Composite Structures 99 (2013) 375–379

Contents lists available at SciVerse ScienceDirect

Composite Structures

journal homepage: www.elsevier.com/locate/compstruct

Fatigue crack growth in nano-composites

R. Jones [a,*], S. Pitt [a], D. Hui [b], A. Brunner [c]

[a] Centre of Expertise in Structural Mechanics, Department of Mechanical and Aerospace Engineering, Monash University, P.O. Box 31, Monash University, Victoria 3800, Australia
[b] Composite Material Research Laboratory, Department of Mechanical Engineering, University of New Orleans, 2000 Lakeshore Drive, New Orleans, LA 70148, USA
[c] Mechanical Systems Engineering, Empa, Swiss Federal Laboratories for Materials Science and Technology, CH-8600 Duebendorf, Switzerland

ARTICLE INFO

Article history:
Available online 2 December 2012

Keywords:
Nano-composites
Fatigue
Hartman–Schijve
Carbon nano-tubes

ABSTRACT

This paper discusses the potential of a variant of the Hartman–Schijve equation to represent fatigue crack growth in a range of nano-composites. It is found that when expressed in this form the exponent of this variant is approximately 2 and, as such, is considerably lower than the exponent in "Paris like" power law representations for delamination growth in composites. As such we see that, in these examples, the present variant of the Hartman–Schijve representation of delamination growth in nano-composites is similar to that seen for crack growth in metals, delamination growth in composites and the environmental degradation of adhesive bonds. This suggests that the present formulation may be useful for the damage tolerant assessment of small naturally occurring defects in nano-composite structures and for the ranking of nano-composites.

1. Introduction

It has long been suggested [1–8] that for the growth of both long and short cracks in metals the crack growth rate da/dN can often be linearly related to $(\Delta K–\Delta K_{th})^{\alpha}$, where α is approximately 2. The paper by Jones et al. [7] has shown that Modes I and II delamination growth in composites can often be accurately described by a variant of the Hartman–Schijve [1] equation, viz:

$$da/dN = D\left(\left(\Delta\sqrt{G} - \Delta\sqrt{G_{th}}\right)\Big/\sqrt{\left(1 - \sqrt{G_{max}}/\sqrt{G_{cy}}\right)}\right)^{\beta} \tag{1}$$

where β is approximately 2 and the terms $\Delta\sqrt{G_{th}}$ and $\sqrt{G_{cy}}$ are perhaps best viewed as parameters that are used to ensure that the entire range of data fits the equation. Jones et al. [7] also revealed that for those problems that can also be modeled using crack closure based approaches [9] such that there exist a function $U(R)$, which is a function of the R ratio, whereby we can define a function $\Delta\sqrt{G_{eff}}$:

$$\Delta\sqrt{G_{eff}}\left(= \sqrt{G_{max}} - \sqrt{G_{op}}\right) = U(R)\Delta\sqrt{G} \tag{2}$$

such that the various R ratio dependent da/dN versus $\Delta\sqrt{G}$ curves reduce to a single da/dN versus $\Delta\sqrt{G_{eff}}$ curve which is (essentially) coincident with the high R ratio da/dN versus $\Delta\sqrt{G}$ curve(s) for the material, see [7], then delamination growth could also be expressed in the form: an alternative relationship, viz:

$$da/dN = H\left(\left(\Delta\sqrt{G_{eff}} - \Delta\sqrt{G_{eff,th}}\right)\Big/(1 - G_{max}/A)\right)^{\beta} \tag{3}$$

Here G_{op} is the value of G at which the delamination first opens and $\Delta\sqrt{G_{eff,th}}$ is the associated effective threshold. Indeed, Pitt et al. [10] have shown how variants of Eqs. (1) and (3) can also be used to describe the effect of the environment on delamination growth in ASTM standard wedge tests.

