ELSEVIER
Available online at www.sciencedirect.com
ScienceDirect
Procedia Engineering 2 (2010) 363–377
Procedia Engineering
www.elsevier.com/locate/procedia

Fatigue 2010

The lead fatigue crack concept for aircraft structural integrity

L. Molent[a]* and S.A. Barter[a]

[a]*Defence Science and Technology Organisation, Melbourne, Australia, 3207.*

Received 4 March 2010; revised 9 March 2010; accepted 15 March 2010

Abstract

Over many years of quantitative fractographic examination of fatigue cracking from in-service and full-scale fatigue tests of metallic airframe components, it has been consistently observed that the largest cracks formed have grown in an approximately exponential manner. These crack growth observations range from the initiation of cracks and their early growth from a few micrometers through to many millimetres in length. It appears that these lead cracks commence growing shortly after the airframe is introduced to the loading environment. Furthermore these cracks usually initiate from production-induced or, less frequently, inherent material discontinuities. Based on these two observations, an aircraft lifing methodology that is based on the results of fatigue testing programs utilising the lead crack concept has been developed and implemented as an additional tool in the determination of aircraft component fatigue lives in several Royal Australian Air Force (RAAF) fleet types.
In this paper the lead crack concept is developed and its strengths and weaknesses are discussed. Examples of crack growth behaviour that are considered typical and representative of lead cracks are presented.

Keywords: quantitative fractographic, fatigue cracks, fatigue lifing, aircraft structural integrity

1. Introduction

The accurate prediction of the fatigue lives of metallic structures under variable amplitude (spectrum) loading, particularly for physically small cracks still presents challenges [1], especially since the demands on the designers and operators are for improved performance with reduced costs during manufacture and service. This leads to an overall requirement for lighter aircraft. The outcome of this requirement are highly stressed and efficient designs where fatigue cracking can arise at features such as shallow radii at the junction of flanges, webs and stiffeners, as well as at holes and tight radii. As a consequence, there are usually many areas that need to be assessed for their fatigue lives, and many potential places at which cracking may occur.

Fatigue is considered a complex phenomenon that is affected by many parameters, including (but not limited to) material properties, environment, geometry, surface treatments, load levels and sequences, load interactions, crack nucleation, growth rate scatter, grain sizes and others. To assess the many critical areas that may crack along with the sensitivity of the structure to the above parameters, the designer uses the results from base-line fatigue testing programs involving simple coupons. These coupons may be loaded with constant amplitude or representative

* Corresponding author. Tel.: 61 3 96267653; fax: 61 3 96267089.
E-mail address: Lorrie.Molent@defence.gov.au

doi:10.1016/j.proeng.2010.03.041

variable amplitude (VA) load cycles, and the configuration may or may not represent some feature of a built up structure.

It is rarely the case that the coupons are fully representative of the built-up structure, particularly the surface finishes that are produced during production. The results of these tests are averaged to give an indication of the life of the structure in a production aircraft. When the actual structure of the aircraft is considered, it will often be seen that each component of the structure has many features that have the potential to crack and it could be argued that each of these is typical of a single coupon. Hence, the real component's life is equivalent to only the shortest life of a group of representative coupons. This size effect is often addressed by other means. One way of addressing this is to test a fully built up structure, either representative of a component of the aircraft or the full airframe itself. Another way of improving the knowledge of how the aircraft will respond in service is to improve the coupon testing to make the coupons as representative as possible of the most fatigue critical details of the component, for example, applying surface treatments that produce representative manufacturing discontinuities. Having generated realistic cracks, further assurance of the coupon's representativeness can be produced by measuring the crack growth (CG) and characterising the initiating discontinuities and their populations. With this information, it is possible to use the CG curves to predict lives from larger or smaller discontinuities and this can be used as a basis for a methodology that can extend the usefulness of coupon, component and full-scale fatigue tests for more accurate life prediction of built up structures.

The fatigue cracking considered in this paper is that of the fastest or lead cracks in production quality aircraft structures, i.e. those cracks that will ultimately limit the life of the structure. A methodology is presented based on the lead crack concept that can be used for the interpretation of fatigue test results for the purposes of lifing aircraft components. This framework has been used to aid in the lifing of several Royal Australian Air Force (RAAF) aircraft and examples will be drawn from some of the fatigue test programs used as a basis for this lifing.

1.1. RAAF lifing criteria

In essence, the RAAF methodology for the lifing of aircraft primary structure requires the establishment of the test life of a full-scale or major component fatigue test loaded under representative loading to a residual strength (RS) requirement of 1.2 times Design Limit Load (DLL) without failure. Whether the test structure fails below 1.2DLL or survives[2], a method is required to determine the equivalent fatigue life as defined by a location's ability to achieve and survive 1.2DLL with cracking present. In effect, the test time (equivalent RS life) to the critical crack length or depth (a_{RST}) at the 1.2DLL point is required. For the case of a crack that fails the structure below the 1.2DLL the life of the critical crack is reduced to a time at which it reached a calculated RS test (RST) critical crack size, but for those cracks that survive the RST load some assessment of the amount of life remaining may also need to be carried out.

