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Using the lead crack framework to reduce durability test duration

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ABSTRACT

Aircraft full-scale fatigue tests are expensive and time-consuming to conduct but are a critical item on the certification path of any aircraft design or modification. This paper outlines a proposal that trades cycling hours for increased detail in the teardown of a metallic test article. A method for determining the equivalent demonstrated crack size (and crack growth curve) at the mandated test life utilising the lead crack framework is demonstrated. It is considered that the test duration can be significantly reduced, whilst still achieving all the desired outcomes of a certification program.

Keywords: Fatigue; Aircraft; Fatigue or Durability tests

1.0 INTRODUCTION

Full-Scale Fatigue/durability Tests (FSFTs) form the cornerstone in the structural qualification and certification of new aircraft types in both the civil and military sectors. Even though conducting these tests consume enormous resources of time, effort and costs, they have thus far proved to be the most reliable and, in many cases, an essential means of establishing the evidence base for an aircraft's safety and durability.

Whilst there is a move towards a more analytical (or virtual test) basis[1], 'FSFTs continue to produce premature fatigue cracking[*contineo*, 2], demonstrating the current inadequacies of the fatigue analysis process. Until a time is reached when FSFTs do not reveal significant premature failures or cracking, designers and regulators alike will not have enough confidence in analysis methods to allow deletion of full-scale testing'. This paper proposes a means of accelerating the progress of the FSFT program, to lower the associated resource burden, whilst avoiding the current shortcomings[1] that are associated with a fully analytical approach.

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FSFT certification programs aim to[(e.g. (3–5))]:

1. Demonstrate the durability of the airframe (i.e. the period of service before structural impairment due to fatigue cracking could occur given a specific value for the allowable probability of failure);
2. Reveal locations prone to cracking (i.e. fatigue hotspots);
3. Provide data to correlate, validate or correct design fatigue crack growth (FCG) codes and damage tolerance analyses;
4. Determine the residual strength (i.e. critical crack size a_{CRIT}) of the airframe in the presence of fatigue cracks;
5. Determine the potential influence of widespread fatigue damage (WFD[(4)]);
6. Provide data to support sustainment, including the measurement of strain data;
7. Provide the baseline for individual aircraft fatigue life monitoring; and
8. Generate Equivalent Pre-crack Sizes (EPS) to facilitate probabilistic analyses[(e.g. (7))].

The data collection aspect described can be achieved early in the program, leaving the crack growth-based items as factors driving the longevity of the test. There are many potential means of accelerating an FSFT to produce a certification outcome[(e.g. (8))], these include: increased cyclic rates; smart spectrum truncation; autonomous non-destructive inspection (NDI); more sensitive NDI[1]; inducing artificial cracks in predicted hotspots[(9)]; and inserting fracture surface-marking loads into the spectrum to aid quantitative fractography (QF). Most of these require further development, which is recommended. This paper only considers an alternative method for determining the equivalent test demonstrated critical crack size (and FCG curve) for a metallic airframe at the mandated/desired test life utilising the lead crack framework. This methodology allows the test program duration to be significantly reduced whilst still achieving all desired outcomes of a certification program.

2.0 THE LEAD CRACK FRAMEWORK METHOD

A fatigue life prediction framework using a lead crack framework has been developed by the Australian Defence Science and Technology (DST) Group for primarily metallic airframe components[(6)]. This framework builds on the observation that (near) exponential (i.e., log crack size versus linear cycles) FCG is a common occurrence for naturally nucleating lead cracks (i.e. those leading first to failure) in test specimens, components and airframe structures subjected to variable amplitude load histories[(6,11–18)].

Lead cracks have the following general characteristics (adapted from[(6)]):

1. They start to grow shortly after testing begins or after the aircraft is introduced into service from material or production discontinuities. Significantly, this implies that the threshold cyclic stress intensity factor (ΔK_{thr}) is small (i.e. close to zero). For more details, see (6,10,16–20). Other cracks where nucleation is time-dependant need separate consideration (e.g. in-service induced damage, poor repairs, corrosion nucleated). Also, it should be noted that a conventional FSFT will not provide information relating to such cracks.

[1] If cracking could be detected at sub-mm depths, then repair options would be simpler than, say, component replacement due to significant cracking. It is also noted that a substantial portion of the FSFT program is consumed by inspection and (later) repair activities.

