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A Review of Australian Aeronautical Fatigue and Structural Integrity Investigations During May 2023 to March 2025

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). This 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 May 2023 to March 2025. The Australian national review was presented during the 39th ICAF Conference and 32nd ICAF Symposium at Xi'an, China, 9–13 June 2025 and published on the Committee's website www.icaf.aero.

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CONTENTS

GLOSSARY

AA	Aluminium alloy
AM	Additive manufacturing
BSIWS	Boeing Space, Intelligence and Weapon Systems
BVID	Barely visible impact damage
CA	Constant amplitude
CF	Carbon fibre
CFRP	Carbon fibre reinforced polymer
DSTG	Defence Science and Technology Group
EIDS	Equivalent initial damage size
FCG	Fatigue crack growth
FCGR	Fatigue crack growth rate
FEA	Finite element analysis
FT	Flash thermography
FSFT	Full-scale fatigue test
HAFT-TD	Helicopter Advanced Fatigue Test – Technology Demonstrator
ICAF	International Committee on Aeronautical Fatigue and Structural Integrity
JSSG	Joint services specification guide
LB-PBF	Laser-based powder bed fusion
LEFM	Linear elastic fracture mechanics
LST	Line scan thermography
NDI	Non-destructive inspection
PAUT	Phased array ultrasonic test
QF	Quantitative fractography

SLAP	Service life assessment program
SPASM	Selective Plasma Arc Spot Melter
STAB	Stabilator
SSC	Structural supercapacitors
TSA	Thermo-elastic stress analysis
USAF	United States Air Force
WAAM	Wire-arc additive manufacturing

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1. INTRODUCTION

This report presents a review of Australian research, engineering and technology activities in the field of aeronautical fatigue and structural integrity from May 2023 to March 2025. The review includes inputs from government, industry and academic institutions 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.

Contributions are arranged by broad field of structural integrity activity. The organisations contributing to this review include:

- 1Millikelvin, https://1millikelvin.com
- Deakin University, www.deakin.edu.au
- Defence Science and Technology Group (DSTG), www.dst.defence.gov.au
- Monash University, www.monash.edu
- MEMKO, https://memko.com.au
- Molent Aerostructures, www.molent.com
- QinetiQ Australia, www.qinetiq.com/en-au
- RMIT University, www.rmit.edu.au
- Swinburne University, www.swinburne.edu.au
- Sydney University, www.sydney.edu.au
- University of Adelaide, www.adelaide.edu.au
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2. FATIGUE CRACK GROWTH AND LIFE PREDICTION METHODS

2.1. Equivalent Initial Damage Sizes (EIDS) for Type 1C Anodised Aluminium Alloy 7085-T7452 under Variable Amplitude Loading – B. Dixon, B. Main and S. Barter (DSTG, RMIT)

Aluminium alloy (AA) 7085-T7452 is a high-strength, low-weight alloy used in large forgings in primary airframe applications. For corrosion protection, airframe components are anodised and this study looks at the influence of type 1C anodising on the fatigue crack growth (FCG) behaviour in AA 7085-T7452 specimens that were fatigue tested with fighter wing spectrum loading. Surface etch pits formed during essential anodising preparatory processes were the primary cause of fatigue crack nucleation. The severity of the early crack growth caused by those etch pits that nucleated the worst crack in each specimen was evaluated using quantitative fractography (QF)-based FCG measurements, and the equivalent initial damage size (EIDS) of those etch pits was determined. Characterising the EIDS distribution of aircraft alloys with representative surface treatments is a fundamental input into linear elastic fracture mechanics (LEFM)-based fleet failure analysis of aircraft structures. In this study, a notable increase in the average EIDS value for the worst crack present was observed whenever material or loading condition changes led to a significantly increased number of crack nucleation sites. In particular, an increase in the cyclic stress spectrum's severity increased the number of nucleation sites in a specimen and significantly increased the average EIDS. It appears that due to variable nucleating discontinuity properties (e.g. size, shape and orientation) and material property variations, an increased number of nucleation sites increases the probability that the worst crack present will nucleate at a location that is conducive for relatively fast early crack growth. This will potentially cause shorter overall airframe component fatigue lives, and so both the mechanisms and conditions of etch pit fatigue nucleation in this material and material finish shall be discussed for the benefit of failure analysts evaluating unexpected fleet fatigue cracking.

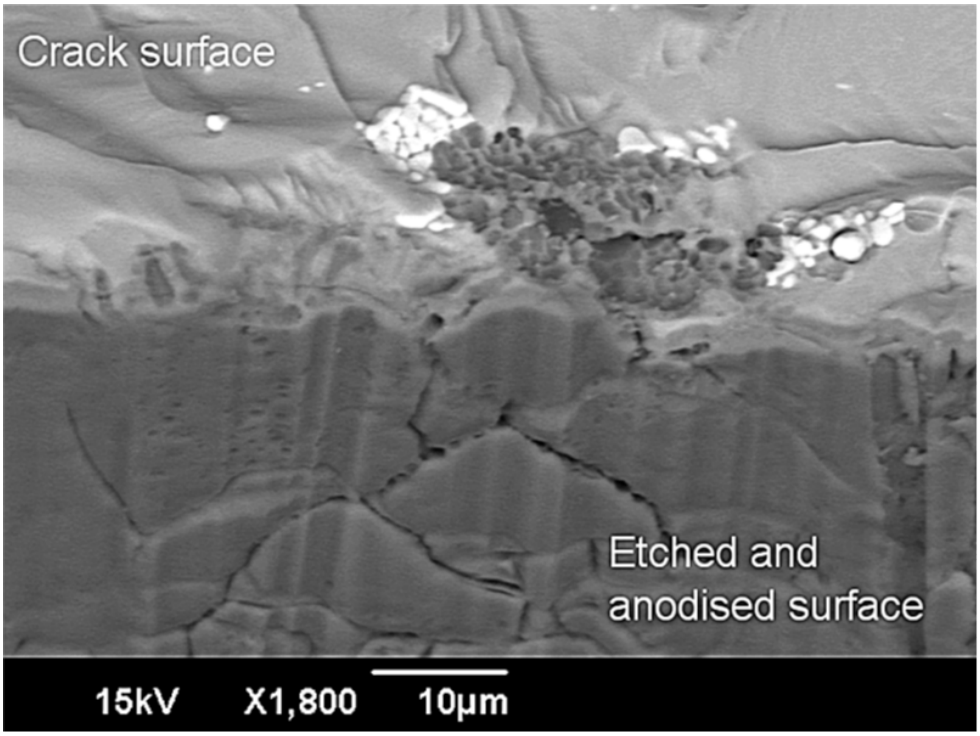

Figure 1 Oblique view of the discontinuity that nucleated the lead crack a specimen (low Kt T-L). This is a scanning electron microscope micrograph taken with back-scattered electrons in the compositional contrast mode so that intermetallic inclusion particles are bright objects. Here the crack nucleated from an etched-out cluster of intermetallic inclusion particles.

Reference

1. B. Dixon, B. Main and S. Barter, 'Equivalent initial damage size (EIDS) for Type 1C anodised aluminium alloy 7085-T7452 under variable amplitude loading', *Engineering Failure Analysis*, 2023. https://doi.org/10.1016/j.engfailanal.2023.107578

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2.2. Materials Characterisation of Anodising Effects on Small Fatigue Crack Nucleation in AA7XXX Alloys – B. Main, L. Jiang, R. K. W. Marceau and S. Barter (DSTG, Deakin, RMIT University)

The precursor cleaning steps in the anodising surface finish process for AA aircraft parts are known to lead to etching of surface-breaking intermetallic particles and grain boundaries, which in turn leads to preferential sites for fatigue crack nucleation. Type 1C anodising per MIL-A-8625F, is a favoured aircraft aluminium part corrosion protection treatment meeting modern health and environmental requirements. Type 1C anodising uses electrolysis in a non-chromic acid bath (typically based on sulphuric, boric-sulphuric or phosphoric acid formulations) to develop a thick aluminium oxide surface layer. In previous studies, the equivalent initial damage sizes of these nucleation sites was calculated from near surface FCG measurements using QF in both Type 1C anodised aluminium alloy 7050-T7451 and 7085-T7452. A notable difference was observed in the values for these 2 materials, with AA7085-T7452 equivalent initial damage sizes being significantly smaller. Since these values are critical crack starting size assumptions in fatigue life analysis for aircraft fleets, the observed differences must be investigated and explained for notionally similar alloys under identical conditions.

In this work, the material surrounding the population of crack nucleation sites in several Type 1C anodised fatigue tested specimens manufactured from AA7050-T7451 and AA7085-T7452 materials has been characterised using precise surface preparation methods and scanning electron microscopy techniques. It has been demonstrated that there is a statistically significant difference in the number, size, aspect ratio and distribution of surface-breaking Al_7Cu_2Fe intermetallic particles beneath the anodising layer that, when etched out during the anodising process, are responsible for fatigue crack nucleation in 7050-T7451 over 7085-T7452 material. The increased size, angular shape and coverage of these intermetallics in AA7050 serves to increase their effectiveness as preferential locations for fatigue crack nucleation through 2 mechanisms, as a highly localised stress concentrating feature, as well as through their ability to sample more of the material's microstructural features, ensuring a discontinuity is present in the most favourable orientation and location for fatigue crack nucleation.

