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The influence of cyclic stress intensity threshold on fatigue life scatter

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ABSTRACT

Understanding factors that contribute to scatter in fatigue lives of metallic structures (particularly airframes) subjected to identical spectrum is critical to maintaining safety and optimising designs. This paper first briefly discusses the sources of scatter, and then concentrates on the effect of variations in the "cyclic stress intensity threshold" (ΔK_{thr}) on fatigue crack growth. It shows that a version of the NASGRO equation can be used to account for the crack growth scatter seen in a number of classical fatigue experiments by accounting for variations in ΔK_{thr}. This is an important outcome for safety and is particularly useful when considering lead cracks for which ΔK_{thr} is small (approaching zero) as these cracks appear to commence growing soon after introduction into service.

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1. Introduction

It is well known that when nominally identical metallic structures are subjected to the same environment, loading and stress level that significant variations in fatigue life can occur. Furthermore, for aircraft design it is normal to apply a scatter factor to the result of a full-scale fatigue test in order to establish the safe – life (or durability life) for a nominated cumulative probability of failure (CPOF) (e.g. see [1]).

Understanding of the fatigue behaviour of metal airframes has progressed to the point where, for a fixed constant amplitude loading or spectrum, stress level and material, it is known that the scatter in the fatigue performance of monolithic metallic airframes is governed by the variability in the metal's material properties and manufacturing quality.[1] These two areas can be quantified, see [2,3], by gaining an understanding of the variability in the:

1. initial discontinuities that lead to fatigue cracking;
2. stress concentrations leading to inter-aircraft variations in local stress;
3. fit-up or residual stresses;
4. fracture toughness of the material;
5. crack nucleation and/or initiation period;
6. cyclic threshold stress intensity factor, referred to here as ΔK_{thr} (see [5,6]).[2] It is noted that short cracks (as well as cracks grown under spectrum loading) do not show a pronounced dip in the da/dN versus ΔK approaching the threshold region as is often seen in long crack data, see e.g. [7]; and
7. fatigue crack growth (FCG) rate in the material being examined.

Items 1–3 are considered to define the aircraft's build quality from a fatigue perspective. Items 4–7 define the metal's material property variability.[3] Thus, if the variation or influence, in these variables can be quantified then a more accurate estimate of the scatter in a metal's fatigue performance can be made. In this context it is interesting to note that [7] stated "Reliable determinations of fatigue crack growth thresholds are important for fatigue crack growth analyses, especially for helicopter airframe components, since the analyses rely mainly on crack growth data in the near-threshold region. This region is often characterized by considerable data scatter, including scatter in the threshold values."

As such the purpose of this paper is to substantiate the premise first stated in [5,18] that the FCG rate variability seen in a number of classical constant amplitude where generally the initial crack length was fixed and a sample of variable amplitude fatigue experiments, can be accounted for by allowing for variations in ΔK_{thr}. This is considered an important outcome for the safety and

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[1] Note that variations in aircraft usage will also lead to significant scatter in fatigue lives. This is one reason why most military agile aircraft are fatigue-usage monitored, see [4].

[2] It should be noted as stated in [5,18,19], the parameter ΔK_{thr} herein should not be confused with the term ΔK_{th}, which the ASTM 647 fatigue test standard suggests to be the value of ΔK at a crack growth rate da/dN of 10^{-10} m/cycle, see Appendix.

[3] Some aspects of item 1 (e.g. production discontinuities) may also be considered as a material processing property.

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optimisation of designs, and is particularly useful when considering lead cracks [8,9] for which ΔK_{thr} is small (i.e. asymptotes to zero).

2. The lead crack concept

A fatigue lifing framework using a lead crack concept has been developed by the Defence Science and Technology Group (DSTG) for primary metallic airframe components [8,9]. This framework builds on the observation that (near) exponential FCG is a common occurrence for naturally-initiating lead cracks (i.e. those leading to first failure) in test specimens, components and airframe structures subjected to variable amplitude load histories [8–18].

