in excess of 14,000 test hours and DSTO test interpretation tool development and validation nearing completion, future DSTO activities will focus on interpreting, exploiting and integrating the results of the WFT into the RAAF ASIP and addressing the RAAF C-130J-30 certification issues. This will initially involve the development of a test interpretation strategy that is compliant with the adopted CSDSTD, of which a key element will be the formulation of a generic test interpretation methodology for reviewing test findings. Another critical issue will be the prediction and analysis of widespread fatigue damage on the test article, and how this will be used to determine the structural life-of-type for airframe components (e.g. centre wing). DSTO will also continue to provide advice in support of all elements of the RAAF C-130J-30 Aircraft Structural Integrity Program, (ASIP), in particular the development of an enhanced usage and fatigue monitoring system for the RAAF C-130J-30, that is both compliant with the CSDSTD and compatible with information generated from the structural testing program.

Figure 1: Collaborative RAF/RAAF C-130J Wing Fatigue Test (Marshall Aerospace, Cambridge, UK)

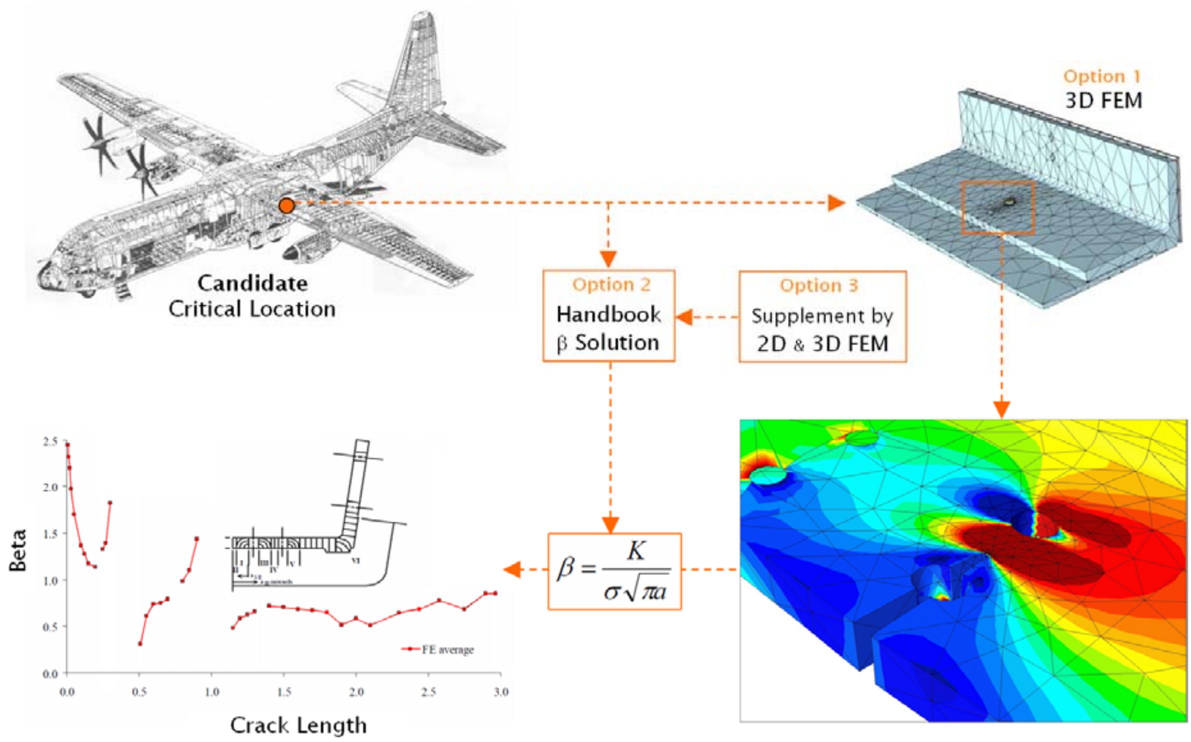


Figure 2: Representation of Options for Beta Solution Development

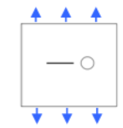
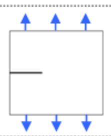
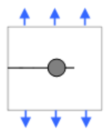
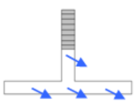
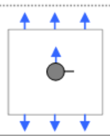
Model		Description	Rationale for Development
	2D	Through Crack Approaching Hole	Requirement driven by limitations in beta solutions within extant literature, particularly as crack approaches the hole.
	2D	Edge Crack Finite Width Plate	Requirement driven by need to (i) create new and useful data, (ii) understand the limiting conditions for the Harris equation, and (iii) ascertain the bounding cases and understand the effect of bending restraint for 3D models.
	2D	Crack from Fastener After Ligament Failure	Existing solutions required the compounding of three handbook solutions. The accuracy of this method was initially unknown and the generic FEM resulted in increased accuracy.
	3D	Cracked Stringer Approaching Junction	The handbook solutions are considered overly simplistic and in need of refinement using 3D modeling.
	3D	Corner Crack Loaded Hole	The current literature does not provide accurate beta values for small crack lengths. Additionally solutions are only applicable for loaded holes in which out of plane bending is unrestrained.

Figure 3: 2D and 3D FEM Developed to Supplement Existing Handbook Beta Solutions

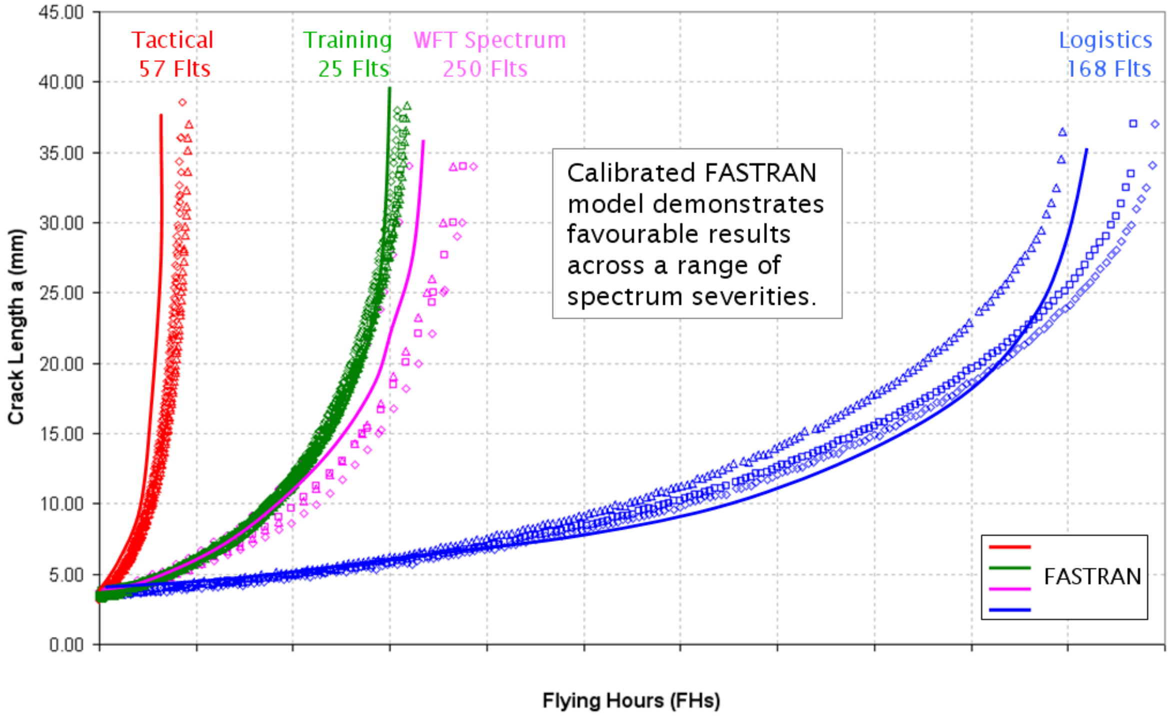


Figure 4: Wing Lower Surface Spectra Coupon Test (7075-T7351) vs FASTRAN Analytical Crack Growth Prediction (Various Spectra Severities)

2.2.2 F/A-18 Flaw IdeNtification through the Application of Loads (FINAL) Program, (G. Swanton, [DSTO])

In the two years since ICAF 2009, DSTO's Flaw IdeNtification through the Application of Loads (FINAL) program has continued to test ex-service "classic" F/A-18 wing attachment bulkheads, also known as centre barrels (CBs) [1]. As of December 2010, a total of 14 CBs had been tested. Five of the test articles have been RAAF CBs, and are part of the 10 CBs that were replaced during the RAAF's Hornet Upgrade Program. Earlier test articles have been made available to DSTO from the USN and Canadian Forces CB replacement programs. The number of quantitative fractography (QF) reports stands at approximately 120 with some 650 individual cracks examined to date.

One of the main reasons for the continuation of the FINAL program has been the requirement to extend the safe life of discrete locations which could not be extended previously using the typical lifing methodology. Specifically, two locations on the Y453 bulkhead have had their safe life extended using in-service crack growth data collected by QF, combined with DSTO's novel lead crack concept methodology [2, 3]. These locations had a high payoff value since any modification work would have been costly due to the development of the repairs and the inaccessibility of the region (requiring removal of the fuel bladder tank). This has allowed decision makers to withdraw them from the list of locations that needed to be modified, allowing the desired safe life to be achieved at less expense.

Furthermore, another location is still the subject of intensive investigation: the lower Y488 bulkhead commonly known as the "kick point". At this location there are two different configurations present in the RAAF fleet. The early configuration has a bonded aluminium doubler on a straight flange, and the latter configuration has a tapered flange (without a doubler). In the former case, there have been several instances of disbonded doublers found in the fleet. This negates the stress reduction benefit the doubler was designed to provide at the critical point, as disclosed by the original equipment manufacturer's (OEM's) full-scale fatigue test. The counterpoint to this is that those doublers that remain fully bonded create a stress concentration at their ends, which results in an increase in the growth rate of cracking that can occur in this location (in particular the inboard end of the doubler), leading to a marginal life for this location. For the later configuration kick point, the thinnest part of the taper also causes a point of stress concentration which in FINAL has also produced cracking with a marginal life. To address this issue, DSTO has developed an extended length doubler, as well as refined the surface preparation and installation process [3], to cover the entire kick point lower flange. Full-scale proof-of-concept validation is currently in progress using the FINAL CB11 test article.

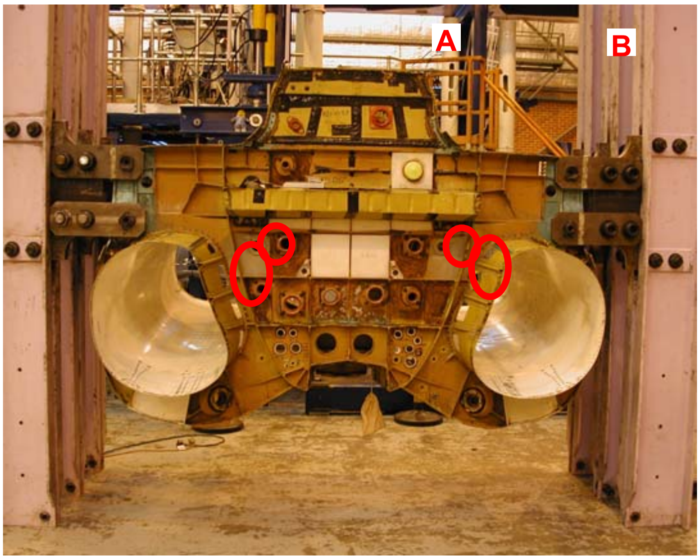


Figure 1: Locations (circled) on the Y453 bulkhead where a safe life extension was possible using DSTO's novel lead crack concept methodology. View looking aft.

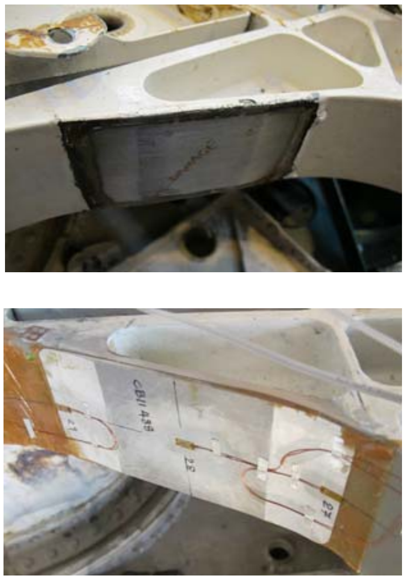


Figure 2: (Top) The OEM bonded aluminium doubler developed to address potential cracking on early configuration kick points (Point "A"). Point "B" indicates the point of stress concentration caused by the inboard end of the doubler on the kick point lower flange. (Bottom) The DSTO developed doubler which essentially extends the inboard end of the OEM doubler to cover the entire kick point pocket lower flange area.

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2.2.3 F/A-18A/B Hornet Outer Wing StAtic Testing (HOWSAT), (W. Foster, [DSTO])

Attempts to develop a through life management strategy for the outer wing based on direct test interpretation of previous full scale fatigue tests including FT93L (OEM test) and FT245 (IFOSTP) have proven unsuccessful. Loading at the outer wing representative of RAAF usage and configuration was not achieved on any of these fatigue tests. A strict application of DEFSTAN 00-970 would therefore require the use of severe life reduction factors, which may lead to unacceptably short Safe Life Limits and Safety-By-Inspection program intervals. Most of the critical structure in the outer wing requires substantial disassembly to allow inspection, including removal of the upper wing skin, which comes with significant costs and impact to aircraft availability. However these tests have been useful in demonstrating the likely fail-safety of the wing structure. Both the test wings sustained loads above limit with damage including severed spars. It is thought that the excess load carrying capacity comes from the carbon fibre/epoxy upper and lower wing skins.

A new representative fatigue test will take at least a few years to input sufficient cyclic loads to make useful conclusions about the actual fatigue life of the outer wing. Considering that the RAAF F/A-18A/B fleet is currently projected to retire in the 2018-2020 timeframe, the results would be available too late to be useful for the Aircraft Structural Integrity Program. Therefore, another method for lifing the outer wing outside of the guidance of the present Certified Structural Design Standard (DEFSTAN 00-970) has had to be found.

F/A-18A/B Hornet Outer Wing StAtic Testing (HOWSAT) [1] is the DSTO component static test program intended to substantiate fail-safe assumptions about the outer wing.

HOWSAT will attempt to quantify the strength of the outer wing torque box under RAAF representative (and factored) loading with severe representative damage (i.e. simulated cracks) introduced to the underlying spars and ribs. The results of this program are intended to substantiate a fail safe through life management strategy, expected to be the most cost effective option for the RAAF.

The status of HOWSAT is that the test rig has been designed, manufactured and assembled (Figure 1). The hydraulic system, control system and data acquisition system are nearing completion and the first test article has been instrumented. Representative damage maps have been created using data from previous tests and in-service data. These maps will assist in identifying critical locations and damage induction sizes for the test. The load cases to apply have been established and the actuator loads have been developed. Current scheduling has static loading being applied in the first half of 2011.

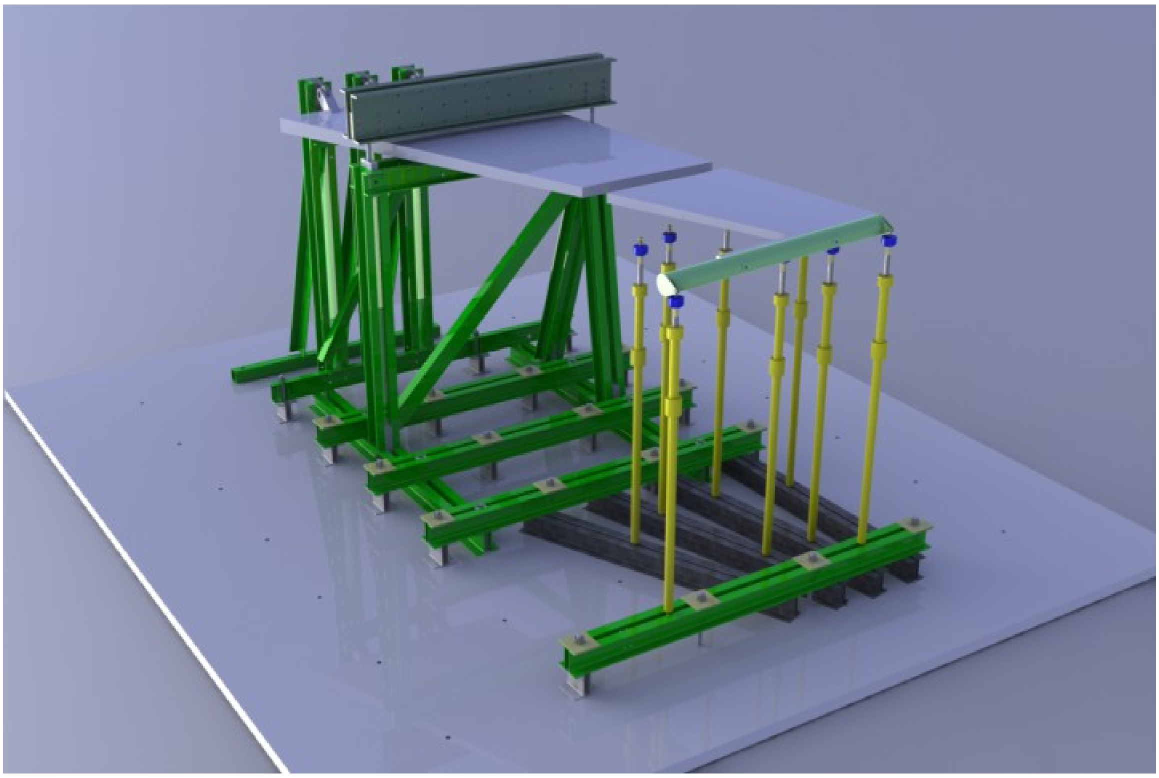

Figure 1: Design of HOWSAT test rig

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2.3 IN-SERVICE STRUCTURAL INTEGRITY MANAEMENT

2.3.1 Sustaining Aircraft Structural Integrity Risk and Value Management (Eric Wilson, [Australian Defence Force Academy, University of New South Wales])

Numerous examples exist in civil and military aviation that highlight the way systemic problems can combine with aircraft maintenance issues to rapidly impact the safety or economic viability of a fleet. In this research program, the technology management and safety regulation and implementation challenge are investigated by drawing on lessons from local and international events. Factors influencing organisational growth and commercial arrangements (eg alliances, outsourcing, global supply chain management etc), together with human factors considerations in safety management systems such as individual participation in formal and informal knowledge networks, are drawn together to establish the foundation for improved management and regulatory approaches to aerospace structure technologies throughout their product life cycle. The research involves development of a risk management framework based on knowledge management principles focused on helping meet the combined need of satisfying continuing structural integrity requirements whilst maximising the value obtainable from continuing to operate a fleet of aerospace platforms for the duration of their product life cycle. The framework developed in the research program builds on the premise that no system can guarantee safety at all times hence, an organisation needs to operate with a continuous state of unease, or 'mindfulness', and be on the alert for the possibility of system failure particularly during the organisation's evolution to meet changing market dynamics. The framework recognises the three discrete perspectives needing to be addressed simultaneously:

- cost of ownership of the technology
- risk issues associated with the evolving needs for supporting the technology
- value to be gained from continuing to employ the technology.
-

It is the way these perspectives combine in a particular technology and operator's circumstances that will govern the overall effort and investment needing to be applied to meet the business objectives against which the technology is being retained in-service.