With this in mind let us now address the question of whether nano-composites offer the potential to enhance the damage tolerance of aerospace composite structures. In this context it should be recalled that composite materials are now widely used in aircraft structural components. Australia's current lead fighter the F/A-18 has approximately 39% of its external wetted surface fabricated from advanced composites. At Boeing, the state-of-the-art 787 passenger aircraft has approximately 50% composite parts on its major components. In the Joint Strike Fighter (JSF), a "next generation" aircraft, a collaboration involving aerospace companies in the USA, the UK, Australia, Canada, etc., advanced composites make up approximately 50% of the aircraft. However, the primary structural members in these aircraft are still metallic. A similar situation arises in the civilian sphere where skins, stabilators, fins, control surfaces, etc., where the strain level in these components are low, now make extensive use of composites. However, for certification purposes, current designs are such that any delamination will not grow. This approach places an artificial limit on the use of composite materials and the components that can be lightened through their use. It also requires an accurate knowledge of the fatigue threshold associated with the particular damage state.

* Corresponding author. Tel.: +61 3 99053809; fax: +61 3 99051825.
E-mail address: rhys.jones@monash.edu (R. Jones).

0263-8223/$ - see front matter © 2012 Published by Elsevier Ltd.
http://dx.doi.org/10.1016/j.compstruct.2012.09.050

In this context the boron epoxy doubler on the upper surface of F-111 aircraft is a good example of an in-service delamination problem associated with a major load bearing component [11]. In this case the doublers were approximately 120 plies thick and took approximately 30% of the load in the critical section of the wing pivot fitting [12]. Even though the doublers passed cold proof load (CPLT) testing, small defects, which were thought to be less than 0.1 mm in size and hence were undetectable, led to extensive delamination/disbonding in under 1000 flight hours [11]. At this stage it should be stressed that the doublers met the RAAF's initial design requirement, viz: to ensure that the doublers and the wings did not fail during CPLT, and that up till the retirement of the F-111C wings there was no further cracking in the critical location post doubler installation. Nevertheless, lessons learnt from this program were 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 predict delamination growth so as to establish the necessary inspection intervals. The in-service delamination growth of small sub mm defects in composites and adhesively bonded joints is not unique to the F-111 and was also observed on the boron epoxy repair to the Canadian CF-5 aircraft [13] as well as on the Canadian CF-5 and the F/A-18 full scale (wing torsion box) fatigue tests [14].

The growth of such small defects under operational load spectra also implies that the fatigue threshold was very low and as such mirrors the findings presented in [15] where it was concluded that under operational loading the fatigue threshold for metallic structures is very low and that crack growth can be assumed to commence on day one.

Given the failure during static testing of the Boeing 787 wing to fuselage joint and the in-service delaminations discussed above the need to be able to assess the fatigue performance of composites with small initial defects takes on an added significance. Indeed, the importance of working to the correct fatigue threshold, for a given set of stresses and initial flaws, was highlighted by Schoen et al. [18], 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".

These examples demonstrate the importance of being able to assess delamination growth in the design of composite aircraft structures.

Fortunately, due to the low stress levels seen in the majority of composite structural members there are only a few instances where there has been in-service delamination growth or disbanding [11,13]. The challenge is to increase these working stress levels. This can only be achieved by knowing the precise boundary of the no growth design envelope, its relationship to the size and nature of the initial defect(s) and how to allow for the growth of small naturally occurring defects. Whereas for metals it is known [15–17] that for small defects subjected to operational flight load spectra the fatigue threshold is very small and that fleet experience has shown that delaminations can grow from small sub-mm initial defects there has, as yet, been no controlled study performed to establish the behaviour of representative small (sub mm) defects in composites. The in-service delaminations seen in the boron epoxy doubler on RAAF F-111 and Canadian CF-5 aircraft and the delamination growth seen in various full scale fatigue tests also highlight the need to develop a methodology that can be used to accurately and reliably predict the inspection intervals associated with delamination growth arising from small naturally occurring defects.

Current approaches to predicting the growth of delamination damage are based on those used to predict crack growth in metals. The difference lies in the fact that whereas in metals the driving parameter is based on the stress intensity factor range ΔK and K_{max}, in composites it is based on ΔG and/or G_{max} [19–27] and the equation for the increment in the delamination size with cycles takes the form:

$$da/dN = C \cdot (\Delta G)^m \tag{4}$$

or variants thereof. Unfortunately, when expressed in this form the exponent m is usually very large, in [23] m has values greater than 10. As such [22] commented

"For composites, the exponents for relating propagation rate to strain energy release rate have been shown to be high, especially in mode I. With large exponents, small uncertainties in the applied loads will lead to large uncertainties (at least one order of magnitude) in the predicted delamination growth rate. This makes the derived power law relationships unsuitable for design purposes".