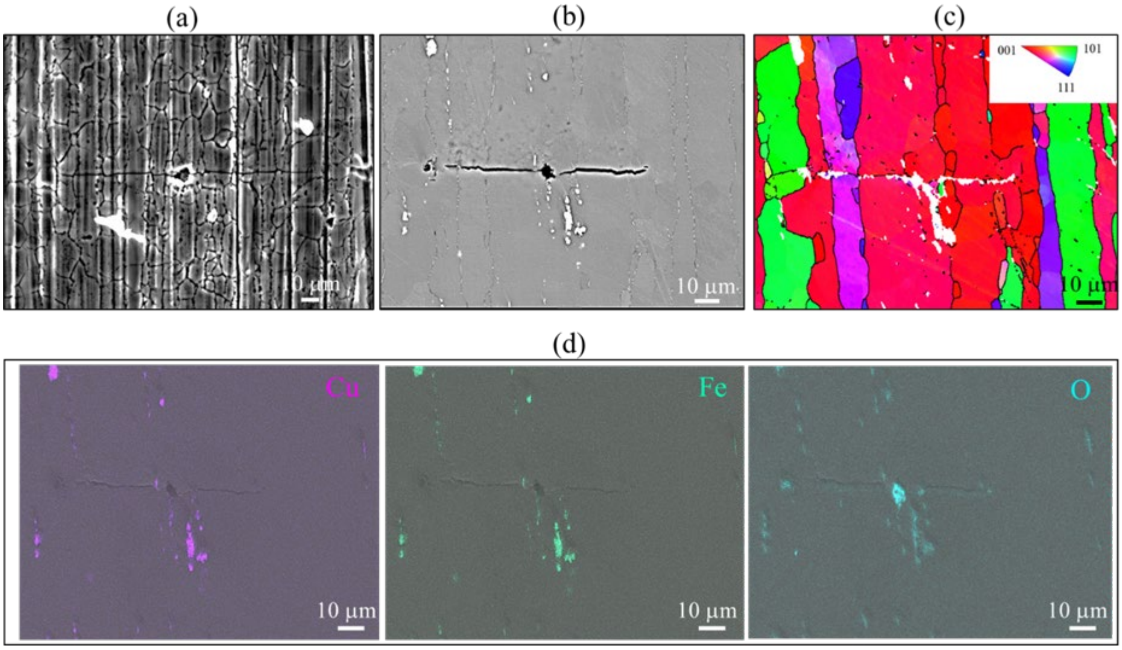

Figure 2 (a) and (b) are scanning electron microscope images of a AA7085-T7452 fatigue crack on the T-S face (Figure 2) before and after polishing, respectively, with the specimen preparation methods discussed in Section 2.4; (c) is the corresponding electron backscatter diffraction grain orientation map; (d) are the energy dispersive spectroscopy maps of Cu (purple), Fe (green) and O (cyan) overlayed with the secondary electron image. Since the inclusion at the crack origin site has been etched out, there is no remaining evidence of the Cu and Fe, only O from the anodising layer within the pit. Meanwhile, the nearby inclusion cluster clearly has both Cu and Fe present, typical of an AlCuFe inclusion.

Reference

1. B. Main, L. Jiang, R. Marceau and S. Barter, 'Material characterization of anodising effects on small fatigue crack nucleation in AA 7XXX alloys', *Materialia*, vol. 33, 2024. https://doi.org/10.1016/j.mtla.2023

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2.3. An Analysis of Fatigue Crack Features at Various Crack Lengths through Mating Fracture Surface Pairs – I. Field, E. Kandare, B. Dixon and S. Barter (DSTG, RMIT University)

This work included a novel high-resolution direct analysis of mating AA7050-T7451 fracture surfaces at crack lengths from 0.1 to 1.3 mm (Kmax of 4.51 to 16.39 MPa$\sqrt{m}$) for the purpose of analysing the fatigue growth mechanisms. The various fractographic features observed at these crack lengths were then compared to the known accelerative effects of an applied loading sequence that contained underloads at regular intervals. A correlation was found between some of the observed features and the local FCG rate (FCGR). This study provides new insight into the prominent mechanisms that dictate crack growth rate and ultimately should assist with developing crack growth prediction methods that more accurately represent the physics of the FCG mechanisms.

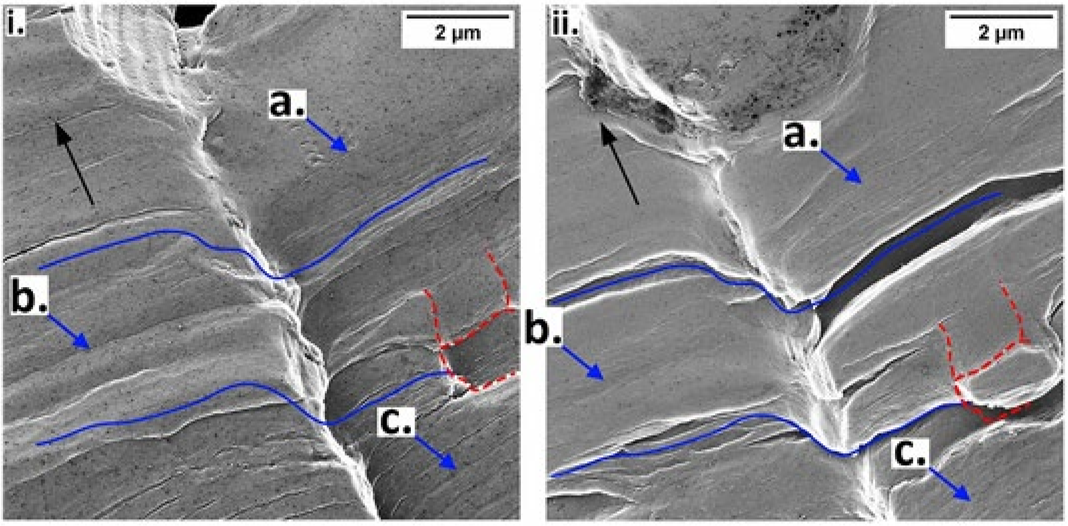

Figure 3 An image showing the region where the R = 0.5 CA cycles transition into the R = -1 underloads. The images were scaled and aligned using the features marked by the red dashed lines. The blue solid lines indicate the largest fissure in the right image (ii), and which features they match with in the left image (i). A small fissure-like crack at (a) can be seen on both fracture halves. A similar small-scale fissure (b) in the left image (i) that does not appear in the right image (ii). The R = 0.5 striation features for both fracture halves are at (c). The nominal direction of crack growth is given by the black arrows.

Reference

1. I. Field, E. Kandare, B. Dixon and S. Barter, 'An analysis of fatigue crack growth features at various crack lengths through mating fracture surface pairs', *Fatigue and Fractures of Engineering Materials and Structures*, 2024. https://doi.org/10.1111/ffe.14276

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2.4. Optimising FCG Predictions for Small Cracks under Variable Amplitude Loading – B. Dixon, H. Fayek, C. Hodge, T. Wiley and S. Barter (DSTG, RMIT University)

The fatigue cracks in a fighter aircraft that pose the greatest threat to structural integrity and availability are usually less than 1 mm for most of their lives. However, it has long been recognised that small cracks can grow significantly faster than long cracks at the same stress intensity range (ΔK). This means predictions for small cracks growing under realistic spectrum loading can be significantly non-conservative when using linear elastic fracture mechanics and long crack-based empirical rate data. Previous work showed empirical rate data based on small cracks grown under constant amplitude (CA) loading could significantly improve predictions. However, such data still have limitations related to an inability to predict the spectrum history effects that influence small cracks. This study presents a method to optimise a model to predict FCGRs for cracks loaded with realistic spectra. This method is demonstrated for small cracks in AA 7050-T7451 grown under fighter wing root spectrum loading. Numerical optimisation is used to select model parameters that minimise FCGR prediction errors for a training dataset. This purpose-built model consistently outperformed a generic model based on the growth of small cracks under CA loading.

Reference

1. B. Dixon, H. Fayek, C. Hodgen, T. Wiley and S. Barter, 'Optimising fatigue crack growth predicitons for small cracks under variable amplitude loading', *International Journal of Fatigue*, vol. 184, 2024. https://doi.org/10.1016/j.ijfatigue.2024.108339

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2.5. Crack-like Effectiveness of Some Discontinuities in AA2024 – L. Molent and M. R. Fox (Molent Aerostructures)

Maintaining aircraft airworthiness to ensure the fleet's safe operation and maintain its readiness is critically dependent on accurate modelling and reliable predictions of FCG. In this process, a knowledge of the representative initial discontinuity sizes that cause fatigue crack nucleation and early growth in aircraft is essential. Here, the effective pre-crack size of AA 2024, from samples of aircraft production material and tested under aircraft spectra, is considered.