Balancing these perspectives in a commercial and highly competitive environment invariably leads to the need for strategies to address the ever present contradictory forces between growth and productivity; the latter also at risk of being increasingly impacted by evolving technical regulatory requirements. Accordingly, organisations seeking to operate new and ageing aircraft need an integrated strategy based approach that addresses in part:

- the technical risks they face,
- an investment management program to maximize the value associated with retaining and supporting aircraft throughout their life cycle,
- a knowledge management strategy tailored to their business strategy
- people strategies for development and retention of the technical competencies they will need to maintain their evolving fleet configuration

- developing and sustaining a high performance organizational culture and its associated mindfulness

The research involves the application of cognitive thinking methodology to define the framework that addresses the interactions of the various factors associated with the organisational and technology life cycle. Figure 1 provides an example of how the research has been applied in the development of strategies for an Aircraft Structural Integrity Program.

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM
STRATEGY MAP

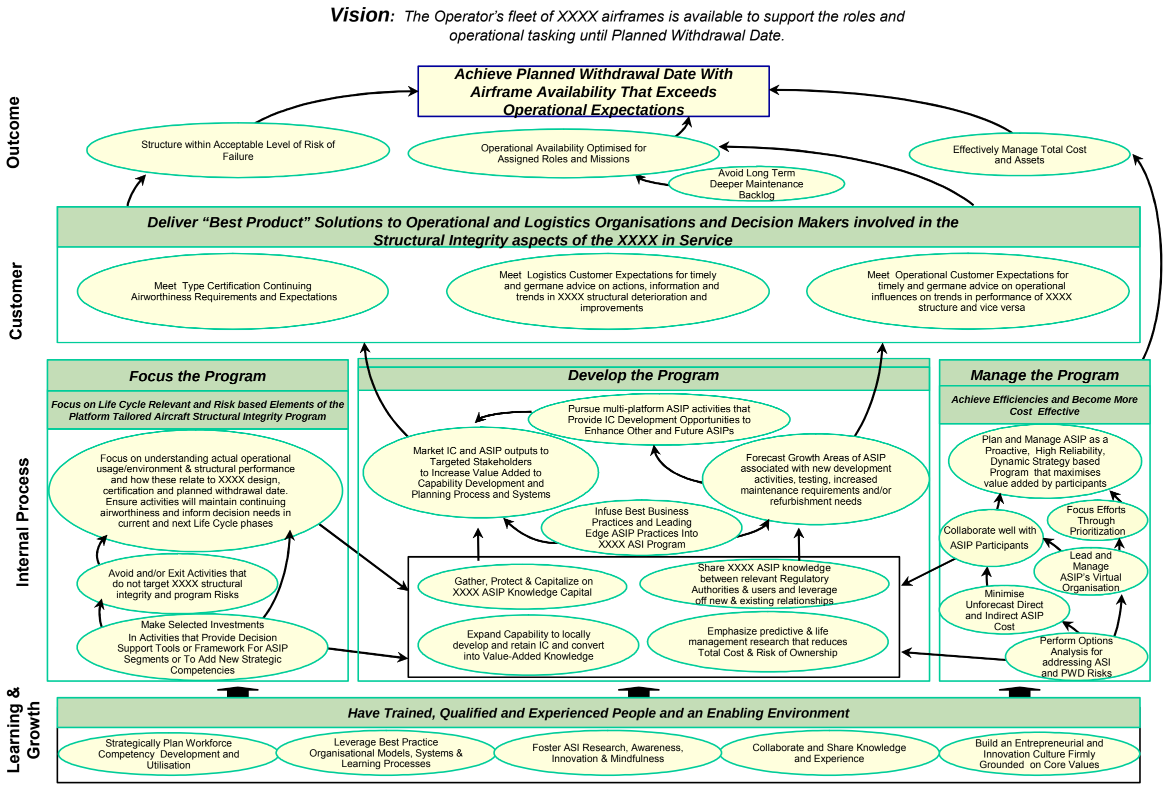

Figure 1: Risk and Value Management Strategies for ASI Program

2.3.2 Computational approaches for the development of improved beta factor solutions for C-130J-30 DTA locations (Evans R, Heller M, Burchill M [DSTO] and Gravina R, Clarke A, Rock C [QinetiQ AeroStructures])

For the C-130J-30 Hercules aircraft in service with the Royal Australian Air Force (RAAF), the airframe is currently being managed on a safety-by-inspection approach based on the manufacturer's analysis, [1]. This analysis will shortly be replaced by an indigenous analysis, calibrated by the full scale fatigue test of the C-130J-30 wing that is being conducted by the RAAF in concert with the Royal Air Force. DSTO is working in conjunction with local industry to provide specialised advice; since it is desirable for

Australia to have an independent life-assessment capability for this aircraft. Determination of beta factors (which account for the effect of geometry on crack growth) is one element of a significant Australian effort, including full-scale wing fatigue testing, operational loads monitoring, component and coupon fatigue testing, and fatigue life prediction modelling. These ongoing activities at DSTO also contribute to and support a wider program of international work as members of The Technical Cooperation Program (TTCP).

One key input in generating crack-growth curves on which to base the inspection intervals is crack-front stress intensity factors; typically referred to as beta (or geometry) factors when given in normalised form. Accurate estimates of the variation of beta factor as a function of crack length are needed. For the wing, 33 Damage Tolerance Analysis (DTA) locations have been identified. Some typical DTA configurations which are relevant include; (i) cracked integral wing skin stiffeners and panels, (ii) cracked panels with fastened hat stringers, and (iii) L-section spar caps. For each location, cracking is typically split into multiple phases, and for each phase beta factors are determined for multiple crack sizes. In previous similar work undertaken at DSTO or by industry partners, handbook methods have been mostly used to derive beta factors. However in recent years an improved capability has been developed based on variable polynomial (p) order finite element modelling (FEM) using the StressCheck© code, see [2]. This approach offers the potential to obtain more accurate results, in a more efficient manner, than usual h-version finite element analysis (FEA). Such a computational approach also offers a reference to assess the quality of the standard handbook methods.

Reference 3 is an excellent summary of the work undertaken by DSTO (up until early 2010) on the determination of beta factors in the context of structural through-life support for the RAAF C-130J-30 Hercules aircraft. This beta factor analysis has given insight into the relative merits of FEA and handbook approaches, along with recommendations for future C-130J-30 work. In this paper, beta factors are initially determined using accurate fully parametric, three dimensional, polynomial order, finite element analysis (FEA). Detailed 3D parametric FEA has provided excellent solutions for specific DTA locations (4 off). However, the sensitivity of predicted results to assumed loading conditions has been highlighted, including pin loading effects. The approach is relatively time consuming and is not necessarily recommended for all locations. It is expected that this approach will be used for fatigue critical locations (as identified on full scale testing or fleet experience), if the expected accuracy of handbook solutions is not considered appropriate for the lifing calculations. For a limited set of relevant Original Equipment Manufacturer (OEM) results available for earlier models of the C130, there is also good agreement. In Figure 1 the DTA location CW-5C with assumed crack growth phases is shown, along with the solid FE model used in the analysis. Figure 2a shows a typical mesh around a phase II crack. The example beta factor variations along the crack boundary are shown in Figure 2b.

Beta factor results based on handbook methods have been determined with and without fastener loading, for many of the 33 DTA locations (this work is continuing). It is being completed in parametric form via Excel spreadsheets (refer to [4] and [5]). The accuracy of some of the handbook solutions have been explored in detail. Where possible, results have been compared to the detailed 3D FEA, and this has highlighted cases where there are deficiencies in certain crack-growth phases due to accuracy or range of applicability. For example, greatest inaccuracies occur using handbook solutions for certain crack phases when there is multiple compounding of 2D solutions to approximate a realistic 3D location. It is realised that OEM's may have more accurate proprietary handbook solutions as compared to those in the open literature. Figure 3 shows handbook solutions compared to the parametric FEA results for each crack phase for one of the hat-stiffened DTA geometries analysed (location CW-5C).

To augment the above handbook approach, new generic beta factor solutions have been obtained via parametric FEA, and include cases of; (i) edge crack in a finite-width plate, (ii) crack in an integral stiffener approaching a junction, (iii) crack approaching a circular hole (refer to Figure 4) and (iv) crack from a hole after ligament failure. Some of these have been used to improve the handbook solutions, while the use of others is pending. It is planned that they will also be used for other locations on the airframe, such as the fuselage. Some initial test cases have indicated that there can be a significant inaccuracy in handbook solutions for pin-loaded fastener holes with corner cracks or through cracks. Also for open holes with corner cracks there appears to be discrepancies in the distributions of beta factor around the crack front. Generic FEA work is currently underway to clarify these issues.

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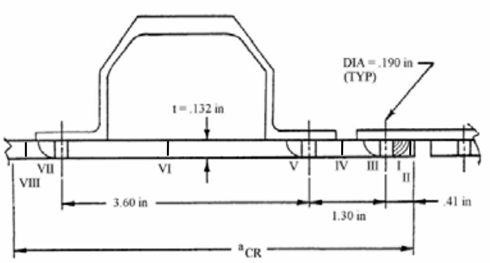
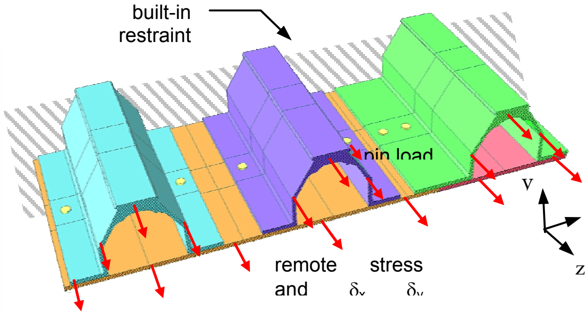
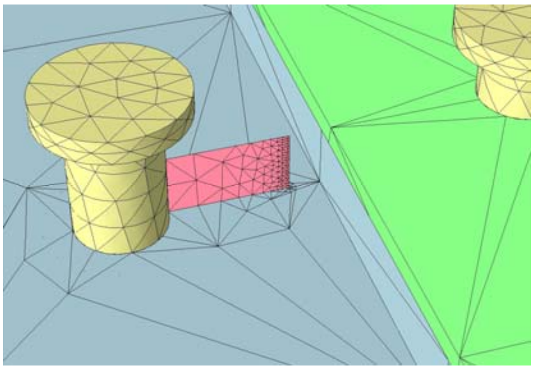


(a)



(b)

Figure 1: Parametric FEA of CW-5C; (a) location geometry and assumed crack growth phases, (b) solid model with loading and restraints.



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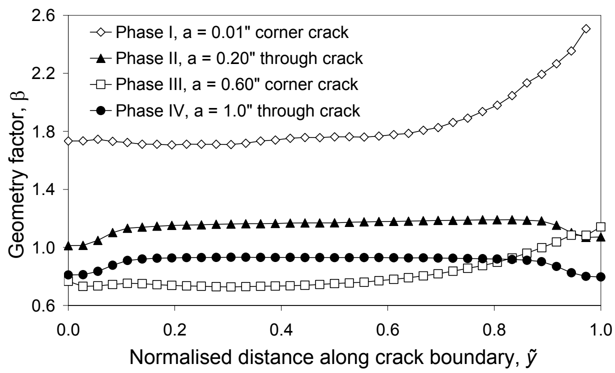

(b)

Figure 2: Parametric FEA of CW-5C; (a) localised detail for phase II crack (interior free surfaces), (b) typical beta-factor distribution along crack boundary for remote-loading case.

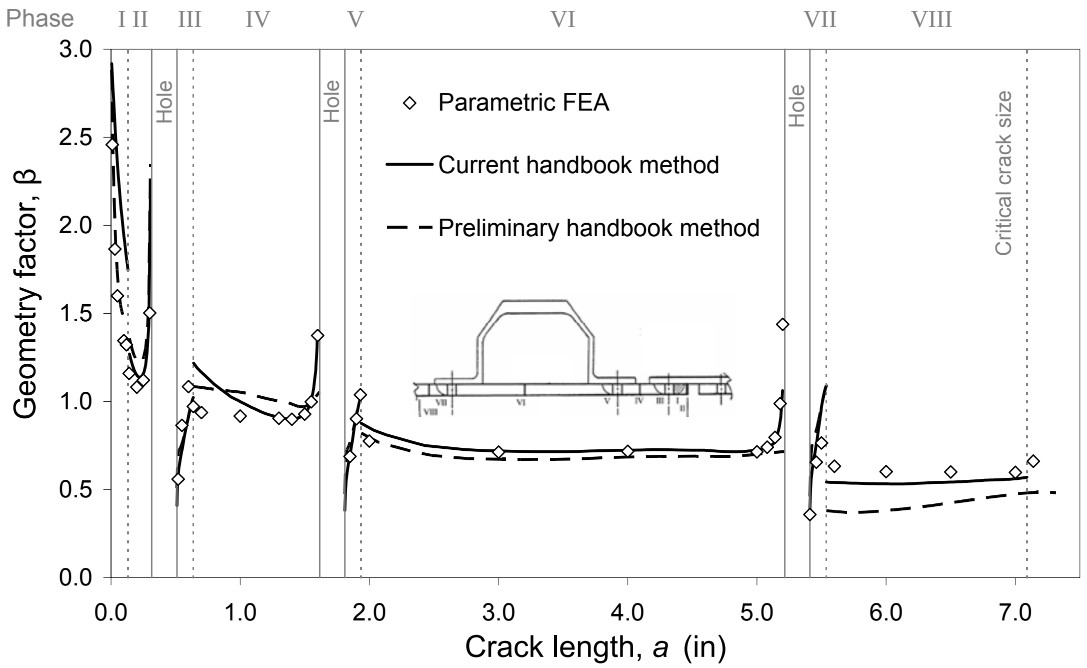

Figure 3: Typical handbook results and comparison with parametric 3D FEA (location CW-5C example, combined pin and remote loading).

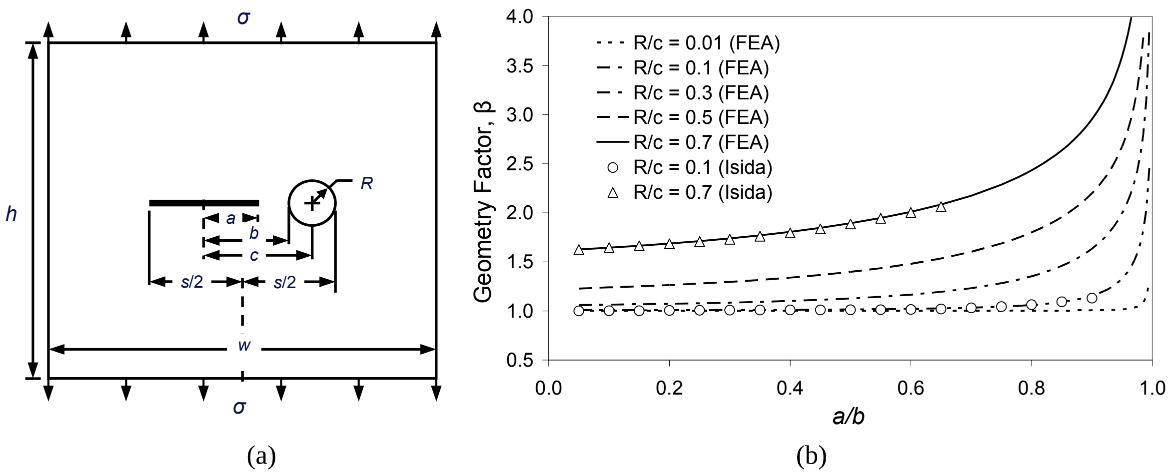

Figure 4: Generic FEA of crack approaching a circular hole; (a) geometry and notation, (b) typical results including comparison with Isida [6].

2.3.3 Remote Synthesis of Loads on Helicopter Rotating Components using Linear Regression (J. Vine, X. Yu, DSTO)

Applying individual usage tracking to fatigue-critical dynamic components on helicopters can lead to reductions in maintenance costs; however, difficulties associated with directly monitoring these components have led to investigations of alternative methods to determine their load-time history.

This research effort continues earlier work [1] in which a number of methods (linear regression, load path analysis, statistical analysis) were investigated to synthesize the load-time history of a main rotor pitch link from measurements made on stationary components. The work used data from 42 level flight runs obtained from a United States Air Force (USAF) and Australian Defence Force (ADF) Black Hawk Flight Strain Survey conducted in 2000.

The current work is focused on improving the linear regression methods previously developed [1] by improving the overall accuracy and reducing the number of stationary component measurements required as inputs. The emphasis has been on improving the method of linear regression in the azimuth domain.

The most significant changes to the earlier method include the implementation of a new technique for optimizing the selection of input data channels and the inclusion of flight parameter data in addition to strain measurement data.

The optimization technique performs regressions on all azimuth locations concurrently, selecting only those data channels with the most positive effect on the accuracy of the regression. This minimizes the number of input data channels required and ensures the input data for all azimuth locations is a common set; greatly improving the practicality of the method. Regression results of 2 azimuth locations using the same input channels are shown in Figure 1.

The flight parameter data provides additional information within the regression analysis which improves the accuracy of the synthesized load-time history. As a result, the effects associated with the existence of reactionless force components (which had previously prevented completely synthesizing the load of a component above the swashplate from strain measurements below the swashplate [1]) were reduced.

