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A Review of Australian Aeronautical Fatigue and Structural Integrity Investigations During April 2019 to April 2021

Defence Science and Technology Group

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EXECUTIVE SUMMARY

This document has been prepared to support the aims and objectives of the International Committee on Aeronautical Fatigue and Structural Integrity (ICAF). The report contains summaries of the research, engineering and technology activities in the field of aeronautical fatigue and structural integrity that have occurred at Australian research laboratories, universities and within industry during April 2019 to April 2021. Due to the novel coronavirus (COVID-19) pandemic, the ICAF Symposium during which these national reviews are usually presented will not occur in 2021. Instead, this document will be published on the Committee's website www.icaf.aero.

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GLOSSARY

AM	Additive manufacture
AFRL	Air Force Research Laboratory (USA)
ARI	Acetate replica inspection
ASI	Aircraft structural integrity
CASG	Capability Acquisition and Sustainment Group
CDM	Continuum damage mechanics
CFRP	Carbon fibre-reinforced polymer
CIC	Corrosion inhibiting compound
CP	Commercially pure
DD	Domain decomposition
DFR	Detail fatigue rating
DIC	Digital image correlation
DLL	Design limit load
DSTG	Defence Science and Technology Group
DUL	Design ultimate load
EBSD	Electron back-scatter diffraction
EDM	Electric discharge machining
EFH	Equivalent flight hours
EIFS	Equivalent initial flaw size
FAA	Federal Aviation Administration (USA)
FCG	Fatigue crack growth
FE	Finite element
FSFT	Full-scale fatigue test
HAFT-TD	Helicopter Advanced Fatigue Test – Test Demonstrator
HCF	High cycle fatigue
HIM	Helium ion microscope
HOWSAT	Hornet Outer Wing Static Test
ICAF	International Committee on Aeronautical Fatigue and Structural Integrity
LAD	Laser additive deposition
LDED	Laser-directed energy deposition
LIF	Lead-In-Fighter (Hawk Mk127)

mCT	micro-focus computed tomography
NDT	Non-destructive testing
OEM	Original equipment manufacturer
OPERAND	Operational Load Analysis and Asset Diagnostics
OW	Outer Wing
PRA	Probabilistic risk assessment
QF	Quantitative fractography
RAAF	Royal Australian Air Force
SFC	Single fibre composite
SLAP/SLEP	Structural Life Assessment/Extension Program
SSP	Structural Substantiation Program
TI	Test interpretation
TSA	Thermoelastic stress analysis
USAF	United States Air Force
VA	Variable amplitude
WFT	Wing fatigue test

1. INTRODUCTION

This report presents a review of Australian research, engineering and technology activities in the field of aeronautical fatigue and structural integrity from April 2019 to April 2021. The review includes inputs from the government, industry and academic institutions listed below. The editor acknowledges each of these contributions with appreciation and encourages readers to approach the listed author(s) directly with any enquiries; contact details are provided at the end of each section.

- Deakin University, www.deakin.edu.au
- Defence Science and Technology Group (DSTG), www.dst.defence.gov.au
- DMTC Pty Ltd, www.dmtc.com.au
- James Cook University, www.jcu.edu.au
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2. AIRCRAFT STRUCTURAL INTEGRITY RESEARCH ACTIVITIES

2.1. The Potential for Structural Simulation to Augment Full Scale Fatigue Testing of Aircrafts – A. Kotousov and A. Khanna [the University of Adelaide]

As a part of a Defence Science and Technology Group (DSTG) funded project undertaken at the University of Adelaide, Australia, a review article 'The Potential for Structural Simulation to Augment Full Scale Fatigue Testing: A Review' has been published in Progress in Aerospace Sciences journal in 2020 [1]. This review article has been aimed at supporting the ongoing development of an integrated analytical and experimental fatigue testing capability at DSTG.

The main body of the paper presents a review of the current state-of-the-art of structural-scale modelling tools for predicting the fatigue life of critical aircraft components. The review is divided into four sections, each covering one domain within the broad theme of virtual fatigue testing, namely

1. Multiscale fatigue damage modelling
2. Full-scale structural modelling of airframes
3. Uncertainty quantification and modelling
4. Experimental methods for fatigue damage characterisation.

These research domains were identified as the most critical in the development of virtual simulation tools in several recent articles, technical reports and books. As part of the review, a relative assessment of the accuracy and consistency of different methodologies was performed, along with an evaluation of the sensitivity of fatigue predictions to a range of relevant inputs. Based on the literature survey, technological gaps are identified, and research initiatives are recommended to address some of the major Aircraft Structural Integrity challenges faced by military and civilian platforms, including but not limited to:

1. Widespread fatigue, e.g. thousands of similarly stressed local details within large airframes
2. Multi-material built-up and multi-axial loading, e.g. composite wing skin/metallic airframe

3. Local/global coupling, e.g. localised damage/repair interdependency with global structural response.

Table 1 summarises the authors' recommendations for the most promising research initiatives to address aircraft structural integrity (ASI) challenges faced by aircraft manufacturers and operators worldwide. The recommendations are divided into two parts, namely (1) well-developed modelling and experimental tools, which are expected to provide immediate benefits upon the implementation and (2) methods currently under development, which are promising but not mature enough to be implemented directly towards virtual fatigue testing of aircraft structures.

Table 1 Down-selection of research initiatives to address ASI challenges

ASI Challenge	Recommendations
Fatigue life estimation	**Short term**: The utilisation of generalised 3D strip-yield model can address issues with necessity to apply empirical constraint factors and improve predictive capabilities of fatigue crack growth models for physically small and long cracks. The development of effective numerical algorithms can allow cycle-by-cycle crack growth simulations avoiding compression or truncation of fatigue load history.
	Long term: (1) Finite element based crystal plasticity models validated by micro-DIC and other advanced micro-characterisation techniques can improve the prediction of the fatigue life consumed in crack nucleation and microstructurally-small growth phases, which constitutes majority of the total fatigue life in the HCF regime. (2) Direct numerical simulations of short and long crack propagation, 3D and plasticity effects.
Widespread fatigue damage	**Short term**: Reliability-based methods, such as the detail fatigue rating (DFR) method or deterministic methods, based on the equivalent initial flaw size (EIFS) concept can be applied.
	Long term: Fatigue life or reliability models must be calibrated using experimental data gathered from tear-down inspections.
Multi-material built-up & multi-axial loading	**Short term**: Data fusion from multiple experimental techniques, such as DIC, TSA and ultrasonic inspection, can provide a complete picture of the stress state and failure mode in multi-material structures subjected to complex loading. This is essential for the validation of structural models.
	Long term: Physics-based multiscale modelling approaches, incorporating experimentally-validated multiaxial failure criteria have been developed for

	composites and metallic alloys. These may be integrated with commercial FE packages to facilitate design and certification activities.
Local/Global coupling	**Short term**: Generalised (extended) finite element methods (G-XFEM) and its modifications can enable direct modelling of fatigue cracks in 2D and 3D, without specifying the crack propagation path.
	Long term: The methods for non-intrusive coupling between local and global scales, based on algorithms originally developed for domain decomposition (DD), are expected to be very beneficial for full-scale structural modelling in the presence of localised defects or stiffness degradation.
Uncertainty quantification	**Short term**: Markov chain and Bayesian methods for calibration of probabilistic distributions using the test data, fatigue damage models as well as for the generation of the variable amplitude sequence for fatigue life evaluation purposes and full-scale testing.
	Long term: Quantification of epistemic uncertainties arising from modelling assumptions via round-robin analyses (blind prediction) of full-scale and coupon level experimental tests.
Residual strength assessment	**Short term**: Surface topology or roughness evolution coupled with the infrared thermography imaging technique as well as linear and nonlinear ultrasonic techniques for the evaluation of material nonlinearities can be utilised for qualitative assessment of fatigue damage accumulation at local/global scales.
	Long term: Multiscale progressive damage models calibrated using the above experimental techniques can provide quantitative assessment of the residual strength of structural components at the end of the design fatigue life.

Acknowledgements

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2.2. Propagation of Short Cracks Exposed to Underloads – I. Field and S.A. Barter [RMIT University]

The current damage tolerance design philosophy that pervades the aircraft industry relies on the prediction of fatigue crack growth rates to dictate inspection intervals. However, due to the inability to accurately predict these growth rates under realistic loading, conservative inspection intervals are often required, imparting an undesirable cost burden on aircraft operators. Further research into the detrimental effects of underloads linked with the variation in growth for short cracks will help more accurate prediction of fatigue crack growth, and ultimately allow for more realistic inspection intervals. This research will use quantitative fractography to extract the growth rates due to blocks of underloads and baseline loads in specially designed spectra. High resolution imaging techniques, such as helium ion microscopy, will be used to extend this quantitative fractography data as close to the crack initiation point as possible. Electron backscatter diffraction (EBSD) analysis of the underlying structure beneath the fracture surface will then allow for insight into potential governing mechanisms, such as crystal orientation, residual stresses, and slip system orientations.

Aircraft are exposed to significant variations in loading over their lifetimes, generally due to manoeuvres, buffeting, or ground-air-ground cycles. Such variable amplitude (VA) loading causes issues for fatigue crack growth prediction, since the order in which loads are imparted is known to significantly impact a cracks' growth rate [1]. To solve this, it is critical that a greater understanding of how underloads effect the growth of fatigue cracks is developed. The current research community acknowledges their detrimental effects, however, there is still much debate as to the exact mechanisms behind them [2-5]. Additionally, the possibility of these mechanisms changing as a crack develops must be explored. For example, it has been suggested that short cracks rely on shear driving force [6], as opposed to the perpendicular driving force for long cracks; as well as there being a reduction in crack closure for short cracks [7]. It is likely that these changes in mechanism are what causes short fatigue cracks to grow at a driving force below the threshold of long cracks [8]. The importance of gathering crack growth data for such shorts cracks to aid prediction has been clearly demonstrated by Main et al. [9].

This research will focus on conducting a range of fatigue tests on AA7050-T7451 coupons that are exposed to specially designed loading spectra which include periodic underloads imparted in baseline cycles, such as the spectrum shown in figure 1. After this, quantitative fractography (QF) analysis of identified blocks of loading will allow crack growth rates to be found. Such a blocked method of crack growth analysis will allow extraction of crack growth during the underload cycles, compared to that of the following baseline cycles. This will provide further insight into the immediate and lasting effects of underloads. The ability for short crack growth analysis to be conducted using this method has been clearly demonstrated by Barter et al. [10].

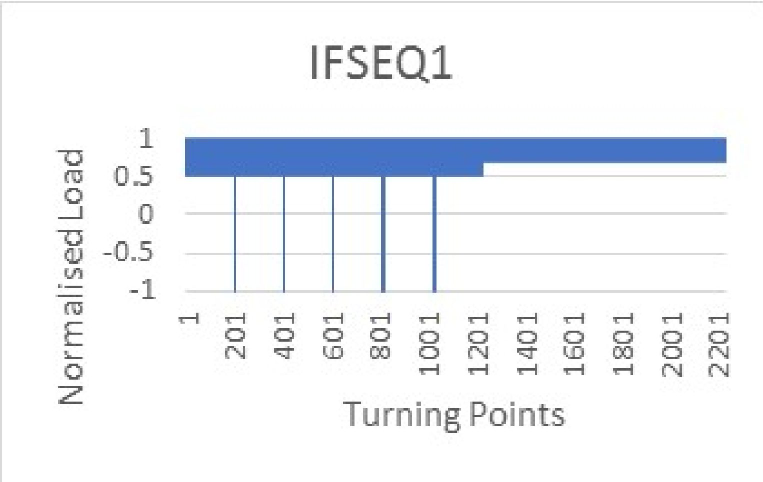

Figure 1 Normalised spectra IFSEQ1, designed to elucidate the effects of multiple underloads in succession. The spectrum consists of increasing numbers of underloads in succession (1-5), each followed by 100 cycles (200 turning points) of R = 0.5 loading. 500 cycles of R = 0.7 loads are applied at the end of each block to act as an additional marker band.

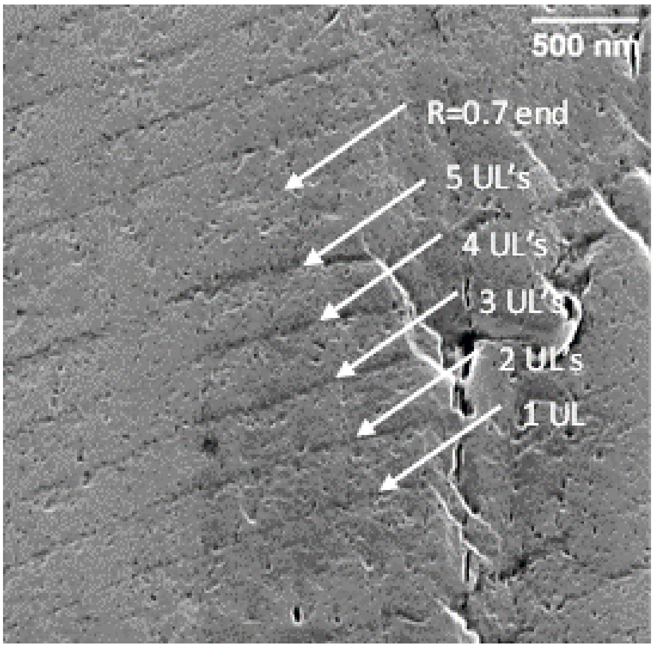

Figure 2 HIM image taken at an overall crack length of 140 µm. This specimen was subject to IFSEQ1 spectrum, and the dark bands clearly define the locations of the underloads. The R = 0.5 cycles are growing the crack at a little more than 1 nm per cycle. Rubbing has damaged the surface, but better images are expected with improved experimental techniques.

In order to gather QF data for cracks that are as short as possible, helium ion microscopy (HIM) will be used to aid with high resolution imaging of the marker bands, as shown in figure 2. To the author's knowledge, this imaging technique has never been used to aid with such crack growth measurements, and may allow for unprecedented short crack growth rate data.

Analysis of the fracture path compared to the underlying residual stresses and local grain crystal orientations will then be conducted using EBSD. This will allow the surface features associated with underload events, such as those seen in Figure 3, to be compared to any local areas of material residual strain, and how these features compare to the likely slip or cleavage planes of the local grains. This should result in a greater understanding of potential governing mechanisms associated with underloads and how they change as the crack grows.

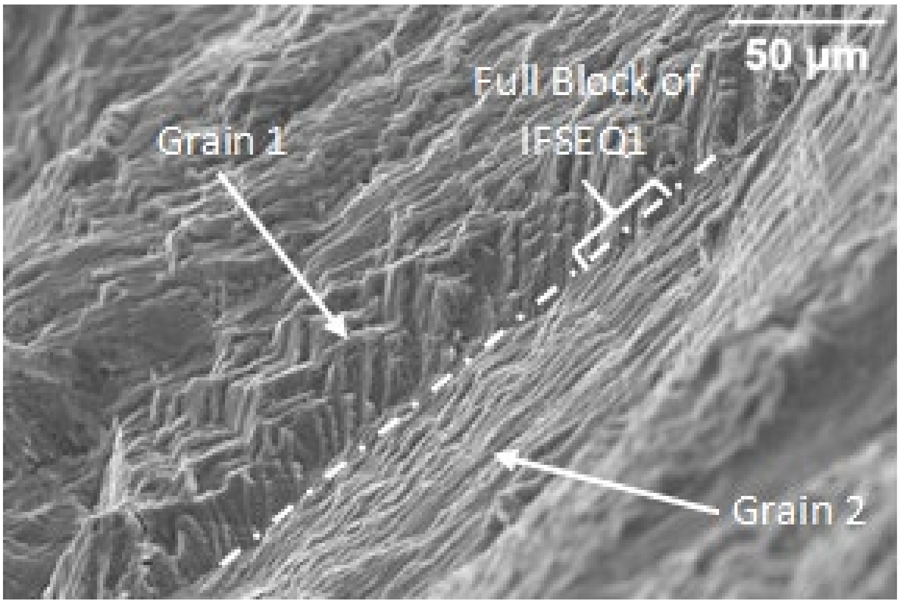

Figure 3 A high angle SEM image of the fracture surface of a specimen subject to the IFSEQ1 spectrum. Clear variations in surface features are observable across two different grains, suggesting a reliance on grain orientation

This research is in its early stages, though some initial data has been gathered using QF, and images taken on the HIM, validating the ability to use the proposed methodology. Additional coupons are yet to be tested with different loading spectra designed to further study the effects of underload spacing, magnitude, and consecutive loading effects. QF data is also yet to be extracted from any of the HIM images.