One result of this is that the common design philosophy is to adopt a no growth design [28,29]. However, as explained in [7] the potential of Eq. (1) which often has an exponent of approximately 2 to represent delamination growth raises the possibility of removing this restriction.

One major difference between the formulation presented in [7], i.e. equations of the form given in Eqs. (1) and (3), and the more traditional Paris based laws lies in their potential to represent the growth of small naturally occurring defects. As we have previously mentioned the design philosophy currently used for composite structures is to ensure that the energy release rate associated with any defect lies beneath the threshold determined from tests on specimens containing large defect. Unfortunately it is known that, for metals, predictions based on Paris like crack growth equations that use data obtained from tests on large cracks produce highly non-conservative estimates for the fatigue life of aircraft structures containing small naturally occurring defects. This phenomenon is aptly illustrated in [30] which studied the growth of a small 0.003 mm initial defect in a 7050-T7451 aluminium bulkhead of an F/A-18 aircraft subjected to a representative fighter load spectra [30]. Here it was found that crack growth predictions obtained using long crack da/dN versus ΔK data gave a highly non-conservative life. In contrast [8] revealed that for 7050-T7451 aluminium alloy the growth of both long and naturally occurring short cracks conform to same the Hartman–Schijve equation. This suggests that Eqs. (1) and (3), with the constants determined as above from the long delamination tests but with the threshold set to near zero, may enable a conservative estimate of the time required for a defect to grow from a small naturally occurring defect to an detectable size to be computed. As such the present paper focuses on examining whether a variant of these equations can be used to represent crack growth in a range of nano-composites.

2. Crack growth in nano-composites

Having seen that delamination growth in composites can be represented by a variant of the Hartman–Schijve equation let us now address the question of crack growth in nano-composites. Current studies on nano-composites appear to be limited to tests on large initial defects [31–37]. However, it has long been known that the use of long crack data can lead to the incorrect[1] ranking of materials [38]. If nano-composites are to address the problem of delamination damage due to small naturally occurring defects in composites then we need to establish that the crack growth versus G relationship can be expressed in a way that the exponent is small and in such a fashion that the information obtained can be used to

[1] Here it was shown that the superior fatigue performance of aluminium lithium alloys in comparison to traditional aluminium alloys vanished when the initial crack size was small.

used to assess the growth of small naturally occurring defects under realistic operational loading. In this context it has recently been shown [7,8] that when the crack growth rate is expressed as per Hartman–Schijve variant:

$$da/dN = D\left((\Delta K - B)/\sqrt{(1 - K_{max}/A)}\right)^{\alpha} \tag{5}$$

where A, B, α and D are constants, then the crack data associated with the growth of long and short cracks in 7050-T451 coincide [8] as do the growth of long and short cracks in 4340 steel [7]. Thus in the present paper we examine whether Eq. (5) and related variants can be used to represent the growth of delamination damage in nano-composites.

To this end we first analysed the data presented in [31] for SiO_2 nano-particles in a standard diglycidyl ether of bisphenol A (DGEBA) epoxy resin as well as for the un-modified DGEBA epoxy.a raw epoxy and for an epoxy containing nano-particles. The da/dN versus ΔK relationships presented in [31] are reproduced in Fig. 1 together with together with a representation of the behaviour obtained using Eq. (5) where the values of A, B and D are as given in Table 1 and where we have set $\alpha = 2$. Zhang et al. [33] presented data for crack growth in a 0.5 wt.% multi-walled MWCNT-epoxy nano-composite as well as for a 0.5 wt.% single wall SWCNT-epoxy nano-composite. The epoxy used in this study was the Epoxy-2000 resin system. The data presented in this study are reproduced in Figs. 2 and 3 respectively along with together with a representation of the behaviour obtained using Eq. (5) with the values of A, B and D as given in Table 1 and $\alpha = 2$. Zhang et al. [36] also presented data for crack growth in a multi-walled MWCNT-epoxy nano-composite where the MWCNT's had a range of different diameters, viz: 5–8 nm, 10–20 nm and 50–70 nm. (The epoxy used was nominally the same as that used in [33].) The data presented in this study are reproduced in Fig. 4 along with together with a representation of the behaviour obtained using Eq. (5) with the values of A, B and D as given in Table 1 and $\alpha = 2$. Data was presented in [32] for crack growth in a diglycidyl ether of bisphenol A (DGEBA) epoxy resin, 'LY556' both with and without 2 wt.% multi-walled carbon nano-tubes (MWCNT). The associated da/dN versus G_{max} relationships, presented in [32], are reproduced in Fig. 5 together with a representation of the behaviour obtained using the Hartman–Schijve variant:

$$da/dN = D\left(\left(\Delta\sqrt{G_{max}} - B\right)\Big/\sqrt{\left(1 - \sqrt{G_{max}}/A\right)}\right)^{\beta} \tag{6}$$

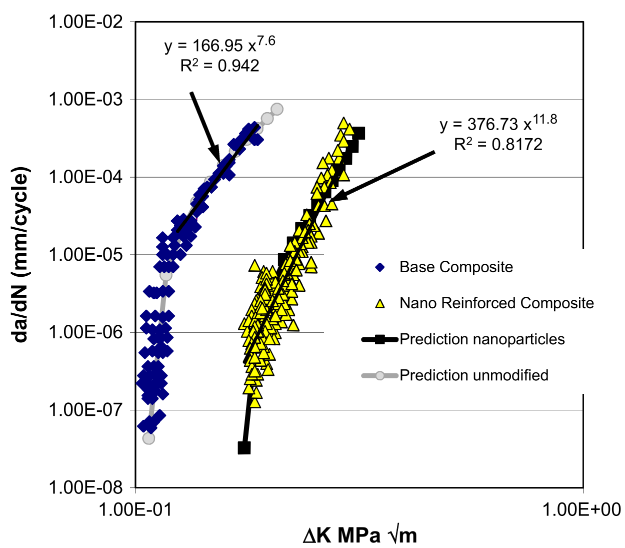

Fig. 1. Comparison with crack growth data obtained on a nano-particle-epoxy nano-composite and crack growth in the unmodified epoxy, from [31].

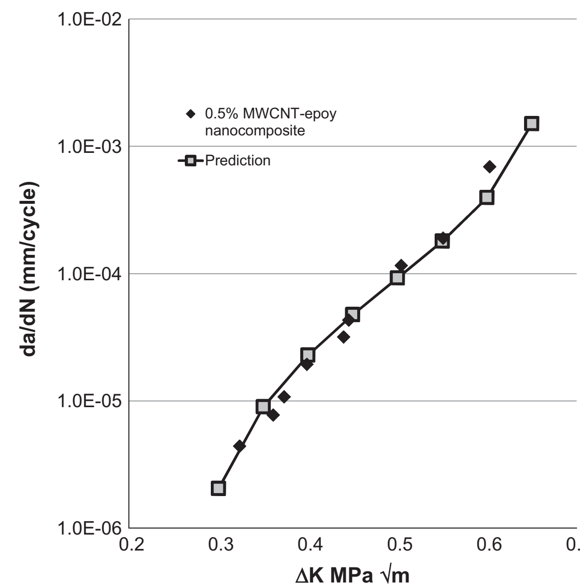

Fig. 2. Comparison with crack growth data obtained on a MWCNT-epoxy nano-composite, from [33].

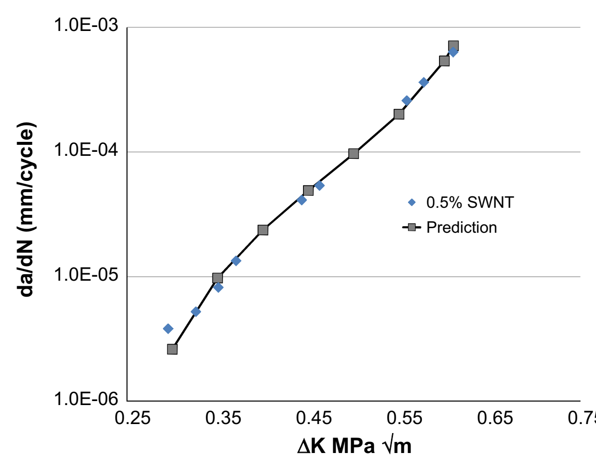

Fig. 3. Comparison with crack growth data obtained on a SWCNT-epoxy nano-composite, from [33].