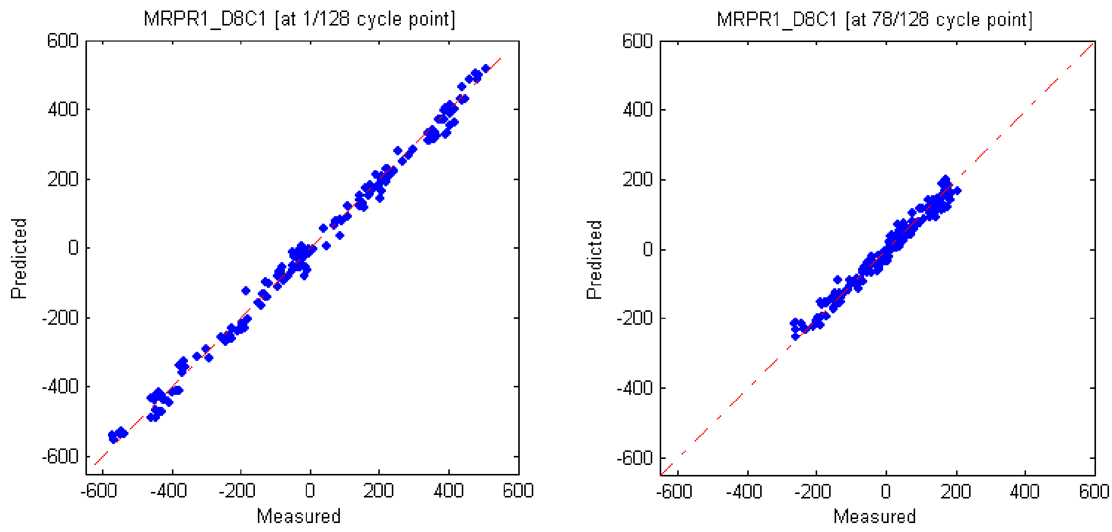

Figure 1: Regression of the Main Rotor Pitch Link loads at 2 azimuth locations using the same 8 stationary component strains and 2 flight parameters as inputs.

REFERENCE

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2.3.4 The lead fatigue crack concept for aircraft structural integrity, (S.A Barter and L. Molent, [DSTO])

Over many years of quantitative fractographic examination of fatigue cracking from in-service and full-scale fatigue tests of metallic airframe components, it has been consistently observed that the fastest growing cracks grow at an approximately exponential rate. These crack growth observations range from the initiation of cracks and their early growth from a few micrometers through to many millimetres in length. It appears that these lead cracks commence growing shortly after an airframe is introduced to the loading environment. Furthermore, these cracks usually initiate from production-induced or, less frequently, inherent material discontinuities. Based on these two observations, an aircraft lifing methodology that is based on the results of fatigue testing programs utilising the lead crack concept has been developed and implemented as an additional tool in the determination of aircraft component fatigue lives in several Royal Australian Air Force (RAAF) fleet types.

The features of lead cracks, should one occur[1], are;

1) That they commence growing soon after the aircraft is introduced into service;

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[1] Careful and often unrealistic (of service structure) surface preparation in test coupons can limit the numbers of cracks that can grow and therefore limit the possibility of a crack growing in the optimal location. This usually results in longer lives due to slower than possible early crack growth.

2) Irrespective of local geometry, they grow approximately exponentially with time (i.e. log *a* (the crack depth or length) versus linear life or cycles) if:

a. little error is made when assessing the effective crack-like size of the fatigue-initiating defect or discontinuity. For example, an error in underestimating the size of the crack-like effectiveness of the initiating discontinuity will lead to a small departure from exponential for a small period near the commencement of crack growth.
b. the crack does not grow into an area of significant change in the component's geometry, particularly if the crack depth is small in comparison to the component's thickness/width either before or after the change in section;
c. no significant load shedding occurs (i.e. the crack is not unloaded as the part loses its stiffness and sheds load to surrounding members, or grows towards a neutral axis due to loading by bending;
d. the crack does not encounter a significantly changing stress field i.e. grows into or from a region having residual stresses;
e. the crack is not retarded by very occasional very high loads (usually in excess of 1.2 x the peak load in the spectrum; and
f. the small fraction of life involved in fast fracture or tearing at the end of the life of a fatigue crack is ignored. This usually represents a small portion of the total life as tearing normally starts on the highest load just before failure. Additionally, the requirement of the 1.2DLL as the failure case for the Design Standard used by the RAAF would tend to negate this period of crack growth from any lifing calculation.

Within the bounds of the above, observations of the formation, growth and failure resulting from these cracks have led to the following generalisations:

a. the geometrical factor β (which depends on the ratio of the crack length to width of the specimen) does not appear to influence the nature of the crack growth significantly as may be expected. For low K_t features the majority of the life is spent when the crack is physically small so that the β does not change much. But even when the crack starts at an open hole and the β is changing rapidly, the lead cracks still appear to grow in an approximately exponential manner. This is not to say that there is no geometry influence, cracks from open holes grow at greater acceleration rates than from low K_t details at the same net section stresses [1]. The expected influence of the high β on the shape of the CG curve appears minimal even when the crack is spending a considerable amount of its life in the region so affected. The reasons for this observation are the subject of further research;
b. typical (mean) initial discontinuities for typical fighter aircraft metallic materials (e.g. high strength aluminium alloys) are approximately equivalent to a 0.01 mm deep fatigue crack. In general a 0.01mm deep crack representing the discontinuity is a good starting point for CG assessment;
c. cracks may also grow exponentially within residual stress fields, with an increased or decreased rate compared to a residual stress free material, depending on the sign of the residual stress;
d. should large loads that affect CG by retardation occur periodically throughout the life, then on the average exponential CG may still result, although a rare overload may lead to a significant departure (or step-change) in the local CG rate; and

e. whilst the critical crack size is easily calculated, it has been observed that for typical fighter aircraft metallic materials the critical crack depth is typically about 10 mm for highly stressed areas[2].

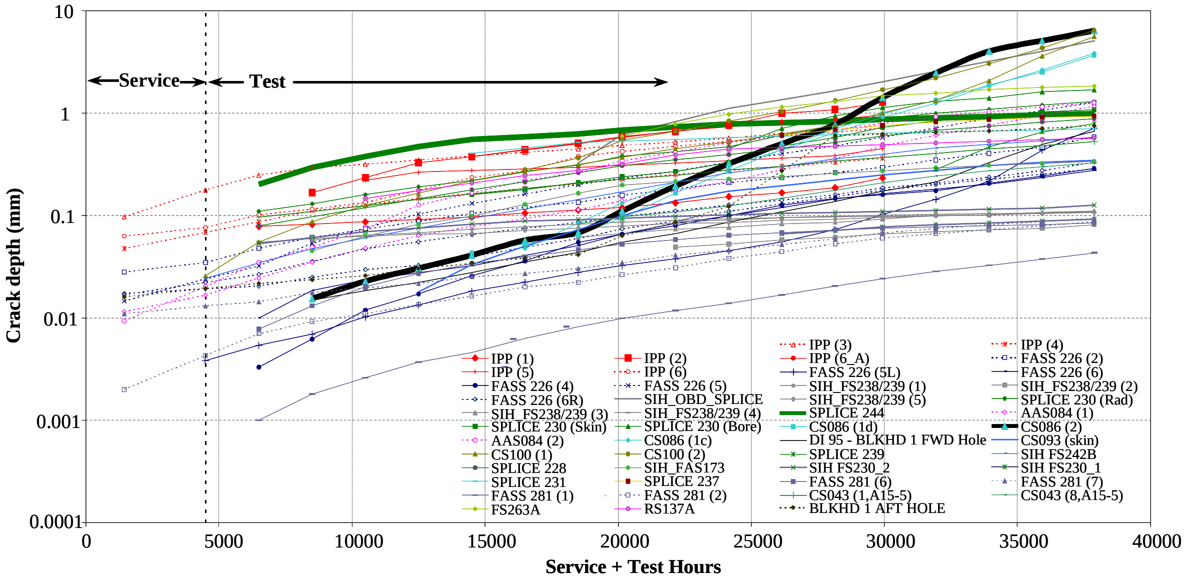

Figure 1: Sample of FCG curves from different locations in the aluminium alloy (AA) 2024-T851 lower wing skin of an F-111 test article removed from service [2]. CS = central spar, BLKHD = bulkhead, FASS = Forward Auxiliary Spar Station inches, IPP = Inner Pivoting Pylon, RS = Rear Spar.

While the lead crack may not occur at every detail in an evenly stressed airframe component due to the size and density of discontinuities at the highly stressed details, there are usually many such details i.e. fastener holes, and indeed many aircraft in the fleet resulting in at least some details growing lead cracks. For example, the crack growth curves shown in Figure 1 are from fastener holes in a wing tested under spectrum loading where the holes were all similarly stressed. One of the cracks grows at an approximately exponential rate that is faster than the others (i.e. the lead crack) and it is this CG rate that would be used in the analysis of the wings life. More details of this approach, with examples may be found in references [2]-[6].

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2.3.5 In Situ Structural Health Monitoring using Acousto-Ultrasonics and Optical Fibre Sensors (S. Galea, N. Rajic, C. Davis, K. Tsoi, C. Rosalie and I. Powlesland [DSTO])

A significant through life support cost driver for military aircraft is time-based non-destructive inspections for corrosion or defects, often in inaccessible regions, which are costly and time-consuming. The introduction of Condition Based Maintenance (CBM) approaches using in situ Structural Health Monitoring (iSHM) systems has the potential to reduce costs and increase aircraft availability. iSHM facilitates the early detection (and possibly characterisation) of defects in high value Defence assets thus providing a critical underpinning technology for efforts to improve the efficiency of contemporary structural integrity management practice.
Stress-wave or acousto-ultrasonic (AU) based iSHM techniques have the potential to offer an effective broad area, low sensor density, low-cost and reliable means of non-destructive inspection [1]. This technique is based on elastic waves produced by a fixed source, typically using piezoelectric materials. These waves are sensed at separate receiving locations where variations from the baseline response are used to identify damage within the wave path. The Defence Science and Technology Organisation (DSTO) has a program of work aimed at developing autonomous, robust, reliable iSHM systems, based on AU, with the specific aim of retro-fitment to existing aircraft. This program has focused on the demonstration of the feasibility of AU based iSHM systems on realistic aircraft components, the development of an AU modelling capability, the assessment of robustness of piezoelectric ceramic transducers and strategies for ensuring system robustness.

AU feasibility studies

One activity within this project has focused on developing and demonstrating the technology on a complex structure, viz. the F-111 lower wing skin (LWS) at the fuel transfer groove (FTG) at FASS281.28 (Fwd Auxiliary Spar Station) see Figure 1. In this case cracks initiate on the inside surface of the wing skin, due to a stress concentration at the FTG, and propagate chordwise along the FTG for some distance before penetrating the wing skin. It is an awkward problem for conventional NDI. F-111 LWS FTG structurallydetailed test coupons were used to assess the feasibility of detecting cracks in this location using AU. Positioning the piezoelectric elements optimally is a fundamental requirement, since these transducers are permanently attached. Ideally the AU response of this structure would be numerically modelled using FE techniques in order to determine the best sensor locations. However, predictions furnished by FE models have been found to be unreliable so transducer placement was experimentally determined using laser vibrometry (LV). After attaching the piezoceramic Lead Ziconate Titinate (PZT) transducers, the coupon was subjected to an F-111 representative loading spectrum, with peak strains up to 3400 □□, and was interrogated at piezoelectric drive frequencies ranging from 300 to 1200 kHz. Experimental studies on the F-111 LWS FTG structural-detailed test coupon showed a significant decrease in the peak response of the AU wave packet with increasing spectrum loading [2, 3,4], see Figure 2. This is more clearly indicated in the evolution of the normalised AU peak signal strength, with a significant reduction in energy after about 1800 simulated flight hours indicating the detection of the fatigue crack, which was independently verified. Thus, the AU technique proved successful in detecting the presence of the crack in the F-111.

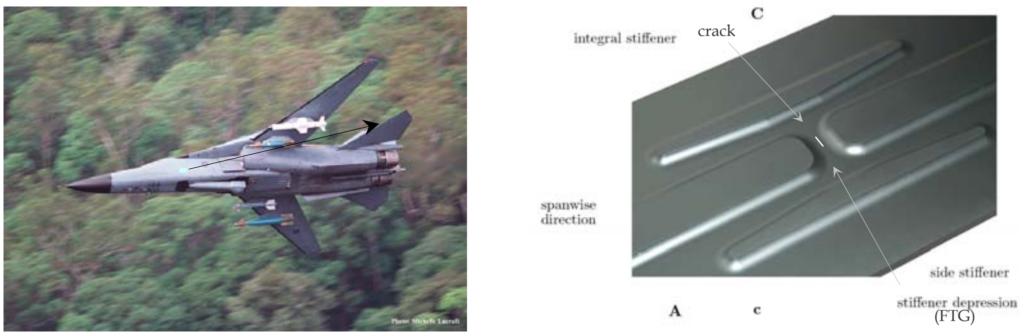


Figure 1: Location of cracking in F-111 LWS (left). Internal view of F-111 LWS around the FTG (right)

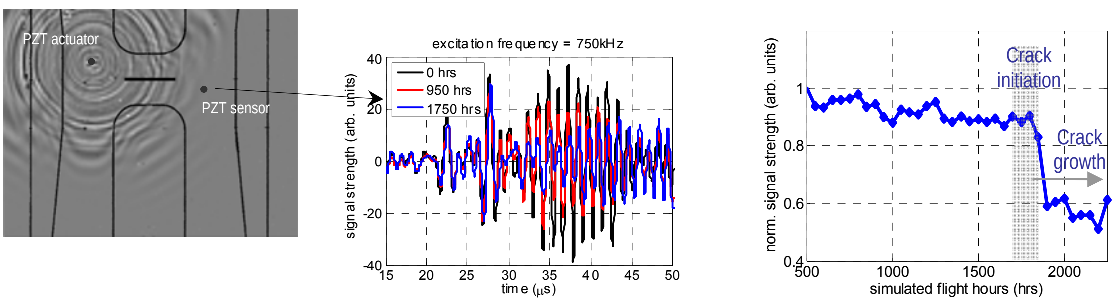


Figure 2: Typical AU response from PZT actuator on the F-111 LWS panel: left –'snapshot'of surface displacements; middle – time history response of PZT sensor, right – variation in PZT sensor signal strength with increasing load spectrum.

The PZT transducers have a high efficiency in the production and reception of acoustic waves, however the performance of these materials is questionable when they are subjected to sustained mechanical loading at even moderate levels (greater then 2000 με) which is the case for many aircraft related structural integrity problems. DSTO is developing a number of strategies to overcome this potential 'show stopper', including (1) a durability testing program to understand PZT performance under various damage modes, (2) developing a novel stand-off element (SoE) approach to reduce the load transfer into surface-bonded PZT elements and (3) the use of optical fibre sensors in highly loaded regions.

Assessment of Transducer Performance:

Transducer durability studies were undertaken using a specimen geometry that allows reference transducers to be placed in a load-free zone, so as to provide a stable reference against which bonded transducers that are exposed to mechanical loads can be assessed (see Figure 3). An initial study was undertaken on a piezoceramic transducer package developed by DSTO and local industry [3,5,6]. Transducers were subject to constant amplitude sinusoidal loading, at a stress ratio of 0.1, for strain amplitudes between 500 to 3500 με (i.e. peak strain of 1100 to 7700 με). The response to a stable acoustic wave field (from a reference actuator) changed significantly as transducers were exposed to long term, high strain cycling. For the transducers tested here, loading above a peak strain of 3300 με produced complex behaviour associated with degradation of the transducer package (Figure 3). It is important to note that degradation, either in the bondline or the transducer itself, does not always lead to a decline in the piezoelectric signal. Overall, an impedance spectrum measurement appeared to be the most reliable in revealing the presence of damage in a transducer. Displacement mapping using a laser vibrometer also proved effective and can provide a quantitative measure of the extent of disbonding under surface-mounted piezoceramic transducers.

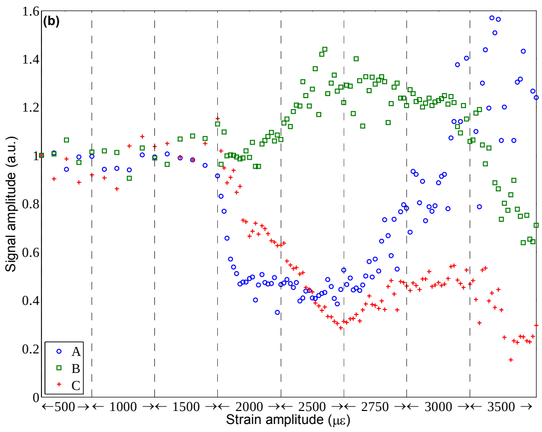


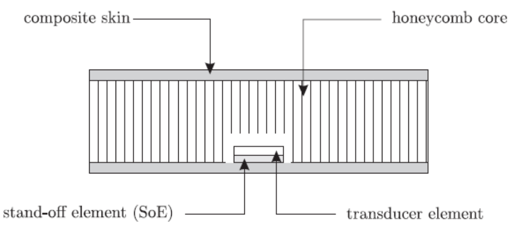
Figure 3: Left - Photograph of specimen showing transducers and switching circuitry required to take impedance and acoustic response measurements. Eight transducers are shown within the load path of the specimen and two reference transducers outside the load path on the 'winged' arms of the specimen. Right - Maximum amplitude of the Hilbert transformation calculated from the FFT of the AU response for three transducers at each of the strain amplitudes indicated.

SoE studies:

The SoE concept relies on the introduction of local out-of-plane or secondary bending in the structural host through a structural augmentation, as shown in Figure 4 [5]. Analytical and numerical modelling with experimental studies have been undertaken to prove the concept for both monolithic and honeycomb structures. One study undertaken considered a 4 ply, 1.1 mm thick carbon-epoxy composite laminate. An optimal SoE thickness of 0.68 mm was deduced from a geometrically nonlinear 3D FEA of the laminate. Three stand-off 12.5 mm diameter elements were incorporated with 10 mm diameter and 0.5 mm thick PZT disc transducers to experimentally evaluate the approach. Theroretical and experimental results, shown in Figure 4, illustrated large reductions in the transducer strains by incorporating the SoE, without adding significantly to the transducer profile or impairing acoustic transduction efficiency, important factors in iSHM. This therefore looks like a feasible approach in improving PZT serviceability of the life of the iSHM system.

AMAP sensor studies:

Optical fibres provide a number of advantages for structural interrogation, including: an immunity to electromagnetic interference, excellent mechanical and environmental durability, the capacity for dense sensor multiplexing, ease of embedment in composite materials, reduced wiring requirements and reduced weight. Optical fibre Bragg gratings (FBG) allow the incorporation of a large number of discrete dynamic strain sensors on a single fibre. A spatially distributed array of FBGs offer significant temporal and spatial information allowing the decomposition of structural plate wave modes which is not possible with piezoelectric receiver arrays [6]. This novel approach was demonstrated by DSTO using a prototype acoustic mode assessment photonic (AMAP) sensor. This was an optical fibre sensor comprising an array of 14 uniformly distributed one mm long Bragg gratings with a centre spacing of 1.5 mm, see Figure 5. To assess this approach an AMAP sensor and PZT elements were installed on an aluminium plate. In this case a small thick aluminium disc was bonded to the panel to produce a structural inhomogeneity in the wave path to simulate damage (see Figure 5). This study focused on symmetric Lamb waves (S_0) generated by surface-bonded PZT elements. The sensor measurements were acquired without (baseline) and with the disc, and were then compared with theoretical dispersion curves for the plate for both conditions. The insertion of the disc caused wavemode conversion from the symmetric (S_0) to the antisymetric (A_0) modes, which were measured by the AMAP sensor (see right hand figures in Figure 5).



