Available online at www.sciencedirect.com

ScienceDirect

Procedia Structural Integrity 28 (2020) 370–380

Structural Integrity Procedia

www.elsevier.com/locate/procedia

1st Virtual European Conference on Fracture

A means for industry to determine the economic life of bonded joints under representative operation flight loads

Rhys Jones[a*], Anthony J Kinloch [b], John G. Michopoulos[c], Daren Peng[a]

[a] *Centre of Expertise for Structural Mechanics, Department of Mechanical and Aerospace Engineering, Monash University, Clayton, Victoria, 3800, Australia*

[c] *Department of Mechanical Engineering, Imperial College London, Exhibition Road, London SW7 2AZ, UK*

[d] Computational Multiphysics Systems Laboratory, Code 6394, Center for Materials Physics and Technology, US Naval Research Laboratory, Washington, DC 20375, United States

Abstract

The present paper shows that the variability in the cyclic fatigue crack growth (FCG) rate in structural adhesives that is frequently observed in laboratory test results can be captured by using a Hartman-Schijve methodology. To this end the Hartman-Schijve equation has been used to access an 'upper-bound' FCG rate curve that (a) encompasses all the experimental data, (b) provides a conservative, 'worst-case' FCG rate curve, and (c) accounts for the experimental scatter that is frequently seen under fatigue loading. The importance of allowing for the variability in the measured FCG rate is illustrated by considering FCG in a bonded double-overlap joint subjected to an industry-standard, combat-aircraft flight-load fatigue-cycle spectrum (i.e. FALSTAFF).

Keywords: CFRP; composites; fatigue; fracture mechanics; modelling; service-life;

1. Introduction

The cracking and delamination found in US Navy [1] and RAAF [2] F/A-18 inner-wing lap-splice joints (IWSLJ) has focused attention on fatigue crack growth (FCG) in adhesively-bonded joints. As noted in the US Defence Department's Composite Materials Handbook CMH-17-3G [3], much of the methodology currently used in the design

* Corresponding author. Tel.: +61-399053809
E-mail address: rhys.jones@monash.edu

10.1016/j.prostr.2020.10.044

Nomenclature

a	total crack length
A	constant in the Hartman-Schijve equation
CFRP	carbon-fibre reinforced-plastic
da/dN	rate of fatigue crack growth per cycle
D	intercept in the Hartman-Schijve crack-growth equation
F_{max}	maximum load applied during the fatigue test
F_{min}	minimum load applied during the fatigue test
FCG	fatigue crack growth
G	energy release-rate
G_{co}	quasi-static value of the interlaminar fracture energy at the onset of crack growth
G_{max}	maximum value of the applied energy release-rate in the fatigue cycle
G_{min}	minimum value of the applied energy release-rate in the fatigue cycle
ΔG	range of the applied energy release-rate in the fatigue cycle, as defined below
ΔG	$= G_{max} - G_{min}$
$\Delta\sqrt{G}$	range of the applied energy release-rate in the fatigue cycle, as defined below
$\Delta\sqrt{G}$	$= \sqrt{G_{max}} - \sqrt{G_{min}}$
$\Delta\sqrt{G_{th}}$	the value of $\Delta\sqrt{G}$ corresponding to a FCG rate, $da/dN = 10^{-10}$ m/cycle
$\Delta\sqrt{G_{thr}}$	range of the fatigue threshold value of $\Delta\sqrt{G}$
n	exponent in the Hartman-Schijve crack-growth equation
N	number of fatigue cycles
R	stress ratio ($=F_{min}/F_{max}$)
R^2	coefficient of determination
USAF	United States Air Force
RAAF	Royal Australian Air Force

and analysis of bonded joints in airframes is based on tools validated as part of the USAF 'Primary Adhesively Bonded Structure Technology (PABST)' programme [4]. One of the primary recommendations contained in [4] was that, to avoid durability issues, the adhesive should not be loaded beyond yield. This recommendation is reflected in MIL-STD-1530D Section 5.2.4 [5] which 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."

However, the belief that an adhesively-bonded joint that meets static-strength requirements will also meet durability requirements was invalidated by the problems associated with the boron-fibre/epoxy composite doubler on the upper surface of the F-111 aircraft when in service with the RAAF [6]. In this case the composite doublers, which were bonded to the F-111 steel wing-pivot fitting using an epoxy-film adhesive, were approximately 120 plies thick and carried approximately 30% of the load in the critical section of the wing-pivot fitting [6,7]. (Doublers were fitted to 39 wings and each wing had two boron-fibre/epoxy composite doublers. This resulted in a total of 78 adhesively-bonded boron-fibre/epoxy composite doublers being installed on the RAAF F-111 fleet [6].) Even though the adhesively-bonded composite doublers passed cold proof-load testing (CPLT) at -40ºF, subsequently there was extensive cracking/delamination in under 1000 airframe flight hours [6]. These cracks initiated in the adhesive under the cyclic-fatigue loadings encountered during flight at the outboard edge of the bonded doubler and, after some inboard growth, moved in and delaminated the first ply of the boron-fibre/epoxy composite doubler. Of the fleet of twenty aircraft to which doublers were fitted cracks/delaminations were detected in seven wings at between 729 and 1233 airframe flight hours. In general, when detected, the cracks/delaminations tended to be quite large, often extending from the outboard edge of the bonded doubler for a distance of approximately 230 mm.