2.1. Lead crack characteristics

Lead cracks have the following general characteristics (derived from [8,9]):

(1) They start to grow shortly after testing begins or after the aircraft is introduced into service. Significantly this implies that ΔK_{thr} is small (i.e. close to zero), see [7–9,16–20] for more details. This is consistent with the statement in Appendix X3 of the ASTM fatigue test standard E647-13a which states:

"It is not clear if a measurable threshold exists for the growth of small fatigue cracks"

(2) Subject to several caveats (see [8,9]) they grow approximately exponentially with consistent loading history, i.e. FCG may be represented by an equation of the form:

$$a = a_0 e^{\lambda t} \tag{1}$$

where:
a = Crack depth
a_0 = Initial crack size (or equivalent pre-crack size (EPS)[4] [10–15])
λ = Growth rate parameter that includes the finite geometrical factor β
t = Cycles/No. of Load Blocks/Simulated Flight Hours

(3) A significant portion of their lives is spent in the short crack regime (i.e. at depths less than approximately 1 mm).
(4) The lead cracks grow in an optimum manner generally unaffected by such factors as crack-closure or material grain size etc. [8].
(5) The fastest possible lead crack is more likely to be revealed in a larger component than in a small coupon (i.e. the area or volume effect). Having a concurrent combination of 'favourable' grain orientation, local stresses and large initial discontinuities is more probable for a larger sample of material.
(6) For a given material, spectrum and item, the λ parameter of the exponential equation, e.g. the slope of the crack growth curve shown in Fig. 1, is approximately a constant for given spectrum and stress level.
(7) The mean equivalent pre-crack size (EPS) for AA7050-T7451 plate is approximately equivalent to a 0.01 mm deep (semi-circular) surface fatigue crack [8–14]. In other words a 0.01 mm deep crack is a good starting point for assessing the average fatigue life using the lead crack framework, see Fig. 1. This EPS value is well below the smallest initial surrogate flaw/crack size usually assumed in the damage tolerant method for durability (i.e. 0.254 mm). This mean EPS is consistent with those reported in [21] and [22] for other aluminium alloy airframe materials.
(8) The metallic materials used in highly stressed areas of high performance aircraft, where load shedding has not occurred, typically have critical crack depths of the order of 10 mm, see [9,12,13] and [17].

[4] This definition of the EPS is effectively the same as the Equivalent Initial Quality Method [23], see [14,15].

3. Scatter in fatigue

When considering the factors influencing scatter for a fixed material, environment, loading spectrum and stress level in monolithic structure it should be noted that:

(1) Past studies have indicated that the most significant variable is the variability in the population of the initiating discontinuities, a_0 (e.g. [2,24]).
(2) Whilst variations in local stress concentration features between aircraft can lead to significant differences in local stresses and thus fatigue lives, for the purposes of this paper it will be assumed that modern aircraft are built to exacting standards and close tolerance, using (for example) computerised numerical control milling. Therefore, the variability in the local stress concentration factors and the variability in the fit-up stresses between (monolithic structure in) aircraft are not considered in this paper.
(3) Similarly, variations in local residual stresses can result in significant variations in fatigue lives. In this paper the effect of residual stresses is not considered.
(4) For aircraft the variation in the fracture toughness of thick plate material has little effect on the variability in total life (if $a_{\text{critical}} \gg a_0$), see Ref. [2] i.e. the crack is growing so fast near the end of its life that small changes in critical crack size make little difference to the total life.
(5) For lead cracks the times to initiation are negligible (see for example [8,9,16,17,23–27,28]) and can be conservatively ignored.[5] This can be achieved (or expressed) by allowing ΔK_{thr} to approach zero as shown in this paper.
(6) Finally, for lead cracks, the variation in FCG rate (for a given environment, spectrum and stress level) is considered to be secondary in comparison to the contribution of the effect of variations in the initiating discontinuity size, see Refs. [2,3]. This statement is consistent with the US Air Force's approach to probabilistic failure analysis [26,27] and, as such, it will be further considered in the following sections.

4. NASGRO equation

Eq. (2) (referred to as the Hartman–Schijve variant [5,18,19,28–31]) is a special case of the NASGRO equation[6] [18]. It can be found in both NASA's crack growth life prediction program NASGRO and in AFGROW, and was first shown [5] to capture the variability noted in crack growth rates associated with the growth of cracks from small initial material discontinuities in AA7050-T7451 by varying the term ΔK_{thr}[7]:

$$\frac{da}{dN} = D[(\Delta K - \Delta K_{thr})/\sqrt{(1 - K_{\max}/A)}]^{\alpha} \tag{2}$$

Here $K_{\max}$ is the maximum stress intensity produced by the load cycle at each crack's tip, α is a constant, which is determined from the slope of the da/dN versus $[(\Delta K - \Delta K_{thr})/\sqrt{(1 - K_{\max}/A)}]$ curve and has been found to be approximately 2 for several metallic

[5] Fretting and corrosion induced cracking are likely exceptions to this.
[6] See Appendix.
[7] It should be noted that a similar equation has been used to capture scatter in growth from discontinuities in composites, nano-composites and adhesive bonds see [30,34–37]. In these studies the term ΔK in Eq. (2) was replaced by $\Delta\sqrt{G}$ where G is the energy release rate.

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