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2.3. Efficient and Effective Fatigue Life Management for Metallic Airframe Components – L. Molent and B. Dixon [DSTG]

Fatigue of metallic airframe components remains one of the main threats to airworthiness and aircraft availability. Further, due to these potential threats, considerable resources need to be committed to prevent these occurrences. Two recent publications by DSTG [1, 2] highlight that a deficit in the current understanding of the metallic fatigue phenomenon forces designers to adopt less optimal designs and operators to accept less than optimum aircraft availability. Reference [1] describes the current understanding of airframe describes of fatigue including:

- Fatigue is a localised material events associated with the nucleation and propagation of fatigue cracks.
- Fatigue cracks in metallic airframe components typically nucleate early in the service life from discontinuities (generally >> 0.1 mm) that are either inherent to the material or that result from manufacture (e.g. hole drilling).
- The main influences on fatigue crack propagation including local stress spectrum and environment, material and nucleating discontinuity.
- The main characteristics of fatigue cracks that may lead to airframe structural.

Both [1] and [2] briefly describe the current frameworks that are commonly used for metallic airframe component fatigue management and seek to build and extend from those foundations. Reference [2] describes a number of innovative approaches based on observations of the growth of real fatigue cracks under service representative conditions that have been successfully used to extend existing management frameworks to allow more optimal fatigue management of Royal Australian Air Force (RAAF) aircraft including:

- The lead crack lifing methodology
- The cubic rule for the assessment of fatigue propagation rate estimation
- The effective block approach to fatigue propagation rate estimation
- The Hartman-Schijve fatigue crack growth variant.

Reference [2] looks to build on such improvements and assesses where current management frameworks can be improved upon and where further research is needed in order to attain greater efficiency for metallic airframe component fatigue management. Recommendations for future research include:

- Development of methods to characterise the size distributions of the manufacturing and production-induced crack nucleating discontinuities.

- Develop better methods for predicting the propagation behaviour of short crack populations from nucleation to detectable sizes.
- Development of methods to improve testing rates for full-scale airframe fatigue tests in order to maximise the amount of service relevant fatigue information that can be attained within limited timeframes and budgets.

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2.4. Low Cycle Fatigue Testing of Aviation Grade Aluminium Alloys – J. Singh, G. Wheatley [James Cook University], R. Branco and F. Antunes [Universidade de Coimbra]

This project is based on the low cycle fatigue testing of the aluminium alloys 6061 and 2024. The aim is to investigate the effect of pre-strain on the fatigue life and to determine the strain ratio i.e. minimum and maximum stress to fatigue. A critical analysis of relevant literature concludes that the alloy undergoes cyclic softening at lower strain amplitudes and cyclic hardening at high strain amplitudes. The experiment is conducted in two parts: Pre-strain tests and Strain ratio tests. The pre strain tests are carried out for 0%, 4% and 8% pre strain tensile respectively and the strain ration tests would encompass strain ratios -1, 0 and 0.5. The strain amplitudes for both tests would be in the interval 0.6-1.75%. The pre-strain tests would be instrumental in determining the relation of tensile pre-strain histories to fatigue life and the strain ratio tests would shed light on the cyclic strain-softening behaviour of the alloys.

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2.5. Aeroacoustic Energy Harvesting from an Aircraft Cavity – D. J. Munk, S.D. Moss, E.J.G. Ellul, J. Flicker [DSTG] and G. A. Vio [University of Sydney]

It is well known that, under certain flight conditions, the dynamic pressure acting on an aircraft structural cavity can produce cavity-flow fluctuations known as aeroacoustic noise. The aerodynamic forces created by this noise can excite structural resonance and fatigue. Typical examples include the weapon bays of military aircraft, landing gear wheel-wells of civil aircraft, and crevices in the surface of air vehicles [1]. Recent work has examined the possibility of harvesting energy from the structural resonances excited by these cavity-flow fluctuations [2, 3]. The aim of this work is to develop and validate an electro-mechanical computational model for predicting the power generation capability of an aeroacoustic energy harvester. Specifically, operating in a dynamic environment and within geometries commonly found on aircraft. Such computational models can then be used to optimise the design of the energy harvesters to maximise their power output, ultimately reducing aircraft downtime and the need for on-board power storage.

The design of the cavity, i.e. its length (L) and depth (D), influences the frequency of the pressure fluctuations. In order to predict the possible frequency range for the acoustic mode for different cavity geometries, the Rossiter and Block analytical models have been employed. An electromechanical model of the energy harvester has been created to predict the voltage produced for different cavity-floor plate lengths. The model consists of a spring-steel plate having a width of 153 mm and a thickness of 0.381 mm. The length of the steel plate can be varied from 75–200 mm. The material properties used for the steel plate are a Young's modulus of 205 GPa, a density of 7850 kg m-3 and a Poisson ratio of 0.28. The single fibre composite (SFC) transducer, based on the relaxor ferroelectric single crystal Mn-PMN-PZT, has dimensions 50 × 25 × 0.175 mm^3 corresponding to the length, width and thickness, respectively, and is placed in the centre of the steel plate. The material properties of the [011] poled anisotropic SFC transducer are taken as follows:

$$E = \begin{bmatrix} 149 & 65.2 & 17.2 & & & \\ 65.2 & 84.5 & 73.4 & & 0 & \\ 17.2 & 73.4 & 107 & & & \\ & & & 51.3 & 0 & 0 \\ & 0 & & 0 & 5.596 & 0 \\ & & & 0 & 0 & 37.3 \end{bmatrix} \text{GPa and } \rho = 7900 \text{ kgm}^{-3}$$

With the following piezoelectric constants:

$$d = \begin{bmatrix} 0 & 0 & 0 & 0 & 19.97(10^{-10}) & 0 \\ 0 & 0 & 0 & 1.45(10^{-10}) & 0 & 0 \\ 4.05(10^{-10}) & -11(10^{-10}) & 8.20(10^{-10}) & 0 & 0 & 0 \end{bmatrix} \mathrm{mV}^{-1}$$

Finally, the capacitance is assumed to be $C_p = 1.42(10^{-9})$ F. A 0.3 mm bond line is included in the model, between the transducer and the plate, where the material properties of the bond line are taken as a Young's modulus of 2.5 GPa, a density of 1420 kg m^{-3} and a Poisson's ratio of 0.34. Finally, a border of double-sided tape, positioned on the underside of the plate, is used to secure the energy harvester to the cavity floor. The double-sided tape has a thickness of 1.2 mm, a Young's modulus of 345 kPa, a density of 175 kg m^{-3} and a Poisson's ratio of 0.3. The expected frequency range for the Rossiter model for cavity lengths of 0.075–0.4 m and an airflow velocity of 60 m s^{-1} is given in Figure 1 (left), with frequencies up to 330 Hz. The fundamental Eigen mode of the energy harvester is also shown in Figure 1 (right).

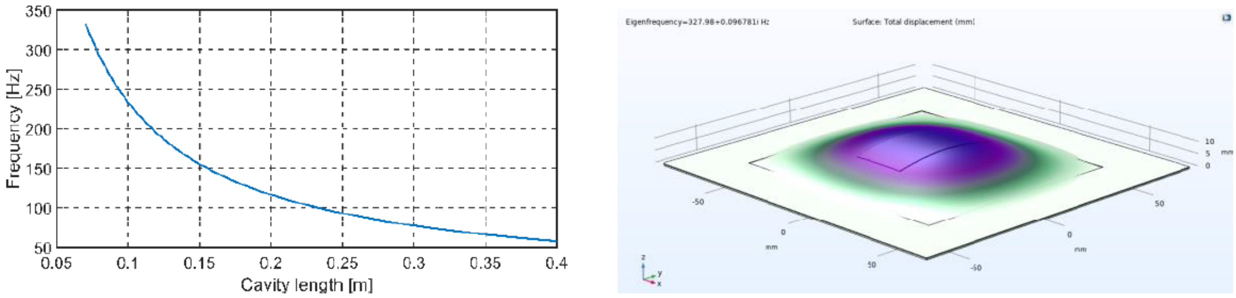

Figure 1 Rossiter analytical model of acoustic frequencies produced by cavity, and modelled fundamental mode at ~330 Hz for cavity-floor with bonded SFC

The fundamental frequency of the energy harvester is predicted to be approximately 330 Hz, which corresponds to the upper bound of the predicted acoustic frequency range (Figure 1 (left)). This may result in large excitations of the structure, which can lead to excessive vibrations and structural failure by fatigue. Predicting and monitoring the dynamic response of the structure in such environments is crucial to ensuring its safe operation over the operational life-cycle of the aircraft.

To predict the power capability of the energy harvester, the SFC open circuit voltage has been modelled using a 1 N load applied to the centre of the steel plate and with the underside of the tape clamped. Figure 2 shows the predicted stress distribution of the SFC transducer for an energy harvester with a free-plate length of 90 mm which is the plate length that generates the largest voltage for the given configuration.

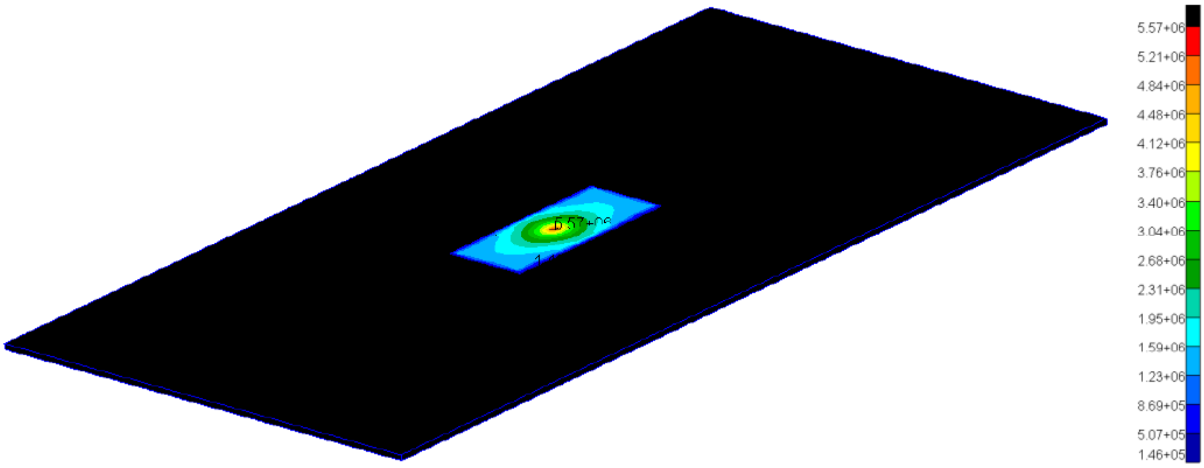

Figure 2 Modelled energy harvester, spring-steel cavity floor with centrally mounted SFC transducer. Stress distribution over 90 mm harvester under (F = 1 N). Note: only stress of transducer crystal material displayed.

This electro-mechanical model, along with aeroacoustic models, will be utilized to predict the performance of the harvesters in aircraft cavities. Experimental validation is currently underway in the low-speed wind tunnel at the University of Sydney. Ultimately, these models will help to development of energy harvesters to power sensors to measure the aeroacoustic noise, and to assist the development of suitable controls.

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2.6. Requirements and Variability Affecting the Durability of Bonded Joints – R. Jones and D. Peng [Monash University], A. J. Kinloch [Imperial College London], J. G Michopoulos, [NRL].

MIL-STD-1530D Section 5.2.4 states: 'Stress and strength analysis shall be conducted to substantiate that sufficient static strength is provided to react all design loading conditions without yielding, detrimental deformations and detrimental damage at design limit loads and without structural failure at design ultimate loads.' The key phrase here is the statement that there be no yielding and detrimental damage at design limit load (DLL). There is a related statement in JSSG-2006. The difference being that the JSSG-2006 requirement is linked to 115% DLL. However, the current approach to designing bonded joints, as delineated in CMH17-3G, is to ensure that the strength of the joint is beneath design ultimate load.

Unfortunately, as shown in [1], it does not follow that, if a joint does not fail at design ultimate load (DUL), it will not yield at 115% DLL, or at 100% DLL. As such, the way in which A4EI is commonly used does not necessarily ensure that a bonded joint will meet the guidelines given in the PABST report [2], or the certification requirements inherent in MIL-STD-1530D and JSSG-2006, see [1] for details. As such, it does not ensure that the adhesive bond will be durable.

It was also revealed in [1] that fatigue crack growth in both nano-reinforced epoxies, and structural adhesives can be captured using the Hartman-Schijve crack growth equation, and that the scatter in crack growth in adhesives can be modelled by allowing for variability in the fatigue threshold. A methodology was also established for estimating a valid upper-bound curve, for cohesive failure in the adhesive, which encompasses all the experimental data and provides a conservative fatigue crack growth curve. The subsequent paper [3] illustrated how this formulation can be used to assess the variability seen in the adhesive crack growth curves on operational life. The problem studied in [3] was a simple double lap joint, see Figure 1, where the inner and outer 2024-T3 aluminium alloy adherends were bonded using FM73. The effect of small variations in the fatigue threshold on crack growth in the joint under a Falstaff flight load spectrum with a maximum remote stress in the inner adherend of 193 MPa is shown in Figure 1.

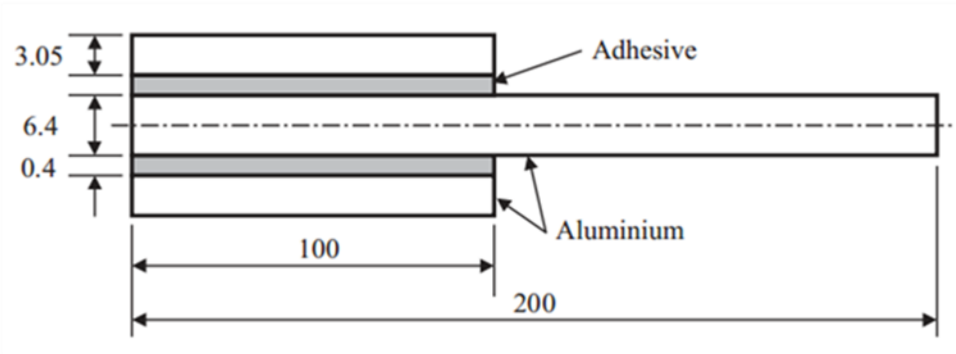

Figure 1 Double lad joint studied in [3]

The analysis used the Hartman-Schijve crack growth equation:

$$\frac{da}{dN} = D\left[\frac{\Delta\sqrt{G} - \Delta\sqrt{G_{thr}}}{\sqrt{\{1-\sqrt{G_{max}}/\sqrt{A}\}}}\right]^n \quad \text{Equation 1}$$

where D, n are constants, A is the cyclic toughness, and Δ√(Gthr) is fatigue threshold. The constants used were taken from a prior study [1]. The results of this analysis are shown in Figure 2, where we see that the service-life of the specimen is a relatively strong function of the value of the fatigue threshold that is employed in the analyses. This reinforces the need to determine a statistically-valid value of Δ√(Gthr)for the fatigue threshold, and hence determine an 'upper-bound' FCG rate curve where the variability of the FCG rate for the adhesive joint is taken into account.