The F-111 adhesively-bonded doubler programme highlighted the importance of designing to the fatigue threshold(s) associated with the inherent initial flaws and the associated stress states, and the need to be able to conservatively predict crack and delamination growth, so as to establish the necessary inspection intervals. Furthermore, the in-service growth of small sub-millimetre defects in adhesively-bonded joints is not unique to the F-111 composite doubler and has also been observed on adhesively-bonded composite doublers applied to Canadian CF-5 [8] and AIRBUS A310 [9] aircraft.

The importance of working to the correct fatigue threshold, for a given set of stresses and initial flaws, has also been highlighted by Schoen, Nyman, Blom and Ansell [10], who stated:

"During certification of the AIRBUS A320 vertical fin, no delamination growth was detected during static loading. The following fatigue loading of the same component had to be interrupted due to large delamination growth. … This demonstrates the importance of using the threshold value instead of the static value for delamination growth in the design of composite structures."

Due to the low stress levels seen in the majority of adhesively-bonded structures there are relatively few instances where there has been in-service crack growth. Nevertheless, a range of examples can be found in [1,6,8,9,11-14], thereby underlining the need for considering delamination growth induced by fatigue loading as well as of delamination due to static loading. Indeed, the disbonding crack and delamination growth in the inner wing step lap joint of an F/A-18 aircraft [1] is an excellent example of this class of problems.

In this context it should be noted that the growth of small naturally-occurring cracks in a structural epoxy-film adhesive was also highlighted [14]. In this instance small cracks in the adhesive, that were less than approximately 0.01 mm in length, initiated and propagated in the adhesive layer from the interface between the adhesive and the aluminium-alloy. This finding, when taken in conjunction with fleet experience and the examples presented in [13, 14], led [14] to conclude that the value of the fatigue threshold for naturally-occurring cracks in adhesives can be very low.

The above examples reveal that, for existing designs, from a sustainment perspective the ability to perform a slow growth assessment of crack growth in operational aircraft can be essential. In this context, a slow crack growth approach to certifying adhesively-bonded structures was introduced in the 2009 US Federal Aviation Administration (FAA) Airworthiness Advisory Circular [15]. It is also contained as part of the JSSG-2006 guidelines [16], as well as in MIL-STD-1530D [5].

Thus, from all the above work and statements, it is clear that a methodology for determining the 'worst-case' FCG curve for structural adhesives is an important requirement (a) for developing conservative designs and (b) for modelling the life and inspection intervals of adhesively-bonded components and structures. As such, determination of the `worst case' FCG is a focal point of the present paper. In this context, recent papers [17,18] have considered the fatigue of carbon-fibre reinforced-plastic (CFRP) composites and proposed a novel methodology, based on the Hartman-Schijve equation [19], which is a variant of the NASGRO equation, to determine a valid 'upper-bound' FCG curve. Such an 'upper-bound' FCG curve may be thought as a 'worst-case' curve and represents a material-allowable property and, in the present work, aims to account for the experimental variability that is observed when adhesively-bonded joints are subjected to cyclic-fatigue loading.

It has been shown [2] that the Hartman-Schijve equation accurately predicts delamination growth of in a double cantilever beam tested under variable amplitude fatigue loads. Consequently, an aim of the present paper is to investigate FCG in adhesively-bonded joints tested under an industry standard combat aircraft flight load spectrum (FALSTAFF). (FALSTAFF is an industry-standard fighter-aircraft flight-load fatigue-cycle spectrum. One load block represents 200 airframe flight hours [20].)

2. THEORETICAL BACKGROUND

Recent work has revealed [14, 17,18,21-23] that the logical extension of the Paris FCG equation for metals to crack growth in adhesive joints is, in fact, to express da/dN as a function of $\Delta\sqrt{G}$, or $\sqrt{G_{max}}$, rather than ΔG, or G_{max}. Where $\Delta\sqrt{G}$ is given by;

$$\Delta\sqrt{G} = \sqrt{G_{max}} - \sqrt{G_{min}} \quad (1)$$

This has led [14, 17,18,21,22] to the development of the Hartman-Schijve crack growth equation

$$\frac{da}{dN} = D\left[\frac{\Delta\sqrt{G} - \Delta\sqrt{G_{thr}}}{\sqrt{\{1-\sqrt{G_{max}}/\sqrt{A}\}}}\right]^n \quad (2)$$

where *D*, *n* are constants, *A* is the cyclic toughness, and $\Delta\sqrt{G_{thr}}$ is fatigue threshold. Now, for structural adhesives, it is often found from experimental tests [e.g. 24-30] that a clearly defined threshold value exists, below which little fatigue crack growth occurs. In this case the value of the threshold, $\Delta\sqrt{G_{thr}}$, may be taken to be the experimentally-determined value. If this is not the case, then the concepts described in the ASTM standard [31], which are widely used by the metals community, may be employed. This standard defines a threshold value which, in the above terminology, may be taken to be the value of $\Delta\sqrt{G_{th}}$at a value of da/dN of 10^{-10} m/cycle. This is termed $\Delta\sqrt{G_{th}}$ and hence, by rearrangement of Equation (1), the value of $\Delta\sqrt{G_{thr}}$ is given by:

$$10^{-10} = D\left[\frac{\Delta\sqrt{G_{th}} - \Delta\sqrt{G_{thr}}}{\sqrt{\{1-\sqrt{G_{max}}/\sqrt{A}\}}}\right]^n \quad (3)$$