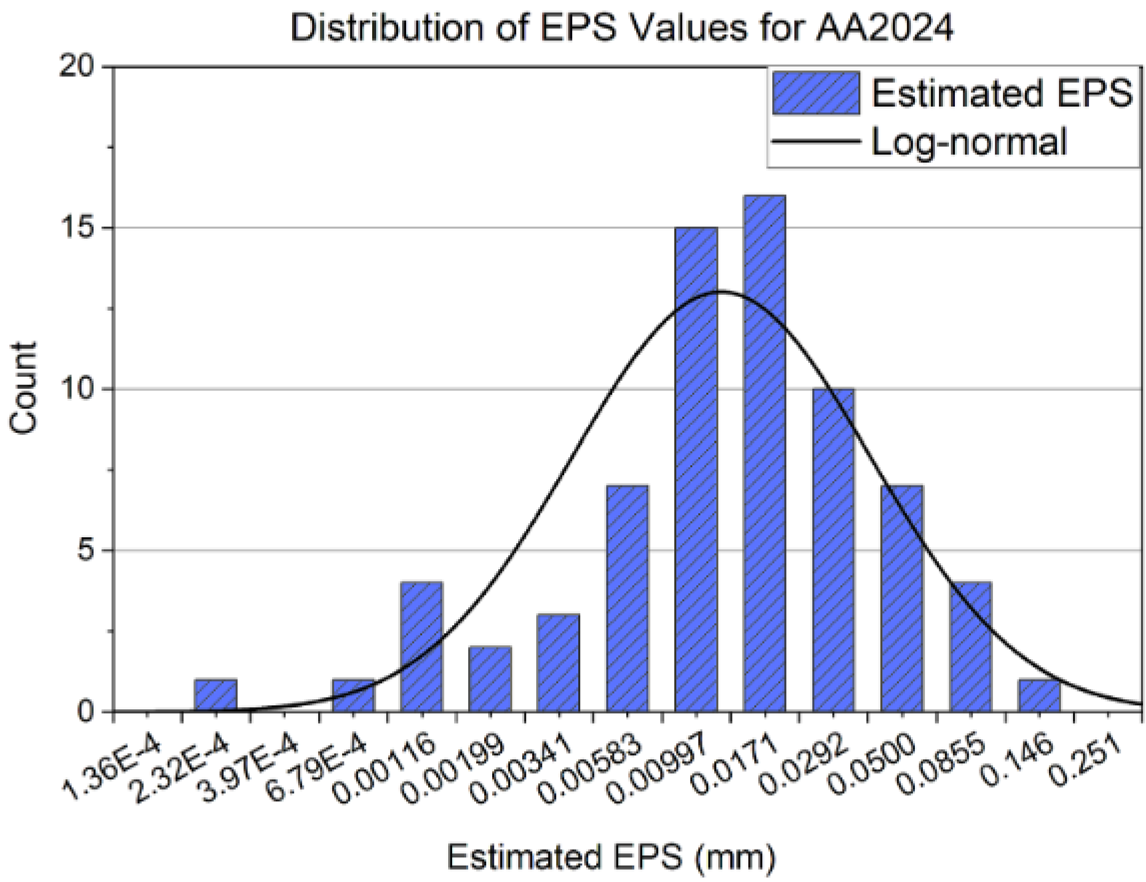

Figure 4 Log-normal distribution of the 71 AA2024 estimated pre-crack size values.

Reference

1. L. Molent and M. R. Fox, 'Crack-like effectiveness of some discontinuities in AA2024', *Fatigue and Fractures of Engineering Materials and Structures*, September 2023. https://doi.org/10.1111/ffe.14145

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2.6. Simulation of FCG in Aluminium Alloy 7050-T7351 under Spike Overload and Aircraft Spectrum Loading – K. F. Walker, A. Grice, J. C. Newman Jr., R. Zouev, S. A. Barter and D. Russell (QinetiQ, RMIT University, DSTG)

The trend towards virtual testing and digital-twin assisted management means that the accurate and reliable simulation of fatigue crack propagation behaviour is more important than ever. Reliable but conservative approaches to support this are in widespread use in the aerospace industry. Nevertheless, the conservatism comes at a significant cost in terms of reduced structural life and an increased ongoing inspection requirement, and as such leads to questions about the economic burden of these approaches. Recent comparisons between blind predictions and test results revealed the extent of the issue for cracking in AA 7075-T7351 coupons with configuration and loading representative of military transport aircraft wing skins. The current models were generally conservative by a factor of 2 or more in terms of crack propagation life. This suggested that there was significant scope to improve the modelling to better reflect all the complex contributing factors. The current work has investigated the issue of changes in the crack front constraint as the crack progresses from a state of high constraint (close to plane strain) to a lower constraint (approaching plane stress). This issue was investigated both experimentally and with the development of an improved analytical model. A test program was conducted on several specimens, loaded under CA, CA with spike overloads and a variable amplitude spectrum. Crack-opening stress levels were measured at key points in the tests and the results were used to develop and evaluate improved modelling approaches. The improved model was generally able to predict crack growth within about ± 30% of that demonstrated, along with the correct form of the crack growth, which is a significant advance and will lead to reduced costs and increased safety.

References

1. K. F. Walker, A. Grice, J. C. Newman Jr., R. Zouev, S. A. Barter and D. Russell, ‘Simulation of fatigue crack growth in aluminium alloy 7050-T7351 under spike overload and aircraft spectrum loading’, *International Journal of Fatigue*, vol. 190, January 2025. https://doi.org/10.1016/j.ijfatigue.2024.108660

2. J. C. Newman and K. F. Walker, 'Fatigue Crack Growth on Several Materials under Single-Spike Overloads and Aircraft Spectra during Constraint-Loss Behavior,' *Materials Performance and Characterization* https://doi.org/10.1520/MPC20230074

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2.7. Developments in Determining the Closure-free da/dN versus ΔK_{eff} Curve – R. Jones, A. S. M. Ang and D. Peng (Swinburne University, Monash University)

Recent studies have derived a simple closed-form analytical formulae for determining the crack closure-free da/dN versus ΔK_{eff} curve from the experimentally measured R ratio-dependent da/dN versus ΔK curves. This approach, which is termed 'Simple Scaling', has been verified for a wide range of tests on metals, polymers and on a medium-entropy alloy. The current study illustrates how to use this simple approach to improve the consistency of the da/dN versus ΔK_{eff} curves given in the open literature for fatigue tests on B9310 steel and solution-treated and overaged Ti-6Al-4V. Advantages of this approach over other approaches in the open literature are that it does not require the use of an assumed constraint factor, strain gauges, adjusted compliance ratio measurements or the measurement of the ASTM E647-defined crack opening force.

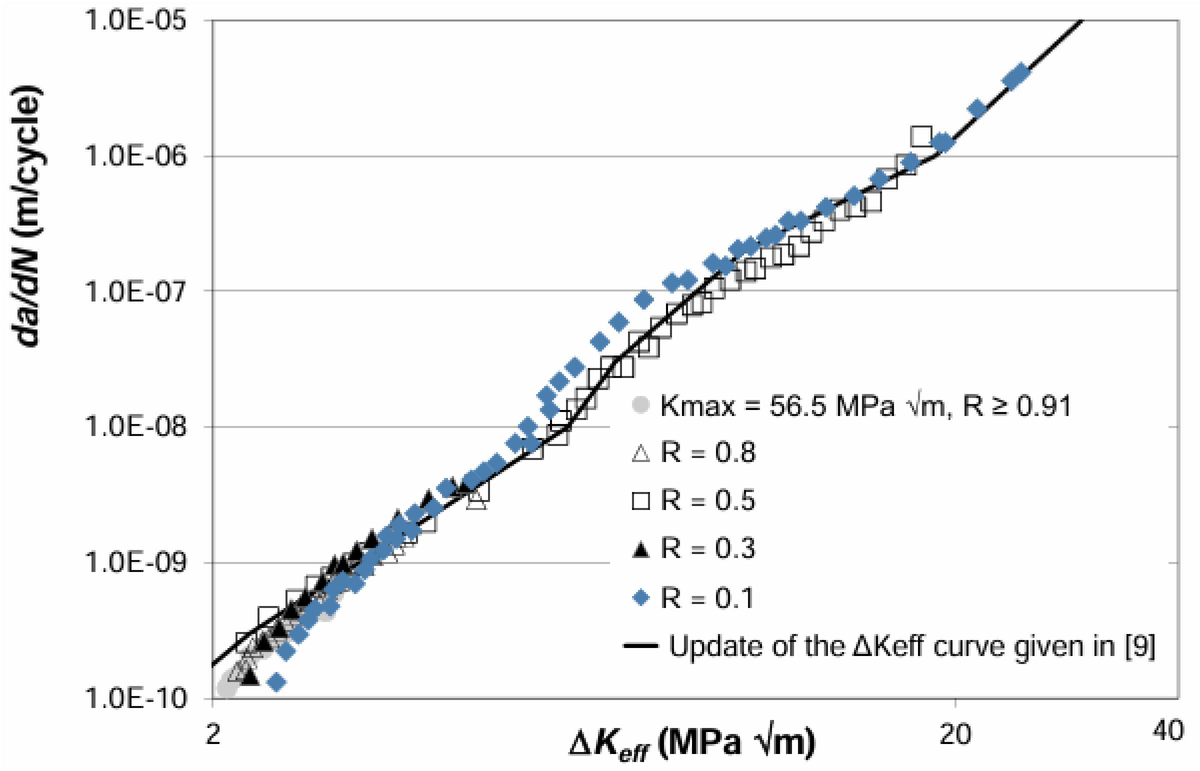

Figure 5 The measured and computed curves associated with the lead crack in Ti-6Al-4V.

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3. STRUCTURAL INTEGRITY OF COMPOSITE STRUCTURES

3.1. Enhancing Damage Tolerance in Tufted Composites: Finite Element Modelling and Predictions – M. Limprapuwiwattana, A. R. Ravindran, E. Kandare, P. Marzocca and R. B. Ladani (RMIT University, Sydney University)

Fiber-reinforced polymer composite laminates can suffer delamination damage from incidents such as low-velocity impacts, overloading and cyclic loading. The introduction of through-thickness 3D reinforcements can promote greater resistance to delamination damage. 3D reinforcement based on tufting can lead to significant improvement in the delamination resistance and damage tolerance of carbon fibre-reinforced polymer (CFRP) composites. In tufted composite laminates, the tufts, which are oriented along the through-thickness direction, resist delamination growth by forming a large-scale crack bridging zone. Recent studies have demonstrated the use of Nitinol (Ni-Ti) wire as a tufting material to improve the interlaminar fracture toughness of composite laminates, while simultaneously promoting the ability to detect and close delamination cracks using the shape memory effect of the Ni-Ti alloy-based tufts. This technology has the potential to replace conventional composite tufting materials made from carbon, glass and aramid filaments. However, the efficacy of Ni-Ti-based tufts at improving the interlaminar fracture toughness of composites has not been compared to conventional tufting materials. This comparative assessment is necessary to promote the uptake of this technology for composite aerostructure applications.