Table 1
Data and threshold values for the various tests.

Ref.	Type of nano-material	B MPa $\sqrt{m}$	A MPa $\sqrt{m}$	D
[31]	SiO_2 nano-particles	0.172	0.35	1.8×10^{-3}
[31]	DEGBA epoxy without nano-particles	0.106	0.35	3.0×10^{-2}
[33]	MWCNT	0.25	0.75	3.0×10^{-4}
[33]	SWCNT	0.24	0.72	3.0×10^{-4}
[34]	FPS	0.44	0.8	2.2×10^{-3}
[36]	5–8 nm diameter MWCNT	0.24	0.8	8.0×10^{-4}
[36]	10–20 nm diameter MWCNT	0.22	0.8	6.5×10^{-4}
[36]	50–70 nm diameter MWCNT	0.18	0.8	5.5×10^{-4}
[36]	Epoxy, no MWNT	0.07	0.8	4.0×10^{-4}
[37]	Discontinuous SiC whiskers, not a nano additive.	4.15	100	2.58×10^{-9}
[37]	2168 Aluminium, no additives	3.50	100	1.64×10^{-9}

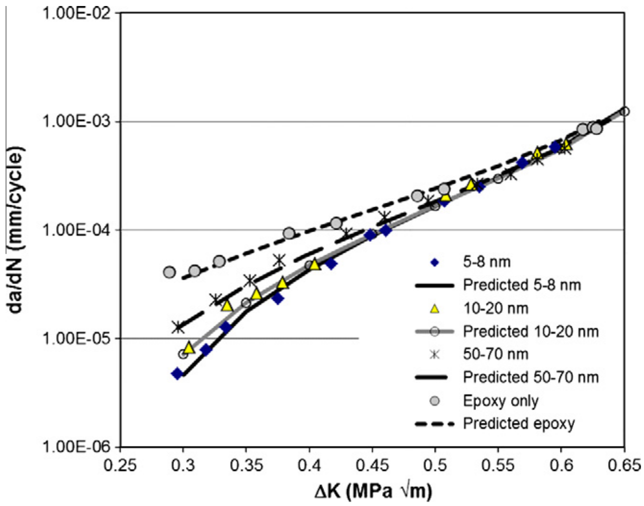

Fig. 4. Comparison with crack growth data obtained for MWCNT-epoxy nano-composites with a range of MWCNT diameters, from [36].

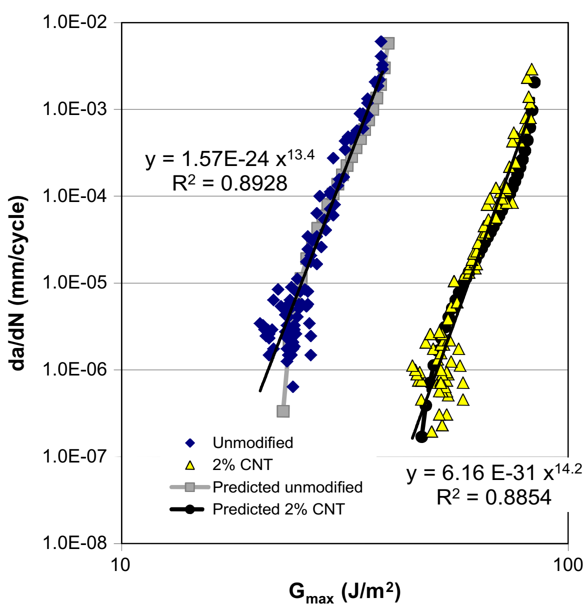

Fig. 5. Comparison with crack growth data obtained on a 2% CNT-epoxy nano-composite and the unmodified epoxy, from [32].

where the values of A, B and D are given in Table 2 and as previously we have set $\beta = 2$. To further investigate the crack growth behaviour of nano-composites Fig. 6 shows the crack growth data presented in [34] for a functionalized (i.e. partially oxygenated) graphene sheets (FGS)–epoxy nano-composite compared with Eq. (5) with the values of A, B and D as given in Table 1. (The particular epoxy used in this study was not reported.) In each case we see that the crack growth relationships are reasonably accurately represented by variants of the Hartman–Schijve equation with an exponent β or α, depending on which equation is used, of 2. In contrast it is interesting to note the large exponents obtained when Paris like equations are fitted to the data shown in Figs. 1 and 5.