with the experimental data having *G* with units of J/m^2 and *da/dN* with units of m/cycle [18]. The review paper by Jones, Kinloch, et al [18] explains how this formulation can be used to determine a valid, 'upper-bound' (i.e. 'worst-case') FCG rate curve which takes account of the material, specimen and test variability that gives rise to the observed experimental scatter; and is therefore valid for material characterisation, material comparisons, and design and lifing studies. To achieve this it was suggested [14,17,18] that the best methodology is to first to use the Hartman-Schijve variant of the Nasgro equation, e.g. Equation (1). The next step is the adoption of the statistical approach suggested in [32,33], i.e. by plotting 'upper-bound' curves obtained from using values of $\Delta\sqrt{G_{thr}}$ corresponding to the mean value of $\Delta\sqrt{G_{thr}}$ minus two standard deviations, or the mean value minus three standard deviations. Of course, for a normal distribution the mean minus two standard deviations is equivalent to a 95% confidence estimate, and a mean minus three standard deviations curve is equivalent to a 99.7% estimate. This suggestion finds support from the work of Niu [32] and Rouchon [33] and they have reviewed the statistical procedures used to derive material allowable properties to input into design and lifing analyses for aerospace applications, under the basic headings of 'A' and 'B':

'A' basis: The mechanical property value indicated is the value above which at least 99% of the population of values is expected to fall with the confidence of 95%. This value is used to design and lifing a single member where the loading is such that its failure would result in a loss of structural integrity.

'B' basis: The mechanical property value indicated is the value above which at least 90% of the population of values is expected to fall with the confidence of 95%. This value is used in the design and lifing of redundant or fail-safe structural analyses, where the loads may be safely distributed to other members.

Obviously the ‘A’ basis is essentially equivalent to the idea of taking the mean value minus three standard deviations, and the ‘B’ basis to the idea of the mean value minus two standard deviations.

3. The effect of variability of the threshold term $\Delta\sqrt{G_{thr}}$ on the predicted fatigue life

The effect of the variability of the threshold term $\Delta\sqrt{G_{thr}}$ on fatigue crack growth was first studied in [34] which analysed the double-overlap fatigue-specimens (DOFSs) tested by Cheuk and et al. [35]. The geometry of the DOFS specimen analysed is shown schematically in Figure 1. The DOFS was symmetric and the width of the specimen was 20 mm. The inner and outer substrates were an aluminium-alloy (Grade 2024-T3) and were bonded using a rubber-toughened epoxy-film adhesive (Cytec, USA, ‘FM73’). The fatigue crack was observed to grow cohesively through the adhesive layer from the end of the bonded overlap via naturally-occurring defects which were present in the adhesive layer. The inner aluminium-alloy substrate was 400 mm long and 6.4 mm thick. The outer aluminium-alloy substrate was 200 mm long and was 3.05 mm thick. The layer of the epoxy-film adhesive was 0.4 mm thick. In the experiments reported in [35] the test frequency was 3 Hz and a simple sinusoidal fatigue-cycle spectrum was employed. The maximum stress in the aluminium-alloy remote from the joint was 193.5 MPa and the R-ratio was 0.0. The modulus and Poisson’s ratio associated with the aluminium-alloy substrates and the ‘FM73’ adhesive are shown in Table 1 [35]. The paper by Hu et al. [34] revealed that, for the rubber-toughened epoxy-film ‘FM73’adhesive, the values of the Hartman-Schijve constants were are as given in Tables 2 and 3.

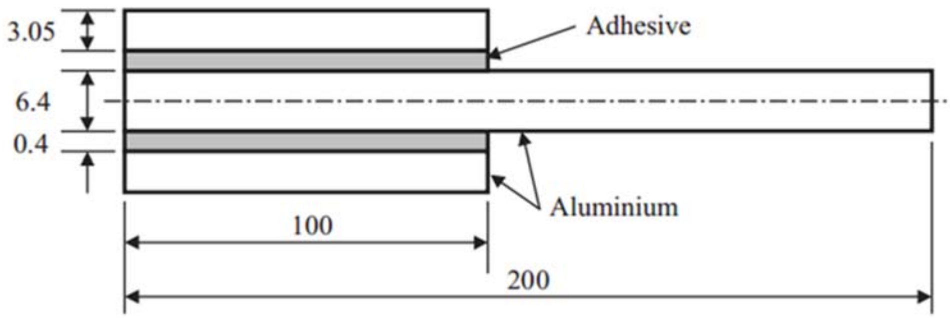

Fig. 1 Schematic diagram of the geometry of the cylindrical test specimen, from [35].

Table 1. Mechanical properties of the aluminium-alloy and epoxy-film adhesive given in [35].

Properties/Units	Aluminium 2024-T3	FM73 adhesive
Elastic modulus, E (GPa)	72	2.295
Poison’s ratio, ν	0.33	0.35

Table 2. Values of the Hartman-Schijve constants for the ‘FM73’ adhesive given in [35].

D (m/cycle)	*n*	*A* (J/m^2)	$\Delta\sqrt{G_{thr}}$) (√(J/m2))
1.9 x 10^{-10}	2.7	2000	7.1

Table 3. Values of the Hartman-Schijve constants for the 'FM73' adhesive given in [34].