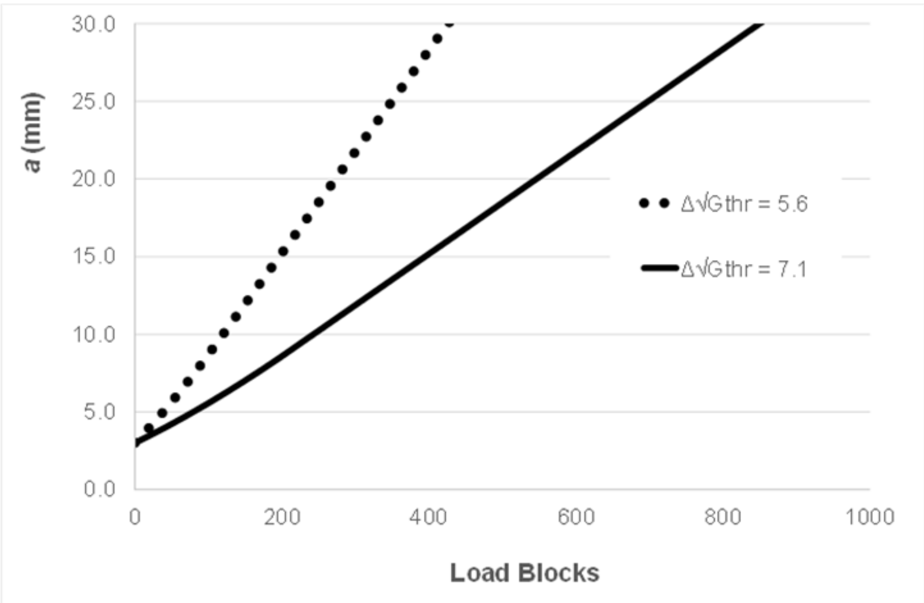

Figure 2 Predicted crack length, a, histories as a function of the number of load blocks for the specimen tested under a FALSTAFF flight-load spectrum with a remote maximum stress of 193.5 MPa

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2.7. Effects of Peen Forming and Interference Fit fasteners on Aircraft Wing Skin Durability – J. Carroll [RMIT University]

Many military and commercial aircraft wing-skins are formed into shape using a die-less forming process called peen-forming. This process consists of bombarding the flat skin panel with small metal spheres, which causes denting, stretching, and large-scale curvature without expensive and cumbersome tooling. However, this process causes the introduction of not only small dents in the material surface but internal stresses due to the deformation of the material. Generally, these ‘residual’ stresses have been assumed to be beneficial to the durability and crack resistance of the skin, as they are compressive at the surface level. Recent fatigue tests of certain aircraft have indicated that this may not always be the case, and fatigue cracks have been observed to preferentially initiate at the base of these dents when located near interference fit fasteners, an example is given in Figure 1. This is wholly unexpected as interference fit fasteners are also designed to improve airframe durability; and hole surfaces, rather than peening dents, are generally considered to be the preferential failure sites.

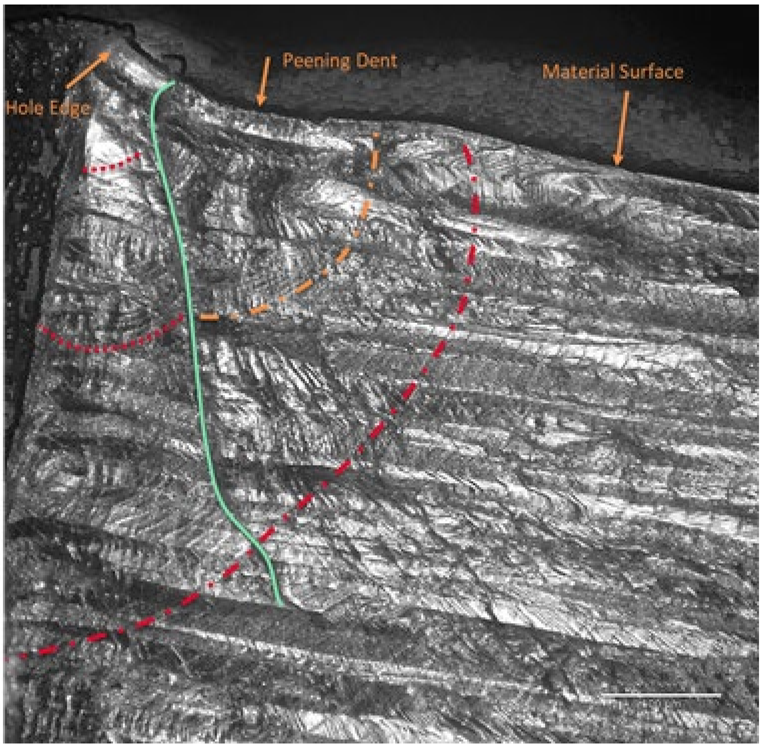

Figure 1 Example of Crack Initiation at the base of peening dent. Crack initiation locations at edge and dents annotated.

Potential cracking away from a fastener hole may be detrimental to the expected life of the aircraft, as well as to typical repair and life extension process, by accelerating the growth of the crack and lessening the likelihood of detection and removal of the crack via typical methods. It is therefore important to characterise the cause of this initiation site, the cracks growth, and the ultimate effect on repair and maintenance schedules. By mapping the initiation and subsequent crack growth to the combination of residual stresses caused by the interference fit fastener and the peening process, maintenance and life extension processes can be designed in response.

Through initial coupon testing using tactical transport spectrum loading with varied distances of peening dents to the fastener; and via thorough analyses of test results using quantitative fractography, the effect of the peening process on both crack initiation and early crack growth can be interrogated, with crack growth curves being compared to fasteners absent of peening. This process is currently underway. Additionally, the critical location that which the peening dents do not affect the coupons fatigue behaviour can be differentiated.

Subsequently, mapping of the residual stress field around the interference fit fastener using highly spatially resolved analysis techniques, including micro x-ray diffraction and focused ion beam milling with digital image correlation, will be conducted, and be used to cross-validate numerically simulated stress fields. The numerical model will then be used to predict possible stress field variation between coupons, as a function of the material and assembly tolerances. Then modern crack initiation prediction methods will be trialled to establish their efficacy of predictions, given a detailed understanding of the initial stress states.

Finally, using the highly spatially resolved stress fields as initial conditions as well as bespoke beta solutions from the numerical model, LEFM predictions of the crack growth will be made, with a specific focus on the small/short crack regime. Direct comparison to the fractographic results will be made, comparing crack front shapes and advancement in known residual stress fields, and the effect of using true short/small crack data as compared to common adjustments of long crack data. This will inform test interpretation of full-scale fatigue tests (FSFTs) by benchmarking observed crack growth against standard industry tools and methods, allowing for prediction and understanding of cracks that may grow in this area.

Currently, coupon testing is underway, with a recreation of the initiation site accomplished. An example fractography of a crack initiating from the base of a dent, with auxiliary crack initiation at the hole edge is shown in Figure 1. Residual stress measurement and mapping is beginning this year. Ultimately, by fully interrogating the residual stress field, and the cracks response to the field, as well as the initiation site, we will be able to not only understand fully this failure mode but also understand the effect of residual stress fields on the growth and shape of small and short cracks.

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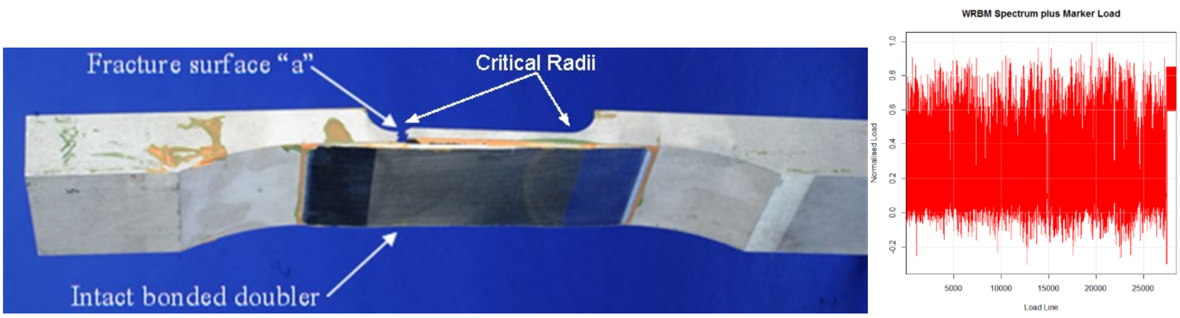
2.8. Assessing the Fatigue Effectiveness of a Life Enhancing Repair/ Reinforcement to an Aging Aircraft Structure – S. A. Barter, L. Robertson [RMIT University] and G. Swanton [DSTG]

The need to minimise the risk of structural fatigue failure of aging aircraft components presents challenges to developing and certifying adequate structural modifications and repairs, while minimising their installation and maintenance costs. An issue associated with any structural repair is the risk of not being able to assure the removal of any existing fatigue cracking via adequate material removal that may be smaller than conventional non-destructive inspection (NDI) methods can detect. This uncertainty can limit the trust placed in a repair and drive short inspection intervals.

The investigation presented here considers a critical short life location in a flange of a large integrally stiffened bulkhead, which is repaired sometime after entering service, and is likely to contain fatigue cracking smaller than typical NDI thresholds that remain after repair. Two repairs were investigated here using aluminium alloy 7050-T7451 coupons (Figure 1 A):

1. Ceramic bead peening applied to the critical radius
2. Ceramic bead peening applied to the critical radius plus a non-ideally located bonded aluminium alloy 7075-T6 doubler, which was located on the opposite side of the coupon to the critical radii where fatigue cracking grew from.

Un-repaired coupons were fatigue cycled with 12 blocks of a representative fighter wing root bending moment spectrum at a representative stress level to induce very small fatigue cracks (Figure 1 B). A marker load was added to the spectrum to aid with quantitative fractography. After repair, the coupons were then fatigue cycled to failure without removing these cracks to assess the repair effectiveness.



A B

Figure 1 (A) The coupon designed to represent a critical radius detail on a bulkhead. This image of a coupon after fracture, showing the coupon had separated, however the doubler remained intact and bonded to each half of the coupon. (B) The fatigue spectrum plus marker load applied to the coupons.

This study found that the bonded doubler produced a marked improvement in fatigue life even though it was not ideally positioned, and despite the presence of undetected cracking, while peening was also beneficial when applied over the top of cracking. The crack growth progression was seen to slow considerably after the repair via peening and a doubler (Figure 2) resulting in a significant life extension.

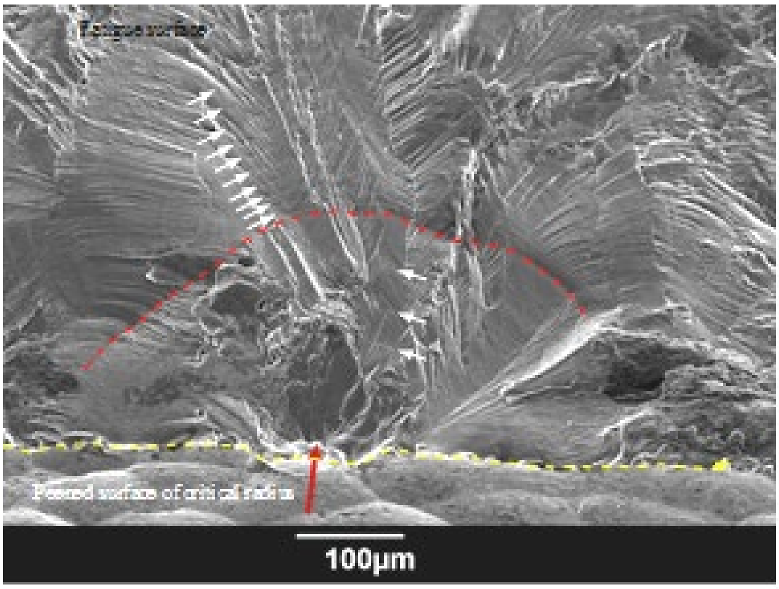


Figure 2 The origin region of crack KDIJ-03a.The original initiating discontinuity is marked with a red arrow, while the size of the crack at the time the repair was applied is indicated with a red dashed line. Crack growth progression marks (indicated with white arrows) have a much smaller spacing after the repair indicating the slower rate of crack growth. The peening dents on the surface of the coupon can be observed in the lower portion of the image.

While peening can be considered to provide an additional level of preventative repair, the size any cracks at the time of peening is crucial, since the peening effect is quite shallow, and larger cracks may escape the residual stress region produced by the peening. Alternatively, the stress reduction resulting from a bonded doubler will continue to provide some extension to the life of the structural detail even while larger cracks are present simply because of the reduction in local stress. A combination of peening and a bonded doubler will provide the greatest benefit to structural life (Figure 3 A), where the doubler can be relied upon to reduce local stresses and thereby reduce the rate of crack growth, including for cracks which may have grown beyond the effective peened depth.

Despite the somewhat disadvantageous location of the bonded aluminium alloy doublers applied to the coupons tested here, and the presence of cracks which were larger than the effective peened depth at the time repairs were introduced (Figure 3 B), the application of these preventative modifications did result in a useful extension in coupon life for each repair configuration.

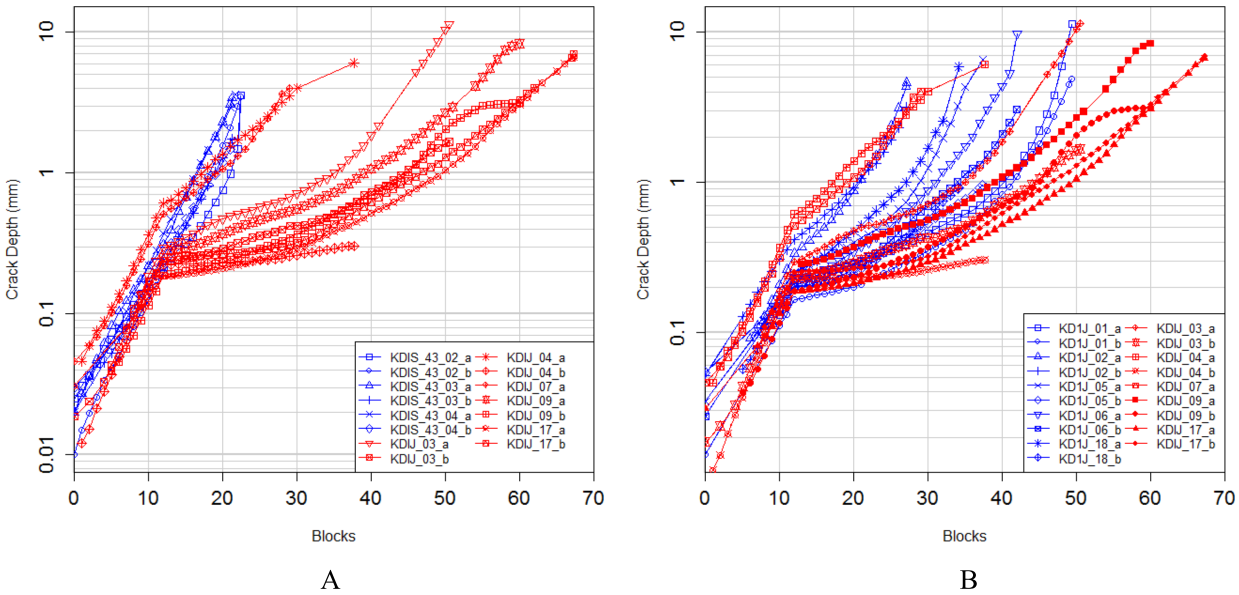


Figure 3 (A) Comparison of crack growth from un-repaired coupons (blue plots) with coupons repaired via peening and a bonded doubler (red plots) where it can be seen that the crack growth rate before peening and the addition of the doubler was similar to the crack growth rates for the un-repaired coupons. (B) Comparison of peen repaired coupons (blue plots) with coupons repaired with a peen and bonded doubler (red plots) shows that the deepest cracks (KDIJ-04a and KDIJ-07a) grew as if the peening was not present after the doublers were applied, whereas all the other cracks showed retardation greater than would be expected from the doubler alone.

