THE LEAD CRACK CONCEPT 30 YEARS ON

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Abstract: Since the early 1990s extensive quantitative fractography (QF) of early nucleation and growth of airframe fatigue cracks that have resulted in accidents, or led to threats to structural safety, have shown that the largest – or "lead" – cracks often show approximately exponential growth. These observations have been formalised in the lead crack concept and the development of the Lead Crack Fatigue Lifing Framework (LCFLF). This has now progressed to a stage where it provides a robust and appropriately conservative method of assessing crack growth, enabling the setting of inspection and maintenance periods for a variety of in-service fatigue cracking problems in aircraft. The framework has been used extensively in airframe life predictions and/or life extensions by numerous airworthiness authorities.

The LCFLF relies on determining the equivalent crack-like sizes of the fatigue-nucleating discontinuities and the crack depths at known points in the fatigue lives. This method is flexible in that it may be pragmatically combined with fracture mechanics models of crack growth, provided they have been verified by actual measurements.

This paper summarises the current knowledge state for the LCFLF and provides some new examples of the framework directed at determining the crack growth history from limited quantitative fractography or in-service crack length measurements. The LCFLF has become an increasingly important tool for aircraft sustainment and fatigue failure analyses.

Keywords: Fatigue; Crack growth; Life prediction; Aircraft

INTRODUCTION

Metal fatigue is still a phenomenon that causes aircraft in-service and certification issues and is of much interest to the scientific as well as the aircraft structural integrity communities. Since the early 1990s (i.e., over 30 years), extensive fatigue crack investigations by quantitative fractography (QF) of cracking that has nucleated and grown in airframes early in a component's life, have shown that the largest – or "lead" – cracks often show approximately exponential growth behaviour. These cracks have sometimes caused accidents or led to threats to structural safety or caused certification test failures. These observations were formalised in the 'lead crack concept' and the development of the Lead Crack Fatigue Lifing Framework (LCFLF) [1]-[4]. The LCFLF has now progressed to a stage where it provides a robust and appropriately conservative method of assessing crack growth that enables the setting of inspection and maintenance periods for a variety of in-service fatigue cracking problems in aircraft, and is a valuable tool, in addition to other fatigue predictive and assessment tools. Numerous airworthiness authorities have used this framework in airframe life predictions and/or life extensions (e.g. [6]).

In the context of this paper, it is instructive to revisit the well-known Paris equation [7], namely:

$$\frac{da}{dN} = C(\Delta K)^m \quad (1)$$

Where a is the crack length at cycle N, ΔK is the stress intensity range (or similitude parameter), and C and m are nominally material constants (note: the units of m and C are interdependent).

Taking the natural logarithms:

$$ln\left(\frac{da}{dN}\right) = \ ln\, C + m\ ln\Delta K \qquad (2)$$

And integrating for a closed-form solution with a constant width correction factor β the following are found for the life of a crack:

$$a_f = a_0 e^{C\pi(\Delta\sigma\beta)^2 N_f} \qquad for\ m = 2 \qquad (3)$$

$$a_f = \left[a_0^{\left(1-\frac{m}{2}\right)} + N_f C(1-\tfrac{m}{2})\left(\Delta\sigma\beta\sqrt{\pi}\right)^m \right]^{\left(\frac{1}{1-\frac{m}{2}}\right)} \qquad for\ m \neq 2 \qquad (4)$$

where a_f is the final crack size and σ is the far field stress.

In the present paper, the long-neglected Equation 3 is of most relevance. This corresponds to the first crack growth equation, attributed to Head [7], an early researcher from the Australian Defence Science and Technology Group (then called the Aeronautical Research Laboratories, ARL). Subsequently Frost, Dugdale et al. [9][10], used Head's observation of self-similar crack growth and expanded Head's equation, thereby reporting that crack growth under constant amplitude loading could be described via a simple log-linear relationship (log crack growth vs linear life). Independently, Shanley also proposed the exponential model [11]-[15], and later the USAF [14][15] suggested this for small cracks to medium length cracks as an approximation of crack growth data available from spectrum fatigue tests.