Mean (√(J/m^2))	7.1
Standard deviation, σ (√(J/m^2))	0.5
Mean - 3 standard deviations (√(J/m^2))	5.6

For the Hartman-Schijve equation to be used to predict crack growth the relationship between G and the crack length, a, for the DOFS was needed. This relationship, which was given by Cheuk et al in [35] and subsequently validated by Hu et al in [34], is shown in Figure 2.

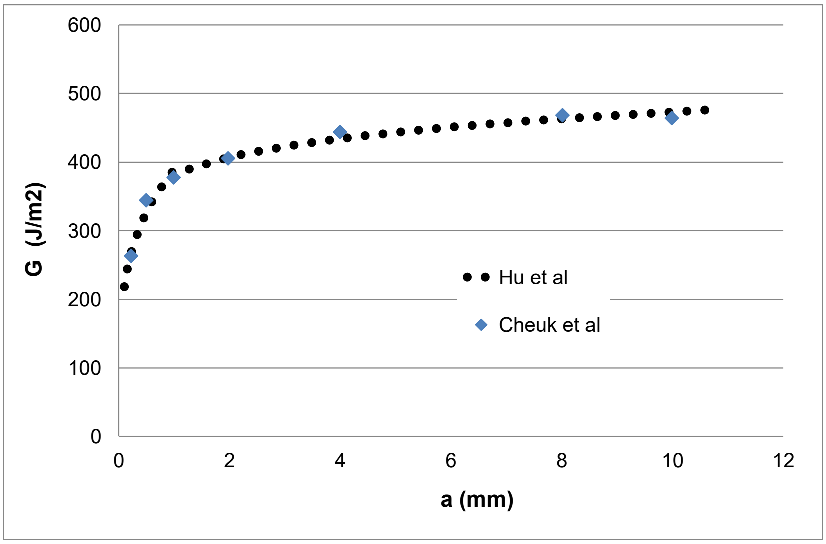

Fig. 2. The calculated values of the ERR, G, as a function of the crack length, a, for the DOFS, from [34].

The tests performed in [35] were intended to mimic the behaviour of an adhesively-bonded composite doubler adhesively-bonded onto a 3.2 mm thick wing-skin. As explained in [13], testing a specimen based upon a 3.2 mm skin with a single adhesively-bonded doubler only on one side results in secondary bending effects that are not seen by a doubler bonded onto an operational aircraft. The DOFS test, which uses an inner adherend of twice the thickness of the wing skin, with doublers adhesively-bonded on either side, was specifically developed to overcome this shortcoming [13,36]. In the tests reported in [35] the remote maximum stress in the aluminium-alloy remote from the joint was 193.5 MPa. The resultant measured and computed crack length histories, from an initial crack length of 0.05 mm, are shown in Figure 3. This stress level represents the approximate, very highest, maximum stress that is likely to be seen in a real 3.2 mm thick wing-skin. To investigate the effect of a somewhat more representative stress, the analysis was repeated but for a more realistic maximum remote stress of 134 MPa [37]. This stress level corresponds to the maximum stress seen at the control point, FCA352, in a 2.6 mm thick wing skin of a P3C Orion aircraft. (Fatigue critical location FCA352 corresponds to the fairing-dome nut holes in the lower surface panels at the inboard nacelle [37].) The resultant predicted crack length, a, versus number, N, of cycles curves are shown in Figure 4, where the initial crack length is again taken to be 0.05 mm [37]. Several noteworthy points may be seen. Firstly, from comparing the predictions shown in Figure 3 and 4, it may be seen that the cracks in the DOFSs are predicted to grow at a significantly slower FCG rate when a maximum stress of 134 MPa is used as opposed 193.5 MPa. This observation holds irrespective of the value of $\Delta\sqrt{G_{thr}}$ used in the analysis, as indeed would be expected. Secondly, it would appear that the effect of varying the value of $\Delta\sqrt{G_{thr}}$ is of more significance at the lower maximum stress of 134 MPa (see Figure 4), compared to when a stress of 193.5 MPa (see Figure 3) is employed in the predictions. Thirdly, the results illustrated in Table 4, taken from Figure 4, reveal that the differences in the fatigue lifetimes associated with achieving a given crack length as predicted when using the mean (i.e. 7.1 √(J/m^2) or the 'mean - 3 σ ' (i.e. 5.6

√(J/m²) values of $\Delta\sqrt{G_{thr}}$ ranges from approximately 160% for a 1 mm long crack to approximately 70% for a 30 mm long crack.

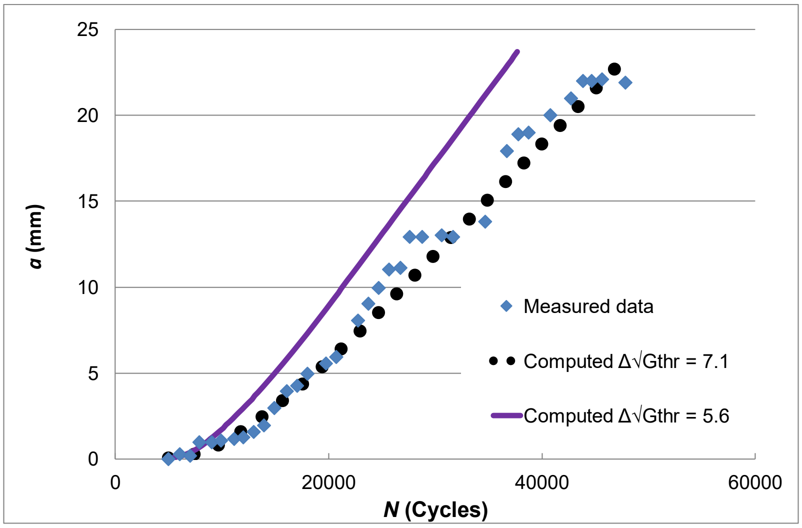

Fig. 3. The measured and computed crack length, a, histories as a function of the number, N, of fatigue cycles for the DOFS corresponding to a remote maximum stress of 193.5 MPa under a simple sinusoidal fatigue-cycle spectrum.