In no case was cracking seen to arrest, rather it was slowed by the lowering of the stresses by the bonded doubler and retarded by the residual stresses of the peening. Therefore the use of these preventative modifications should not be treated as ‘terminating’ repairs, since the possibility of further crack extension is still present, and recurring inspections may still need to be maintained to mitigate against the risks that such cracks could become a risk to the airframe’s structural integrity. However, these repairs may still be a cost-effective solution when compared to more significant structural repairs or component replacement. A representative coupon test program, such as the one conducted here may provide extended inspection intervals leading to reduced on-going maintenance costs.

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2.9. Helicopter Advanced Fatigue Test – Technology Demonstrator: Developing Advanced Spectrum Editing Techniques for Future Full-Scale Fatigue Tests – B. Krieg [DSTG]

DSTG is currently addressing the challenge of helicopter structural fatigue via the Helicopter Advanced Fatigue Test-Technology Demonstrator (HAFT-TD) program. The aim is to mature the technologies required to perform a FSFT of a helicopter airframe, which will see the application of accelerated, representative helicopter loading to a Seahawk test article. Previous DSTG studies have successfully demonstrated novel spectrum compression approaches that compress load sequences whilst preserving damage content at individual airframe locations. However, spectrum compression for a FSFT presents the complexity of performing this operation for multi-channel test load inputs, rather than stress sequences at individual locations. Researchers at DSTG are currently exploring ways to adapt the Compression Strain-Life algorithm with Selective Scaling (CSL-SS) algorithm, which has been previously demonstrated to be effective at compressing spectra at individual locations (local application) to the whole airframe (global application).

The CSL-SS algorithm assigns and removes the lowest damaging cycles, preserving the most damaging cycles unmodified, and scaling the amplitudes of the remaining cycles. Analytical damage models and coupon test programs can determine the combination of omission and scaling that works at the local level to maintain the desired fatigue response; however, when translating to the global level, the following complications need to be addressed:

1. Removing a load line from a single location removes it from all locations.
2. The level of scaling required to preserve damage response is different for all locations.
3. Scaling at a global level has different effects at the local level, as determined by the relationship between test input loads and local airframe responses.
4. Damage effects from truncation algorithms need to generalise well to all locations, including those that are not directly considered by the spectrum compression process. This requires the preservation of load shapes throughout the airframe.
5. When scaling the amplitudes of specific cycles, the peak of the cycle is increased while the valley is decreased. When locations are out of phase with each other, this can mean scaling can increase damage at one location, whilst decreasing it at another.

The effects are demonstrated in the following figures. Figure 1 shows the baseline, rainflow stress cycle densities for three different airframe locations. Figure 2 shows the effect of the omission of small cycles from each location on damage. The points show the resulting life and load line density for each location when omission is performed either individually by location, or simultaneously between them. This figure demonstrates the loss of efficiency in the omission algorithms when applied globally. Finally, Figure 3 was produced by applying the CSL-SS algorithm at scaling factors of 1 to 1.5 to the input loads. This demonstrates the irregularity of the effects of scaling between locations.

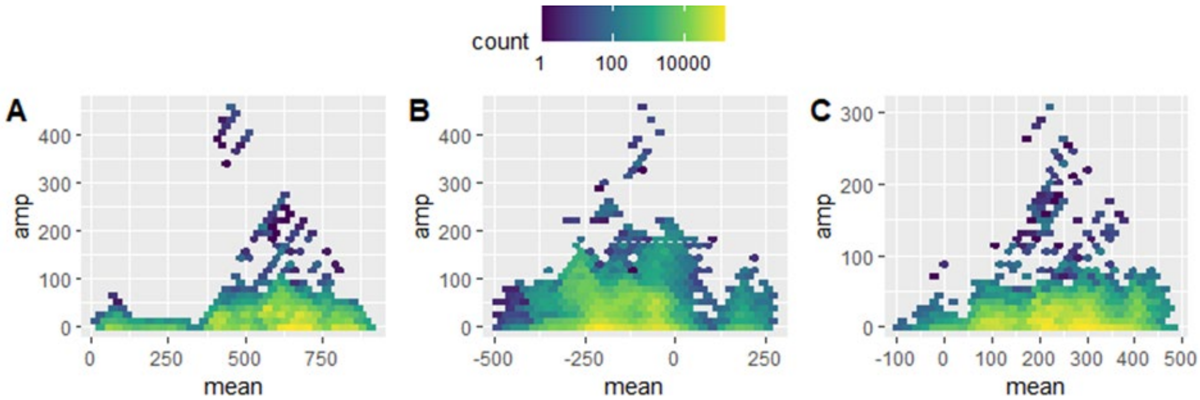

Figure 1 Rainflow density plots for three helicopter airframe locations (units MPa)

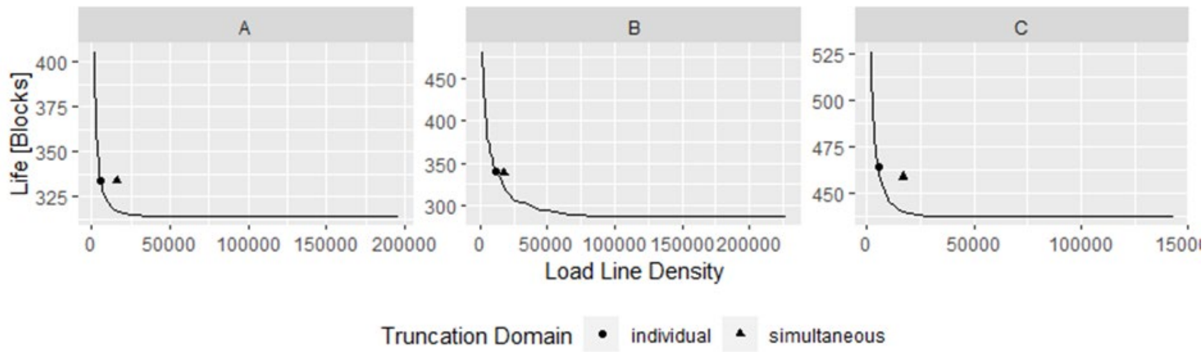

Figure 2 Effect of omission on life for various locations. Points how the effect of individual (local) verse simultaneous (global) omission for a select load line density.

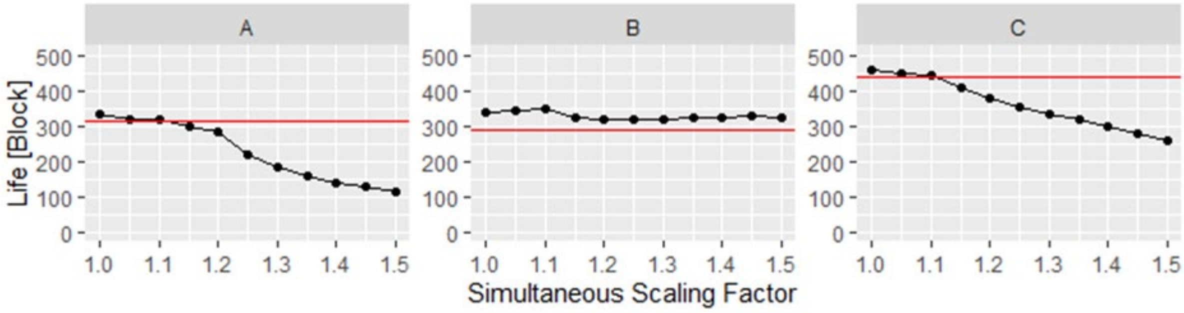

Figure 3 Effect of simultaneous (global) scaling on Life for various airframe locations (red line represents target life)

Ongoing development of these algorithms is supported by a significant analytical modelling effort. In particular, an effective model has been developed using an in-house code called 'easigro'. This code is computationally fast, efficient and ideal for batch processing. Using this model, four distinct airframe locations which are life limiting under crack growth or crack initiation assumptions have been identified, and are being used to drive the spectrum compression process. The next steps will use this code to iteratively optimise these spectrum compression algorithms for the FSFT scenario. This analytical effort is also supported by an experimental coupon test program, the results of which will ultimately support the development of a compressed load set to be applied to the FSFT article during the technology demonstrator phase of the program.

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2.10. High Fidelity Residual Strength Assessment for Composite Aircraft Sustainment – K. Jacobs, R. Ladani, R. Das, A. Orifici [RMIT University] and A. Harman [DSTG]

The sustainment of composite aircraft requires analysis tools and knowledge suitable to support airworthiness decision-making. Compressive residual strength assessment following impact is particularly challenging, given the complex damage field at the impact location, multiple damage modes, and structural phenomena such as buckling. The use of high fidelity simulation tools has significant potential to address this challenge, as it can allow tracking of the development and interaction of damage modes and incorporate complex physical phenomena and geometries. One aspect that requires further study is the strategy for incorporating the complex impact damage field into the model prior to the residual strength assessment, particularly as this can involve techniques and properties that are specific to the analysis tool and as high fidelity techniques provide opportunity for a wide variety in modelling options and representations.

In this work, a methodology for compressive residual strength assessment is investigated that uses the high fidelity computational analysis tool BSAM. The tool is capable of representing fibre fracture, matrix cracking and delamination at a ply-level, and capturing the initiation, progression and interaction of these modes. The effect of different strategies for incorporating the impact damage field are investigated, which includes the geometry and properties of the impact indentation, the representation of fibre fracture, the arrangement of mesh-independent cracks and material property degradation to capture matrix cracking, and the size and placement of delaminations that are modelled with cohesive elements. Results are compared with experimental results from literature for compression after impact coupons, to drive insight into the characteristics of each modelling strategy, as well as the damage mechanisms and interactions occurring in the experiment.

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2.11. Two Normalisation Approaches for Accounting for the Scatter in Fatigue Delamination Growth Curves – R. Jones, R.K. Singh Raman and D. Peng [Monash University], A. J. Kinloch [Imperial College London] and J. G Michopoulos [NRL]

The damage tolerance and durability requirements associated with composite and bonded US military aircraft are delineated in the USAF certification standard MIL-STD-1530D and the United States Joint Services Specification Guidelines JSSG-2006, and guidelines for achieving these requirements are given in US Composite Materials Handbook CMN-17-3G. MIL-STD-1530D explains that damage growth must be predictable and that the objectives of full-scale aircraft structure durability tests is to validate, or correct, the damage tolerance analysis. The question thus arises: How can we determine the necessary 'worst-case', i.e. upper bound, delamination curves needed to predict the growth of the fastest possible delamination? To answer to this question a means for accounting for this scatter and for determining the worst case delamination growth curves are given in [1], [2].

Another method that has been suggested is to use a the normalisation approach, whereby the energy release rate is divided by the resistance to delamination growth GR. Unfortunately, as shown in [3] whilst this approach does help reduce scatter it doesn't eliminate the scatter in the near threshold region. This is illustrated in Figure 1 where da/dN is plotted against G' = G'max (= (Gmax/GR(a))GC0), where GC0 is the initiation value of G, for IM7/977-3. The data in this plot is from [4].

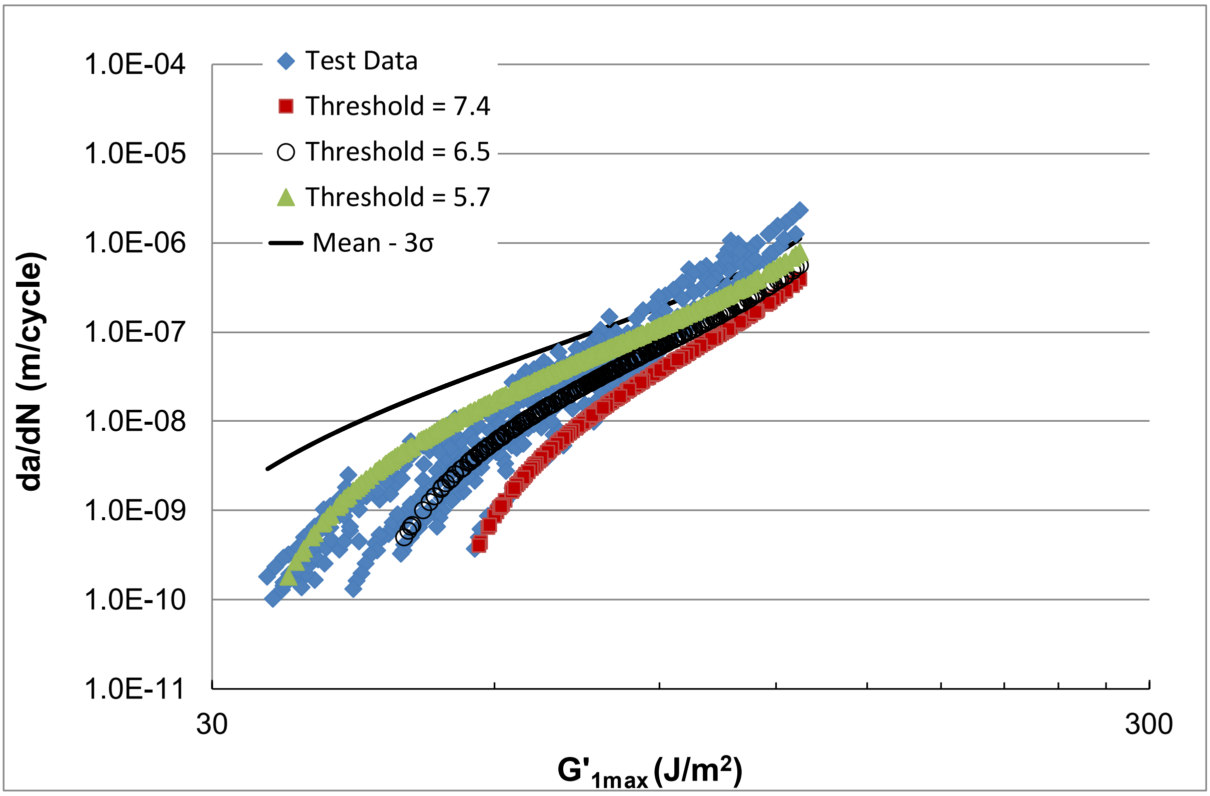

Figure 1 The 'toughness' normalised delamination growth curves for IM7-9773 and the upper bound (mean - 3σ curves, from [3].

This shortcoming is largely alleviated if G is normalised by dividing by its threshold value and da/dN is plotted against G''max (=(Gmax/Gmax,th) x Gthr), where Gthr is the worst case value of the fatigue threshold as determined using the Hartman-Schijve delamination growth equation, see Figure 2.

In [3] this example is supplemented by a range of other examples associated with both delamination growth in composites and crack growth in both conventionally and additively manufactured metallic parts.