It should be noted that whereas it has been stated that the exponents α (or β) in Eqs. (1), (3), and (5) are approximately 2 they typical have values that range from 1.8 to 2.2, see [8]. To illustrate this we analysed the data presented in [37] for crack growth in a metal matrix composite (mmc), a 2168 aluminium alloy that was reinforced with 15 wt.% SiC whiskers. The resultant crack growth

Table 2
Parameters associated with the various tests.

Ref.	Type of nano-material	B ($\sqrt{}$J/m)	A ($\sqrt{}$J/m)	D
[32]	MWCNT	45	85	3.0×10^{-6}
[32]	DEGBA epoxy without MWNT	22.5	40.5	3.0×10^{-6}

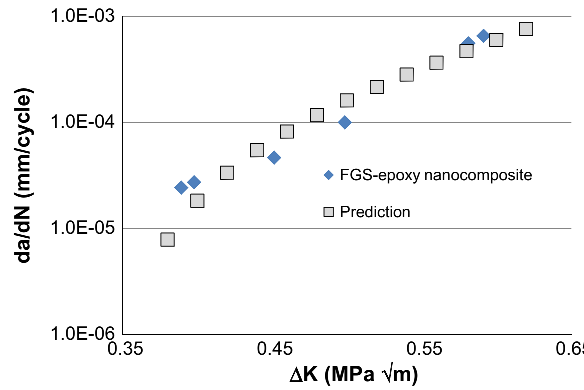

Fig. 6. Comparison with crack growth data obtained on a FGS-epoxy nano-composite, from [34].

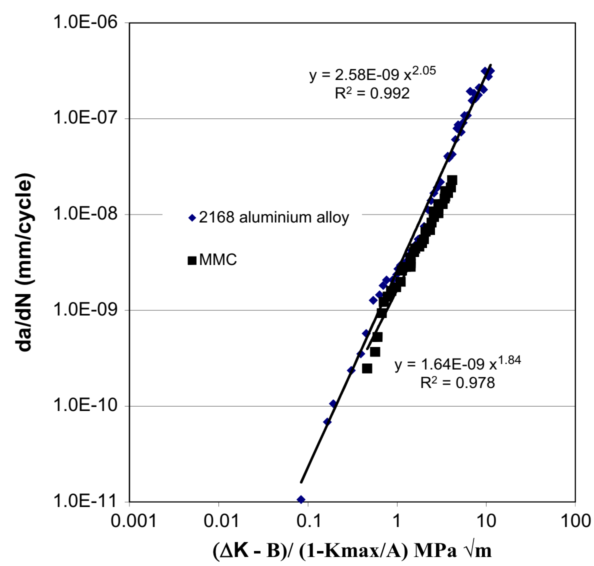

Fig. 7. Crack growth in a SiC whisker reinforced metal matrix composite and the base 2618 aluminium alloy, from [37].

data is shown in Fig. 7, together with the data, also provided in [37], for the case when there was no reinforcement. This figure reveals that, in both cases, the experimental data is very well represented by Eq. (5) and that the "best" fit to these data sets yield values of $\alpha = 1.84$ for the metal matrix composite and of $\alpha = 2.05$ for the base 2618 aluminium alloy. However, in the nano-composites examples studied (above) we have chosen, for simplicity, to use a fixed value of α (or β) = 2 for these various exponents. The exponent could be varied (slightly) to further improve the representation.