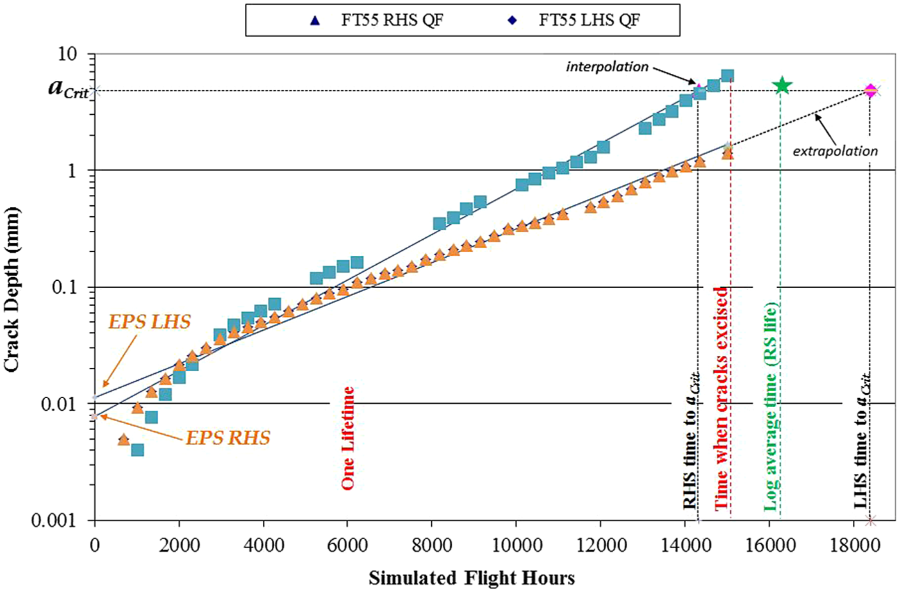

Figure 1. Schematic of typical (lead) crack growth (note: each point represents one spectrum block of growth derived from QF).

2. Subject to several caveats[(6)], they grow approximately exponentially with consistent loading history, i.e. an FCG may be approximated by an equation of the form:

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

where a = Crack depth (or length)
a_0 = Initial crack size (or EPS)[(11–15)]
λ = Growth rate parameter that includes the finite geometry factor β
t = Fatigue life in terms of Cycles/Number of Load Blocks/Simulated Flight Hours (SFH)
3. A significant portion of their lives is spent in the physically short crack regime (i.e. at depths less than approximately 1mm).
4. They grow in an optimum manner generally unaffected by such factors as crack-closure or material grain size, etc.
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 or a component containing multiple fastener holes.
6. For a given material, spectrum, peak stress level and structural detail, the λ parameter of the exponential equation, e.g. the slope of the FCG curve shown in Fig. 1, is approximately a constant.
7. The mean EPS for a 7050-T7451 aluminium alloy (AA) plate is approximately equivalent to a 0.01 mm deep (semi-circular) surface fatigue crack[(6,7,11–14)]. In other words, in this material, a 0.01 mm deep crack is a good starting point for estimating the average fatigue life using the lead crack framework, see Fig. 1. Note that this EPS value is

well below the surrogate initial flaw/crack size usually assumed in the damage tolerant method (i.e., 1.27 mm[4]).

8. The metallic materials used in highly stressed areas of high-performance aircraft, where load shedding has not occurred, typically have a_{CRIT} of less than 10 mm deep[6,9,13,14,17].
9. The framework is designed to produce a conservative FCG curve (e.g. the quasi-static crack tearing close to end-of-life is ignored).

Exponential FCG represents an optimum path for the crack to reach a_{CRIT} for a given spectrum, stress level and detail – thus analyses based on the framework will tend to result in conservative fatigue life management outcomes.

3.0 THE LEAD CRACK BASED METHOD

Let us assume for illustrative purposes that the FSFT is ready to commence cycling and the program's aim was to demonstrate structural durability via two lifetimes of cycling. The lead crack framework predicts that cracks (with lives short enough to pose a meaningful risk of failure within the service life of the airframe) will commence growing from inherent material discontinuities shortly after the test starts.

The proposal is to terminate cycling early, for example after one lifetime, and then perform a thorough teardown and detailed inspection of the test article (noting that the regulations already contain a requirement for a teardown inspection). The teardown is driven (but not limited) by a pre-knowledge of hotspots and from collected strain data to prioritise the order in which details are inspected. It is envisaged that the hotspots and surrounding details will be subjected to loads that would cause fracture to occur and reveal the largest crack (as well as, in many instances, the surrounding smaller cracks). Many hotspots can be assessed in parallel subject to availability of resources. When cracking is detected, QF is conducted to derive an FCG curve, see example in Fig. 1. This step also confirms the appropriateness of the lead crack framework or informs the need for the use of an alternative model to predict the FCG at the hotspot under consideration. The a_{CRIT} can be calculated from the geometry and material fracture toughness or separate tests.