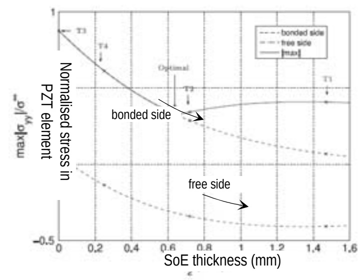


Figure 4: Schematic showing the SoE scheme (left). Predicted reductions in PZT stress with SoE thickness (right)

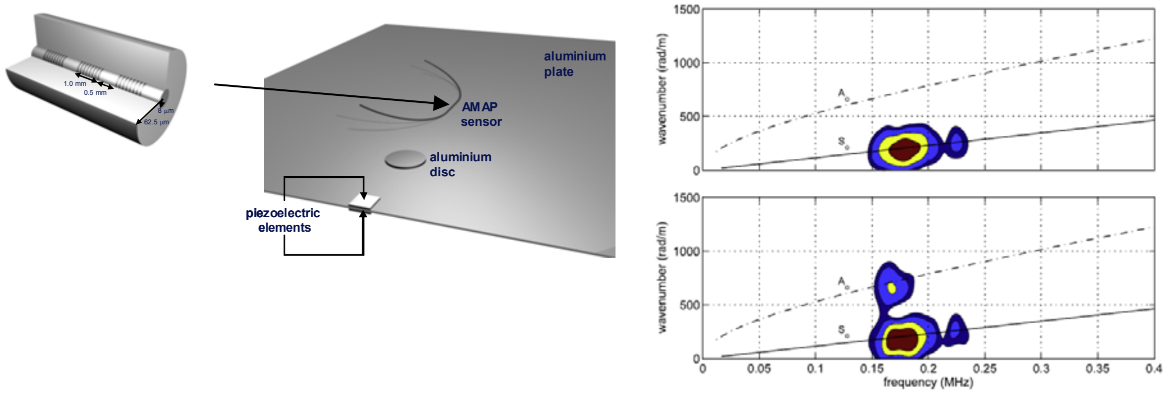


Figure 5: Schematic showing AMAP sensor approach (left) and results for an AMAP sensor on an aluminium plate, without (top right) and with (bottom right) the aluminium disc.

Conclusions

The feasibility of AU-based iSHM has been successfully demonstrated on complex aircraft structure. This study has provided confidence in the technique, and approaches to enhance the robustness and capability of the system have been demonstrated through the SoE concept and the optical fibre AMAP sensor. Work is currently underway to address other critical aspects including, reliability, sensitivity and ease of design, installation and interrogation.

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2.3.6 P-3C Repair Assessment Manual (K. Watters, M. Ignatovic [QinetiQ AeroStructures] and SQNLDR B. Krutop [RAAF])

The Royal Australian Air Force (RAAF) operates 19 P-3C aircraft. Federal Airworthiness Regulations (FAR) 25.571 has been approved for fatigue management of the RAAF P-3C fleet, including the management of repairs. Interpretation of FAR 25.571 requirements, supported by the guidance provided in associated Advisory Circulars (ACs) including AC120-93, has led to the development of a Repair Assessment Manual (RAM).

The RAM provides procedures for engineers to assess both new and existing repairs applied to AP-3C aircraft, without the need for individual Damage Tolerance Analyses (DTAs). The output of the RAM is inspection requirements for the repair. These include inspection thresholds and inspection intervals.

Repairs are mapped to representative fatigue critical areas (FCA) in the structure for which damage tolerance analysis data is available from full-scale test interpretation. The severity of the repair relative to the mapped FCA is characterised by peak and far field stress ratios. These stress ratios are used to adjust the inspection threshold and interval of the FCA for application to the repair. All the critical locations in a repair are analysed and the most critical location determines the repair inspection management.

Subject to some limitations, the RAM applies to all repairs to P-3C primary structure. It includes all types of damage that might instigate a repair (cracking, corrosion, accidental, manufacture), and all repair types. Repair types include rework, blend out and doubler reinforcement (Figure 1). Even leaving damage such as non-blueprint manufacturing damage unrepaired, and developing an ongoing monitoring inspection programme for it, falls within the scope of the RAM. The RAM has been issued by the RAAF and is being used routinely for repair assessment.

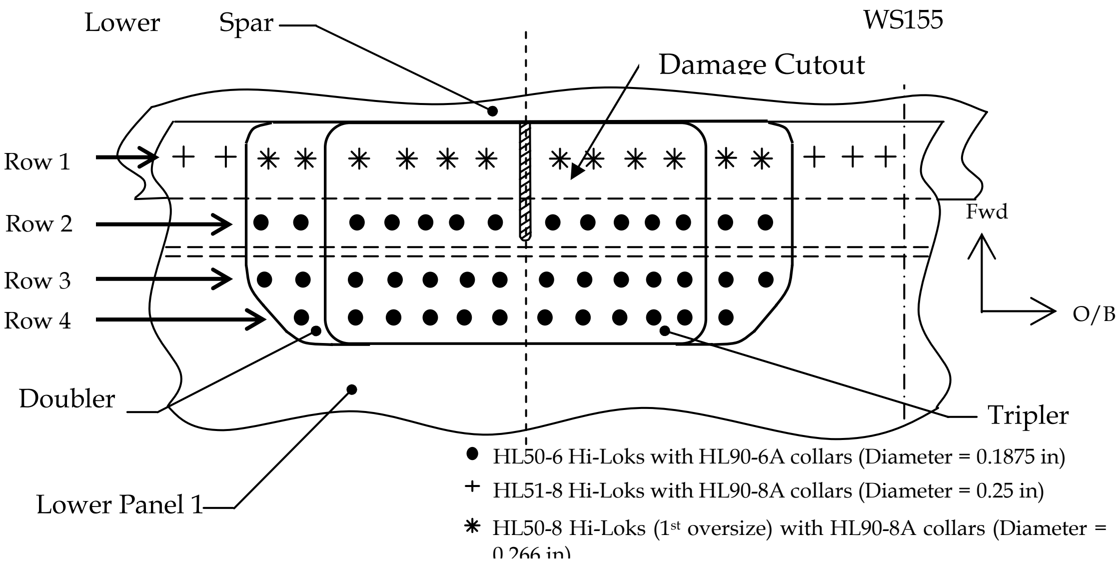

Figure 1: Example Repair for RAM Analysis

2.3.7 Development of a Revised Helicopter Usage Spectrum (K. Jackson, R. Buller, J. Lamshed and B. Hindmarsh [QinetiQ AeroStructures])

In 1993 Sikorsky developed a usage spectrum called the Australian Unique Usage Spectrum (AUUS) for the Australian Defence Force (ADF) Black Hawk fleet. This spectrum was developed using a variety of data, including mission monitoring forms filled out by Black Hawk pilots after each flight and the witnessing of army operations by Sikorsky test pilots. The final spectrum detailed the percentage time spent in various flight regimes, along with occurrence rates and durations for various manoeuvres and operations. The spectrum included flight regime prorating in various gross weight and altitude bands.

Using this spectrum, Sikorsky calculated safe lives for a number of critical components. This process followed standard industry practice whereby a working S-N curve was produced for each component using fatigue test failure data coupled with suitable safety factors on stress and life. For each flight regime, Sikorsky obtained suitable component flight loads from their UH-60 loads database. Fatigue damage calculations where performed using Miners rule to produce component safe lives.

In 1995 the ADF introduced a paper based usage monitoring system for the Black Hawk fleet. Fleet usage was monitored using EE 360 data sheets that are completed after each flight and record a number of usage parameters such as gross weight and altitude bands along with occurrences of certain fatigue damaging manoeuvres.

In 2010 the ADF commissioned a task to develop a new Black Hawk usage spectrum based on the collected EE 360 data. QinetiQ Aerostructures, with support from DSTO, reviewed the EE 360 data and developed representative occurrence rates for a number of parameters including autorotations, landings, rotor starts and time spent in various altitude and gross weight bands. These EE 360 based occurrence rates were then used to modify the AUUS to develop a new spectrum representative of ADF usage. Finally, the new spectrum was employed to calculate component safe lives by adjusting the damage rates for components assessed by Sikorsky using the AUUS.

2.3.8 Critical Reviews of Helicopter Paper-Based Usage Monitoring Systems (B. Hindmarsh and T. Cooper [QinetiQ AeroStructures])

The Australian Defence Force (ADF) has traditionally employed a paper-based usage monitoring (UM) system for helicopter platforms, which requires individual helicopter usage to be manually recorded after each flight.

Usage data is compared to an approved baseline usage spectrum to assess whether the basis for lifing remains valid. To make an adequate assessment, it is necessary to capture the fatigue damage drivers (FDDs) for critical structure. Figure 1 shows the proportionate fatigue damage contribution for the Black Hawk FDDs.

Critical reviews have been conducted to assess the adequacy of the ADF helicopter UM systems, and determine whether the FDDs for critical structure are being captured. The helicopters reviewed were Black Hawk, Seahawk, Chinook and Squirrel.

As an example, for the S-70A-9 Black Hawk, 48% of fatigue damage is captured by the current EE360 sheet (GAGs, DSP, Take Offs, Landings, Autorotations); 32% is partially captured (Turns, Pullouts); and 20% is not captured (Control Reversals, Dives, Sideslip, and the remaining flight regimes). Gross Weight prorating was found to play a significant part in the total fatigue damage for some components.

The reviews concluded that a paper-based UM system is adequate for tracking manoeuvres and flight regimes that can be detected, recalled and recorded with reasonably fidelity by aircrews. For parameters that cannot be captured with reasonably fidelity using a paper-based UM system, alternative methods for capturing those parameters should be investigated to ensure the FDDs for all critical structure are adequately monitored.

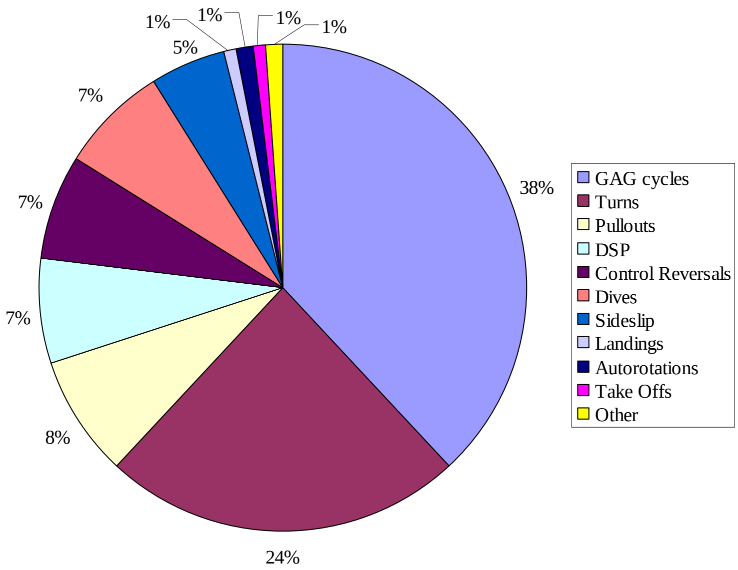

Figure 1: S-70A-9 Black Hawk Critical Structure Fatigue Damage Drivers

2.3.9 P-3 RAM Fastener Modelling Study (S. Norburn and G. Papp [QinetiQ AeroStructures])

In support of fatigue assessment of repairs involving mechanically fastened reinforcements, a study has been performed to compare five empirical approaches to modelling fastener stiffness against a high fidelity Finite Element modelling approach, for a simple generic doubler and tripler repair configuration (Figure 1). This high fidelity approach is referred to as the Rutman method and is used as a benchmark for the five empirical fastener stiffness methods, to assess how conservative and how representative the empirical approaches are for the simple repair configuration considered.

The empirical methods include two variants of a Grumman method, two variants of a Boeing method, and the Douglas-Swift Method. All the empirical methods proved to be stiffer than the Rutman FE method. The Grumman methods were the least stiff and most accurate.

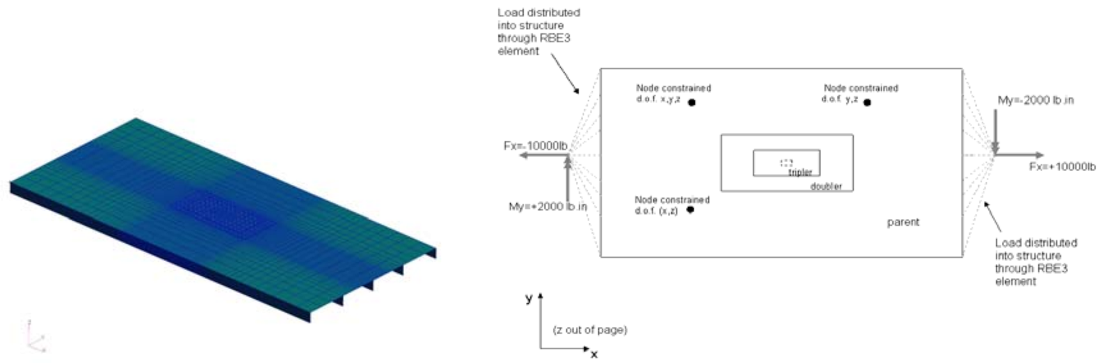

Figure 1: Finite Element Model of Test Problem for the Study

2.3.10 C-130H Centre Wing Lower Surface Tang Blend Repair Management (A. Ng, S. Trezise, P. East, R. Stewart, S. Norburn and K. Jackson [QinetiQ AeroStructures])

Corrosion damage to the Royal Australian Air Force C-130H Centre Wing Lower Surface (CWLS) panel tangs is an ongoing and widespread problem. To repair this damage, a number of doubler, splice, spot face and blend repairs have been incorporated on the fleet. The crack growth implications of spot face and blend repairs, and the impact of these repairs on the current Aircraft Structural Integrity Management Plan (ASIMP) have been assessed. The aim of the assessment was to provide detailed, fleetwide inspection instructions, including inspection procedures and initial and recurring inspection intervals, for spot face and blend repairs on the CWLS panel tangs.

A Finite Element model (FEM) to determine the bearing and by-pass loads in each fastener row of the panel tangs was built (Figure 1). Additionally, the FEM considers the effect on loads following failure of a single tang.

Damage Tolerance Assessment (DTA) was carried out on the blue print configuration for each panel tang fastener row to determine the crack growth life relative to the control point, and to determine the relative severity of various cracking options.

Stress intensity correction factors were developed for cracks growing from or to a spot face using Stress Check (Figure 1). These correction factors were applied on top of geometry factors for the baseline configuration. The Stress Check correction factors identified spot face diameter had a greater impact than spot face depth on crack growth.

Using the blueprint DTA results, damage maps of the tangs and Stress Check results, six repair cases were assessed that covered the extremes of the damage at the tangs.

Based on the results for the repair cases, it was shown that the repairs identified in the damage maps can be managed using the existing ASIMP Volume 2 inspection interval. A management strategy for future repairs was outlined, defining future repair limitations where no change to ASIMP Volume 2 is required.

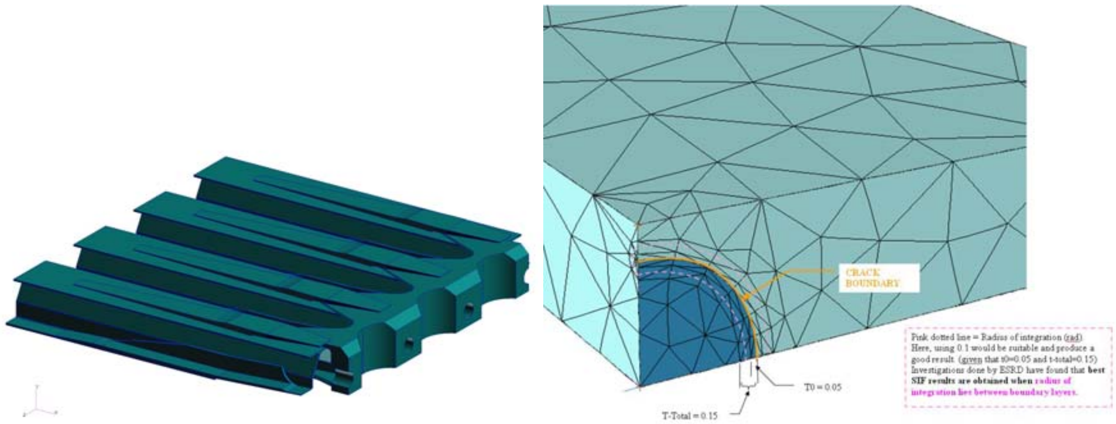

Figure 1: Finite Element and Stress Check Modeling of Cracked Tangs

2.3.11 RAAF PC-9/A Empennage and Aft-fuselage Recertification and Life Assessment Program (W. Zhuang and L. Molent, [DSTO])

PC-9/A trainer aircraft have been operated by the Royal Australian Air Force (RAAF) since 1986. The aircraft was originally certified against US Federal Aviation Administration, Federal Aviation Regulations Part 23 (FAR 23) by the Swiss Federal Office for Civil Aviation. To better understand the structural life of the PC-9/A fleet in Australian usage, and to provide more fatigue management flexibility the RAAF adapted DEF STAN 00-970 as a supplemental Certification Structural Design Standard (CSDSTD) for the PC-9/A. A critical element of this approach was the conduct of a full scale fatigue test (FSFT) to establish baseline information for long-term fatigue management of the PC9/A fleet.

The PC-9/A FSFT was carried out from January 1996 and completed in January 2000 after the empennage was tested to 100,000 Simulated Flying Hours (SFH). However, recent fleet condition data has shown that the RAAF PC-9/A fleet has experienced unexpected fatigue cracking in aft fuselage primary structure. Some of the fatigue damaged locations found in the aft fuselage structure in the PC-9/A fleet are inconsistent with that found in the FSFT; furthermore, the growth rate of fatigue cracks where the locations are consistent with the FSFT are significantly faster than that demonstrated by the FSFT. These inconsistencies suggest that the previous FSFT results can not be relied upon to substantiate the safe lives for aft fuselage primary structure. To ensure the safe operation of the RAAF PC-9/A fleet through to its planned withdrawal date, the RAAF has launched the PC-9/A Empennage and Aft-fuselage Recertification and Life Assessment (PEARLA) program.

The primary objective of the PEARLA program is to develop a revised basis for certification of the structural management strategy applied to the PC-9/A aft fuselage, based on the outcomes from an Operational Loads Measurement program. Key to this will be the identification and life assessment of all structural "hot spots", which are likely to be susceptible to fatigue, using analytical approaches and aircraft teardown. Currently, OLM aircraft and equipment are being sourced and preliminary fatigue assessments are underway.