3. Conclusion

The experimental data presented in this paper reveals that crack growth in a range of nano-composites can be reasonably well represented by a variant of the Hartman–Schijve equation that has recently been developed to account for both crack growth in metals, Modes I and II delamination growth in composites and the environmental degradation of adhesive bonds. We have found that when expressed in this form the exponent of this variant is approximately 2 and, as such, is considerably lower than the exponent in "Paris like" power law representations for delamination growth in composites and crack growth in (unmodified) epoxies. As such we see that, in these examples, the present variant of the Hartman–Schijve representation of crack growth in nano-composites growth is similar to that seen for crack growth in metals. (Note that the constant of proportionality is significantly bigger for nano-composites than it is for metals.) This suggests that, since it has been shown that crack growth equations that relate da/dN to $(\Delta K - \Delta K_{th})$ can be used to model the growth of both small and larges, this formulation may be useful for the damage tolerant assessment of small naturally occurring defects in nano-composite structures. This is an important topic since it is well known that the ranking of materials and surface treatments based on long crack data can lead to incorrect results and since it is now known that the operational performance of aircraft structures is often determined by the growth of damage from small naturally occurring defects. However, given that there is no published data associated with the growth of small naturally occurring defects in nano-composites and no published data associated with the growth of cracks in nano-composites under variable amplitude spectra this subject clearly needs further investigation.

References

[1] Hartman A, Schijve J. The effects of environment and load frequency on the crack propagation law for macro fatigue crack growth in aluminum alloys. Eng Fract Mech 1970;1(4):615–31.

[2] Smith IFC, Smith RA. Fatigue crack growth in a fillet welded joint. Eng Fract Mech 1983;18(4):861–9.

[3] Ramsamooj DV. Analytical prediction of short to long fatigue crack growth rate using small- and large-scale yielding fracture mechanics. Int J Fatigue 2003;25:923–33.

[4] McEvily AJ, Eifler D, Macherauch E. An analysis of the growth of short fatigue cracks. Eng Fract Mech 1991;40:571–84.

[5] Endo M, McEvily AJ. Prediction of the behaviour of small fatigue cracks. Mater Sci Eng A 2007;468–470:51–8.

[6] Ishihara S, Yoshifuji S, Mcevily AJ, Kawamoto M, Sawai M, Takata M. Study of the fatigue lifetimes and crack propagation behaviour of a high speed steel as a function of the *R* value. Fatigue Fract Eng Mater Struct 2010;33:294–302.

[7] Jones R, Pitt S, Bunner AJ, Hui D. Application of the Hartman–Schijve equation to represent modes I and II fatigue delamination growth in composites. J Compos Struct 2012;94(4):1343–51.

[8] Jones R, Molent L, Walker K. Fatigue crack growth in a diverse range of materials. Int J Fatigue 2012;40:43–50.

[9] Elber W. The significance of fatigue crack closure. Damage tolerance of aircraft structures. ASTM STP-486. 1971, p. 230–42.

[10] Pitt S, Jones R, Peng D. Characterization of the durability of adhesive bonds. Fatigue Fract Eng Mater Struct 2012;45(11):998–1006.

[11] Chalkley P, Geddes R. Service history of the F-111 wing pivot fitting upper surface boron/epoxy doublers DSTO-TN-0168. Department of Defence; 1998.

[12] Molent L, Callinan RJ, Jones R. Design of an all boron epoxy doubler for the F-111C wing pivot fitting: structural aspects. Compos Struct 1989;11(1):57–83.

[13] Raizenne D. Case history: CF116 upper wing skin fatigue enhancement boron doubler. In: Baker A, Rose LRF, Jones R, editors. Chapter, Advances in the bonded composite repair of metallic aircraft structure. Elsevier Applied Science Publishers; 2002. ISBN 0-08-042699-9.

[14] van Blaricum TJ, Bates P, Jones R. An experimental investigation into the effect of impact damage on the compressive strength of step lap joints. Eng Fract Mech 1989;32(5):667–74.

[15] Wanhill RJH. Characteristic stress intensity factor correlations of fatigue crack growth in high strength alloys: reviews and completion of NLR investigations 1985–1990, NLR-TP-2009-256, September 2009.

[16] Molent L, Barter SA, Wanhill RJH. The lead crack fatigue lifting framework. Int J Fatigue 2011;33:323–31.

[17] Schijve J. Fatigue life until small cracks in aircraft structures. Durability and damage tolerance. Advanced structural integrity methods for airframe durability and damage tolerance. In: Harris CE, editor. Proceedings of FAA/NASA international symposium, Hampton, 1994. NASA Conference Publication 3274; 1994. p. 665–81.