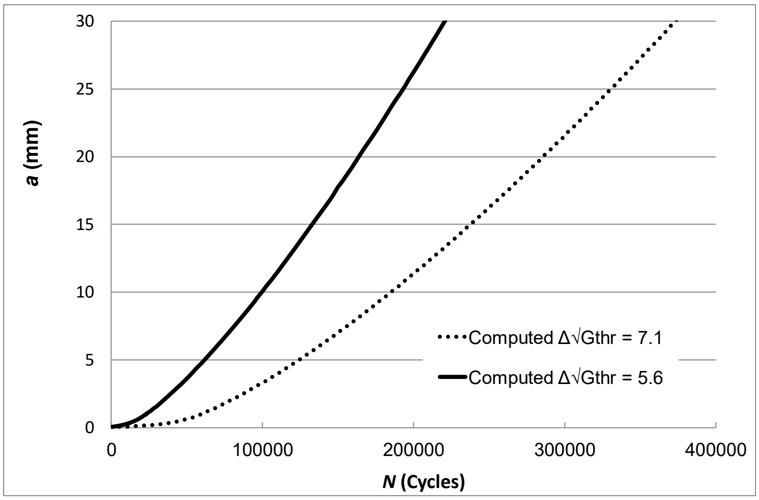

Fig. 4. The measured and computed crack length, a, histories as a function of the number, N, of fatigue cycles for the DOFS corresponding to a remote maximum stress of 134 MPa under a simple sinusoidal fatigue-cycle spectrum.

Table 4. Crack growth histories, from an initial crack length of 0.05 mm, associated with the curves shown in Figure 4.

$\Delta\sqrt{G_{Ithr}}$ (√(J/m²))	Number, *N*, of cycles needed to attain a given crack length of:					
	1 mm	3 mm	5 mm	10 mm	20 mm	30 mm
5.6	23,000	44,050	61,650	99,650	164,000	220,750
7.1	59,500	95,100	124,100	185,240	285,240	373,100

A current concern is related to predicting the service-life of an adhesively-bonded component containing a crack of a length that can be reliably detected by non-destructive inspection (NDI). The examples presented in [13] suggest that it may not be possible to reliably detect cracks in an adhesive joint beneath a crack length of approximately 3mm.

Consequently, Figure 5 presents the crack growth histories shown in Figure 4 but now re-plotted for an initial crack length of 3 mm. As in Figure 4, the results shown in Figure 5 reveal that a significant difference exists for the relationships between the crack length, a, versus the number, N, of cycles depending on whether the mean (i.e. 7.1 √(J/m^2) or the "mean - 3σ" (i.e. 5.6 √(J/m^2) values of $\Delta\sqrt{G_{thr}}$ are employed for the predictions. Since the inspection intervals needed to ensure continued airworthiness depends on the crack length history, this difference has potential implications on the necessary inspection intervals.

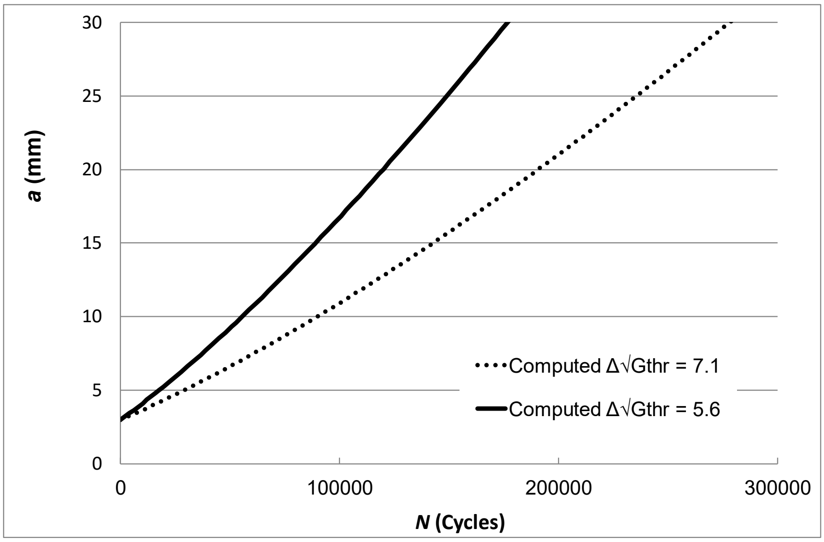

Fig. 5. The measured and computed crack length, a, histories as a function of the number, N, of fatigue cycles for a remote maximum stress of 134 MPa under a simple sinusoidal fatigue-cycle spectrum. (The initial crack length was taken to be 3 mm.)