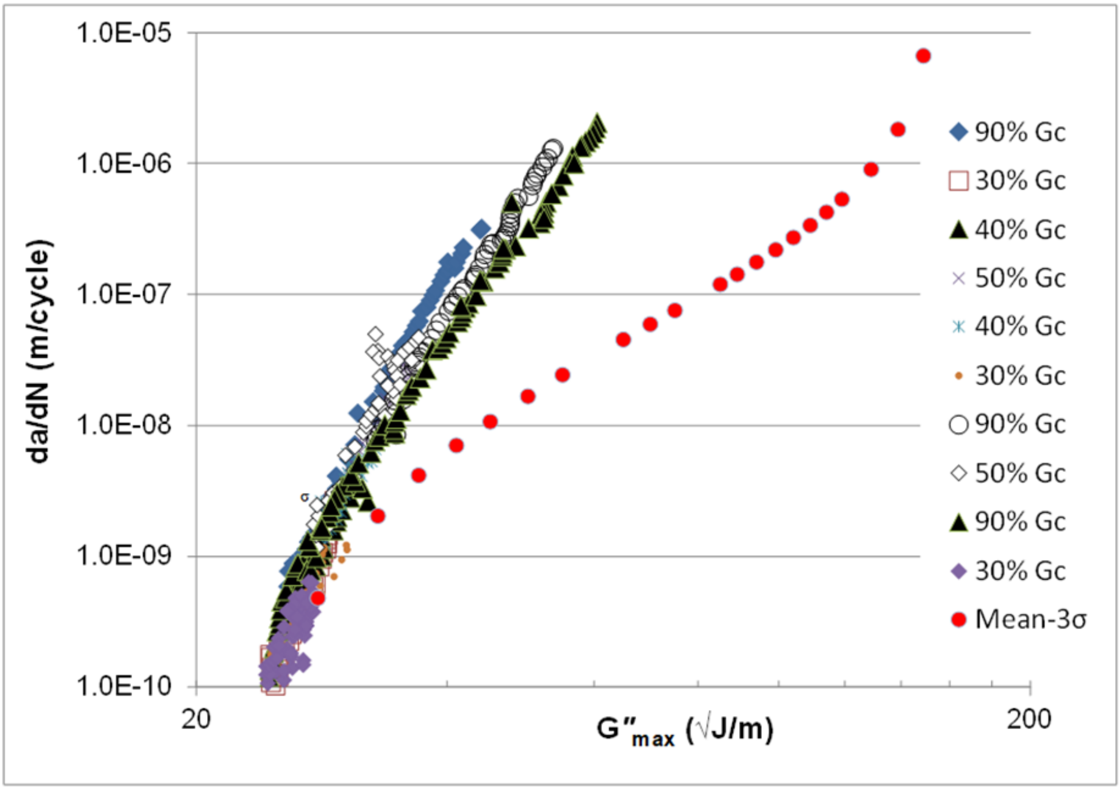

Figure 2 The ‘threshold‘ normalised delamination growth curves for IM7-9773 and the upper bound (mean - 3σ) curves, from [3]. This is the same data as shown in Figure 1.

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2.12. Efficient Damage Prediction in Fibre Reinforced Composites – J. Reiner [Deakin University]

Fibre-reinforced composites are widely used in a large range of applications thanks to their exceptional specific strength and stiffness. The properties of advanced composites can be tailored effectively to satisfy specific requirements by careful selection of raw materials, fibre architectures, lay-ups and processing parameters.

This large design space comes at a cost. To certify composites for aerospace structural applications, industry must rely on the building-block approach for which comprehensive testing programs must be undertaken. Designers require a high degree of confidence on the performance of advanced composites, in particular their actual capacity to safely carry loads. Hence a robust assessment of damage evolution and fracture is a critical aspect of material development.

Finite element (FE) simulation promises the possibility to reduce costs and efforts associated with these testing programs. However, despite tremendous efforts and partial success in establishing FE models and failure theories for advanced composites, the search for physics-based damage models, validated for industrial applications, is still ongoing.

The latest developments about a highly efficient FE-based continuum damage mechanics (CDM) simulation strategy is presented to predict the inelastic structural response of carbon fibre reinforced laminates. CDM maintains the continuity of the finite element mesh while accounting for damage by means of smearing the local stiffness reduction within the constitutive model.

Based on virtual calibration of damage in quasi-isotropic laminates subjected to over-height compact tension and compact compression tests, it is shown that the CDM approach is able to predict various load cases such as size effects in open-hole coupon test samples as well as the transverse impact behaviour and residual strength in compression after impact tests. Unlike time-consuming high fidelity damage simulation techniques, the presented simulation strategy can be applied to large-scale composites thanks to its high efficiency by capturing the essential physics with the least amount of complexity.

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2.13. An Innovative High-Fidelity Approach to Structural Health Monitoring – O. Levinski [DSTG], R. Carrese, P. Marzocca and S. Koschel [RMIT University]

Defence air platforms' structural health management is traditionally driven by scheduled inspection intervals and pre-emptive maintenance based on interpretation of structural certification test results or reactive based on unique or unexpected fleet in-service incidents. However, in the current environment of budget constraints and shrinking resources, a major shift towards actionable and pro-active condition-based maintenance is required to ensure the safe and efficient operation of aircraft and to significantly reduce fleet management and sustainment costs.

The airframe life management process's efficiency depends upon the reliability and accuracy of the fatigue life monitoring system, which aims to estimate the airframe fatigue accrual to manage usage of individual aircraft and meet the overall fleet capability. One of the most critical parts of the fatigue accrual evaluation process is the estimation of airframe service loads and stresses to predict the degradation of the aircraft life relative to the test substantiated service life. Detecting and tracking the global airframe health as well as local structural anomalies caused by fatigue and wear is a crucial component towards the development of a smart structural diagnostics capability to support the sustainment of the current and future air platforms.

This work is a collaborative effort between DSTG and RMIT University in developing a novel high-fidelity approach to structural health monitoring and individual aircraft tracking inspired by the aircraft digital twin concept. Operational Load Analysis and Asset Diagnostics (OPERAND) is a multi-physics analysis suite for structural health monitoring based on integrating current state-of-the-art software techniques, data-driven methods, and model-based approaches. This innovative structural diagnostics and prognostics framework aims to provide substantial savings to aircraft operators and optimise aircraft availability for improved operational effectiveness. It aims to enable pro-active condition-based aircraft maintenance by significantly improving airframe load and stress predictions for any flight condition or aircraft configuration.

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2.14. Buckling of Circular Cylindrical Shells of Finite Length under Radial External or Hydrostatic Pressure – L. Hart-Smith [Royal Aeronautical Society]

New analyses are presented for buckling of circular cylindrical thin shells of finite length under external radial pressure and under hydrostatic compression. Solutions cover both very long and very short cylinders with a transitional solution in-between. The problem is analysed both by a flat-plate analogy and as a thin-walled cylinder. The simpler flat-plate analysis is shown to be accurate for short and long cylinders, but it is also shown that the analysis as a cylinder is needed for cylinders of intermediate length. Two-different thin-shell theories are needed. The classical analysis, which is limited in validity to mode shapes for which radial displacements normal to the shell mid surface must be very much smaller than in-plane displacements, is needed to establish relationship between the longitudinal and circumferential wavelengths. This constrained mode shape is shown to refer not to the lowest buckling mode, but to some higher mode, for which the predicted bucking stress is higher than actual buckling stress associated with an unconstrained mode shape, for which the normal displacements are very much larger than the in-plane displacements. This lowest buckling mode is shown to be resisted by induced bending moments alone; the induced membrane forces needed to satisfy equilibrium are shown to have only a negligible effect on the buckling stress. The key difference between the two theories is that the classical strain-displacement relations would violate St. Venant's compatibility-of-strains equation if they were applied to the real first buckling mode, while the author's analysis does satisfy compatibility because the new expression for in-plane strain includes an additional term accounting for the linear influence of the normal displacement that was excluded from the classical analysis by the constraint on the mode shape. The results of the new analyses are compared with both previous analyses and with the experimental data.

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2.15. Root Cause Analyses of Damage Growth Indications Observed In-Service within Critical Titanium to Composite Bonded Joints – A. Harman, E. Yeo, M. Ibrahim, K. Tsoi [DSTG], A. Groszek [QinetiQ] and D. Slater [RAAF]

In 2017, damage growth was observed within critical bonded step lap joints that connected the composite wing skins to a specially designed titanium fitting attached to the fuselage (Figure 1). The original airworthiness certification basis for the structure provided by the original equipment manufacturer (OEM) did not permit growth of this nature (i.e. no growth permitted) during service. As a result, the operator in consultation with the airworthiness regulator prematurely retired the component from service. The retired structure was quarantined for further investigation into the root cause, and also for the purpose of developing an enhanced sovereign capability for sustaining bonded composite structures for future platforms.

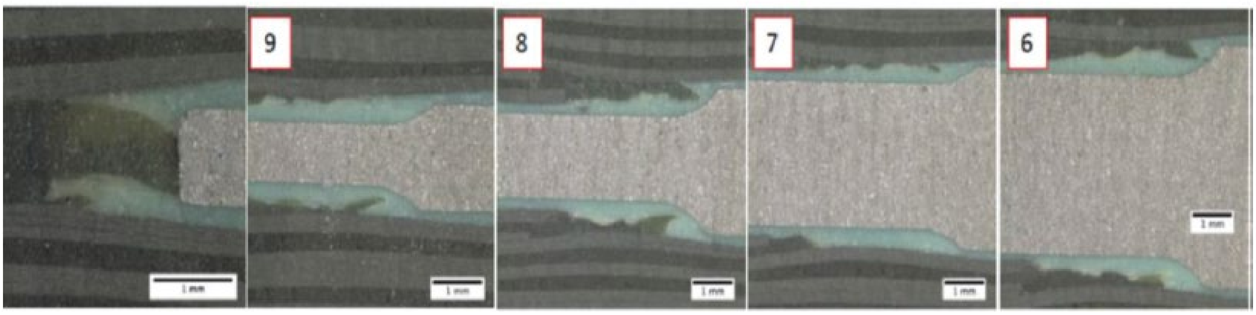

Figure 1 Optical micrograph of section through titanium to composite bonded step lap joint

Commencing in 2020, DSTG took custodianship of the affected components, and in partnership with QinetiQ and DASA began teardown inspections in an attempt to establish conclusively the root cause for the damage growth indications. The project entailed multiple phases, with the initial phase dedicated toward non-destructive inspection, with subsequent phases becoming progressively more detailed and destructive in nature. With reference to Figure 2, the non-destructive inspection phase utilised the same methods deployed at the time of discovery in 2017 but in more favourable laboratory conditions, and trialled new inspection methods currently in development at DSTG.

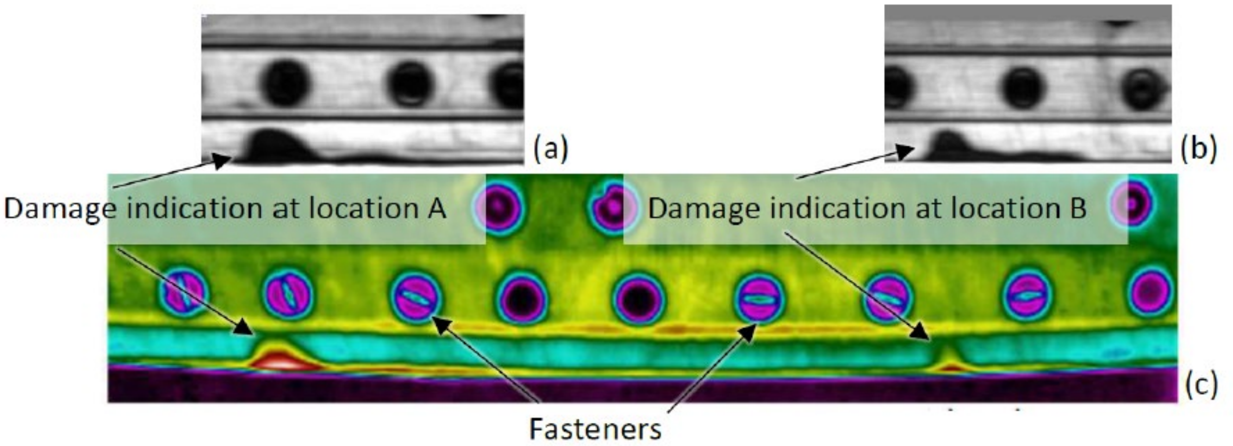

Figure 2 Non-destructive inspections of areas on bonded composite joint with known damage indications using (a)/(b) pulse echo ultrasonic inspection and (c) flash thermography

Following completion of the non-destructive inspection phase, teardown commenced to permit extraction of samples for inspection first with X-ray and neutron micro-focus computed tomography (mCT). Both XmCT and NmCT have been utilised successfully for characterising damage morphologies and composite materials and hybrid structures inclusive of composite and metallic components [1]. A number of subsequent inspections and evaluations, including titanium surface contamination assessment to further understand the root cause of the damage growth indications and develop new capabilities to assist future sustainment. In addition, test specimens are being designed for extraction in cooperation with NAVAIR and the Air Force Research Laboratory (AFRL) from the United States to measure locally the residual mechanical performance for comparison with predictions using high fidelity finite element models. The models will utilise new capabilities developed at AFRL permitting input of damage characterised using non-destructive methods for more accurate predictions.

The work so far has led to significantly enhanced levels of cooperation and mutual understanding between DSTG technologists, industry supporting fleet sustainment, DASA, AFRL and NAVAIR. This will assist greatly with guiding technology and capability development so as to maximise outcomes for RAAF operations.

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2.16. The Effect of CICs on the Fatigue Life of 1&1/2 Single-Shear Lap Joints with Hi-Lok and Rivet Fasteners – R. Ferber [the University of Adelaide]

Aircraft often operate in environments where they are exposed to moisture and other contaminants, leading to susceptibility of fastened joints to corrosive damage. This susceptibility can be attributed to the many mating surfaces and crevices within the joints being able to retain moisture and debris, as well as the breakdown of the primary protective coatings due to wear and tear. To combat corrosive damage, primary and secondary corrosion prevention methods are utilized. Corrosion inhibiting compounds (CICs) are widely used as a secondary corrosion prevention method in the sustainment of platforms. However, CICs can provide lubrication of the joint, which has been found to alter the fatigue behaviour.

The mechanical effects of CICs and faying surface sealants were investigated though experimental fatigue testing and FE modelling. A 1&1/2 single-shear lap joint design was used, with both Hi-Lok fasteners and solid rivets being considered as well as faying surface sealant. These joint configurations were designed in accordance with industry standards for aircraft structures and made with 7075-T6 aluminium. Each joint configuration was tested under constant amplitude fatigue loading until failure. Complimentary analysis includes strain measurement with digital image correlation (DIC), thermoelastic stress analysis (TSA) and strain gauges.

The addition of CICs and sealant was found to alter the fatigue life in certain configurations. For the unsealed Hi-Lok and rivet samples the application of CIC caused an increase in the fatigue life. The sealed Hi-Lok samples showed no change in life, whereas the sealed rivet samples had a decrease in life. FE modelling supported the experimental findings. The findings from this study will allow for more accurate and reliable fatigue life prediction and damage assessment.

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2.17. Acetate Replica Inspection for Aircraft Structural Integrity Management – B. Main [DSTG], S. Barter, D. Russell [RMIT University] and J. Niclis [QinetiQ]

DSTG demonstrated acetate replica inspection (ARI) as a viable, high-resolution inspection technique for aircraft structural integrity (ASI) management and critical decision making in high value components. Whilst not a new inspection method, this research takes the technique out of the laboratory and into the field for aircraft applications in aluminium, steel, and titanium. ARI can complement conventional non destructing testing (NDT), allow repairs to be optimised and extend re-inspection intervals.