In this study, a comparative assessment of 4 different tufting materials, including Ni-Ti wire, copper wire, carbon fibre and aramid fibre, has been conducted to investigate their efficacy at improving the mode I interlaminar fracture toughness. Additionally, the study presents a finite element analysis (FEA) technique for predicting the fracture properties of tufted composite laminates. The experimental results revealed that all tuft types generated some improvement in mode I interlaminar fracture toughness. The extent of the improvement depends on the type of material, with stronger materials such as Kevlar resulting in greater enhancement. The bridging mechanism of the tufts can vary based on material characteristics, with different materials absorbing varying amounts of energy during crack propagation. Some tufting materials, like thin metal filaments, exhibited a significant amount of pull-out during propagation, resulting in additional energy absorption and a greater improvement in mode I interlaminar fracture toughness. The proposed finite element technique is capable of simulating the crack bridging response of the 4 different types of tufting materials and the resulting crack growth resistance (R-curve) behaviour of tufted composite laminates. This finite element model can be utilised in the design of tufted laminates with superior fracture toughness properties.

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3.2. FEA Digital Twin of a Scarf Repair for a Composite Component – C. M. T. Tien, T. Shelley, K. Pattarakunnan and X. Zeng (University of Southern Queensland, MEMKO)

As the use of composite materials in contemporary commercial aircraft increases, the demand for efficient composite repairs has also risen significantly. Major maintenance, repair and overhaul facilities face various challenges, including ensuring damage tolerance, improving repair efficiency, maintaining structural integrity and achieving cost effectiveness. This study presents a parametric FEA of failure mechanisms associated with bonded scarf repairs on carbon fibre composite panels. Through numerical analysis, it is demonstrated that selecting adhesives with appropriate material properties and thicknesses can significantly mitigate stress concentrations within the repaired panel. Additionally, developing a digital twin of the numerical model facilitates automatic updates to the digital representation of the actual repaired component under different repair scenarios, such as variations in ply thickness and stacking configurations. This innovation has the potential to significantly save both time and costs.

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3.3. Post-impact Multi-axial Load Response of Aero Representative Stiffened Composite Structures – C. Swann, L. Doxey, A. Orifici, B. Main, B. Falzon, S. Barter and R. Das (RMIT University, DSTG)

The response of impact-damaged composite aerostructures under loading is often informed by tests undertaken on smaller specimens with idealised boundary and loading conditions. In isolation, these types of tests cannot capture the complex behaviour of realistic composite structures subjected to more representative multi-axial loading. They do, nonetheless, provide useful data as part of a pyramidal building-block approach, but may still fail to anticipate certain modes of failure further up the test pyramid. In response to this, an experimental and numerical investigation seeks to characterise the influence of representative structural and loading configurations on the post-impact load response of stiffened composite aerostructures. To achieve this, a bespoke multi-axial testing facility, designed and built by DSTG, was used to deliver multi-axial load combinations to composite specimens that were configured as an upper wing skin of a loaded wing box section. Initial tests considered low-velocity impact on middle-skin regions that produced barely visible impact damage (BVID). These were then tested for the post-impact multi-axial load response under a combination of direct compression and in-plane shear – simulating typical flight loads on an upper wing skin. The main objective of this research is to assess the influence that the additional in-plane shear loading had on the compression after impact response of the composite laminates used. Preliminary experimental and high-fidelity modelling results provide rich insight into the damage initiation and its progression in a complex aerostructure under multi-axial loads.

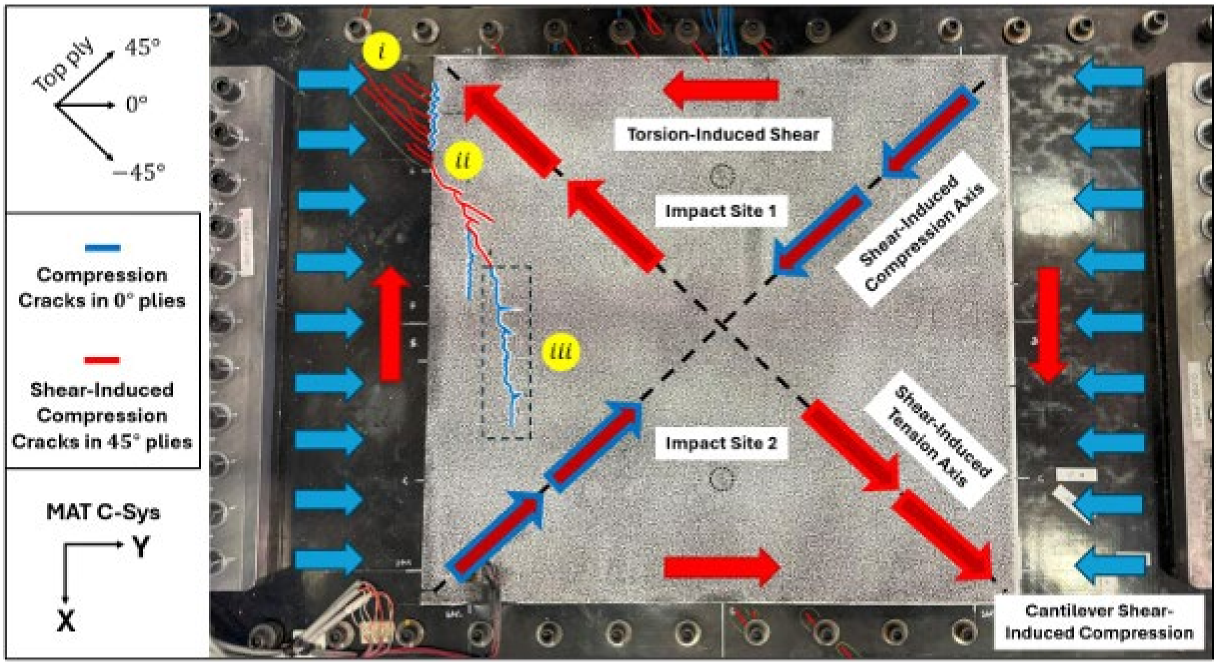

Figure 6 Far-field loadings and observed failures in the unstiffened panel.

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3.4. Effect of Hygrothermal Ageing Temperature on the Mechanical Degradation of Aerospace-grade Carbon Fibre Epoxy Laminates – K. M. Grigoriou, A. Farukh, L. Wood, M. Sarra (Monash University)

The Federal Aviation Administration mandates that aircraft structures be protected against deterioration from environmental factors such as weathering, corrosion and abrasion, and that materials used must withstand service environmental conditions like temperature and humidity (§ 25.609, § 25.603). Real-time weathering tests for composite materials are impractical due to prolonged exposure times and inconsistent environmental conditions. Commonly, composites are conditioned in hygrothermal chambers to accelerate moisture uptake, a method that is both time consuming and costly. This research investigates accelerated conditioning methods for composite laminates, utilising a hygrothermal chamber (60–80 °C, 90% relative humidity) and an immersion bath (30–50 °C). Specimens will be post conditioned mechanically tested under compression at elevated temperatures after achieving a moisture content of 1.0% by weight. The study aims to determine if higher conditioning temperatures, which are known to accelerate moisture uptake, influence the extent of mechanical degradation. Additionally, it examines whether immersion bath conditioning affects mechanical degradation similarly. If mechanical degradation is found to be solely dependent on absorbed moisture content rather than the conditioning method, these accelerated techniques could be employed to significantly reduce the conditioning time for composite laminates. This research has the potential to streamline the conditioning process, providing a more efficient and cost-effective approach for evaluating the durability of composite materials in aircraft structures.

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3.5. Thoughts on the Importance of Similitude and Multi-axial Loads when Assessing the Durability and Damage Tolerance of Adhesively Bonded Doublers and Repairs – R. Jones, R. Chandwani, C. Timbrell, A. J. Kinloch and D. Peng (Swinburne University, Monash University)

Adhesively bonded doublers and adhesively bonded repairs are extensively used to extend the operational life of metallic aircraft structures. Consequently, this paper focuses on the tools needed to address sustainment issues associated with both adhesively bonded doublers and adhesively bonded repairs to (metallic) aircraft structures, in a fashion that is consistent with the building-block approach mandated in the United States Air Force (USAF) airworthiness certification standard MIL-STD-1530D, and also in the United States (US) Joint Services Structural Guidelines JSSG-2006. In this context, it is shown that the effect of biaxial loads on cohesive crack growth in a bonded doubler under both constant amplitude fatigue loads and operational flight loads can be significant. It is also suggested that as a result, for uniaxial tests to replicate the cohesive crack growth seen in adhesively bonded doublers and adhesively bonded repairs under operational flight loads, the magnitude of the applied load spectrum may need to be continuously modified so as to ensure that the crack tip similitude parameter in the laboratory tests reflects that seen in the full-scale aircraft.