The lead crack framework is then used to extrapolate (or interpolate) the FCG to a_{CRIT} or a_{RST} (Residual Strength Test (RST)) depending on certification requirements, as shown in Fig. 1. The example in Fig. 1 from the FT55 fighter aircraft FSFT[21] shows the crack depth (log scale) versus time history (linear scale) for a complex AA 7050-T7451 fuselage detail, as determined via QF. The growth curve for the left-hand-side crack has been extrapolated to the estimated critical crack size for RST loading conditions (a_{RST}) and to estimate the achievable life to a_{RST}. For the right-hand-side crack, the crack growth curve to the critical crack size has been interpolated back to a_{RST}.

The demonstrated fatigue life was calculated from the geometric mean of the lives to a_{RST} for both sides. Note the approximately exponential crack growth, which was used to estimate the EPS values for each crack. The early departure from exponential is thought to result from the transition of a discontinuity into a crack. Nonetheless, the use of the lead crack concept coupled with QF enables the generation of EPS to facilitate any probabilistic or risk analyses.

Once FCG curves are available, the certification requirements outlined above can be evaluated. Given that the QF of the principal cracks can be conducted in a significantly shorter timeframe than the continuation of cycling for another lifetime, it is suggested that this approach could make the required certification data available to the regulator and the original

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equipment manufacturer at an earlier stage, enabling the introduction of any necessary modifications during production or at least more efficiently in-service. This facilitates a more agile approach to introducing new capability to service.

FSFTs are used to validate damage tolerance analysis including the generation of test-based data for in-service inspection programs to detect damage in the fleet. Damage tolerance analysis is typically performed using deterministic methods based on the crack growth information generated from FSFTs. In this instance, the lead crack concept along with the interpolation and extrapolation methods shown in Fig. 1 could be applied to generate the master crack curves required for deterministic analysis. The master crack curves could be developed by either using correlated or corrected FCG codes against fatigue crack data from QF or using the λ parameter of the exponential equation with the rogue flaw as the initial crack size. Increasingly, damage tolerance analyses are being conducted employing risk analysis based on probabilistic methods[(4,5)]. The single flight probability of failure (SFPOF) methodology employed by DST[(22,23)] is one approach that could be employed to conduct such analysis. Again, the lead crack concept is well-suited to generate key input data to conduct SFPOF assessments of a particular structural component (e.g. using EPS to generate Equivalent Initial Flaw Size (EIFS) distributions).

The lead crack concept could also be used to assist in WFD assessments, possibly triggering this effort much earlier. Most certification requirements[(3–5)] typically favour an FSFT, which provides essential evidence to assist in assessing if a structure is susceptible to WFD and determines an appropriate operational life limit equivalent to a Limit of Validity (LOV)[2]. The use of the lead crack concept could be applied, for example, after one lifetime of testing to generate the necessary data to ascertain evidence of global fatigue damage in large areas of structure, whether characterised as multiple site damage or multiple element damage or both[3]. This will aid in a much quicker identification of WFD-susceptible structures. The data generated from the lead crack concept may then allow the manufacturer or regulator to determine whether sufficient data is available (supplemented with analysis where necessary) to establish the LOV of that structure or if further, more dedicated testing is required. If further test evidence is required, then it is possible to consider testing a full-scale portion of the structure, or a component in question subjected to representative loading rather than testing the entire airframe[(5)]. This may allow the manufacturer or regulator to consider multiple full-scale component-level tests to fully explore the influence of WFD. Testing components would also enable a_{Crit} to be determined.

As with good practice, a mid-life teardown and detailed inspection[(e.g. (24))] of a high-life airframe would mitigate many potential concerns with the above proposal.

4.0 CONCLUSION

FSFTs are a costly but necessary part of an airframe certification program. This paper has proposed and demonstrated a method to reduce the duration of test cycling required to achieve structural certification requirements based on the lead crack framework. It is argued

[2]LOV is defined as ‘the period in time up to which it has been demonstrated by test evidence, analysis and, if available, service experience and teardown inspection results of high time airplanes, that widespread fatigue damage will not occur in the airplane structure’[(5)].

[3]Possibly the most contentious issue is fastened lap-joints. The lead crack behaviour noted[(18)] is at odds with calculated nucleation times of similar joints reported in Ref.[(25)].

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that by trading cycling time for more a detailed inspection and fractographic examination, certification requirements can be met in a shorter timeframe with reduced costs. This would be a more cost effective and agile means of achieving certification.

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