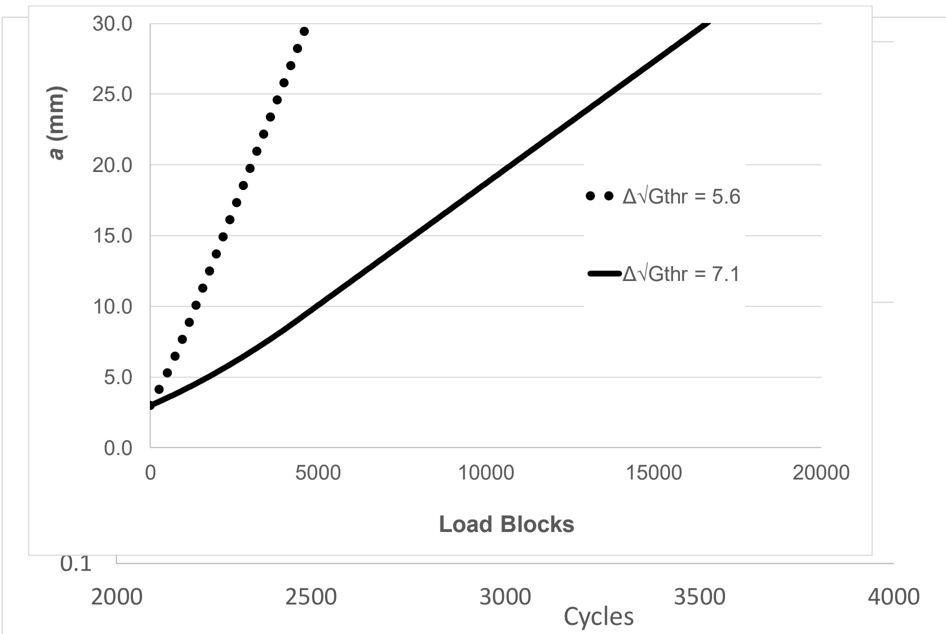

Fig. 6. Predicted crack length, a, histories as a function of the number of load blocks for the DOFS under a FALSTAFF flight-load spectrum with a remote maximum stress of 134 MPa.

To continue this study we next investigated the growth of a 3 mm crack in the DOFS shown in Figure 1 subjected to a FALSTAFF flight-load fatigue-cycle spectrum with a maximum remote stress of either 134 MPa or 193.5 MPa. (FALSTAFF is an industry-standard fighter-aircraft flight-load fatigue-cycle spectrum. One load block represents 200 airframe flight hours [20].) To illustrate the importance of using a 'worst case' (i.e. 'mean - 3σ') value of the threshold, $\Delta\sqrt{G_{thr}}$ an analysis was performed using both the mean (i.e. 7.1 $\sqrt{(J/m^2)}$ and the 'mean - 3σ' (i.e. 5.6 $\sqrt{(J/m^2)}$ values of $\Delta\sqrt{G_{thr}}$. The resultant predicted histories of the crack length, a, versus the number of load blocks for peak stresses of 134 MPa and 193.5 MPa are shown in Figures 6 and 7, respectively. Firstly, again but now using the FLASTAFF loading, it may be seen that the cracks in the DOFSs are predicted to grow at a significantly slower FCG rate when the spectrum has a maximum stress of 134 MPa as opposed to a maximum stress of 193.5 MPa. Indeed, as would be expected this is true irrespective of the value of $\Delta\sqrt{G_{thr}}$that was employed in the analysis. Secondly, the predictions shown in these Figures reveal that, as a result of the large number of small amplitude load cycles in the FALSTAFF flight-load spectrum, the FCG rate, and hence 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 $\Delta\sqrt{G_{thr}}$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.

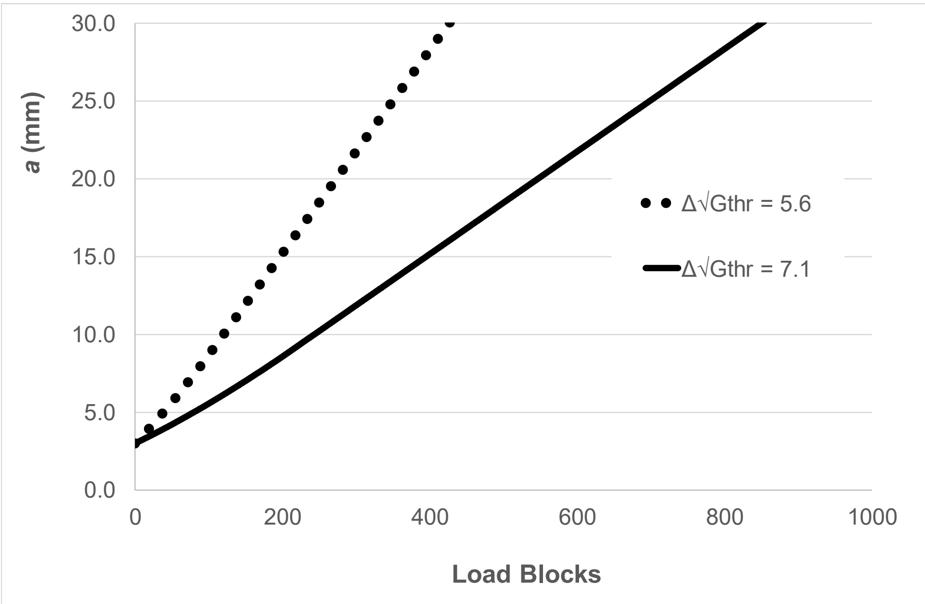

Fig. 7. Predicted crack length, a, histories as a function of the number of load blocks for the DOFS under a FALSTAFF flight-load spectrum with a remote maximum stress of 193.5 MPa.