2.3.12 P-3C Structural Management Update (K. Walker, E. Matricciani and L. Meadows, [DSTO], J. Duthie [QinetiQ], J. Lubacz [Fortburn])

Structural integrity assurance and certification for the RAAF P-3C fleet was originally achieved through a safe life approach under CAR4b regulations. About 10 years ago the fleet were approaching their calculated safe life and other operators around the world were facing similar issues. Australia then entered into a program called the Service Life Assessment Program (SLAP). The SLAP was a comprehensive international collaborative program involving Australia, the United States, Canada and the Netherlands. Key aspects of the SLAP included full scale fatigue tests, coupon testing and analytical model development. In Australia the results from the SLAP were interpreted [1] using a Test Interpretation (TI) methodology suitable to then construct a Structural Management Plan (SMP) [2] which would achieve a transition of the fleet to a safety by inspection certification basis under FAR 25.571. The SMP has now been implemented on the RAAF fleet and about half the fleet have so far been inducted into the inspection regime.

A program of work at DSTO to follow on from the SLAP was begun in 2007. It consists of a comprehensive set of coupon and component tests and analytical model development in order to refine the current SMP. In late 2009 the RAAF formalised their objectives in terms of specific increases in thresholds and inspection intervals. Achievement of these goals will both enable the fleet to meet the Planned Withdrawal Date (PWD, currently 2019+), and to realise major cost savings and enhanced fleet availability. The feasibility of achieving these goals was based on the premise that the additional coupon and component test results in conjunction with refined and improved analytical models could remove conservatism perceived to be present in the current results and plan. The work program now underway on this is known as the 2010 SMP Review.

The approach taken in the current 2010 SMP Review work is consistent with the previous work in that the fatigue process is considered as an initiation or nucleation phase followed by a growth phase. The initiation phase is modelled using local notch strain methods. A comprehensive coupon test program was conducted to determine the best material life data curve for use, and this is also being incorporated into the updated Individual Aircraft Tracking (IAT) program now in place for the fleet. Coupon test programs have also been undertaken (some are still in progress) to quantify the benefits gained through modification programs which introduce cold working and interference fit fasteners.

Coupon testing to quantify the interaction between intergranular corrosion and fatigue are also planned.

As reported in the previous National Review, six aircraft in the fleet are fitted with a strain sensor based Operational Loads Monitoring System (OLMS) capability. These systems require ground calibration under known loading. The first aircraft (see Figure 1) was subjected to the calibration in April 2010 and the other 5 will be calibrated throughout 2011 and 12. the data obtained from the OLMS aircraft will be used to validate the load and stress sequences which are otherwise determined though a parametric system.

Figure 1: RAAF AP-3C in preparation for OLMS calibration

Development and verification of the crack growth analysis modelling aspect of the 2010 SMP review has a very solid foundation in that three levels of testing were conducted; simple centre crack coupons, components containing the wing span-wise splice detail, and the Full Scale Fatigue Test conducted previously under the SLAP program. The wing span-wise splice components measure about 1000 mm long and 350 mm wide and were sourced from a retired P-3B wing. The P-3B lower wing skin material and geometry is the same as for the P-3C. The set up for the testing is shown in Figure 2. As was the case under the SLAP, the modelling is conducted using FASTRAN [3], with improvements to both the code itself and key inputs such as the rate curve and the constraints. Code enhancements were relatively minor, but they did correct some anomalous behaviours under certain conditions, such as very small amplitude cycles having a larger than expected effect on the crack growth. The main change was to the rate curve and constraints. The results from the updated model compared with the experimental results for the same wing location for the three levels of testing are shown in Figures 3-5. The correlation is considered to be very good, and when combined with the updated fatigue initiation life analyses and consideration of fleet data it is likely that most, if not all, of the objectives for life and inspection interval extension will be satisfied. Further details are available in [4].

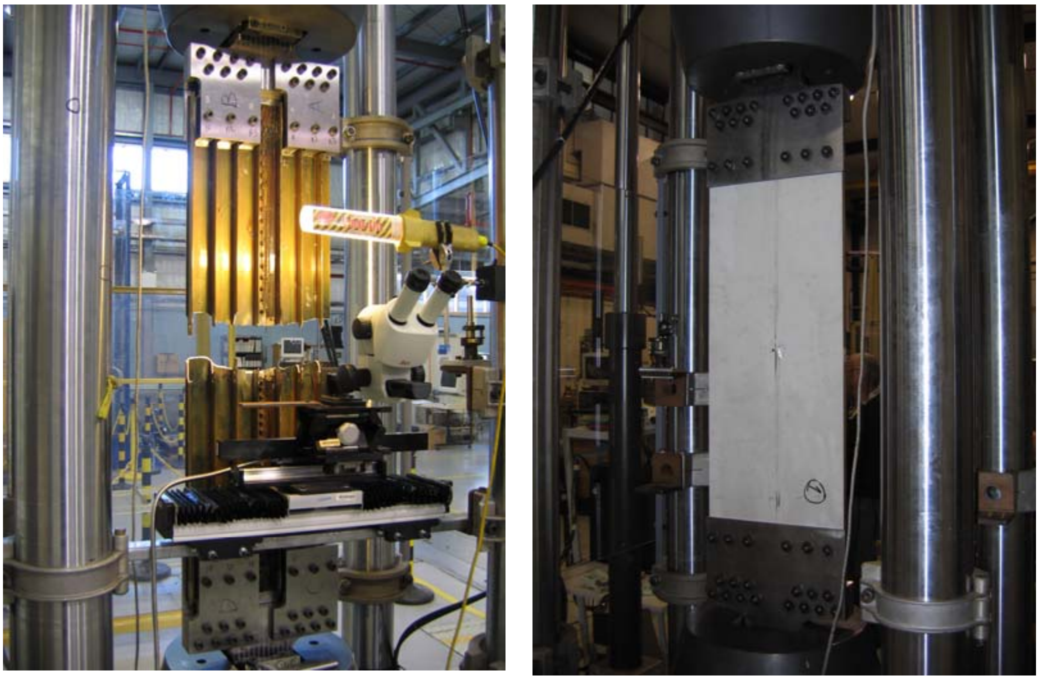

Figure 2: Wing span-wise splice panel component test set-up

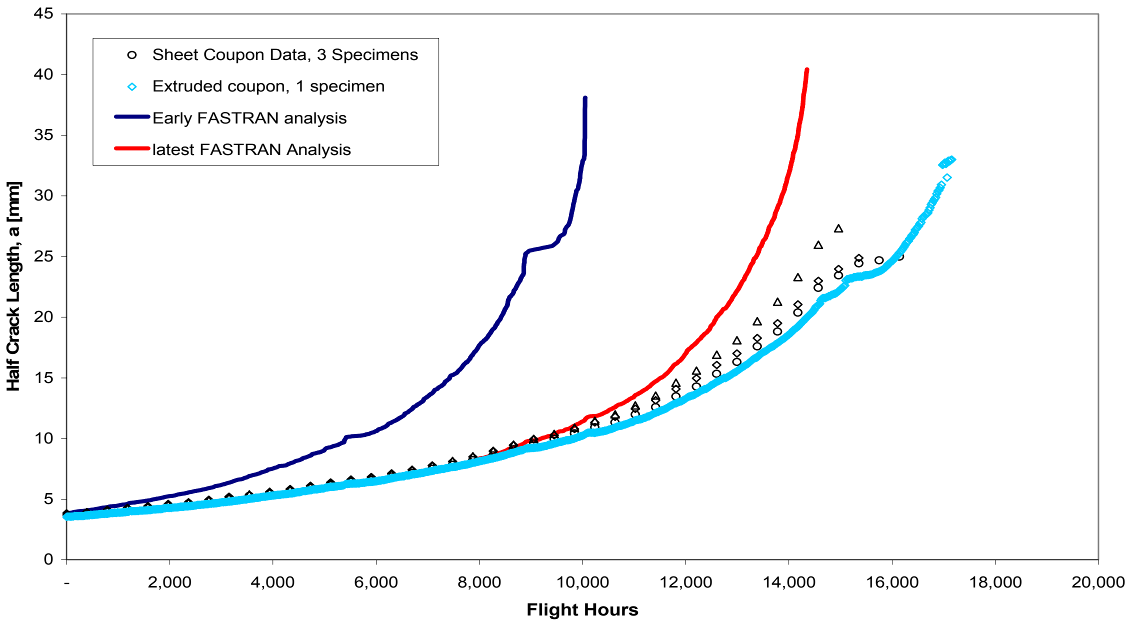

Figure 3: Comparison between analysis and test for centre crack coupons for a typical lower wing location

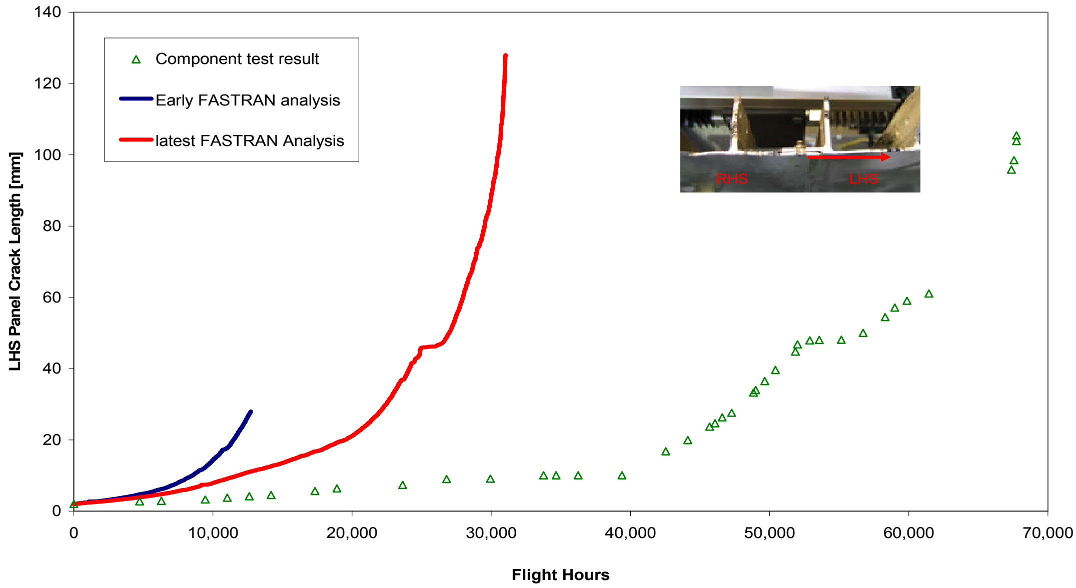

Figure 4: Comparison between analysis and test for spanwise splice component for the same typical lower wing location

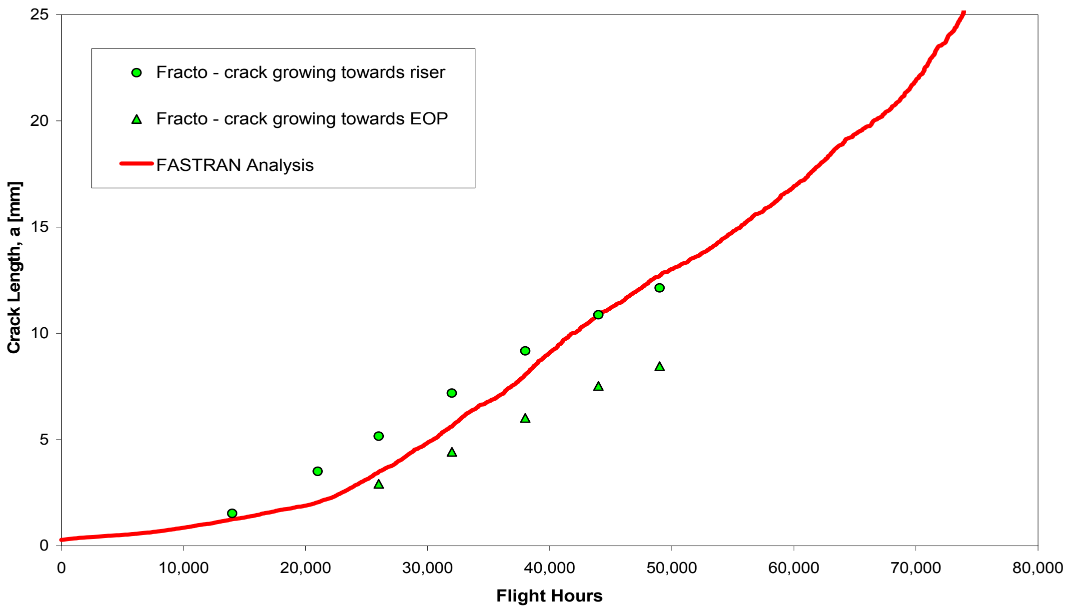

Figure 5: Comparison between analysis and test results for the same wing location from the Full Scale Fatigue Test

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2.3.13 Support to PA 5402 - KC-30A Aircraft Project (G. Chen, R Ogden & L Meadows [DSTO])

The Structural Health Monitoring System (SHMS) and Operational Loads Monitoring System (OLMS) form an integral part of the airworthiness management for the KC-30A aircraft in Australian Defence Force (ADF) Service. DSTO has been tasked to conduct independent validation and verification of the SHMS and OLMS on the KC-30A. To define the details of this program of work DSTO has drawn lessons learned from other ADF platforms. A three phased approach is proposed:

Phase 1 involves ensuring the notional system design is fit for purpose and that acceptable data is being harvested from the aircraft. This is critical for the award of an aircraft Special Flight Permit (SFP) prior to entering service. Based on SHMS IV&V functionalities, DSTO has conducted IV&V activities involving:

- SHMS Data Recording,
- Raw Data File Integrity,
- Multi-Functional Display (MFD),
- SHMU Download,
- SHMU Configuration Control, and
- SHMU File Explorer.

With the majority of Phase 1 activities completed, sufficient confidence has been gained that the system is capable of performing its core roles of capturing and recording valid parametric and strain data; and that the data storage and management procedures and devices are adequate to prevent loss of data, and to allow reprocessing.

Phase 2 involves initial testing of the post flight data processing for compatibility with the extant aircraft lifing philosophy and for compliance with ADF requirements. It will ensure the Ground Station (GS) and the Transfer Function functionality and the alignment of embedded processes with the underlying ongoing airworthiness management processes. The work on Phase 2 is the current focus at DSTO. Detailed plans have been developed, with activities divided into the following four groups:

- **Validation of GS Functionality**: testing the results of the Damage Calculation Module (DCM) to ensure that signals with predefined errors produced the required warnings etc; testing a real data signal to ensure any errors aren't missed; and checking DCMs database, its report generation capability, backup procedures and configuration capability.

- **Validation of Fatigue Damage Calculation:** validating the DCMs spectrum generation capabilities, rainflow counting algorithm, crack initiation analysis algorithm and crack growth analysis algorithm by using DSTO 'in-house-tools". The 'in-house-tools" (which may also require further development) also need to be validated against some accepted fatigue damage calculation tools.

- **Validation of Transfer Function Calculation Functionality:** aiming at ensuring that the trained Artificial Neural Network (ANN) has learned the correct data, has converged to a global minimum and is able to handle data outside the extant training set.

- **Validation of Transfer Function Calculation:** validating the transfer function by assessing the correlation of the ANN produced spectrum at the stress generation points shown in Figure 1 and the strain data for the same flight. If the discrepancies fall outside an appropriate error margin, the transfer functions would need to be redefined.

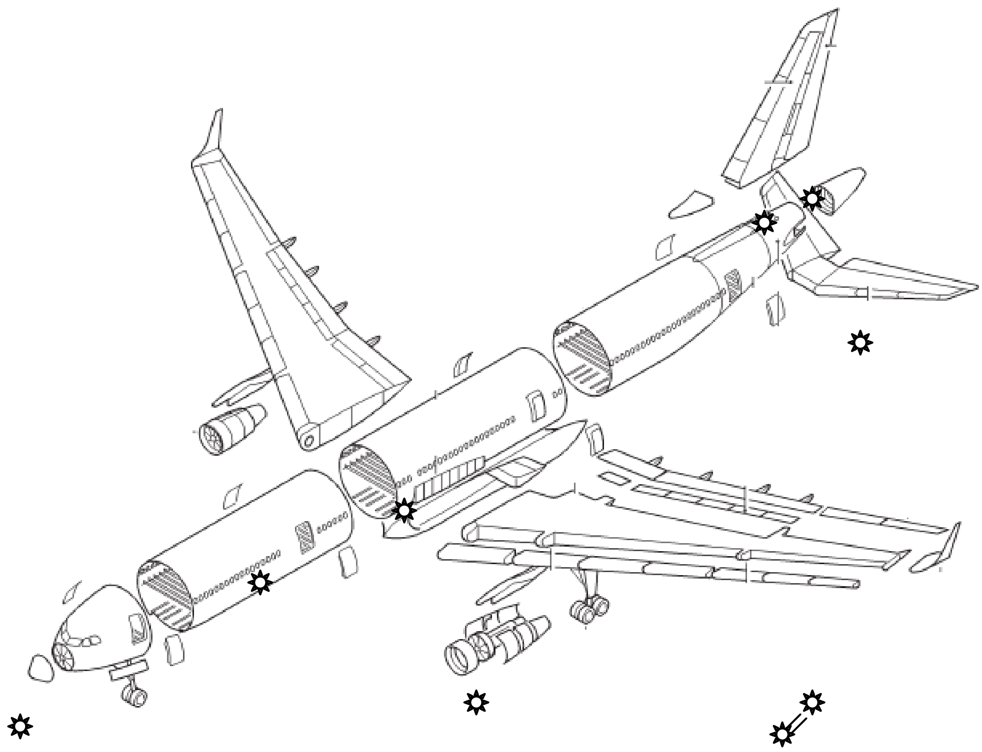


Figure 1: A Few Sample Locations of the Stress Generation Points

2.4 FATIGUE INVESTIGATIONS OF MILITARY AIRCRAFT

2.4.1 Advanced rework options for crack-prone penetration in P-3C floor beam web at FS 515 and LBL 49.5 (R. Kaye and M. Heller [DSTO])

The international P-3 Service Life Assessment program (P-3 SLAP) conducted between 1999 and 2006 consisted of a number of full scale fatigue tests and subsequent analyses. One of the fatigue deficiencies revealed in the full scale test of the fuselage was cracking in an under-floor beam. This cracking has resulted in an inspection program, however DSTO has been undertaking work to determine of a structural re-shaping modification can be applied in order to alleviate the inspection requirement.