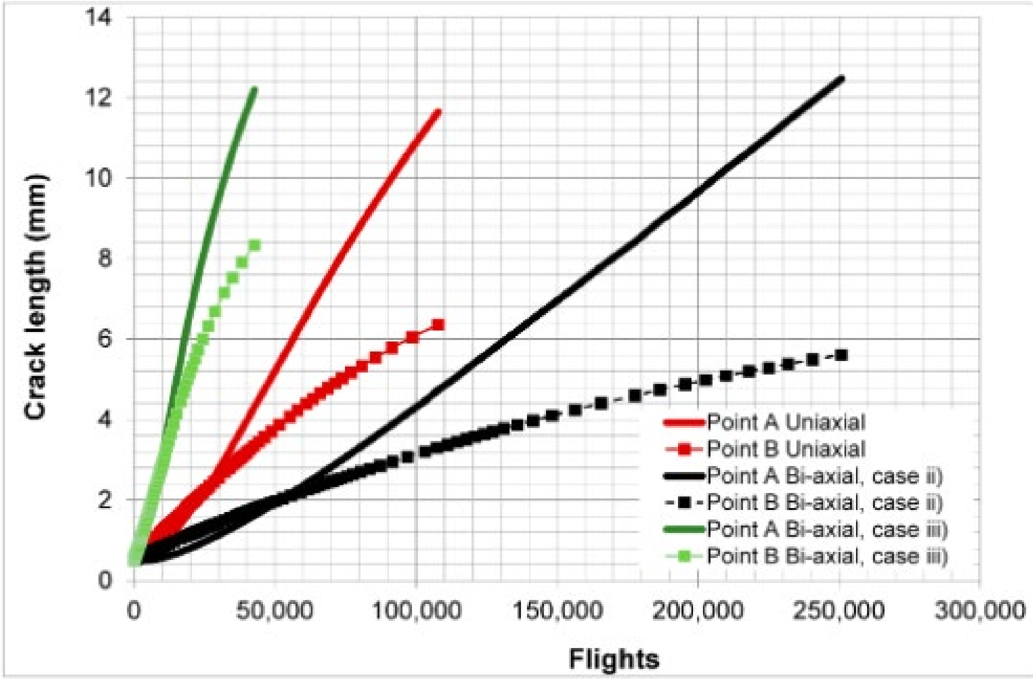

Figure 7 The crack length versus cycles histories at the 2 ends of the crack for the plate with the adhesively bonded doublers growing under a FALSTAFF flight load spectrum.

Reference

1. R. Jones, R. Chandwani, C. Timbrell, A. J. Kinloch and D. Peng, ‘Thoughts on the importance of similitude and multi-axial loading when assessing the durability and damage tolerance of adhesively bonded doublers and repairs’ *Aerospace*, November 2023. https://doi.org/10.3390/aerospace10110946

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4. ADVANCED MATERIALS AND INNOVATIVE STRUCTURAL CONCEPTS

4.1. Predicting the Growth of Small Cracks in Wire Arc Additively Manufactured (WAAM) CP-Ti – D. Peng, A. S. M. Ang, M. B. Nicholas, V. K. Champagne, A. Birt, A. Michelson, S. Langan and R. Jones (Swinburne University, Monash University)

The USAF airworthiness certification standard MIL-STD-1530D requires an ability to predict the growth of small cracks in conventionally manufactured parts in a way that is consistent with the 'building block' approach that is also required in the US Joint Services Guidelines JSSG2006. USAF Structures Bulletin EZ-SB-19-01, which presents the guidelines for additively manufactured (AM) parts, says the same thing. The authors have previously presented examples of how to perform such analyses for AM Ti-6Al-4V, wire arc additively manufactured (WAAM) 18Ni 250 Maraging steel and Boeing Space, Intelligence and Weapon Systems laser bed powder fusion (LPBF) Scalmalloy®, which is an AM aluminium-Scandium-Mg alloy, using the Hartman-Schijve crack growth equation. In these studies the constants used were as determined from ASTM E647 standard tests on long cracks, and the fatigue threshold term in the Hartman-Schijve equation was set to a small value (namely 0.1 MPa √m). This paper illustrates how this approach can also be used to predict the growth of small cracks in WAAM CP-Ti (commercially pure titanium) specimens built by Solvus Global. As in the prior studies mentioned above, the constants used in this analysis were taken from prior studies into the growth of long cracks in conventionally manufactured CP-Ti and the fatigue threshold term is set to 0.1 MPa √m.

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4.2. Enhancing the Fatigue Performance of AM Components with Minimal Intervention – J. Rogers, M. Watson, J. Elambasseril, C. Wallbrink, M. Qian, M. Brandt and M. Leary (RMIT University, DSTG)

AM is intended to reduce manufacturing lead times; however, when post processing is used to modify functional material properties, this time advantage is lost. This research has investigated a pre-process workflow to enhance the fatigue resistance of laser-based powder bed fusion (LB-PBF) components by optimising the build orientation. The inherent surface roughness of LB-PBF varies with surface orientation and in the presence of support material. High surface roughness has been linked with poor fatigue performance; LB-PBF surfaces having mutual contact with support material and low-angle, downward-facing surfaces will exhibit high surface roughness. Therefore, mathematical expressions between both surface orientation and surface roughness, and surface roughness and fatigue life were derived from an extensive experimental coupon fatigue test program. An AM pre-processor was then developed using these expressions to advise the AM build operator of the optimal build orientation to enhance fatigue life based on the intended loading condition and local surface roughness. This pre-processor was experimentally validated using 3-point bend fatigue testing on topology optimised specimens, and these results will be presented.

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4.3. Redesign of Structural Aerospace Components for Metal AM using Multi-objective Topology Optimisation – C. Dionyssopoulos, B. Krieg and C. Wallbrink (DSTG)

The availability of replacement parts is critical to the operational readiness of aerospace platforms. Metal AM enables the on-demand printing of replacement parts, alleviating logistical challenges such as long supply chains, long lead times and part obsolescence. However, the use of metal AM likely necessitates a change in material from the original part, as many metal alloys developed for conventional manufacturing are unsuitable for AM. For structural applications, a redesign may then be necessary to preserve the load transfer mechanisms of the original part and avoid the introduction of new failure modes to the structure. In this work, a redesign workflow is developed that utilises a multi-objective bidirectional evolutionary structural optimisation method. The deformation response is optimised to match the original part while reducing the maximum stress and the amount of support material required for printing. An adaptive weighting scheme and adaptive volume constraint are proposed to improve the ability of the optimisation algorithm to match the local response of a part with multiple connections to the surrounding structure. An exemplar part is used to demonstrate the ability of the proposed redesign workflow to design a printable AM part with minimal supports. Nonlinear FEA is used to validate the performance of the optimised part and demonstrate the part matches the original part behaviour with the benefit of a reduction in maximum stress. The redesign workflow offers the potential to aid qualification and enable rapid design of 'drop-in' AM replacement parts for aerospace applications.

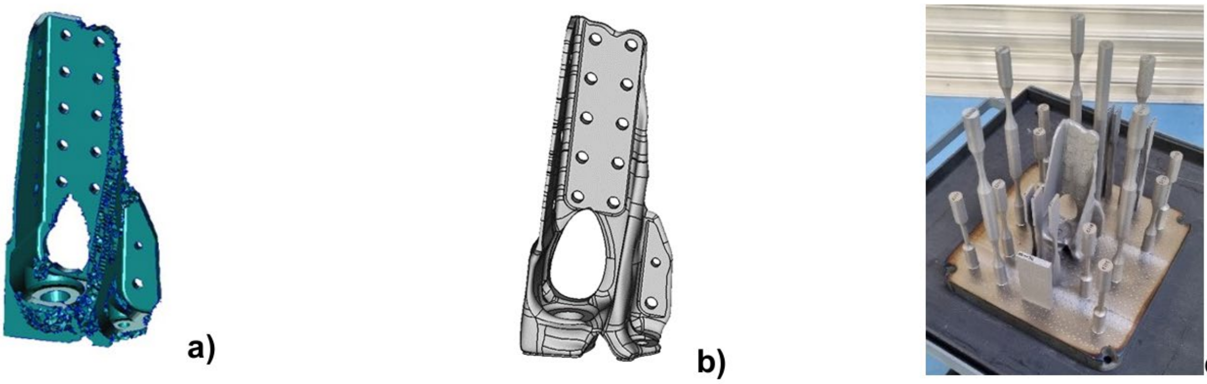

Figure 8 Output geometry a) of the fitting topology optimisation, smoothed optimised fitting geometry b) obtained from post and c) optimised printing using LB-PBF with Ti-6Al-4V powder.

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4.4. A New Approach for the Analysis of Long Cylinders Buckling under External Pressure – L. J. Hart-Smith

It is shown here that the classical analysis of buckling of very long cylinders under external pressure is accurately described as a succession of 3 unnecessary false assumptions leading to a buckling mode shape that is physically impossible. The first assumption is that the buckling mode is inextensional, which eliminates the possibility of satisfying equilibrium around the circumference. The equilibrium equation is conspicuously absent from the classical analysis. The second assumption is that there is only one possible buckling mode shape with a single wavelength around the circumference, and that this is merely a rigid-body motion. There are actually multiple single-wavelength mode shapes that are legitimate buckling modes, when the radial and circumferential strains are unequal. The third assumptions is that, since the single-wavelength option had been ruled out of consideration, albeit incorrectly, there must be 2 wavelengths around the circumference, but that mode shape is physically impossible unless 2 full-length diametrically opposite stiffeners are added to prevent radial motion along the stiffeners.

A new analysis of this problem is presented here, involving no assumptions. This reveals that the mode shape is actually extensional. Also, it is predicted that there is only one wavelength around the circumference – and this is not a rigid-body motion. The new buckling pressure is only one third of the classical prediction. This new buckling pressure would have been a factor of 4 lower if the classical analysis had not used 2 different expressions for the change in curvature around the circumference, during buckling, as was discovered by the NACA researchers using Donnell's equations in the late 1940s. Only shorter cylinders, in which the end effects influence the buckling stress, can exhibit 2-wavelength buckling modes.