Examination of the microscopic surface topography of structure can reveal cracking significantly smaller than the detectability threshold of conventional NDT. However, this approach has two major challenges:

1. To identify small cracks significant magnification is required, the surfaces need to be free of artefacts that could obscure a crack and contrast enhancement is usually needed for positive identification of cracks.
2. It is difficult or impossible to achieve the magnification or depth of field (for curved surfaces in difficult to get to locations) required to make the observations in-situ.

ARI records microscopic surface topography as a negative relief on plastic (acetate) film or rubber impression that may be taken from the inspected structure, and in the case of a plastic replica, be flattened and viewed under a microscope. While inspection surfaces can be in various starting conditions the quality of surface preparation and the contrast enhancement used dictate inspection fidelity. A practical lower threshold of 0.1mm for crack detection was demonstrated. Smaller cracks can be identified and crack morphology observed, providing further information for ASI management.

Optimal ARI conditions and methods are discussed. Remarkable improvements in surface breaking discontinuity resolution can be achieved, illustrated by DSTG case studies managing valuable assets, including close management and evaluation of cracking in full-scale fatigue tests and fleet assets and post-repair evaluation enabling valuable assets to return to service.

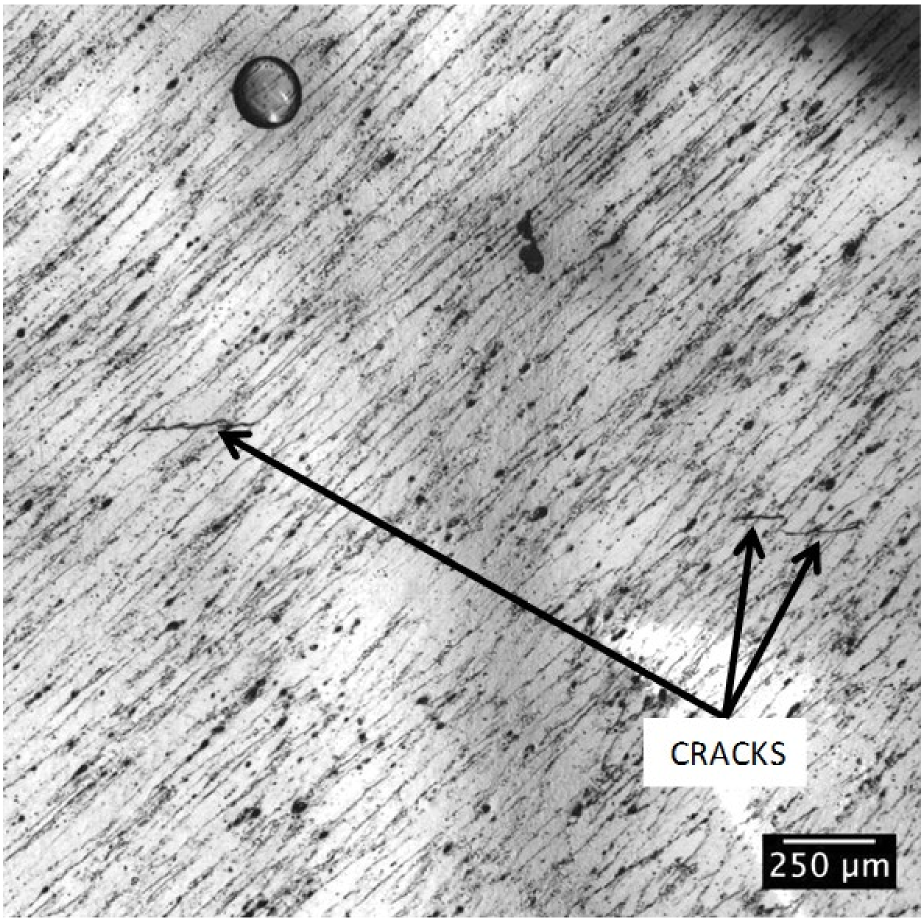

Figure 1 Fatigue cracks detected by acetate replica inspection in aluminium service structures.

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2.18. Finite Element Modelling of Secondary Bending Effects on Fatigue in Aircraft Lap Joint Test Samples – L. Button and J. Codrington [the University of Adelaide]

Lab-based testing of aircraft lap-joint samples is commonplace for investigating mechanical behaviour such as fatigue initiation and growth. Where eccentricities in joint geometry with respect to the load path are present, simple tensile loading results in an out-of-plane deformation known as secondary bending. Numerous past experiments have investigated fatigue behaviour and dependencies on joint parameters. However, the effect of secondary bending in these experiments has not been isolated from other effects, which makes the extension of the predictions to in-service aircraft structure unsuitable without further study.

Typically, secondary bending in sample joints exacerbates the through-thickness stress variation near the fasteners. Which shifts the failure initiation point to be from the faying surfaces and favours the eyebrow mode over the bearing mode. This reduces applicability of the lab-tests to in-service structure and makes crack growth monitoring problematic as the crack tip becomes 'hidden' in the faying surface of the joint. Other design parameters such as clamping force, interference fit, and friction factors can also contribute to the failure mode shifting.

Three previously tested joint configurations (double-shear, single-shear and 1.5-shear-lap-joints) have been investigated in this study. A parametric FE investigation has been performed in ANSYS, analysing the effects of friction factor, clamping force, and interference fit and their relationships to secondary bending. Joint configurations were assessed experimentally, under various tensile loadings, and measurements of secondary bending obtained using DIC, TSA and strain gauges. Experimental and numerical solutions are also compared against an analytical model.

Results show good agreement between analytical, numerical, and experimental solutions. This quantification and understanding helps isolate the effects of secondary bending from other load transfer mechanisms and design parameters, allowing the effects of faying surface treatments on the fatigue behaviour of aircraft joints to be observed and the results utilized with more clarity.

2.19. QinetiQ – Aircraft Structures Fatigue Activities – A. Groszek and J. Moews [QinetiQ Australia Pty Ltd]

C-130J-30 Wing Fatigue Test – Test Interpretation. Test interpretation (TI) of fatigue findings found through the C-130J-30 Wing Fatigue Test (WFT) has continued over the past two years. Analysis of damage found in the centre wing structure has been completed and the program has continued with analysis of damage found in the outer wing structure.

The C-130 outer wing has historically been less prone to fatigue damage, reflected in fewer damage findings in the outer wing at the completion of the WFT compared to the centre wing. Through the WFT TI process, established so that outcomes are compliant with C-130J-30 type certification requirements, analysis of the outer wing WFT findings has been used to support increases to inspection intervals where possible.

All inspection requirements developed through the WFT TI program must meet the certification safety requirements. Further to this, consideration of how inspections are to be integrated into the routine aircraft maintenance schedule, including the impact of how in-service crack size findings can affect repair options, have been used to shape the structural integrity program without compromising the primary safety outcome.

Using results from the deterministic damage tolerance analysis, application of probabilistic risk analysis (PRA) allows further development of the structural integrity program. The objective being to maintain the required level of aircraft safety set by certification requirements at minimum cost and with maximum aircraft operational availability.

The outcome of the program will determine fleet modification and inspections that together provide a certified structural life of type that will enable the C-130J-30 to continue to meet RAAF service objectives.

NDT and Fractography teams. The QinetiQ NDT and Fractography team has provided valuable fatigue related ASI support to its clients in recent years. Specifically, the team has provided embedded support to DSTG focusing on fractographic analysis of fatigue cracking in full-scale fatigue test articles. This has included previous work on the P-8A FSFT and current support on the Hawk LIF FSFT and F-35 AJ-1 FSFT. Analysis involves using quantitative fractographic principles to examine overall fatigue cracks, crack nucleation discontinuities, and extract fatigue crack growth data. The Hawk LIF FSFT support is one of the largest fractography projects undertaken by QinetiQ, the support team comprises of three embedded engineers with a total project timeline of approximately 18 months.

Aircraft Structural Integrity. The Fighters and Trainers team has provided support to the Directorate of Aviation Engineering (within the Defence Aviation Safety Authority) and Systems Program Offices, particularly in relation to Classic Hornet and Super Hornet lifing and fatigue management. This has included close out activities for the RAAF Classic Hornet structural life extension, such as revised lifing for a critical detail on the wing carry through bulkhead based on assessment of data generated through fractographic analysis by DSTG. Assessment was also performed to substantiate extension to RAAF Super Hornet interim life limits until the results of the RAAF Service Life Assessment Program, being conducted by the aircraft OEM, are available.

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3. LARGE AND FULL-SCALE TESTING

3.1. C-27J Structural Substantiation Program Full-Scale Fatigue Test – W. Foster [DSTG]

The RAAF are conducting a Structural Substantiation Program (SSP) for the C-27J Spartan Light Tactical Fixed Wing aircraft in order to address airworthiness and through life support management accounting for the Australian Defence Force configuration, role and environment. DSTG, under the scope of the SSP, are undertaking a full-scale fatigue test (FSFT) of C-27J wing and fuselage. The FSFT activities will be undertaken at the DSTG Fishermans Bend facility in Melbourne, Australia.

A fully production representative version of the RAAF C-27J wing and fuselage will be subjected to 3 lifetimes of cyclic spectrum fatigue loading. The C-27J FSFT will include a tip-to-tip wing on a full fuselage. Representative steady-state loading will be applied to the wing and fuselage, and the complete fuselage will also be subjected to pressurisation cycling. The empennage will not be attached to the fuselage.

Loads and spectrum development work in support of the FSFT are well advanced, and detailed planning for the C-27J FSFT is progressing. Wing and fuselage loading will be applied by approximately 68 servo-hydraulic actuators, and the response of the test article will be measured by up to 650 strain gauges. A possible configuration of test actuator locations is shown in Figure 1. The current schedule has test rig design commencing mid-2021 with assembly currently planned for 2023, prior to test commencement in 2024.

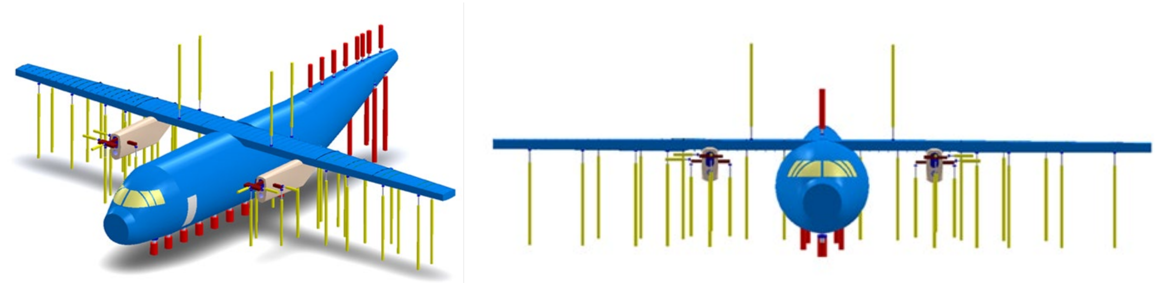

Figure 1 Representation of possible test actuator locations for the C-27J FSFT

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3.2. Design and Build of a Large-Scale Aerospace Multi-Axial Structural Testing Capability for Emerging Aerospace Technologies – B. Main, K. Muller and K. Maxfield [DSTG]

Aerospace Division (AD) of DSTG, working with RMIT University and inspired by the work of the US Federal Aviation Administration (FAA) Technical Centre, have designed and are building and commissioning a large scale, multi-axial aerospace structural test capability at DSTG Fishermans Bend.

Recognising the significance of replicating multi-axial flight loading states for engineering and research into new aerospace technologies such as composite structures and additively manufactured repairs, a test rig has been built that can simultaneously apply, statically or in fatigue, wing bending and torsion loads for example, to large panels or a built up wing structural element. With maximum torsional stresses of the order of 30% of wing bending loads in some aircraft, the integrity and durability of contemporary high performance aerospace structures with service degradation or structural repairs can only be thoroughly evaluated for airworthiness purposes under representative multi-axial load states.

After engaging research, airworthiness and industry stakeholders through a series of working groups, rig requirements were develop and a detailed design activity was initiated and completed (Figure 1). Procurement of servo-hydraulic hardware, control systems, materials and a series of instrumented flat plate carbon-fibre laminate specimens followed. The rig's steel structure is currently being manufactured, hydraulic actuators have been functionally tested and the commissioning specimens manufactured. Rig build and commissioning is on track for completion in June/July of 2021 with a series of compression after impact composites experiments likely to be the first research conducted using this new capability.

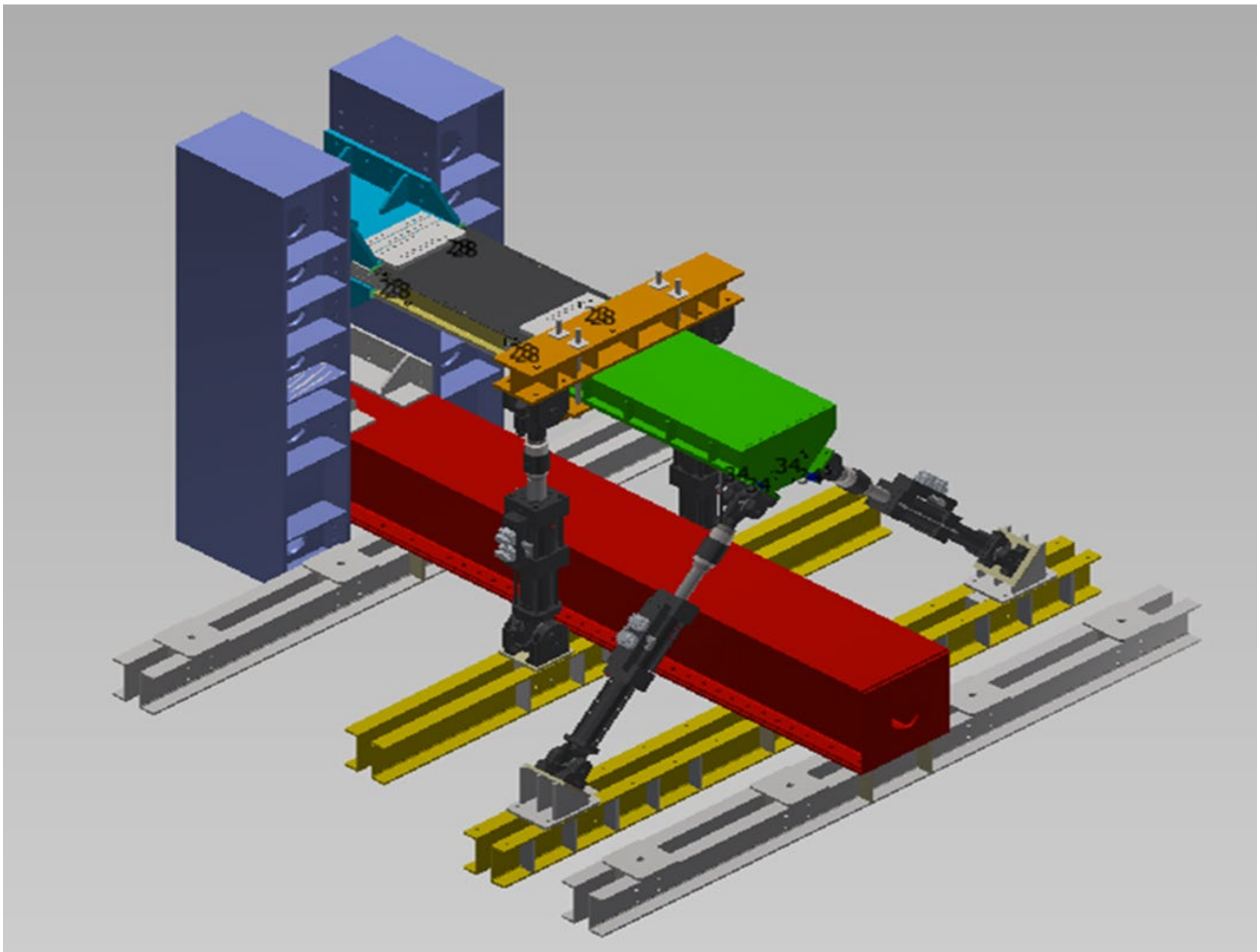

Figure 1 The large-scale aerospace multi-axial structural test capability design.