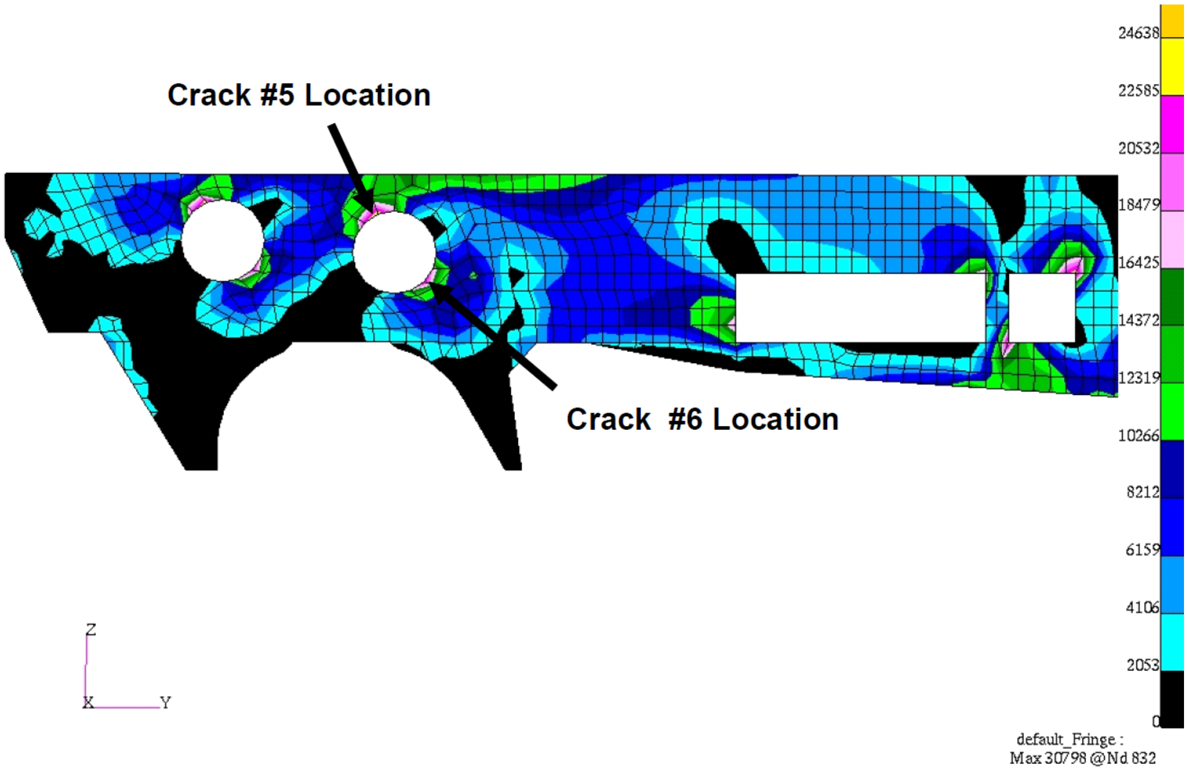


Figure 1: Site location with plot of major principal stress(psi)

Adaptive shape optimisation principals have been used to develop rework shapes for the above circular web penetration (2nd from left). The core of the method involves small hole boundary movements using iterative finite element solutions. The boundary remains smooth with accurate boundary stresses used at each iteration to move the boundary outwards from the centre by small amounts inversely proportional to the boundary stress magnitude. A minimum radius of curvature is applied as well as multiple load cases at a range of stress orientations so that the final hole shape is not compromised by small errors in stress orientation. The final shape is not sensitive to stress magnitude. Reductions of 17 – 33% in peak boundary stress have been achieved.

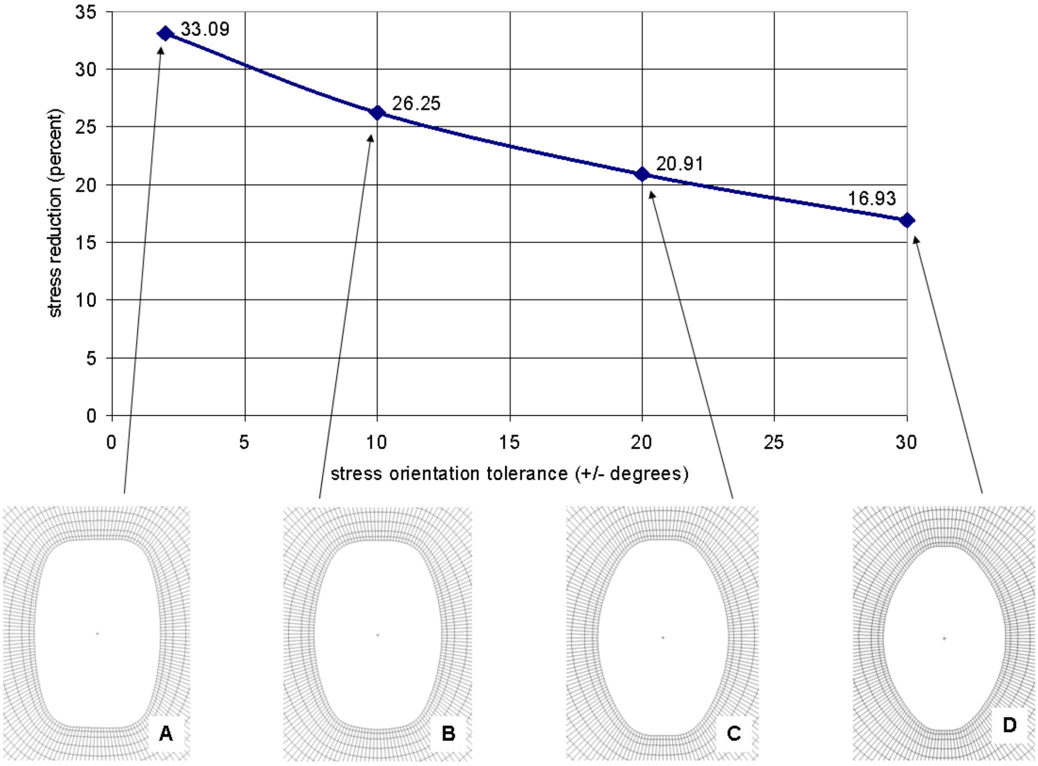


Figure 2: Final hole shapes with stress reductions

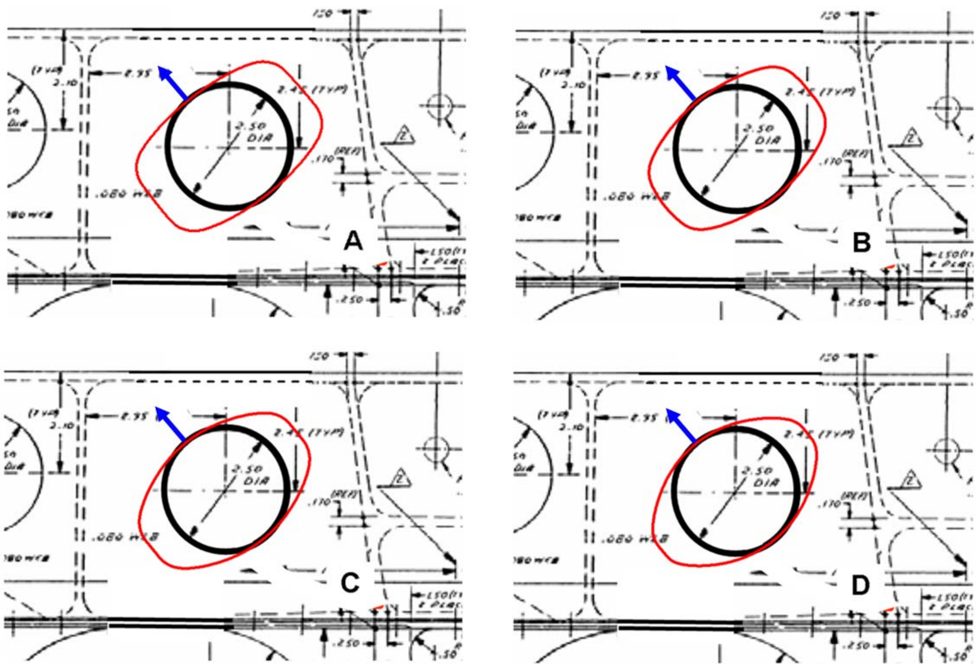


Figure 3: Final hole shapes shown with surrounding structure

2.4.2 Failure Analysis Examples – Military Aircraft (N Athiniotis, [DSTO])

The following sections contain examples of failure investigations conducted in the last two years by DSTO:

2.4.2.1 Helicopter Brake Disc Failure

The brake disc (shown in Figure 1) suffered heat tinting and crazing on the working surfaces (see Figure 2), indicating exposure to high temperature and severe temperature cycling. Numerous larger radial fatigue cracks (up to approximately 13mm in length) were present around the outer diameter of one side of the disc. Failure of the brake disc was due to thermo-mechanical fatigue, with the crazing cracks providing the initiation sites for the larger mechanically driven fatigue cracks. Heat tinting and corrosion of the fracture surfaces indicated the brake disc was operated for some time after failure had occurred at the largest of the radial fatigue cracks.

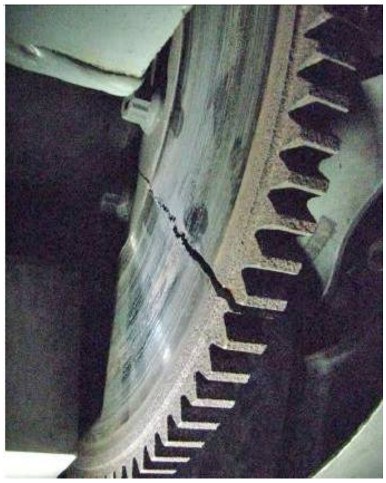


Figure 1: Cracked Main Rotor brake disc in-situ

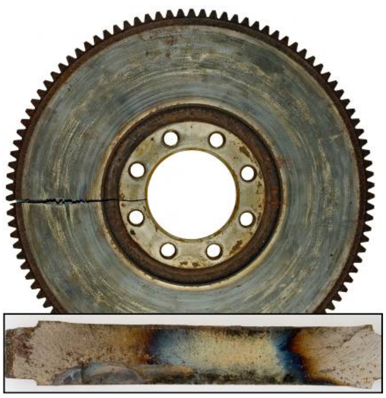


Figure 2: Fracture surface was tinted and corroded. A semi-elliptical crack was present towards the outer diameter of the disc

2.4.2.2 *Maintenance-Induced Rudder Lever Cracking*

Maintenance-induced damage to the surface of the torque tube breached the corrosion protection scheme, resulting in pitting of the material and environmental ingress (see Figure 3). The corrosion pits acted as stress raisers for fatigue crack initiation. Crack propagation was predominantly intergranular via corrosion assisted fatigue.

(a) *(b)* *(c)*

Figure 3: (a) Severe mechanical damage adjacent to the fastener nut, (b) Photograph showing the crack below the attachment point of the torque tube flange and (c) close up view showing multiple crack planes extending from the corrosion pitting

2.4.2.3 *Aircraft Alternator Drive Shaft Failure*

Failure occurred due to excessive fretting wear of the alternator drive shaft external splines and the hydraulic pump/ engine driven compressor idler drive spur gear internal splines (see Figures 4 and 5). The wear was most likely a result of a combination of absence of case hardening of the shaft external splines and absence of interference fit between the shaft and gear splines. The absence of interference fit developed from a lack of adhesion and subsequent exfoliation of silver flash from the shaft external splines which produced a clearance fit between the gear internal and shaft external splines (see Figure 6). The higher loads experienced by the alternator drive shaft when driving the engine driven compressor may have also contributed to the failure.

Figure 4: Photograph providing an exploded view of the Alternator Drive Assembly

Figure 5: Photograph of the alternator drive shaft assembly showing the heavily worn external splines

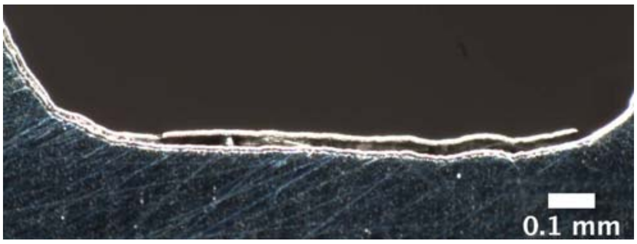
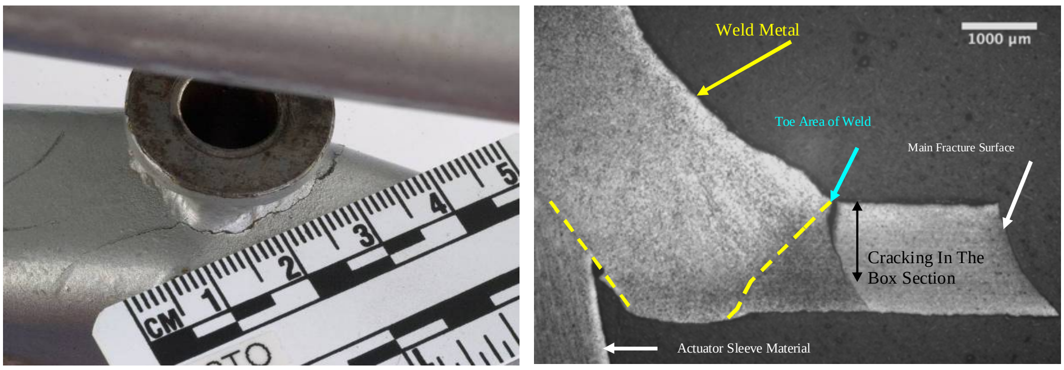


Figure 6: Photograph providing an example of the exfoliated silver flash observed at the root of the alternator drive shaft external spline

2.4.2.4 Nose Landing Gear Folding Strut Cracking

Cracking initiated at the toe of the weld used to join the actuator support sleeve to the triangular shaped welded box-section frame of the Nose Landing Gear Strut (see Figure 7). Fatigue initiated at multiple points in, and propagated from, the toe of the weld used to attach the support sleeve to the welded box-section frame.



(a) (b)

Figure 7: a) crack at the toe of the actuator support sleeve weld and continuing into the box section and b) Photomicrograph of the weld and plate material showing the secondary crack that had also initiated at the toe of the weld and progressed partially through the box section material

2.4.3 Marker Bands - Quantitative Fractography for the Assessment of Fatigue Crack Growth Rates in Metallic Airframes (M. McDonald and R. Boykett, [DSTO])

Reading the microscopic deformations 'recorded' on the fracture surface of a metal fatigue failure, or Quantitative Fractography (QF), can reveal information about the rate at which the fatigue crack grew under the applied in-flight loading. This is done by destructively opening up the cracked component and, under the microscope, associating the fracture surface deformations back to the individual in-flight loading events, thus enabling a plot of the crack size against flight-hours to be generated. This can provide previously unavailable data for the purpose of fatigue design validation and fatigue modelling improvement, particularly for shorter cracks (< 1 mm) which can not be reliably measured by traditional crack size monitoring methods.

QF data is not always easy to obtain as different metals have different propensities for creating visible fracture surface deformations. Figure 1 shows an example of ‘easy to read’ naturally occurring marks for a simple ground-air-ground sequence. A recent joint-study by DSTO & NLR on Aluminium Alloy 7050 revealed that the deformations can be made more visible by forcing the crack path to change direction for small periods of time by inserting special load patterns into the applied flight load spectrum. These load patterns were exploited to force the fatigue crack to ‘grow’ a microscopic light-reflector on the fracture surface at a particular point in the flight spectrum. These reflectors are significantly easier to identify under an optical microscope compared to the deformations caused by the normal flight loading. Multiple path changes can also be forced to effectively create a 'barcode' if needed as can be seen in Figure 2.

This work presents load patterns that were found to create optical reflectors for Titanium Ti-6Al-4V, which has traditionally been very difficult to read and so there is very little available data on the growth rate of small cracks in this material. Fatigue tests for both Titanium and Aluminium Alloy 7050 were performed to fine-tune these load patterns such that they produced optical bands small enough that they did not significantly affect the overall component life (<1%) and yet were still visible. Both of these materials are used extensively in the skeleton of modern combat airframes, and hence this technology is currently being used to provide new, more comprehensive, crack growth rate information that will improve the accuracy and efficiency of independent validation and verification assessments on the durability of these new combat aircraft.

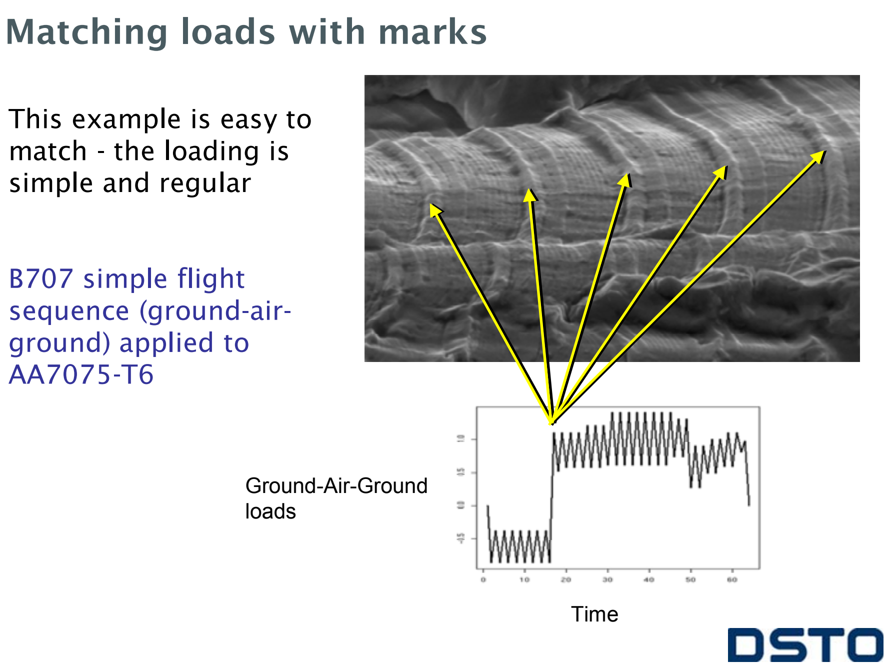


Figure 1: An example of easy to read marks on the fracture surface from applying GAG cycles.