4. Conclusions

It has been shown that a methodology is needed for estimating a valid 'upper-bound' curve capable of encompassing all the experimental data and providing a conservative, 'worst-case' FCG curve for adhesives when subjected to cyclic-fatigue loading. Such a valid, 'upper-bound' curve can then employed for (a) the characterisation and comparison of adhesive materials, (b) a 'no growth' design, (c) for assessing if a crack, that is found in an in-service aircraft, will grow, and (d) the design and lifing of in-service adhesively-bonded aircraft structures where material allowable properties have to be inputted into a FCG analysis. (Although the problem studied is associated with cracking in an adhesively bonded joint it is believed that the conclusions should also apply to delamination growth in a composite airframe.)

A novel methodology, based on using the Hartman-Schijve approach, has been proposed to access this valid, 'upper-bound' FCG rate curve, which may be thought of as a material allowable property It has been firstly demonstrated that the variability seen in the cyclic-fatigue results from fracture-mechanics tests for different adhesives may be taken into account by considering the experimentally-measured values of $\Delta\sqrt{G_{thr}}$, which is the range of the fatigue threshold value of $\Delta\sqrt{G}$, since the values of $\Delta\sqrt{G_{thr}}$ do vary for replicate test specimens. Secondly, the mean value of $\Delta\sqrt{G_{thr}}$ minus three standard deviations, σ, has been used to deduce an 'upper-bound' FCG rate curve which encompasses all the experimental variability observed in the FCG rate data for the different adhesives studied in the present work.

Finally, this proposed methodology has been applied to predict the FCG rate curves for the growth of a crack, a, versus the number, N, of fatigue cycles for double-overlap fatigue-specimens (DOFSs) subjected to a simple, sinusoidal fatigue-cycle spectrum or an industry-standard combat-aircraft flight-load fatigue-cycle spectrum. For the former simple fatigue-cycle spectrum, the predictions based upon using the 'mean - 3σ' value of $\Delta\sqrt{G_{thr}}$ did indeed give a 'worst-case', more conservative, prediction which encompassed all the variability observed in the experimental results. When predictions using the combat-aircraft flight-load fatigue spectrum were undertaken the results revealed that, as a result of the large number of relatively small amplitude load cycles in this flight-load spectrum, the FCG rate, and hence the service-life, of the adhesively-bonded joint was a relatively strong function of the value of the fatigue threshold that was employed in the analyses. This reinforces the need to determine the statistically-valid value of the $\Delta\sqrt{G_{thr}}$ for the fatigue threshold and hence to determine an 'upper-bound' FCG rate curve where the variability of the FCG rate for the adhesive is taken into account.

Acknowledgments

Rhys Jones acknowledges support via the Office of Naval Research NICOP Grant N62909-19-1-2011-P00001. John Michopoulos acknowledges support for this work by the Office of Naval Research (ONR) through the Naval Research Laboratory's core funding.

References

1. Mueller E.M., Starnes S., Strickland N., Kenny P., Williams C., The detection, inspection, and failure analysis of a composite wing skin defect on a tactical aircraft, Composite Structures, 145, 186-193, 2016.
2. Jones R., Kinloch A.J., Michopoulos J., Iliopoulos AP., Phan N., Goel K., Lua J., Singh Raman R.K., Peng D., Assessing failure and delamination growth in composites and bonded joints under variable amplitude loads, Proceedings Twenty-Second International Conference On Composite Materials (ICCM22), 11th-16th August, 2019, Melbourne, Australia. , Editor R. Das, pp 401-410, ISBN: 978-1-922016-65-2.
3. CMH-17-3G, Composite Materials Handbook, Volume 3: Polymer Matrix Composites Materials Useage, Design and Analysis, Published by SAE International, March 2012.
4. Potter D.L., Primary Adhesively Bonded Structure Technology (PABST), Design Handbook for Adhesive Bonding, Technical Report AFFDL-79-3129, Final Report, 15 March 1977 - 14 January 1979.
5. MIL-STD-1530D, 2016, Department of Defense Standard Practice, Washington, DC, USA, 2016.
6. Molent L., Jones R., The F111C wing pivot fitting repair and implications for the design/assessment of bonded joints and composite repairs, in Aircraft Sustainment and Repair, Edited by R. Jones, A.A. Baker, N. Matthews and V. Champagne Jr., Elsevier Butterworth-Heinemann Press, pp. 511-543, 2018.
7. Molent L., Callinan R.J., Jones R., Design of an all boron epoxy doubler for the F-111C wing pivot fitting: Structural Aspects, Composite Structures, 11, 57-83, 1989.
8. Raizenne D., Case History: CF116 Upper wing skin fatigue enhancement boron doubler, Advances in the Bonded Composite Repair of Metallic Aircraft Structure, Edited by A.A. Baker, L.R.F. Rose and R. Jones, Elsevier Applied Science Publishers, 2002.
9. Loss Of Rudder In Flight Air Transat Airbus A310-308 C-GPAT, Miami, Florida, 90 nm S, 6 March 2005, Transportation Safety Board of Canada, Report Number A05F0047, 2005.
10. Schoen J., Nyman T., Blom A., Ansell H., A numerical and experimental investigation of delamination behaviour in the DCB specimen,