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3.3. Running Rate Improvements for the Hawk Lead-In-Fighter Full-Scale Fatigue Test – M. Jones [RMIT University], M. Attia, L. Doxey, B. Main [DSTG] and R. Aaron [BAE Systems UK]

The Hawk Mk 127 Lead-In-Fighter (LIF) is a fast jet trainer currently operated by the RAAF. To support sustainment of the RAAF Hawk fleet, Defence collaborated with BAE Systems UK to conduct a full-scale fatigue test (FSFT) of the Hawk Mk127 airframe to 50,000 equivalent flight hours (EFH). Testing occurred at DSTG Fishermans Bend between 2006 and 2020 (Figure 1).

As the FSFT approached 40,000 EFH in 2018, DSTG and BAE Systems UK undertook an investigation into test running rates with the aim of expediting the final phases of testing, reducing down time and reducing unscheduled test shutdowns through a systematic root cause and corrective action process. The focus of the investigation spanned test operations and procedures, the test loading control systems, test spectra content and load sequencing.

The investigation found that the cycling rate had remained relatively constant over the life of the test; however, both the number of unscheduled shutdown events and the average shutdown time had both increased. At the time, the investigation identified several key areas of the control system setup that could be modified to reduce the number of unexpected shutdown events and improve the overall running rate of the test without compromising test accuracy or outcomes. With the successful completion of the FSFT now achieved in 2020, this paper reviews the running rate analysis and the impact it had on the final 10,000 EFH of testing with the aim of sharing several key lessons that may be relevant to future full-scale fatigue test programs.

Figure 1 The Hawk Mk127 LIF FSFT

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3.4. Rapid Development of a Novel Method of Damage Introduction in Full-Scale Damage Tolerance Testing in Support of F/A-18 Hornet Sustainment – D. Russell and W. Foster [DSTG]

DSTG conducted F/A-18A/B Hornet Outer Wing Static Testing (HOWSAT) to substantiate the through life management strategy for the F/A-18A/B Outer Wing (OW) internal metallic structure. Static proof testing with artificially introduced damage representing severe, but representative, fatigue cracking allowed the RAAF, Royal Canadian Air Force and the United States Navy to reduce inspections, validate repairs and avoid expensive fatigue testing or OW replacements.

Damage was introduced in 95 locations without major disassembly of the OW structure. At some locations use of conventional cutting tools was impossible due to lack of access.

One such location was at the wing fold where a titanium rib, aluminium spars and carbon-epoxy skins are nested. Damage representing large cracks extending from holes in the lower flange of the wing fold rib was required, whilst avoiding damage to the lower skin. There was no direct access to the upper surface of the stack.

DSTG rapidly developed an electric discharge machining (EDM) process to cut narrow slots through the lower flanges of the spar and rib without damaging the lower skin. The EDM machine was a standard tap removing machine; however, DSTG developed bespoke electrodes and guides (Figure 1).

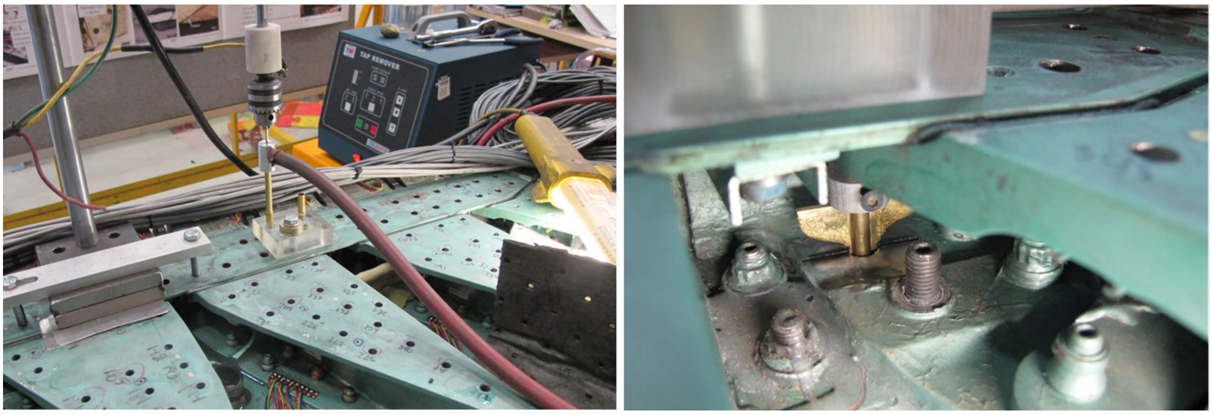

Figure 1 EDM setup (left) and a local view of the electrode and induced damage at a fastener hole (right).

The introduced damage was cycled in the test rig to grow small fatigue cracks at the ends of the EDM slots before static testing. This crack sharpening cycling ensured the artificially introduced damage was fatigue crack-like.

Quadrant shaped damage over 30 mm long was introduced at five hot spot holes. Fatigue cracking resulting from the crack sharpening cycling was observed at the EDM slot tips during post-test fractographic analysis. The OW survived static proof testing.

Rapid development of a novel EDM method to artificially introduce and test fatigue representative damage in a location impossible to access using conventional tooling was successfully demonstrated.

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3.5. Design Development, Manufacture and Commissioning of a Full-Scale Helicopter Airframe Test Rig – G. Swanton [DSTG], L. Robertson and M. Gillet [RMIT University]

DSTG is currently researching new technologies which, if successful, would make a helicopter airframe full-scale fatigue test a viable proposition from a test duration and loads fidelity perspective. Known as the Helicopter Advanced Fatigue Test – Technical Demonstrator (HAFT-TD), this project, as the name suggests, seeks to demonstrate the feasibility of full-scale helicopter aircraft fatigue testing in support of certification and sustainment of future fleets. In this case, the driver for HAFT-TD is the US Navy's future considerations for their MH-60R fleet (Sikorsky Seahawk 'Romeo') Service Life Assessment Program / Service Life Assessment Program (SLAP/SLEP) planning activities. The US Navy have provided DSTG with a retired Seahawk test article whilst Australia's Capability and Sustainment Group (CASG) have provided the financial backing.

Although HAFT-TD aims to mature the technical readiness level of various inputs for generic helicopter airframe testing, this paper presents the developments associated with the design philosophy, manufacture and commissioning aspects of a bespoke test rig to accommodate the Seahawk airframe for HAFT-TD objectives (Figure 1). DSTG has partnered with academia and engaged Australian industry throughout the various stages of requirements determination, rig design and rig manufacture. Needing to consider the various design elements of form, fit and function, other considerations such as modularity, floor plan flexibility, manufacturing costs and assembly efficiencies were also major factors in the evolution of the test rig. In addition to this, a separate design activity was undertaken for the custom fixtures required for the transfer of applied loads from the actuators into the test article. Finally, with the rig assembled, test article installed and control and load systems integrated, commissioning of the entire system was required to demonstrate that functionality and safety requirements had been satisfied prior to commencing the technical demonstration phase.

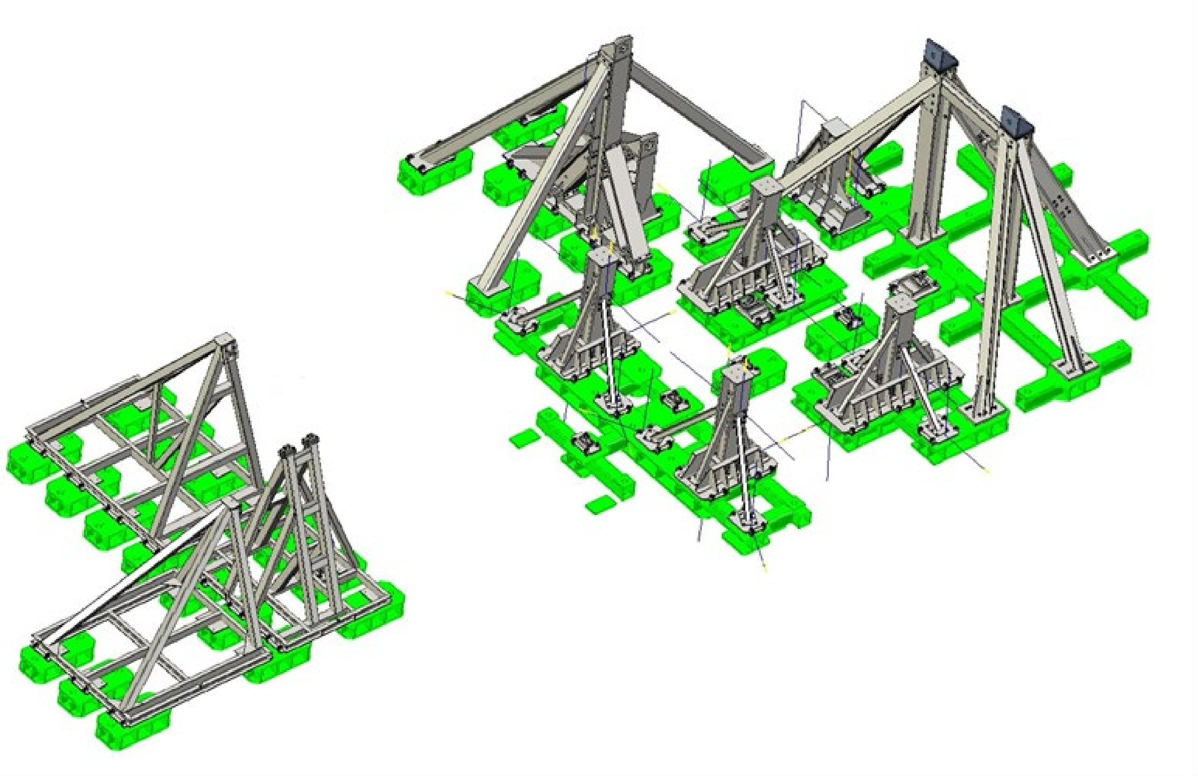

Figure 1 The HAFT-TD full-scale rig design

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4. ADDITIVE MANUFACTURING

4.1. RUAG approach to the durability analysis of laser additive repairs to Aermet100 steel – R. Jones [Monash University] and N. Matthews [RUAG Australia]

RUAG Australia has for many years been providing innovative repair and reengineering solutions to the Australian Defence Force. In this context RUAG Australia has numerous applications of laser additive deposition (LAD) to RAAF F/A-18 aircraft. However, before this solution can be applied to load bearing components a durability analysis capability must be developed. To establish this capability RUAG opted to study the LAD repair to AerMet100 described in [1]. The Hartman-Schijve crack growth equation [2-5]

$$da/dN = D\,[(\Delta K - \Delta K_{thr})\,/(1-K_{max}/A)1^{/2}]^{p} \qquad \text{Equation 1}$$

were D and p are constants, A is the cyclic fracture toughness, and ΔKthr is the fatigue threshold, see [2-5], was used in this study. The constants used were taken from a prior study into additively manufactured materials [2]. As outlined in [3] for small cracks in conventionally manufactured materials the fatigue threshold ΔKthr is set at 0.1 MPa √m and the cyclic toughness at 140 MPa √m. The measured and computed crack growth histories for the baseline specimen are shown in Figure 1 where we see excellent agreement. Having established the ability to perform the necessary durability analysis for Aermet100 the analysis was repeated for the LAD repaired specimens. As shown in Figure 2 the variability in the measured crack growth histories can be reasonably well captured by allowing for changes to the term ΔKthr. The values of ΔKthr used are shown in Table 1. The methodology recommended in [5], i.e. the use of a crack growth curve corresponding to the mean value of ΔKthr minus three times the variance (3σ), was then used to enable an estimate of the worst case curve to be determined. This ‘worst case‘ curve is also shown in Figure 2. As would be expected the resultant ‘worst case‘ crack growth curve yields a fatigue life that is significantly shorter than the fatigue lives of the three (clad) test specimens.

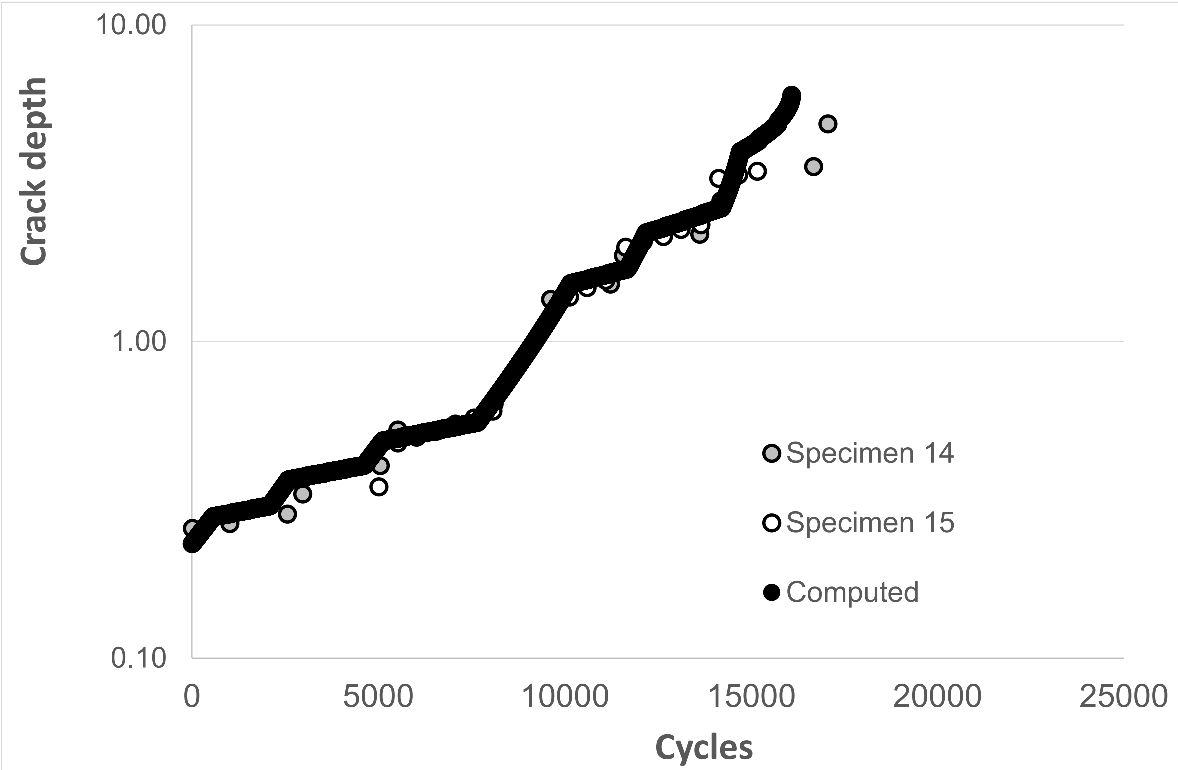

Figure 4 Durability analysis results for the baseline AerMet100 steel tests

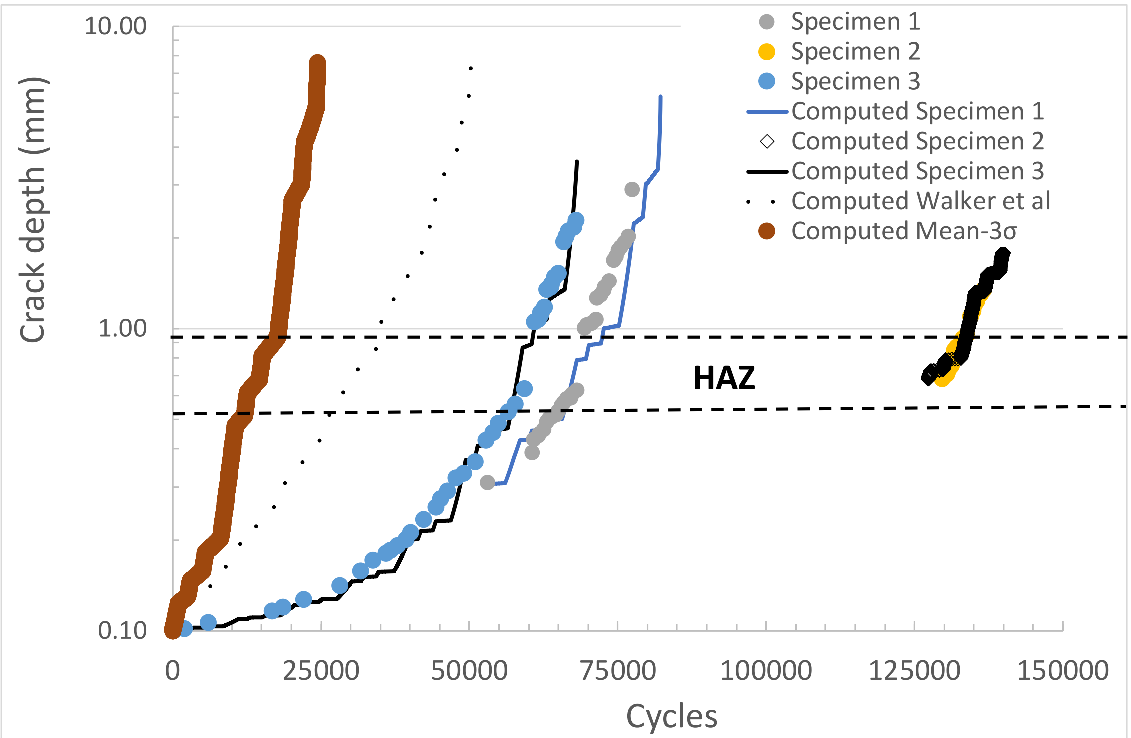

Figure 5 Durability analysis results for the LAD repairs to AerMet100 steel

Table 1 Values of ΔK_{thr} and A used in Figure 2.