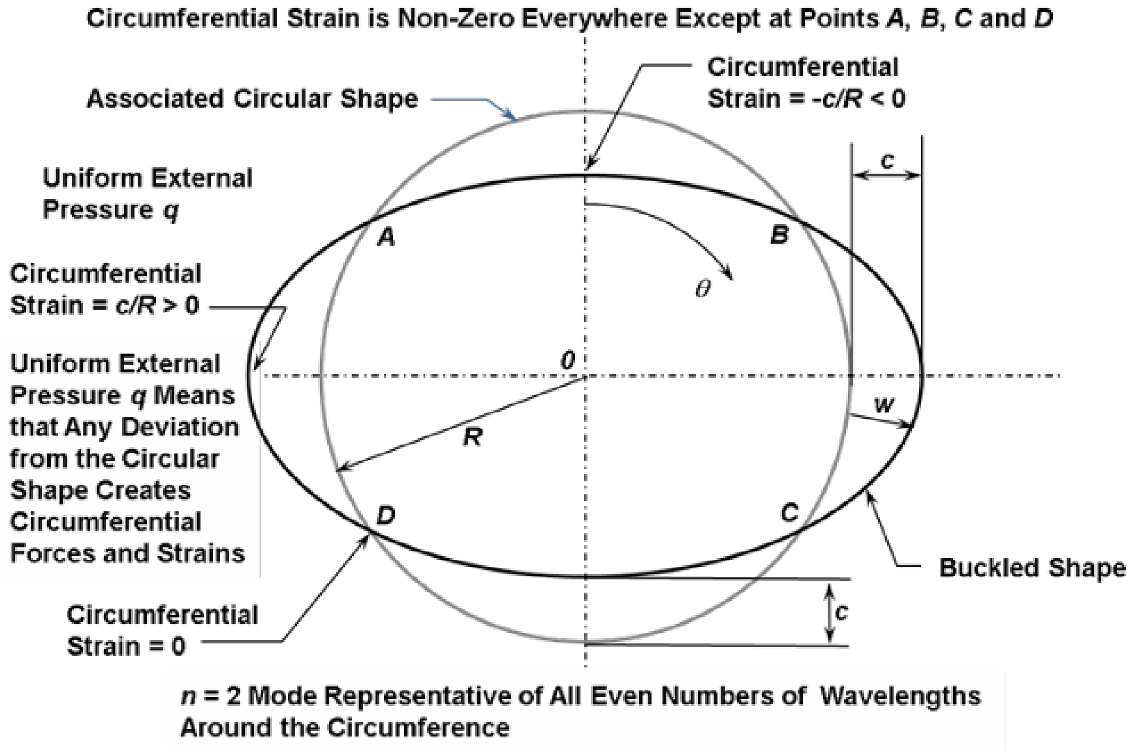

Figure 9 Impossibility of an inextensional even-numbered-wavelength buckling mode for cylindrical shells buckling under external pressure.

Contact

L. John Hart-Smith (dec.)

4.5. Energy Storage Composites with Nanomaterial Functionalisation – V. Gangipamula, K. Subhani, P. J. Mahon and N. V. Salim (Swinburne University)

In the current era, the performance of electric vertical take-off and landing (eVTOL) vehicles is defined by the available onboard power to meet their energy requirements for operations. Therefore, the focus is on increasing payload capacity and flight endurance. This is accomplished by utilising structural composite parts that serve as energy storage devices. This capability is provided by the multifunctionality of the structural supercapacitors (SSC) with carbon fibre (CF) reinforcements, which offer a blend of structural strength and energy storage capacity to realise the benefits of weight and energy savings. Despite extensive research, the energy and power outputs remain inadequate. One of the contributing factors is the hindered ion conductivity resulting from the insufficient active surface area of pristine CFs. Therefore, there is a need to enhance the surface area of the CFs to improve conductivity by functionalising their surface with carbon-based nanomaterials.

In this work, we have fabricated an SSC using surface-modified CF electrodes with a coating of dual materials graphene (G) and activated carbon (aC), resulting in excellent improvement in surface area by 210 times compared to uncoated CFs. A solid polymer electrolyte consisting of a polymer part and ionic liquid combined with lithium-based salts is impregnated through electrodes separated by a cellulose-based separator using a hand layup technique to prepare SSC. The CF electrodes demonstrated the excellent performance of a specific capacitance of 172 F. g^{-1} in aqueous electrolyte and the SSC device with a specific capacitance of 155 mF. g^{-1}, realising the production of high-performing SSCs with the reported method, making them suitable for applications in aerospace and automobile structures.

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4.6. On the Growth of Small Cracks in 2024-T3 and Boeing Space, Intelligence and Weapon Systems AM LPBF Scamalloy – R. Jones, A. Ang, R. W. Aston, N. D. Schoenborn, V. K. Champagne, D. Peng and N. D. Phan (Swinburne University, Monash University)

The desire to use AM parts to ensure the availability of military aircraft, and to build limited-life uncrewed aerial vehicles (drones), coupled with the United States Air Force

(USAF) approach to the airworthiness certification of AM parts has focused attention on durability analysis/assessment, and hence on the growth of small cracks in AM parts. Previous studies have shown that LB-PBF Scalmalloy has: i) a yield stress and an ultimate strength that are greater than that of AA2024-T3 and comparable to that of AA7075-T6; and ii) a resistance to crack growth that is better than that ofAA7075-T6 and comparable to that of AA2024-T3. However, since the ability to predict the durability of a part is essential for its airworthiness certification, the present paper illustrates how to perform a LEFM-based durability assessment of Boeing Space, Intelligence and Weapon System (BSIWS) LB-PBF Scalmalloy. The durability study includes specimens with both machined surfaces and surfaces left in the as-built condition. As a result, it would appear that BSIWS AM LB-PBF Scalmalloy is an ideal candidate for building limited-life AM replacement parts for fixed and rotary wing aircraft and drones.

References

1. R. Jones, A. Ang, R. W. Aston, N. D. Schoenborn, V. K. Champagne, D. Peng and N. D. Phan, 'On the growth of small cracks in 2024-T3 and Boeing Space, Intelligence and Weapon Systems AM LPBF Scamalloy', *Fatigue and Fractures of Engineering Materials and Structures*, vol. 48, pp. 31-43, September 2024. https://doi.org/10.1111/ffe.14468

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5. FULL-SCALE FATIGUE TESTING

5.1. Observations of Fatigue Crack Nucleation and Growth in Ti-6Al-4V Full-scale Structures under Combat Aircraft Loading – I. Field, S. Barter, M. Jones, B. Main, R. F. Rosario and M. Figliolino (DSTG, RMIT University)

Defence Science and Technology Group and RMIT University have a strong foundation in the application of QF techniques to the study of fatigue crack nucleation and growth mechanisms in common aerospace alloys. In this presentation, the authors share their collective observations and experience in undertaking QF of post-teardown recrystallised annealed titanium combat aircraft structures. Naturally occurring and artificial crack nucleation sites were investigated, equivalent initial damage sizes (EIDS) were developed and FCG rates measured through small and long crack regimes. Key findings relate to the faceted and directional nature of Ti-6Al-4V FCG, the importance of material microstructure on fracture surface appearance and fatigue resistance, the care that must be taken when trying to identify marker bands in a material with fine grains where grain boundaries can appear similar to the markers, and how the crack front evolves through nano, micro and macro crack depth scales.

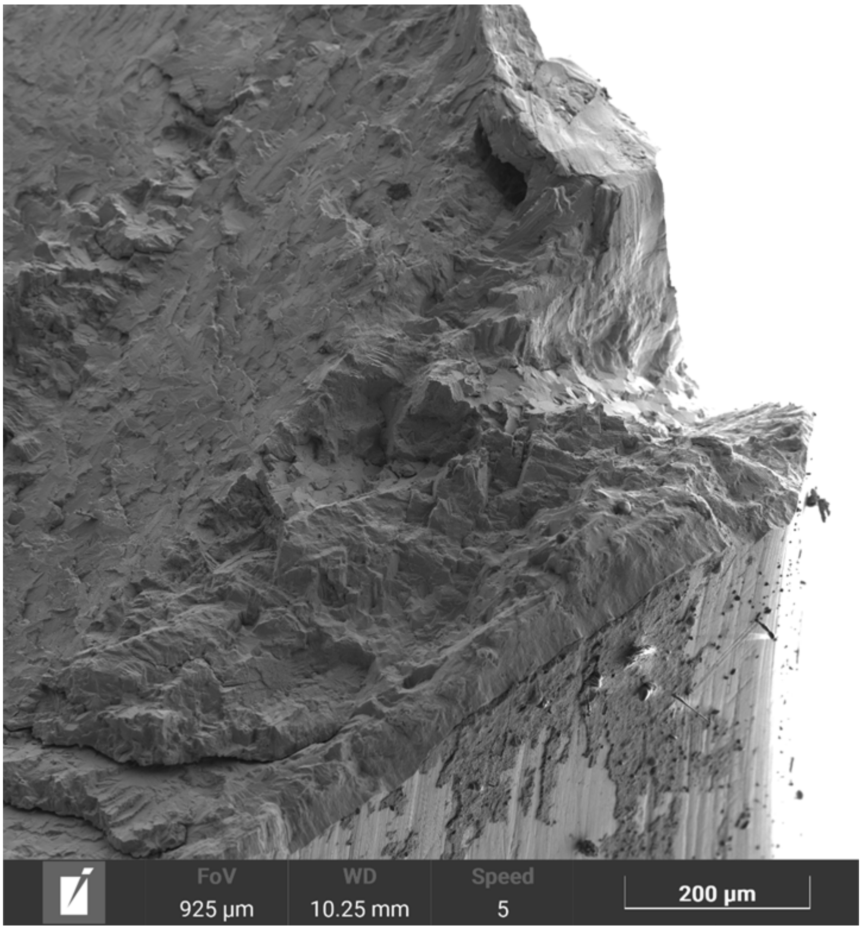

Figure 10 A fatigue crack nucleating from the bore of a fastener hole in a Ti-6Al-4V combat aircraft bulkhead.