The LCFLF relies on determining a crack stating point based on the equivalent crack-like sizes of fatigue-nucleating discontinuities [16]-[18] and the crack depths at known points in the fatigue lives. This method is flexible in that it may be pragmatically combined with fracture mechanics models of crack growth, provided they have been verified by actual crack growth measurements. This is why QF can be an important source of these data after fracture has occurred.

From the LCFLF several related practical fatigue prediction tools have been developed, including the cubic rule [3][19], the spectrum block-by-block approach [20]-[22] and the Hartman-Schijve equation [23][24]. These have been adequately described in the preceding references and will not be discussed further in this paper.

This paper summarises the current knowledge state for the LCFLF and provides some new examples of the framework directed at determining the crack growth history from limited QF data or in-service crack length measurements. The LCFLF has become an increasingly important tool for aircraft sustainment and fatigue failure analyses.

THE LEAD CRACK FATIGUE LIFING FRAMEWORK

Many years of QF of fatigue cracks in metallic airframe components from service and Full-Scale Fatigue Tests (FSFTs) have consistently shown that the dominant fatigue cracks (those that lead to the first failure) grow in an approximately exponential manner (e.g. [1]-[4][25][26]). These QF-based observations cover the nucleation of cracks and their early growth from a few micrometres to many millimetres in size. Evidence suggests that these 'lead cracks' commence growing shortly after the airframe components are introduced to the service-loading environment. Furthermore, these cracks usually nucleate from surface or near-surface production-induced defects or, less frequently, inherent material discontinuities (e.g. [16]).

If a particular area of a structure has the propensity to crack, usually due to high service stresses in that area, it is possible that cracks will nucleate and grow[1]. The fastest growing crack in this area is the (local) dominant or lead crack, given that they all have similar sized nucleating features that may be typical of these areas. There will most likely be several lead cracks within the entire structure, and one of these will ultimately cause failure of the structure unless appropriate measures are taken, such as repairs or replacements.

The general characteristics of these lead cracks can be summarised as follows:

1) They commence growing in high stress areas from (most usually surface) discontinuities soon after the aircraft is introduced into service. This implies that the cyclic stress intensity threshold (ΔK_{THRS}) in these areas is low;
2) Irrespective of local geometry, they grow *approximately* exponentially with time (i.e. log *a* (the crack depth or length) versus linear life or cycles) if:
   (a) little error is made when assessing the effective crack-like size of the fatigue-nucleating defect or discontinuity. For example, an error in underestimating the effective crack-like size of the nucleating discontinuity will cause an *apparent* departure from exponential behaviour over a short period of early fatigue crack growth (FCG);
   (b) the crack does not grow into an area of significant change in component thickness or geometry, particularly when the crack depth is small in comparison to the component thickness/width, and either before or after a change in thickness;
   (c) no significant load-shedding occurs (i.e. the crack is not unloaded as the component either loses stiffness and sheds load to surrounding members, or grows towards a neutral axis owing to the predominance of loading by bending);
   (d) the crack does not encounter a significantly changing stress field, e.g., it does not grow into or from a region containing residual stresses;
   (e) the crack is not retarded by very occasional and very high loads (usually more than 1.2 X the peak load in the spectrum); and
   (f) the small fraction of life involved in fast fracture or tearing near the end of the fatigue life is ignored (in modern alloys significant tearing usually begins during the highest load just before failure). In addition, the general failure/residual strength criterion of 1.2 X DLL (Design Limit Load), as required by DEF STAN 970 or Maximum Spectrum Stress (JSSG2006), would tend to eliminate this period of FCG from fatigue lifing calculations.

Within the limits given in 2) above, many observations of lead cracks have led to the following generalisations:

[1] Of course, it is possible that large or very large, but not very crack-like, discontinuities may exist that allow even a non-lead crack to reach failure prior to a lead crack growing from a small crack-like discontinuity. Such flaws can be better understood as rogue flaws, and tend to be rare, and should not be confused with the physically large natural discontinuities such as porosity, which is not usually crack-like [27].