Specimen	ΔK_{thr} (MPa √m)	A (MPa √m)
1	7.0	140
2	9.0	140
3	5.8	140
Mean – 3σ	0.19	140

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4.2. The State of the Art in the Durability and Damage Tolerance Analysis of AM Parts and Attritable Aircraft – R. Jones [Monash University], O. Kovarik [Czech Technical University], S. Bagherifard [Politecnico di Milano], J. Cizek [The Czech Academy of Sciences], V. Papyan and J. Lang [Titomic]

Assessing the damage tolerance and durability of additive manufactured (AM) materials is a key factor in the airworthiness certification of AM parts and limited life airframes. This paper outlines a state of the art building block approach that is consistent with both MIL-STD-1530D and USAF Structures Bulletin EZ-19-01 and accounts for the anisotropy and variability in the crack growth curves associated with AM and how these curves relate to the equivalent conventional material. This approach is validated via a durability analysis of specimens built using laser bed powder fusion. The experimental data reveals that the reduced strain to failure of AM and cold spray additive manufactured (CSAM) parts left in the as built condition may not significantly affect the economic life. The examples presented reveal that AM and CSAM are attractive for use in attritable aircraft.

It is further shown that, for both conventionally and AM specimens, there is a unique relationship between:

1. da/dN and the change in the potential energy per cycle (dU/dN)
2. da/dN and the (Schwalbe) crack tip parameter Δκ in the Hartman-Schijve crack growth equation
3. Δκ and dU/dN.

The experimental data reveals that these relationships are independent of both the build direction and post-processing conditions.

Hence we have two sets of independent experimental measurements, one yields a power law relationship between da/dN and Δκ, and the a power law relationship between the change in the potential energy per cycle and Δκ. These results highlight that Δκ is a valid parameter for characterising durability of AM and CSAM parts.

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4.3. Effect of Powder Drying and Oxygen Shielding on the Tensile and Fatigue Performance of 300M Repaired through Laser Directed Energy Deposition – C. Barr, R. A. R. Rashid, M. Easton, S. Palanisamy, N. Matthews and M. Brandt [DMTC Pty Ltd]

Laser directed energy deposition (L-DED) is emerging as a key tool in the sustainment of high value aerospace components, as the ability to restore worn or damaged material allows the process to repair previously unsalvageable parts. The subsequent extension of component lifetimes and the ability to quickly return parts to service thus represents significant cost savings and increases the readiness of air platforms. L-DED repair is attractive to high strength steels used in landing gear, as extreme loading and low fracture toughness makes them sensitive to fatigue cracking and foreign object damage. Such steels can be challenging for L-DED repair, as in-situ tempering effects can lower the strength in the repaired region, while process induced defects may reduce fatigue life. In this investigation, L-DED repair was conducted on 300M substrates to investigate the role of powder quality and oxygen shielding on tensile and fatigue performance. 300M powder was prepared with and without pre-baking at 120 ºC to determine the role of moisture contamination on porosity. Deposition was then completed using either local nozzle gas shielding or shielding in an inert chamber to examine the role of oxygen contamination. Combining powder drying with chamber shielding provides significant enhancement to ductility, with almost twice that achieved with local shielding and without baking. No difference in strength was found between specimens, reflecting the similar thermal histories and in-situ tempering conditions in the deposit. Dried and chamber shielded specimens also show dramatic improvements in fatigue life, which is linked to the elimination of fine porosity found in locally shielded and without baking. In addition, chamber shielding eliminates the risk of coarse gas pores originating from oxide build-up, which becomes prevalent in locally shielded specimens due to retained heat. These results demonstrate the need to avoid environmental contamination to ensure optimal properties following L-DED repair.

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4.4. Crack Growth in Commercially Pure Titanium and its Potential for Use in Military Aircraft – R. Jones and D. Peng [Monash University], A. Ang [Swinburne University], J. Lang, [Titomic Pty Ltd] and J. Lua, [GEM]

One of the challenges in aircraft sustainment is to develop AM replacement parts for legacy aircraft. This is particularly important for fixed and rotary wing aircraft that operate in in aggressive environments, i.e. off carriers, in a marine environment, etc. On the other hand the United States Air Force (USAF) have now adopted the concept of using AM to rapidly field limited-life unmanned air platforms. Whilst the temptation is to use AM Ti-6Al-4V for these purposes, Ti-6Al-4V powder is both costly and its supply is somewhat restricted. To address this problem [1] revealed that the yield and ultimate strengths, the strain to failure of commercially pure (CP) titanium, which is highly corrosion resistant, and it's resistance to crack growth is superior to that of the commonly used aluminium ally AA7050-T7451, which is used in the F/A-18 Hornet, Super Hornet and F-35 (Joint Strike Fighter), see Figure 1.

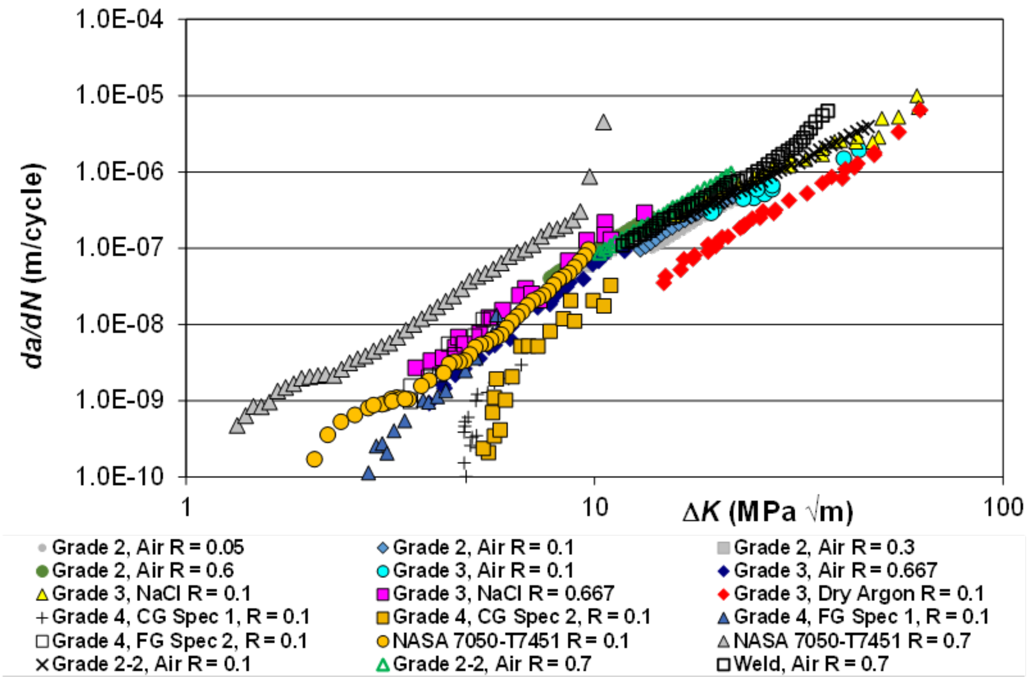

Figure 1 Crack growth in CP titanium and 7050-T7451

Interestingly, when allowance is made for the differences in fatigue threshold and toughness then, if the crack growth rate da/dN is expressed as per the Hartman-Schijve crack growth equation, the resultant crack growth curves for Grade 2, 3 and 4 Titanium, and AA7050-T7451 all fall onto (essentially) the same master curve, see Figure 2. Here da/dN is plotted against the Schwalbe similitude parameter

$$\Delta\kappa = (\Delta K - \Delta K_{thr})/(1 - K_{max}/A)^{1/2} \qquad \text{Equation 1}$$

where A is the cyclic fracture toughness, K is the stress intensity factor, Kmax and Kmin are the maximum and minimum values of stress intensity factor seen in the cycle, ∆K= (Kmax − Kmin) is the range of the stress intensity factor that is seen in the cycle, and ∆Kthr is the fatigue threshold.

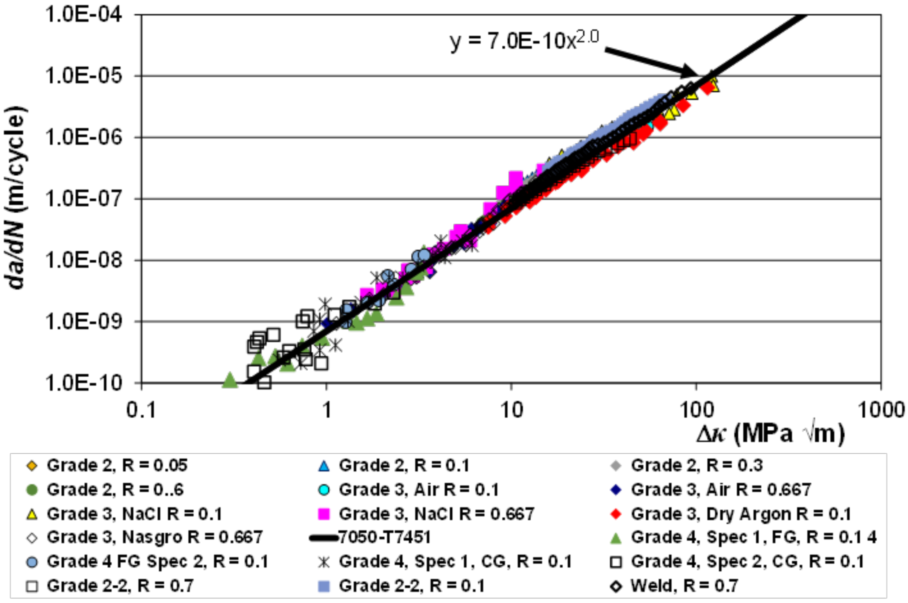

Figure 2 Crack growth curves replotted

The round-robin study [2] into the damage tolerance of a representative helicopter lift frame was then analysed assuming that the component was built using CP Grade 2 Titanium. It was found that the life of the lift frame was substantially greater than that of a 7010 test component.

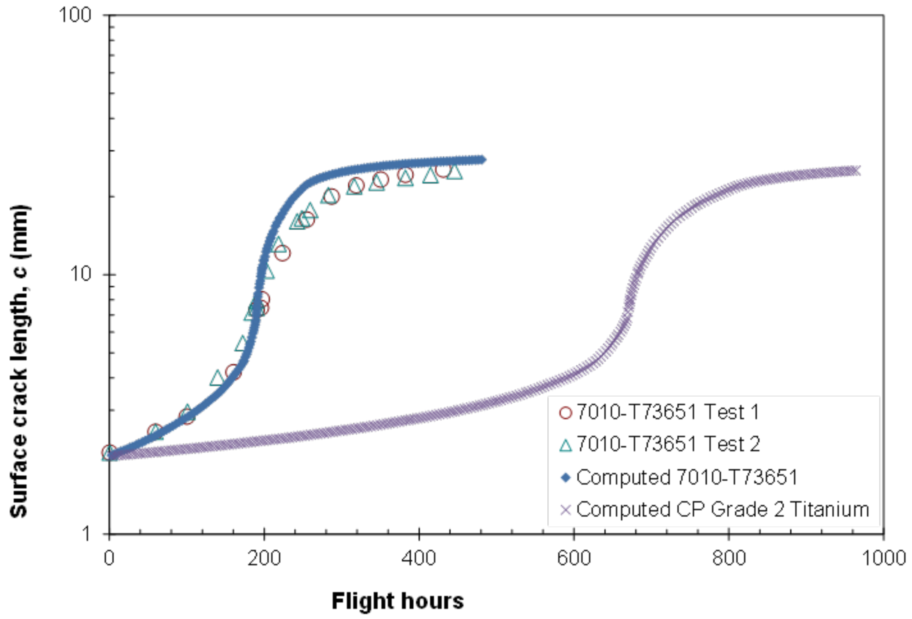

Figure 3 Computed lives for a 7010 and a CP titanium lift frame

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4.5. Additively Manufactured Pinning Technology for Titanium-to-Composite Joints – R. Ladani, E. Kandare, A. Ravindran, S. Inverarity, A. Mirabendini, A. Orifici, A. Mouritz [RMIT University], P. Chang and J. Wang [DSTG]

The extensive use of composites with metallic structures on military air, sea and land platforms requires innovative metal-to-composite joining technologies which can overcome the inherent problems with existing joining methods. For instance, some aircraft use a bonded joint to connect a carbon fibre-reinforced polymer (CFRP) skin to titanium fittings, making this a safety-critical structure that requires a high-degree of damage tolerance. Inspection of such a joint on an aircraft with relatively high flight hours has revealed disbonding of the adhesive at the titanium-to-composite interface and delamination damage in the composite. This problem remains a critical challenge to the integration of composites in future Defence platforms.

This work investigates a novel titanium-to-composite joining method for improving joint strength and damage tolerance. The study developed a hybrid joint which makes use of through-thickness pins to derive superior mechanical performance even in the presence of typical manufacturing or in-service flaws. The joint uses selective laser melting to additively manufacture a titanium adhered with integrated surface pins. These surface pins are inserted in the through-the-thickness direction of uncured composite using an ultrasonic device. Upon curing, the pins create a mechanical interlock between the titanium and composite adherends. The strength and fracture properties of the pinned titanium-to-composite joints are investigated using standard testing protocols and compared to an unpinned joint. Experimental and finite element modelling results of the strength and fracture properties of the hybrid joint are reported and discussed along with the mechanisms responsible for the improvements. The joining technique can be applied for metal-to-composite joints with enhanced strength and damage tolerance, and the modelling capability developed can support the design and sustainment of these joints.

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