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5.2. A Fractographic Study of Fatigue Failures in Combat Aircraft Trailing Edge Flap Hinge Lug Bores in both Test and Service Assets – B. Main, S. Barer, I. Kongshavn, R. F. Rosario, J. Rogers and M. Figliolino (DSTG, RMIT University)

This work compared and contrast the outcomes of 2 fractography investigations of cracking in combat aircraft trailing edge flap hinge lug bores. One, a retired service asset and the second, a full-scale durability test article in a fleet repaired configuration. Since these hinge lugs are primary structure with the potential for fatigue failure to lead to flap separation from the wing in-flight, a detailed understanding of the root cause, crack morphology and growth were required to manage these structures in-service.

The root cause of fatigue crack nucleation and growth in the service assets was found to be corrosion, whilst in the full-scale durability test, it was mechanical damage or fretting associated with the application of the fleet representative bushing repair scheme. The implications of each root cause were examined along with the crack morphologies, and proposals for in-service inspection and the prevention or the delay of damage formation are discussed.

Reference

1. B. Main, S. Barter, I. Kongshavn, R. F. Rosario, J. Rogers and M. Figliolino, ‘A fractographic study of fatigue failures in combat aircraft trailing edge flap hinge lug bores in both test and service assets’, *Engineering Failure Analysis*, vol. AA, 2025.

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5.3. A Method for Imparting Small-scale Damage for Damage Tolerance Testing – I. Field, J. Rogers, M. Jones, B. Main, K. Muller and S. Barter – (DSTG, RMIT University)

One challenge facing aircraft certification under MIL-STD-1530D is the ability to impart damage to test structures in a representative manner for damage tolerance testing. This is particularly true if small scale-damage (of the order of 0.01”) is required to be controllably imparted. The Selective Plasma Arc Spot Melter (SPASM) is a device that has been developed to impart precise and adjustable damage to metallic test articles. The intended purpose of this device is to allow small crack-like features to be applied to various different locations on test articles. These features act as fatigue crack initiators which can be used to assist with damage tolerance full-scale fatigue tests (FSFTs) and related coupon studies. This presentation provides an overview of the SPASM method, including coupon and full-scale test results, verified by QF that show the successful implementation of this technique.

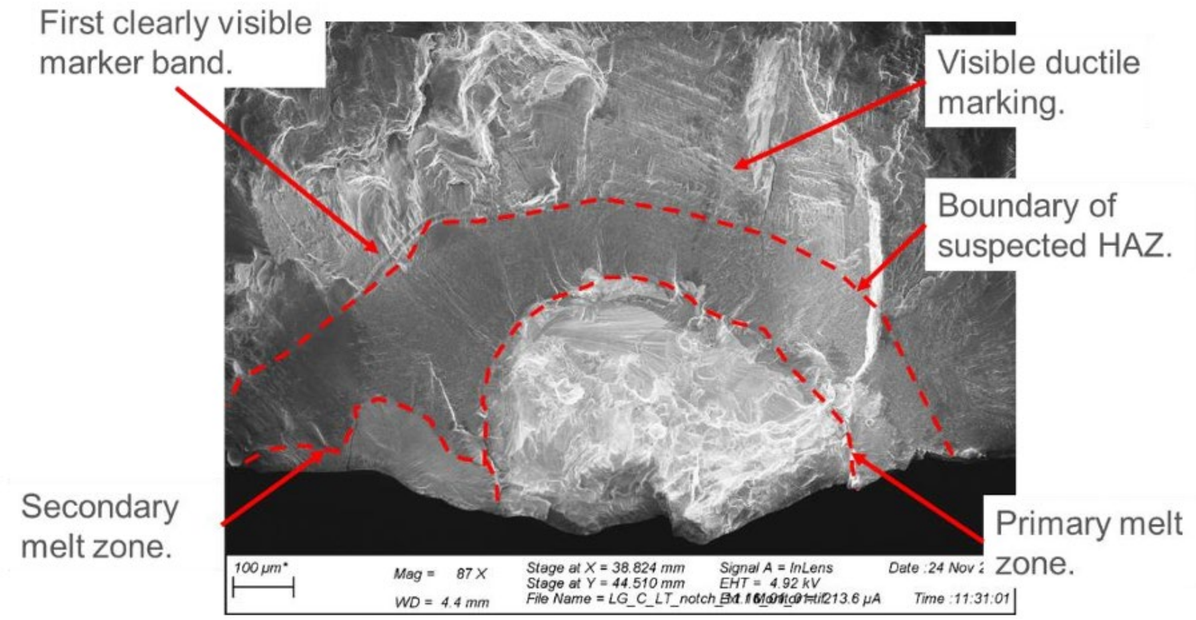

Figure 11 The exposed cross-section of a Ti-6Al-4V coupon exposed to an 11.16J SPASM burn after breaking through fatigue.

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5.4. Technical Outcomes from the Helicopter Advanced Fatigue Test – Technology Demonstrator (HAFT-TD) Program – G. Swanton, A. Manning, M. Chipper, A. Walliker, B. Evans and J. Moonen (DSTG, RMIT University)

With a 75 year heritage in conducting structural testing of fixed-wing military aircraft, DSTG was tasked with the challenge of designing, developing and delivering its first prototype test rig for structural fatigue testing of an entire helicopter airframe. Until recently, the dynamic nature of helicopter flight loading meant that the quantity of load content in any representative test spectrum was considered to be the main technical impediment to executing an independent FSFT in a practical timeframe using extant technologies. Known as the Helicopter Advanced Fatigue Test – Technology Demonstrator (HAFT-TD), this Australian-US collaboration was instigated to assess the viability of this proof of concept to advance test speeds and thus provide the US Navy's MH-60R Seahawk Service Life Assessment Program (SLAP) with options for undertaking a potential FSFT in support of its SLAP activities. Over some 6 years, DSTG partnered with naval aviation groups, industry and academia to mature and tailor the technologies necessary to conduct a helicopter FSFT in a comparable timeframe to typical fixed-wing test campaigns. Research and development efforts were directed at the 4 focus areas required as inputs to realise the demonstrator phase, namely: (1) external loads development, (2) load spectrum compression techniques, (3) advanced model-assisted control methodology for servo-hydraulic actuation, and (4) bespoke test rig and load application systems. At the conclusion of the initial demonstration phase, the test results were assessed against the acceptance criteria, which centred on multiple performance metrics involving load application rate, load control accuracy, airframe strains, demonstration of multiple flight conditions, and the accrual of representative fatigue damage using a manipulated spectrum. Despite the many technical and non-technical challenges encountered, HAFT-TD has made significant strides against all criteria, meeting or exceeding the key measures of test cyclic rate and accuracy therefore underpinning a future potential helicopter FSFT.

Figure 12 The HAFT-TD test article and rig at DSTG Fishermans Bend.

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6. FATIGUE LIFE ENHANCEMENT METHODS AND REPAIR SOLUTIONS

6.1. A Fractographic Study of FCG from a Cold-expanded Fastener Hole at an Engineering 'Crack Initiation' Scale – B. Main, D. Russell and S. Barter (DSTG, RMIT University)

The effects of cold expansion on FCG from fastener holes in aircraft structure is a well-studied problem, with many contributions focussing on both residual stresses and 'long crack' experimental data. In this study, the authors have used QF techniques to measure FCG from naturally occurring discontinuities on an engineering 'crack initiation' scale.

FCGRs were measured through the full range of crack depths allowing the effects of beneficial residual stresses to be directly observed. The almost identical early crack growth rates in cold expanded and non-cold expanded specimens was demonstrated up to crack depths of approximately 0.1–0.2 mm. Beyond this depth, FCG in cold-expanded specimens was markedly slowed to the point at which failure ultimately arose from cracks away from, or growing into, the fastener hole. The available life improvement factor for cold expansion is therefore maximised through a crack growth-based lifing approach.