[18] Schoen J, Nyman T, Blom A, Hans Ansell. A numerical and experimental investigation of delamination behaviour in the DCB specimen. Compos Sci Technol 2000;60:173–84.

[19] O'Brien TK. Analysis of local delaminations and their influence on composite laminate behaviour. Delamin debonding of materials. ASTM STP 876; 1985.

[20] Baker AA, Jones R, Callinan RJ. Damage tolerance of graphite epoxy structures. J Compos Struct 1985;4:15–44.

[21] Tay TE, Williams JF, Jones R. Characterization of pure and mixed mode fracture in composite laminates. Theor Appl Fract Mech 1987;7:115–23.

[22] Martin RH, Murri GB. Characterization of mode I and mode II delamination growth and thresholds in graphite/peek composites. NASA Technical Memorandum 100577, AVSCOM Technical Memorandum 88-B-011, April 1988.

[23] Khan R, Rans CD, Benedictus R. Effect of stress ratio on delamination growth behavior in unidirectional carbon/epoxy under mode I fatigue loading. In: Proceedings ICCM. Edinburgh; 2009.

[24] Brunner AJ, Murphy N, Pinter G. Development of a standardized procedure for the characterization of interlaminar delamination propagation in advanced composites under fatigue mode I loading conditions. Eng Fract Mech 2009;76:2678–89.

[25] Mall S, Yun KT, Kochhar NK. Characterization of matrix toughness effect on cyclic delamination growth in graphite fiber composites. In: Presented at the 2nd ASTM symposium on composite materials: fatigue and fracture. Cincinnati, Ohio, April 1987.

[26] Prel YJ, Davies P, Benzeggagh ML, de Charentenay F. Mode I and Mode II delamination of thermosetting and thermoplastic composites. In: Presented at the 2nd ASTM symposium on composite materials: fatigue and fracture. Cincinnati, Ohio, April 1987.

[27] Brunner AJ. Experimental aspects of Mode I and Mode II fracture toughness testing of fibre-reinforced polymer–matrix composites. Comput Meth Appl Mech Eng 2000;185:161–72.

[28] Johnson RW, McCarty JE, Wilson DR. Damage tolerance testing for the Boeing 737 graphite-epoxy horizontal stabilizer. In: Proceedings of 5th DoD/NASA conference on fibrous composites in structure design. NADC, 81096-60, New Orleans, January 1981.

[29] Fawcett J, Trostle S, Ward, 777 empennage certification approach. In: Proceedings 11th international conference on composite materials (ICCM-11). Australia, July 14–18 1997.

[30] Jones R, Molent L, Pitt S. Crack growth from physically small flaws. Int J Fatigue 2007;29:1658–67.

[31] Blackman BRK, Kinloch AJ, Sohn Lee J, Taylor AC, Agarwal ER, Schueneman G, et al. The fracture and fatigue behaviour of nano-modified epoxy polymers. J Mater Sci 2007;42:7049–51.

[32] Hsieh TH, Kinloch AJ, Taylor AC, et al. The effect of carbon nanotubes on the fracture toughness and fatigue performance of a thermosetting epoxy polymer. J Mater Sci 2011;46:7525–35, ISSN:0022–2461.

[33] Zhang W, Picu RC, Koratkar N. Suppression of fatigue crack growth in carbon nanotube composites. Appl Phys Lett 2007;91:193109.

[34] Zhang W, Srivastava I, Zhu YF, Picu CR, Koratkar NA. Fracture and fatigue in graphene nanocomposites. Small 2009;5:1403.

[35] Srivastava I, Koratkar NA. Fatigue and fracture toughness of epoxy nano-composites. JOM 2010;62(2):50–7.

[36] Zhang W, Picu RC, Koratkar NA. The effect of carbon nanotube dimensions and dispersion on the fatigue behavior of epoxy nano-composites. Nanotechnology 2008;19:1–5.

[37] Lorca JL. Fatigue of particle and whisker-reinforced metal matrix composites. Prog Mater Sci 2002;43:283–353.

[38] Venkateswara Rao KT, Yu W, Ritchie RO. On the behavior of small fatigue cracks in commercial aluminum-lithium alloys. Eng Fract Mech 1988;31(4):623–35.