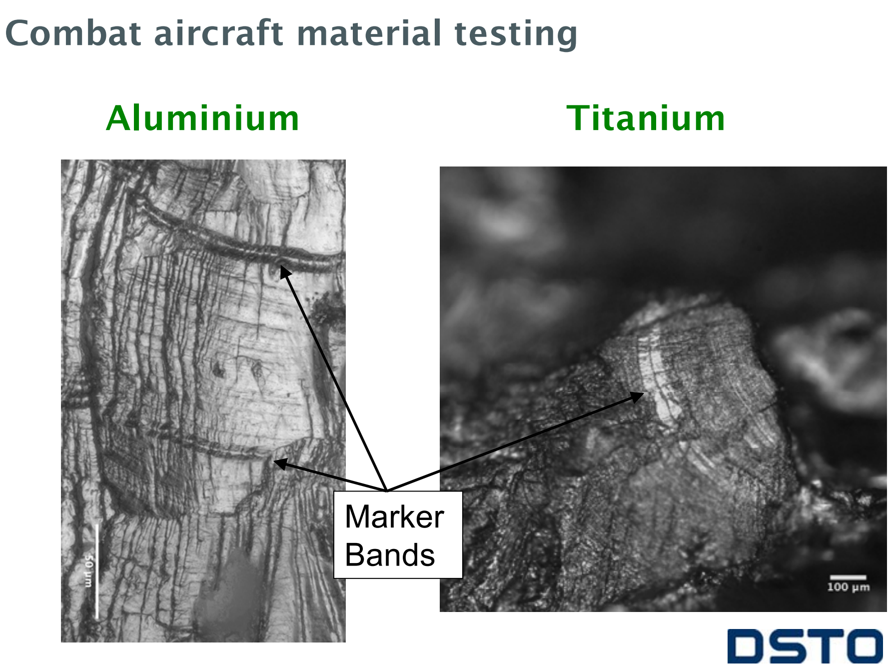


Figure 2: Marker Bands shown on typical combat aircraft Aluminium and Titanium alloys

2.4.4 Life Extension of F/A-18 LAU-7 Missile Launchers Using Rework Shape Optimisation (M. Heller, J. Calero, S. Barter, R. Wescott and J. Choi, [DSTO])

For more than twenty years, the Royal Australian Air Force (RAAF) has reported cracking of the housing guide rails of the LAU-7 missile launchers, in their F/A-18 aircraft. The LAU-7 missile launcher (Figure 1 (a)), which is positioned at the wing tip, carries the Advanced Short Range Air-to-Air Missile (ASRAAM). Multiple fatigue cracks have been found to propagate in the guide rail at the corner adjacent to the missile forward hanger (Figure 1 (b)). The RAAF has been managing the issue by replacing rails when the cracks reach a surface length of 19 mm. This typically corresponds to a crack depth of 0.5 mm or less.

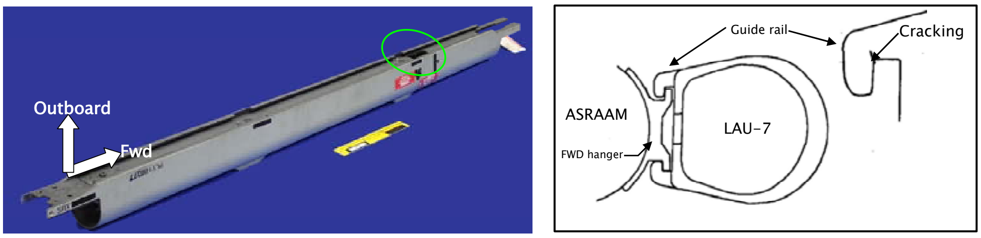
DSTO has developed a relatively unique life extension technology over the last 13 years. The technology uses an FEA-based stress optimisation approach to create optimised rework shapes that significantly reduce peak operating stresses. A very important aspect of this work has been the development of effective in-situ manufacturing methods to accurately achieve the optimised shapes. An overview of this DSTO technology was presented at ICAF 2009.

In the current work DSTO has designed a repair approach for the launcher rails based on optimum rework shaping at the critical fatigue location. This approach removes the existing crack and reduces the stress concentration factor relative to the initial uncracked component. This can provide significant economic benefits by both avoiding the need for component replacement as well as increasing the interval between costly periodic in-service inspections. Two options have been developed. First, an optimised blend shape that can remove existing cracks from 0.5 mm to 1.1 mm depth. Second, a very shallow optimised blend shape that can be used as a pre-emptive measure to increase the fatigue life of new launchers, which the RAAF has stockpiled.

The following stages in launcher repair development have been successfully completed: (i) FEA to design the optimised shapes, (ii) making prototype in-situ tooling to manufacture the new shapes, and (iii) fatigue testing of coupons and Quantitative Fractography (QF) to assess crack growth. A procedure to cold-roll coupons has been developed and cold-rolled coupons have undergone fatigue testing. Also, development of an NDI procedure using a specialised eddy current probe is near to completion.

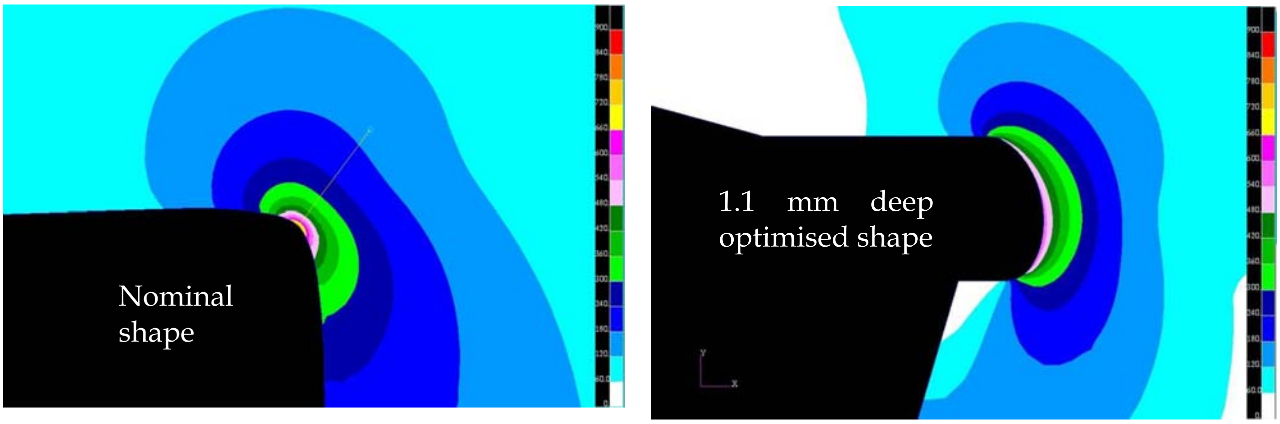
The shape and typical stress distribution for the nominal geometry and the 1.1 mm deep rework are shown in Figure 2. The 1.1 mm deep rework offers a reduction in stress concentration of more than 30%. The shallow rework option (not shown) offers a stress concentration reduction of more than 45%. Fatigue tests have been conducted on coupons cut from a section of the rail (Figure 3) with the nominal shape and both rework options. One loading block in the fatigue test represents a spectrum of 55 flight hours. A typical fracture surface for coupons with optimised profile is shown in Figure 4. Results show an increase in component life of at least three times for the 1.1 mm deep rework (Figure 5) up to a crack depth of 0.5 mm. Equivalent tests for the shallow rework predict an approximately five fold life increase. DSTO is currently collaborating with the RAAF and

Australian industry to develop the most effective logistical approach to undertake these repairs.



(a) (b)

Figure 1: LAU-7 missile launcher (a) complete and (b) cross-section with crack location



(a) (b)

Figure 2: Shape and stress distribution for (a) the nominal geometry and (b) the 1.1 mm deep rework

Figure 3: Fatigue test coupon in test machine

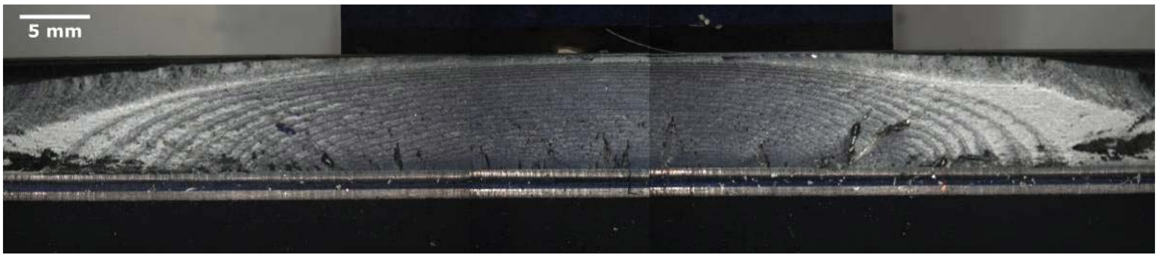


Figure 4: Typical fracture surface for coupons with optimised profile

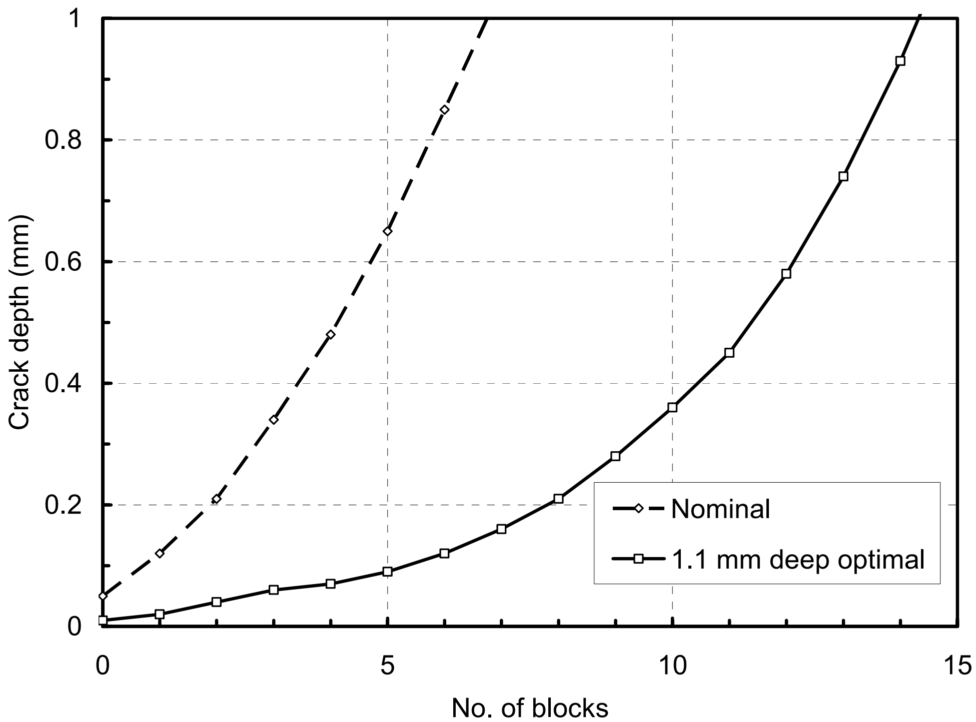


Figure 5: Average fatigue crack depth (over five tests) for coupons with nominal and 1.1 mm deep optimal profiles as a function of loading block number

2.5 FATIGUE INVESTIGATIONS OF CIVIL AIRCRAFT

2.5.1 Fatigue and Structural Integrity of Light Aircraft (D. Morris, Civil Aviation Safety Authority)

The Australian Civil Aviation Safety Authority (CASA) has been involved with Australian industry in a number of projects related to the life extension or damage tolerance evaluation of light aircraft structure. Two specific projects being a fatigue and damage tolerance evaluation of the wing carry through structure of the Cessna 210 aircraft [1], and a life extension program for the Cessna C441 aircraft [2]. Details of both projects can be found in the references

Figure 1: Cessna 441 aircraft, from [2]

REFERENCES:

1. Eric J. Whitney, Kim White AEA Aerospace Group, *Fatigue and Damage Tolerance Evaluation of Cessna 210 Wing Carry-through Structure*, 2010 Aircraft Airworthiness and Sustainment Conference, Brisbane, 17-18 August 2010

2. Kim White, Graham Swannell AEA Aerospace Group, *Cessna 441 life extension*, 2010 Aircraft Airworthiness and Sustainment Conference, Brisbane, 17-18 August 2010

3. NEW ZEALAND

3.1.1 Investigations into the Fatigue Enhancement Provided by the Hole Cold Expansion Process Using Accurate 3D FEA Simulations and Fatigue Testing (S.J Houghton, S.K. Campbell and A.D James [Defence Technology Agency, Auckland])

Enhancing the fatigue performance of aging aircraft structures is of significant concern for military and civil operators worldwide. One such method involves cold expanding the fastener hole to exploit a region of residual compressive stresses in the region surrounding fastener hole. The beneficial effect derived from this process depends on the magnitude and distribution of the residual stress surrounding the hole, therefore accurate identification of residual stress profiles is critical to the evaluation of the life of aircraft structure containing cold expanded fastener holes.

A 3-D Finite Element (FE) simulation of the hole cold expansion process was created. Advances in FE technology allowed this simulation to closely represent the physical expansion induced by the commercial split-sleeve process. The simulation included a post-expansion loading step that applied a remote tensile load to the cold expanded hole to estimate the interaction of the residual stresses and stress concentration of the open hole.

The FE simulations indicated a significant 3-D nature of the residual stress field, with notable variations through the thickness of the specimen. The magnitude of compressive residual stress was lowest at the mandrel entry face for the cold expansion process. It increased to a maximum at the mid-plane of the specimen, before decreasing to an intermediate value at the mandrel exit face. Application of a remote tensile load reduced the compressive residual stress at the bore of the hole. At a tensile stress threshold, the analysis showed the stress state at the mandrel entry face reaches a state at which fatigue damage and crack development can occur if the imposed stresses include a cyclic component.

A constant amplitude fatigue testing programme was conducted to ascertain the correlation of predicted residual stress fields and the fatigue properties of cold expanded fastener holes. For constant amplitude fatigue loading below the tensile threshold, small fatigue cracks initiated and then arrested, with no crack growth after 10 Million fatigue cycles. There was good correlation between the size and shape of these cracks and the residual stress field predicted by the FE simulation.

For constant amplitude fatigue loading above the threshold, similarly shaped fatigue cracks would initiate and grow comparatively slowly. The initial crack front was typically a full through crack at the bore of the fastener hole but crack growth was faster at the mandrel entry face, in essence behaving like a corner crack. This corner crack eventually transitioned to a traditional through crack well outside the region of compressive residual stress. This through crack rapidly propagated to failure.

The combination of the stress distributions from the FEA and the fatigue testing results have provided insights into the behaviour of fatigue cracks in the presence of residual stress fields induced by the hole cold expansion process. This knowledge will assist in the ongoing management of aircraft structure containing cold expanded fastener holes.
The FEA and fatigue testing confirmed the mandrel entry face as the critical fatigue location. Unfortunately for a number of aerospace applications this represents the least inspectable location. The testing results also confirmed the existence of an effective fatigue endurance limit.

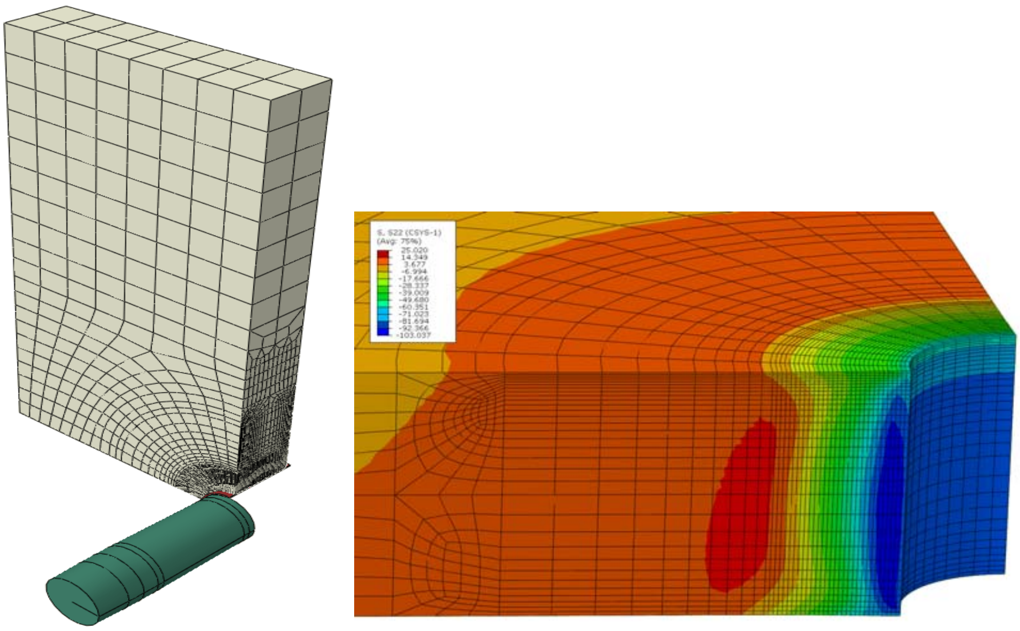


Figure 1: Finite Element model (left) and residual stress profile for the cold expansion process

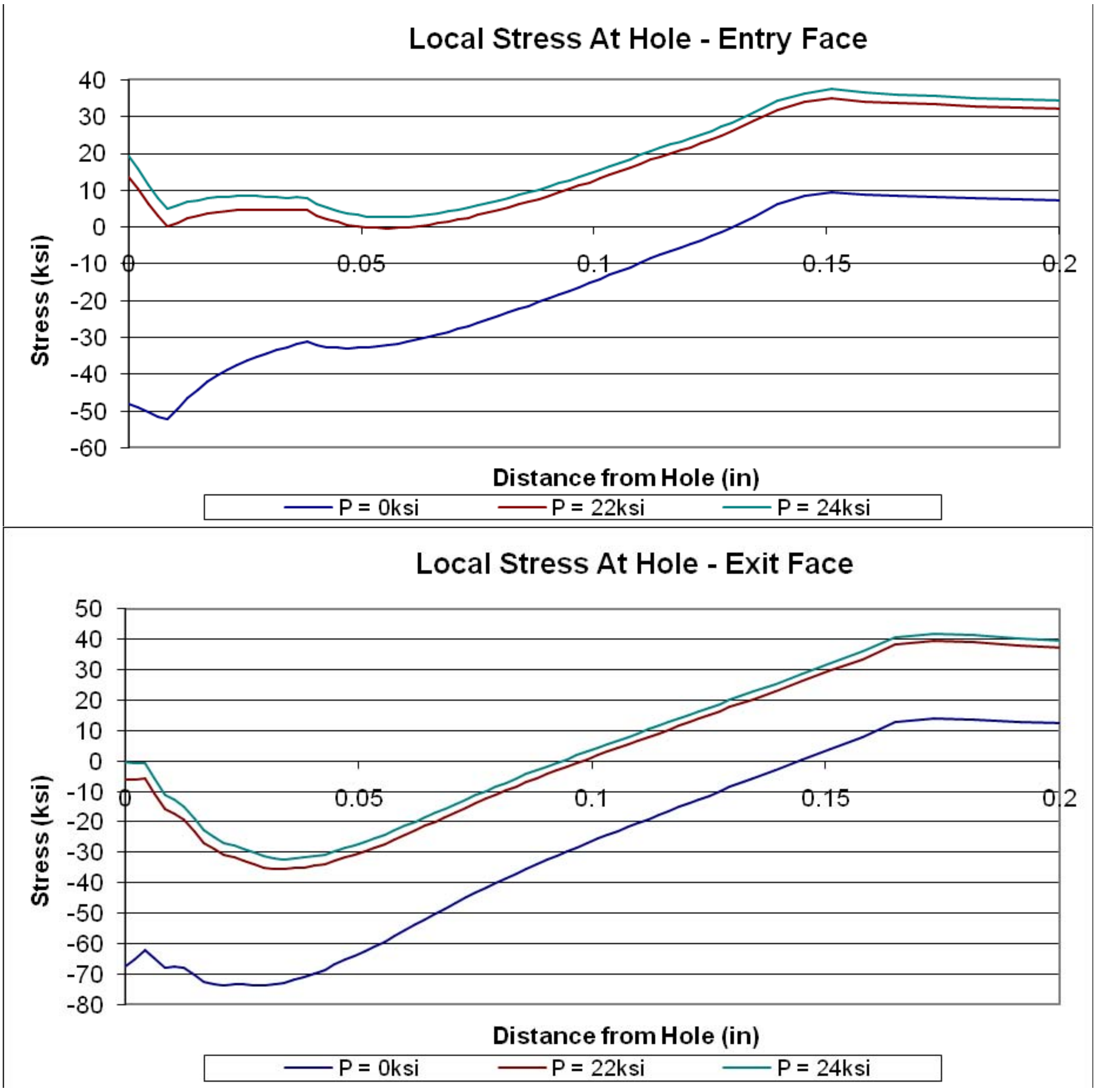


Figure 2: Tangential residual stress profiles for measured at the mandrel entry face (upper) and mandrel exit face (lower) for each of three different levels of remote tensile load.