3) The change-in-geometry factor β (which depends on the ratio of the crack length to width and the component geometry) with crack depth/length does not appear to significantly influence the FCG behaviour. For low K_t features, most of the life is spent when the crack is physically small so that β does not change much. However, even when a lead crack starts at an open hole, where β changes rapidly, it still appears to grow in an approximately exponential manner. (This is not to say that there is *no* geometry influence: under the same net section stresses, cracks from open holes grow faster than cracks from low K_t details.) The unexpected small or negligible influence of a high β gradient on the shape of a lead crack curve requires further research;
4) Typical nucleating discontinuities and defects for combat aircraft metallic materials (e.g. high strength aluminium alloys) are approximately equivalent to a 0.01 mm deep fatigue crack (see for example [18]). Thus, a 0.01mm deep flaw is generally a good starting point for FCG assessment in these materials[2];
5) despite limitation 2) (d) above, lead cracks often appear to grow exponentially within residual stress fields albeit at faster or slower acceleration rates depending on the sign of the residual stress;
6) if high loads that retard FCG occur periodically and fairly regularly throughout the life, then the average FCG behaviour will still be exponential;
7) although the critical crack size should be readily calculable, it has been observed that the typical critical crack depth for highly stressed areas in combat aircraft metallic components is usually about 10mm[3]. This can be a convenient approximation for use in life assessment. The same *appears to hold* for military transport aircraft in the absence of load-shedding.

EXAMPLES

In [26] twenty-three examples of FCG in FSFTs or incident aircraft were provided to demonstrate the exponential behaviour of lead cracks in airframes. The examples covered 23 spectra and stress levels as well as cracking in several materials. Table 1 and Figure 1 present additional FSFT FCG curves from the literature. Again, approximately exponential (i.e. lead crack) behaviour was observed. (Note that only one or two cracks from each FSFT were presented. The additional cracks in the references cited were all approximately exponential). Many of the cracks did not cause failure but represent end of test time. The exception was the F111 WDET AAS078 crack that failed the wing. From these data, of particular interest are:

a. For the C130J test two (2) example cracks are presented from a total of about 30: one that adheres to the lead crack concept; and one that has a delay or at least a period of slower early growth typical of a non-lead crack, and a subsequent period of slower growth due to load shedding when it became large. This ‘rollover’ is also seen at long crack lengths (>10 mm) for the F16 B11 Spar4 NLR crack, see point (c) from the general characteristics of lead cracks, noted above.

b. The Pc9 curve demonstrated the small fraction of the total life involved in crack tearing near the end of life (at crack depths that were more than 10 mm); and
c. The exponential region spanned the short crack to long crack regimes.

[2] To obtain good predictions using conventional FCG models from such a small initial crack size will need small crack growth rate data that have been validated for the fatigue predictive tool to be used. While such data are now becoming available, for example see [28]-[31], in the first instance the LCFLF does not require it.
[3] While many cracks in highly stressed components may exceed this size at final failure, it is usually observed that significant tearing has occurred well before failure. In other words, the FCG period beyond a crack size of 10mm is generally negligible.

Table 1: FSFT Crack Growth Data

	Aircraft FSFT	Material	Reference/Comment
1.	Aust Vampire	BS L65	[32] Wing
2.	C130J (RAAF)	7075-T7351	[33] CW1 fastener holes, fastest FCG. Evidence of load shedding. Multi-site damage scenario.
3.	B52G-H	2024-T351	[34] Lower wing skin from 38mm saw cut. (Note 2a measured includes fastener hole diameter). 1 load block was approximately 262 flight hours).
4.	F18-E/F Super Hornet Y383	Ti 6Al 4V (recrystallisation annealed)	[35] Wing splice
5.	Pc9 (RAAF)	2024-T351	[36] Spar lower flange aft hole 95
6.	F16 Block 15 (RNLAF)	7475-T735	[37] Spar cap and web cracks
7.	Aust F111 Wing	2024-T851	[38][39] Lower wing skin
8.	Aust F/A-18 centre barrel	7050-T74511	[40] 3500 hrs of in-service loading prior to loading with the mini-FALSTAFF spectrum