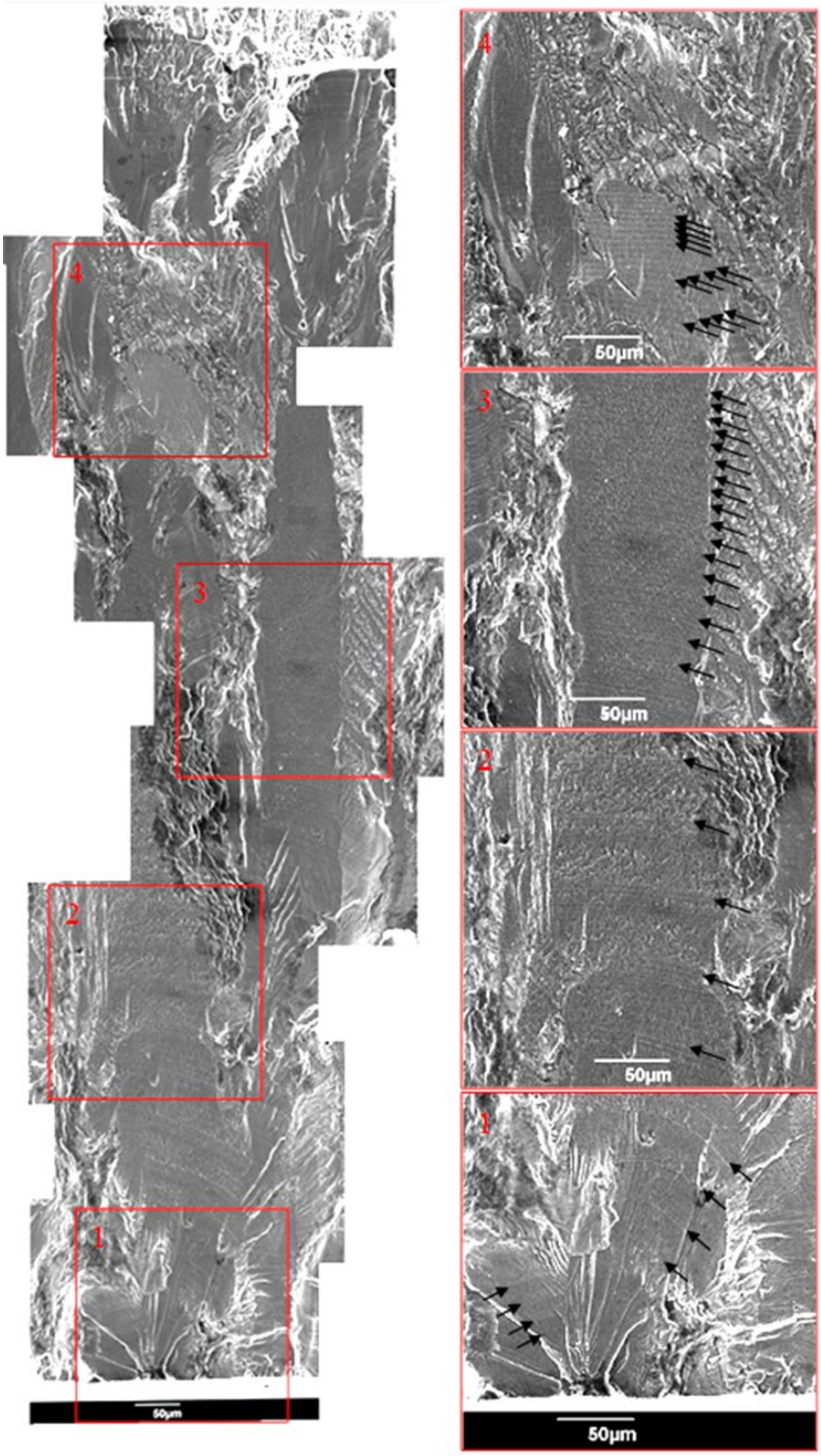

Figure 13 A montage showing the full depth of the fatigue crack perpendicular to the fastener hole bore from a cold-worked specimen origin (right). Expanded views of segments of the fracture surface along the cracks depth are shown on the left. Some of the block markings are indicated with small black arrows. The change in the relative spacing gives an indication of the change in the FCGR.

Reference

1. B. Main, D. Russell and S. Barter, 'A fractographic study of fatigue crack growth from a cold expanded fastener hole at an engineering 'crack initiation' scale', *Fatigue and Fracture of Engineering Materials and Structures*, vol. 47, iss. 4, pp. 1331-1344, 2024. https://doi.org/10.1111/ffe.14245

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7. NDI AND STRUCTURAL HEALTH/LOAD MONITORING

7.1. A Review of the Improvements Made to the F/A-18 Fatigue Tracking System: Individual Aircraft Tracking with a Safe Life Philosophy – M. Phillips and T. Mills (DSTG)

The Boeing Company uses proprietary engineering software and strain-life data to predict the fatigue life expended for each tracking location on the F/A-18 Hornet. This proprietary software undergoes a continual improvement cycle each year with oversight from the US Navy. The improvements made over the past decade to the tracking system of the F/A-18 Hornet, Super Hornet and Growler have been wide ranging. DSTG's expertise in fatigue tracking and long-lasting relationship with the US Navy has provided recommendations that have contributed to significant improvements to this tracking software functionality and how it processes data to assess data quality. These improvements made to data quality and fatigue assessments will be explored and provide a case study for improving other aircraft tracking systems.

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7.2. Barely Visible Impact Damage Detection on an F/A-18 Stabilator using Line Scan Thermography – S. Prasad, C. Rosalie, P. Muir, D. Bitton and K. Tsoi (DSTG)

The strength of CFRP structures is crucial for aircraft safety and sustained performance. BVID poses significant challenges to the structural integrity of these structures. Existing non-destructive inspection (NDI) techniques, such as phased array ultrasonic testing (PAUT), can detect various defects but can be time-consuming for large components. In contrast, thermographic techniques like flash thermography (FT) and line scan thermography (LST) offer rapid, wide-area inspection for sub-surface defects; however, thermographic methods are known to have limitations with respect to flaw depth. A multi-modal approach combining the strengths of each NDI technique allows an optimised solution for detecting and characterising BVID.

LST applies a narrow strip of heat to the inspection surface and the thermal signatures are recorded using an infrared detector. In this study, an F/A-18 horizontal stabilator

(STAB) was impacted with varying energy levels to produce BVID of different size and depth profiles. The STAB was inspected using an overhead cable-suspended LST system, which enabled the identification of the BVID locations by manoeuvring the system through 3 dimensions and with varying detector velocity. The resulting inspection data was analysed and all but the lowest energy BVID (3J) were detectable. The STAB was inspected using FT and PAUT and the lateral sizing and depth of the BVID compared with that detected using LST.

The successful detection of BVID on the F/A-18 stabilator using the overhead LST system underscores the effectiveness of a multi-modal strategy in maintaining the structural integrity and safety of critical composite components in aerospace applications. Initial FT and LST scans enable efficient location of BVID, after which the higher resolution and detailed characterisation offered by PAUT can be employed.

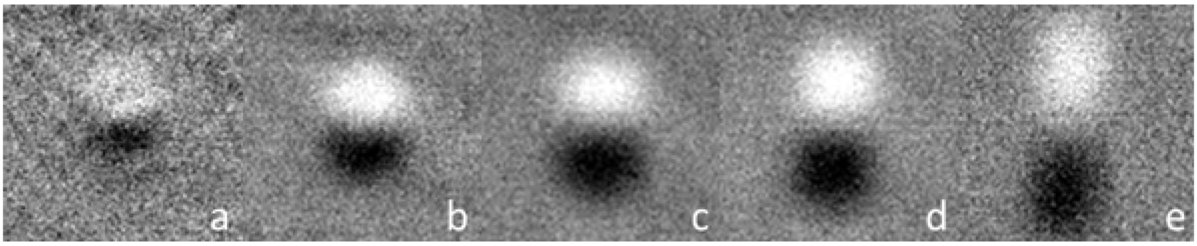

Figure 14 Thin section 4J defect of LST dynamic pulsed phase thermography with scan speeds of a) 25 mm/s, b) 50 mm/s, c) 75 mm/s, d) 100 mm/s and e) 150 mm/s.

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7.3. A Capability for Rapid Experimental Validation of Geometrically Complex and Safety-critical Aerospace Structural Components – N. Rajic, C. Brooks, K. Khauv, A. Mukhaimar, R. Tennakoon, F. Zambetta and P. Marzocca (RMIT University, 1Millikelvin, DSTG)

Recent innovations in digital engineering and advanced manufacturing provide a promising basis for accelerated design and development of new, more efficient, agile and sustainable aerospace platforms. Using a combination of presently available computer aided design, FEA, artificial intelligence and 3D printing capabilities, it is entirely conceivable now to transition new structural concepts to working prototypes in a matter of days, a pace unimaginable only a few years ago.

However, for safety-critical structures the scope for accelerated development is tempered by an airworthiness requirement for experimental validation of any new airframe design. This includes, and typically culminates in, a physical full-scale durability test of an airframe, which can often take many years to complete. Unfortunately, the digital engineering and manufacturing innovations that have done much to foster accelerated design and development have done comparatively little, hitherto at least, to alleviate or accelerate such testing. In fact, to the extent these innovations lead to increasingly complex, i.e. more highly optimised, structural designs, experimental validation is likely to become even more important and concomitantly more technically challenging. The development of new tools and approaches capable of supporting acceleration of this testing is consequently important.

The work describes a pervasive in situ stress imaging approach that facilitates rapid experimental validation of geometrically complex structural components at scales from small coupons to complete airframes. The approach consists of complementary hardware and software elements. The former comprises a compact stress/visual imaging device and the latter a combination of advanced thermoelastic stress analysis algorithms and a 3D stress superimposition capability that transforms pervasive 2D stress imagery into a 3D digital representation of stress state compatible with virtual and augmented reality environments. The core elements, utility, performance and practicality of this new capability are demonstrated using an experimental case study involving a geometrically complex 3D-printed structure.

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7.4. A Thermoelastic Stress Analysis and Coupled Modelling Approach for Improved Structural Testing and Evaluation – J. Codrington, S. Wildy, L. Button, K. Khauv and N. Rajic (University of Adelaide, 1Millikelvin, RMIT University)

To maintain aircraft safety, aviation regulatory authorities mandate that safety-critical components be subjected to rigorous structural design, analysis and experimental testing. However, this process can contribute significant cost and delay to aircraft development programs and subsequent modification and life extension. The primary driver of structural failure is mechanical stress, and one of the key root causes of unexpected failure is an inaccurate or incomplete understanding of the stress state. Thus, experimental stress measurement capabilities along with validated finite element models are therefore critically important to help reduce the timeline and costs associated with aircraft structural integrity programs.

This study presents an overview of recent developments in Thermoelastic Stress Analysis (TSA) and accompanying coupled field finite element modelling. A TSA software tool is being developed to enable visualisation of the stress state in real time, including fully automated detection, tracking and evaluation of stress-driven FCG. The tool is based on processing the in-phase and quadrature thermoelastic response fields measured through thermal imaging. Experimental testing has been undertaken with a wide range of standard test coupons and built-up lap joints with fasteners. The experimental datasets are supported by extensive finite element modelling using both one-way (linear) and direct (non-linear) thermal–structural coupled field simulations. The finite element models match closely to the measured fields and demonstrate the accuracy and also limitations of the TSA capability.

The new TSA and modelling capability will enable industry to more effectively exploit design, manufacturing and rapid prototyping innovations. This will result in shorter development cycles and accelerated material qualification and structural certification to support more effective utilisation of existing aircraft.

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