Composites Science and Technology, 60, 173-184, 2000.
11. Baker AA., Structural Health Monitoring of a Bonded Composite Patch Repair on a Fatigue-Cracked F-111C Wing, DSTO-RR-0335, March 2008.
12. Hu W., Barter S., Wang J., Jones R., Kinloch A.J., On the USAF 'risk of failure' approach and its applicability to composite repairs to metal airframes, Composite Structures, 167, 103-111, 2017.
13. Jones R., Hui D., Analysis, design and assessment of composite repairs to operational aircraft, in Aircraft Sustainment and Repair, Edited by R. Jones, N. Matthews, A.A. Baker and V. Champagne Jr., Butterworth-Heinemann Press, pp. 325-456, 2018.
14. Jones R, Peng D, Michopoulos JG, Kinloch AJ. Requirements and Variability Affecting the Durability of Bonded Joints, Materials. 2020 Mar 23;13(6). doi:10.3390/ma13061468
15. Federal Aviation Authority. Airworthiness Advisory Circular No: 20-107B. Composite Aircraft Structure, 09/08/2009.
16. Department of Defense Joint Service Specification Guide, Aircraft Structures, JSSG-2006.
17. Jones R., Kinloch, A.J., Michopoulos, J.G., Brunner A.J., Phan, N., Delamination growth in polymer-matrix fibre composites and the use of fracture mechanics data for material characterisation and life prediction, Composite Structures, 180, 316-333, 2017.
18. Yao L., Alderliesten R., Jones R., Kinloch A.J., Delamination fatigue growth in polymer-matrix fibre composites: A methodology for determining the design and lifing allowables, Composite Structures, 96, 8-20, 2018.
19. Hartman A., Schijve J., The effects of environment and load frequency on the crack propagation law for macro fatigue crack growth in aluminium alloys, Engineering Fracture Mechanics, 1, 615-631,1970.
20. Van Dijk G., de Jonge J.B., Introduction of a Fighter Aircraft Loading Standard for Fatigue Evaluation (FALSTAFF), Report NLR MP 75017, National Aerospace Laboratory NLR, The Netherlands, 1975.
21. Jones R. Kinloch A.J., Hu W., Cyclic-fatigue crack growth in composite and adhesively-bonded structures: the FAA slow crack growth approach to certification and the problem of similitude, International Journal of Fatigue, 88, 10-18, 2016.
22. Jones R., Hu W., Kinloch A.J., A convenient way to represent fatigue crack growth in structural adhesives, Fatigue and Fracture of Engineering Materials and Structures, 38, 4, 379–391, 2015.
23. Rans C., Alderliesten R., Benedictus R., Misinterpreting the results: How similitude can improve our understanding of fatigue delamination growth, Composites Science and Technology, 71, 230–238, 2011.
24. Jethwa, J.K., Kinloch, A.J., The fatigue and durability behaviour of automotive adhesives, Part 1: Fracture mechanics tests. Journal of Adhesion, 61, 71-95, 1997.
25. Curley A.J., Hadavinia H., Kinloch A.J., Taylor A.C., (2000) Predicting the service-life of adhesively-bonded joints, International Journal of Fracture, 103, 41-70, 2000.
26. Kinloch A.J., Little M.S.G., Watts J.F., The role of the interphase in the environmental failure of adhesive joints, Acta Materialia, 48, 4543-4553, 2000.
27. Azari, S., Papini, M., Schroeder, J., Spelt, J.K., Fatigue threshold behaviour of adhesive joints, International Journal of Adhesion and Adhesives, 30, 145-159, 2010.
28. Ashcroft I., Shaw S.J., Mode I fracture of epoxy bonded composite joints 2. Fatigue loading, International Journal of Adhesion and Adhesives, 22, 151-167, 2002.
29. Mall S., Ramamurthy G., Rezaizdeh M.A., Stress ratio effect on cyclic debonding in adhesively bonded joints, Composite Structures, 8, 31-45, 1987.
30. Kinloch A.J., Adhesives in engineering, Proceedings of the Institution of Mechanical Engineers, Part G, 211, 307-334, 1997.
31. ASTM, Measurement of fatigue crack growth rates. ASTM E647-13a, USA, 2013.
32. Niu M.C.Y., Composite Airframe Structures: Practical Design Information and Data, Conmilit Press, Hong Kong, 1992.
33. Rouchon J., Fatigue and damage tolerance evaluation of structures: the composite materials response, 22nd Plantema Memorial Lecture, 25th ICAF Symposium, Hamburg, Germany, Rotterdam, The Netherlands, National Aerospace Laboratory NLR, NLR-TP-2009-221, 2009
34. Hu W., Jones R., Kinloch A.J., Computing the growth of naturally-occurring cracks in adhesively-bonded patches to metallic structures, Engineering Fracture Mechanics, 152, 162-173, 2016.
35. Cheuk P.T., Tong L., Rider A.N., Wang J., Analysis of energy release rate for fatigue cracked metal-to-metal double-lap shear joints, International Journal of Adhesion and Adhesives 25 181-191, 2005.
36. Baker A.A., Crack patching: experimental studies, practical applications, in Bonded Repair of Aircraft Structure, Edited by A.A. Baker and R. Jones, Martinus Nijhoff Publishers, The Hague, pp. 107-173, 1988.
37. Lo M., Jones R., Bowler A., Dorman M., Edwards D., Crack growth at fastener holes containing intergranular cracking, Fatigue and Fracture of Engineering Materials and Structures, 40, 1664-1675, 2017.