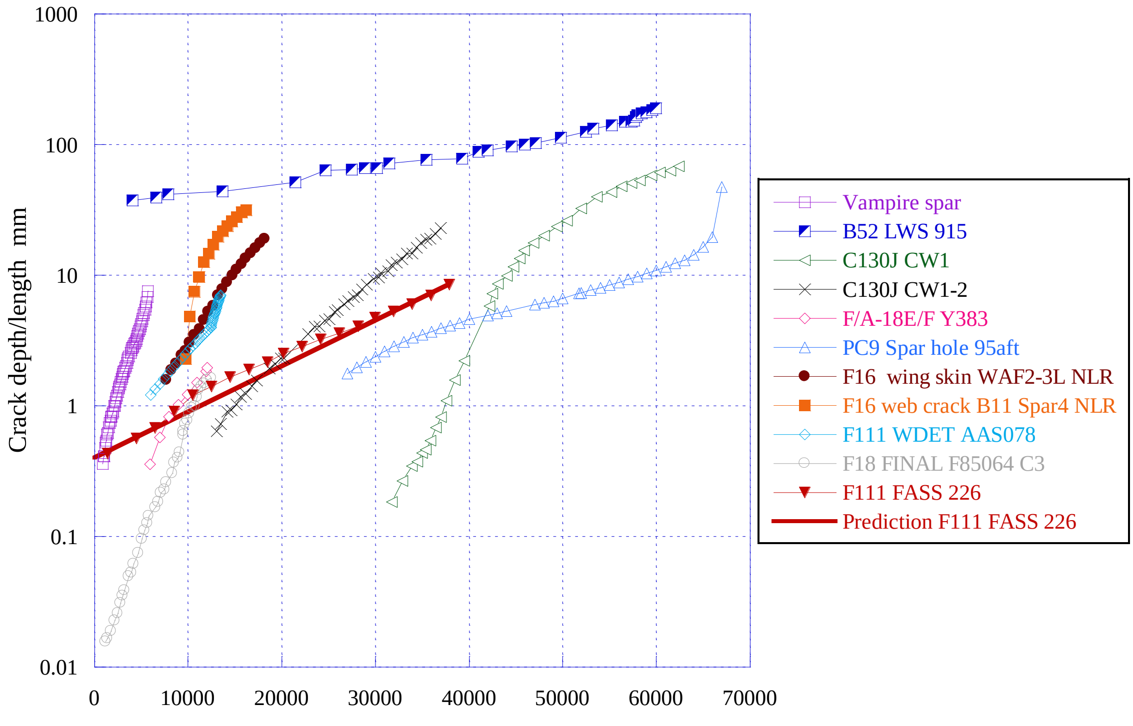

Figure 1: Selection of FSFT FCG curves

An example *prediction* for the F111 FWELD FASS226 crack is also provided in Figure 1 – the solid red line. The following was provided during the assessment of the cracking by the fractographer:

(a) At failure after 37,888 hours, the crack depth was 8.446 mm.

(b) The crack nucleated from a significant machining tear that occurred during preparation of the tapered hole. This tear was crack-like and approximately 0.4mm deep.

The growth model joins these two points in Figure 1 and provides a good fit to the actual data curve. More examples are provided in [3][19]-[27][41][42]. If the discontinuities that are usually present in a structure can be characterised in the form of crack-like features or crack depths in a structure, these starting points can be used to produce simple predictions of the lives of structures as shown above. From the LCFLF and its derivates mentioned above, estimates of the growth rates from stress measurements at the locations of interest; a known loading spectrum from FSFTs, service cracks, or even coupon data; and critical crack size estimates, may be combined to give predictive capabilities for these structural locations both in individual aircraft and fleet-wide..

CONCLUSIONS

After more than 30 years of data analyses and research, it is now clear that lead crack exponential growth behaviour is the norm, subject to the caveats provided above, for naturally nucleating fatigue cracks in military airframe structures under realistic loading conditions. This is an important practical observation that has led to programmes enabling significant life extensions in addition to solving fleet cracking problems. Furthermore, development of the Lead Crack Fatigue Lifing Framework (LCFLF) has led to several other fatigue life assessment tools that can be applied to aircraft fleets.

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