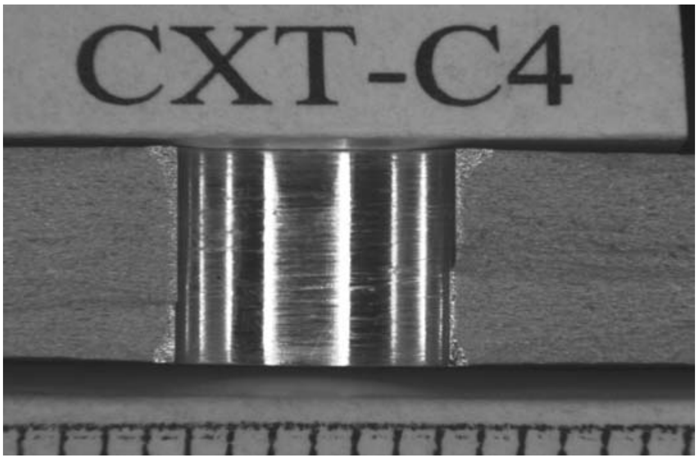


Figure 3: Fracture surface for a non-propagating fatigue crack (after 40 million cycles with a peak stress of 18 ksi at R=0.1).

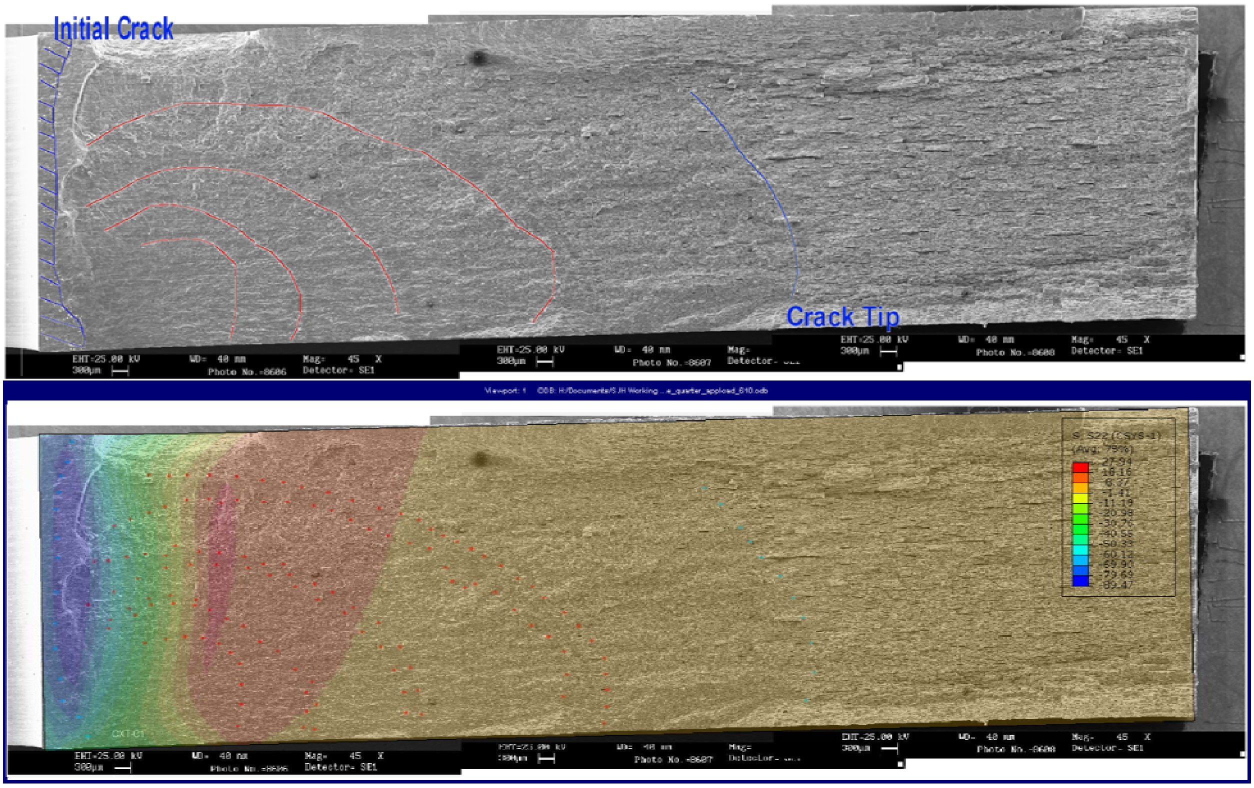


Figure 4: Fracture surface for a propagating fatigue crack (1.5 million cycles, peak stress 24 ksi at R=0.1). Marker bands (coloured in the top image) show successive positions of the crack front. The lower image overlays the FE predicted tangential stresses for an applied load of 24 ksi

3.1.2 Feasibility Study on the Effectiveness of Digital Image Correlation as an Improved Non-Destructive Inspection Technique for Adhesively Bonded Helicopter Rotor Blades (D.A Johns [Defence Technology Agency, Auckland] and M. Battley [University of Auckland])

The Royal New Zealand Air Force (RNZAF) operates a fleet of Bell UH-1H helicopters. In 2001 while operationally deployed, a significant fatigue crack was discovered on a rotor blade. Subsequent investigations revealed that the fatigue crack initiated as a result of disbonding between a steel grip pad and the aluminium blade structure on the underside of the blade surrounding the main rotor attachment pin.

Since 2001 the continued airworthiness of the RNZAF fleet has been managed on a Safety by Inspection (SBI) method, employing a novel Non-Destructive Inspection (NDI) technique developed by the Defence Technology Agency (DTA). This technique has been previously reported to ICAF conferences. The novel NDI technique involves placing a strain gauge across the steel/aluminium bondline to measure relative displacement between the two. A baseline displacement is measured at blade delivery, and then measured at frequent intervals during the service operation of the blade. However, the technique is labour intensive, requiring the removal of blades from the aircraft, application of strain gauges, and re-fitting of the blades. After re-fitting the blades, a rotor balance must be performed.

Recently the University of Auckland has developed a Digital Image Correlation (DIC) strain measurement system, using low cost commercially available hardware and University developed processing software. DIC involves applying a random “speckle” paint pattern to the surface for which strain field measurement is required. Image processing and analysis software then tracks the motion of the speckles as loads are applied to the structure. Application of image change tracking software enables a full field representation of the displacements of the imaged surfaces.

DTA and the University of Auckland have undertaken a series of trials to determine the feasibility of using DIC to measure the relative movement of the bonded elements of rotor blade end fittings. A critical requirement for the test is to be able to determine the integrity of the critical adhesive bond without removing the blade.

Results to date have been obtained using a 5 megapixel camera at distances of 230mm, 400mm and 630mm, trialled on a sample of three blades. The technique shows promise. The trials have provided indications out of plane displacement, which affect the image tracking algorithm employed by DIC. Techniques to help resolve this by improving the image sharpness (using adjustments of lens focal length, lighting and exposure times) are underway.

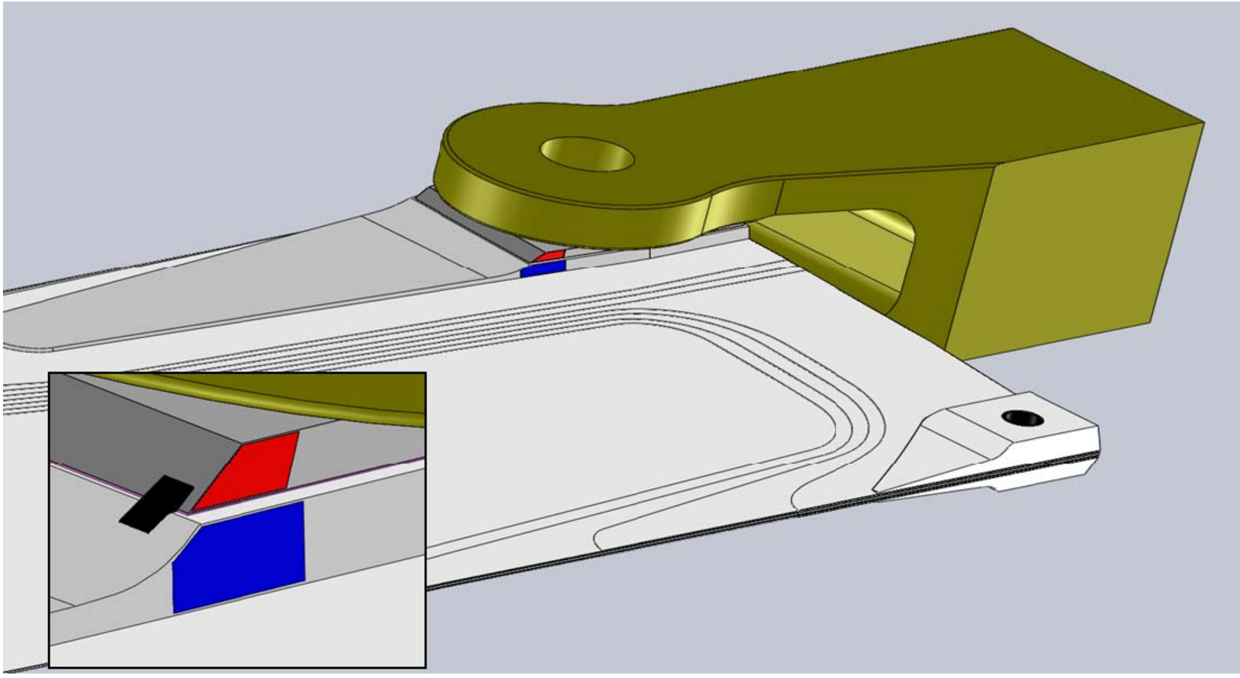


Figure 1: The underside of the UH-1H Rotor Blade and attachment clevis. The inset shows the Strain Gauge Technique gauge location (black) and the areas where a DIC speckle pattern was applied (Red and Blue)

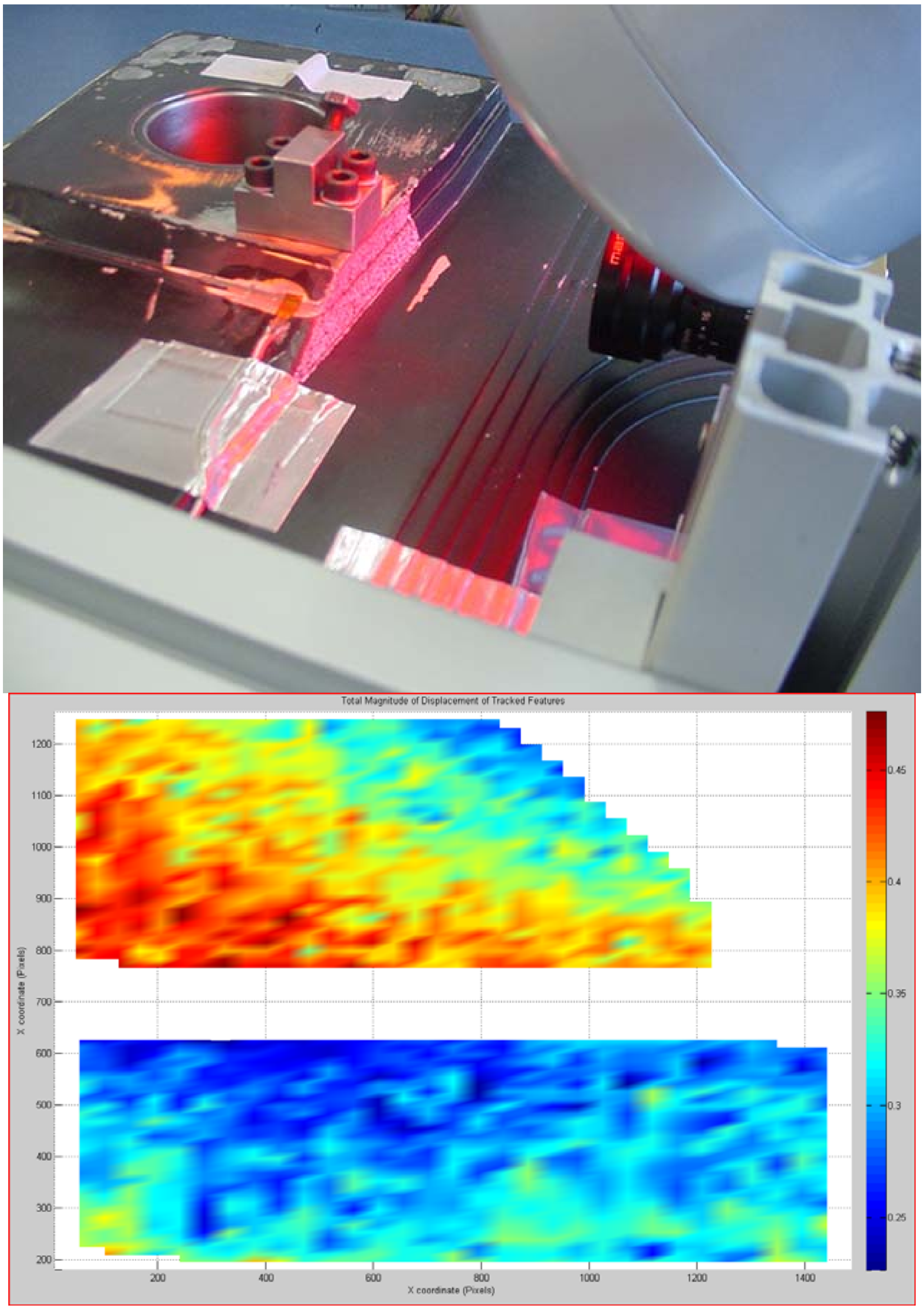


Figure 2: A preliminary trial of the DIC, using a low resolution camera and unrepresentative loading and no rotor attachment clevis (upper image) showed promise in measuring displacement fields (lower image).

3.1.3 Efficient Numerical Determination of Complex 3-D Stress Intensity Factors (S.K. Campbell and S.J Houghton [Defence Technology Agency, Auckland])

The use of fracture mechanics based methods to predict the fatigue life of aircraft has become standard practice. One of the key parameters in such methods is the Stress Intensity Factor (SIF) which is used to describe the stress field near the crack tip. Historically, significant effort has been made to develop analytical, closed form solutions for the SIF for different crack configurations (geometry and loading). These solutions are often available for a limited range of crack configurations, however through the principle of superposition it has been possible to estimate the SIF for an idealised model of most crack configurations experienced in the aircraft industry.

With the advances in computational analysis, it has been possible to further expand the crack configurations for which SIFs can be developed. However, the determination of SIF by computational analysis has traditionally required the time consuming and difficult process of developing a carefully designed computational model (such as a Finite Element mesh).

With recent significant advances in computational analysis tools, it is possible to quickly develop sophisticated computational models of cracking problems. These models do not rely on superposition, and are capable of capturing the effects of contact (loaded fasteners), residual stresses, complex stress fields near the cracked region and non-standard crack front geometries.

This investigation demonstrated and compared the capability of three different computational tools (conventional h-Finite Element models in ABAQUS, p-Finite Element models in StressCheck and Extended Finite Element (XFEM) models in ABAQUS) to determine the Stress Intensity factor for three different complex 3-D crack configurations. The configurations considered are a non-standard (sigmoidal) crack tip at the edge of an open fastener hole, an elliptical crack in a pin loaded lug (including contact between the pin and lug) and an elliptical corner crack in the hole of an unbalanced lap splice (including the fastener and contact between the fastener and hole).

The SIFs were estimated from path independent contour integrals (J-Integral) around the crack tips for a number of points along the crack front. Despite the models being linear elastic, it was found that for accuracy, the path selected for evaluating the contour integral needed to be larger than the size of the zone of crack tip plasticity.

The predicted SIFs demonstrated the need for a controlled computational mesh around the crack tip if a smooth curve of SIF along the crack front is to be achieved. For the p-Finite Element models, the best solutions were attained with three rings of geometrically graded elements around the crack tip. The best results for the h-Finite Element models were achieved using meshes containing a ring of wedge elements at the crack tip surrounded by a number of rings of geometrically graded orthogonal brick elements. While XFEM accounts for the crack tip independent of the computational mesh, it was found that the best solutions were achieved when the crack tip split a brick element into two equal parts.

Therefore, the best results were achieved with brick elements that were orthogonal to the crack.

Computational stability of contact problems where an edge of the crack contacts with another surface was found to be sensitive to the computational mesh selected.

It was found that all of the computational tools tested offered a viable method for efficiently determining the SIF for crack configurations that are not available in traditional closed form solutions or handbooks. Each of the three tools evaluated offered different strengths but was limited in some way. The p-Finite Element method as implemented in Stress Check significantly simplifies the mesh design process eases the creation of models with complex crack fronts. However, the contact models in Stress Check were found to be less robust than those in ABAQUS.

Mesh design within ABAQUS is greatly aided by the pre-processor ABAQUS CAE, but required significant effort to design an acceptable mesh for complex geometries. The h-Finite Element models offered robust stable contact and acceptable computational costs when used with second order elements.

The ABAQUS implementation of XFEM is a relatively new feature, and is limited to first order elements. Therefore, the computational meshes required were highly refined. The extra refinement was difficult to achieve for complex crack geometries, presenting additional meshing difficulties to maintain acceptable element shapes and can result in additional computational overheads.

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

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18. DSTO RESEARCH LIBRARY THESAURUS http://web-vic.dsto.defence.gov.au/workareas/library/resources/dsto_thesaurus.shtml

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

19. ABSTRACT
This document has been prepared for presentation to the 32nd Conference and 26th Symposium of the International Committee on Aeronautical Fatigue scheduled to be held in Montreal, Canada May 29th to June 3rd 2011. 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 2009 to March 2011. The review covers fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.