In some case due to concerns about test article survival during a complex full-scale fatigue test, it is often necessary to remove cracks or modify the areas that crack when the cracks are smaller than the size at which they would have failed under the RST. These areas are now the subject of fleet action prior to the desired life that was to be demonstrated by the fatigue test. This makes it possible to reassess the life for those areas modified during a test if the pre-modified CG of the area can be extrapolated to the RST critical crack size to establish what maybe thought of as a virtual test point for lifing purposes.

Additionally, the test may demonstrate adequate lives for those areas that had representative loading applied but it is often not possible to produce a test that is representative in all areas. So, when cracks form at non-representatively loaded areas, it may be necessary to establish the correct CG and RST level for these cracks by analysis, again necessitating the establishment of further virtual test points.

Finally, it may be found that the loading of the fleet has changed during the time between the fixing of the test loading sequence and the establishment of mature fleet loading. Again, this may result in further analysis of the cracks found in the test to establish new equivalent test lives. Each of these scenarios needs a framework of rules under which CG predictions can be made with the CG data that is available from representative coupon, component and full-scale fatigue tests. It is considered far better to utilise the CG results from a full-scale or large component

[2] This point could be conservatively used without further analyses.

fatigue test than to attempt a blind prediction from a VA spectrum using constant amplitude CG data, uncertain stresses, calculated geometrical factors and a CG model.

2. The lead crack

If a particular region of a structure has the propensity to crack, it is possible that a number of cracks will nucleate and grow. The crack in this region which grows the fastest is the lead crack. There will be most likely a number of lead cracks across the entire structure and one of these will ultimately cause the failure of the structure.

It has been observed by researchers at DSTO [2]-[5] (and elsewhere e.g. [6]-[9]) that approximately exponential growth is a common occurrence for naturally initiating lead cracks (i.e. those that lead to failure where the potential for many cracks exists) in airframes subjected to VA loading. Generally the features that these cracks display during their growth and failure may be summarised as:

1) They commence growing soon after the aircraft is introduced into service;
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-initiating defect or discontinuity (see Section 5.1). For example, an error in underestimating the size of the crack-like effectiveness of the initiating discontinuity will lead to a small departure from exponential for a small period near the commencement of CG.
    b. the crack does not grow into an area of significant change in the component's thickness (see Section 5.1), particularly if the crack depth is small in comparison to the component's thickness/width either before or after the change in section;
    c. no significant load shedding occurs (i.e. the crack is not unloaded as the part loses its stiffness and sheds load to surrounding members, see Section 5.2) or grows towards a neutral axis due to loading by bending;
    d. the crack does not encounter a significantly changing stress field i.e. grows into or from a region having residual stresses (see Sections 5.2);
    e. the crack is not retarded by very occasional very high loads (usually in excess of 1.2 x the peak load in the spectrum, see Sections 5.4); and
    f. the small fraction of life involved in fast fracture or tearing at the end of the life of a fatigue crack is ignored (see Section 5.5). This usually represents a small portion of the total life as tearing normally starts on the highest load just before failure. Additionally, the requirement of the 1.2DLL as the failure case would tend to negate this period of CG from any lifing calculation.

Within the bounds of the above, observations of the formation, growth and failure resulting from these cracks have led to the following generalisations:

3) the geometrical factor β (which depends on the ratio of the crack length to width of the specimen) does not appear to influence the nature of the CG significantly as may be expected. For low K_t features the majority of the life is spent when the crack is physically small so that the β does not change much. But even when the crack starts at an open hole and the β is changing rapidly, the lead cracks still appear to grow in an approximately exponential manner. This is not to say that there is no geometry influence, cracks from open holes grow at greater acceleration rates than from low K_t details at the same net section stresses. The expected influence of the high β on the shape of the CG curve appears minimal even when the crack is spending a considerable amount of its life in the region so affected. The reasons for this observation requires further research;
4) typical (mean) initial flaws for typical fighter aircraft metallic materials (e.g. high strength aluminium alloys) are approximately equivalent to a 0.01 mm deep fatigue crack (see for example [3][10]). In general a 0.01mm deep flaw is a good starting point for CG assessment;
5) cracks may also grow exponentially within residual stress fields (see for example [11][12] and Section 5.3);
6) should large loads that affect CG by retardation occur periodically throughout the life then on the average exponential CG may still result [2], although a rare overload may lead to a significant departure (or step-change) in the local CG rate (see Sections 5.4); and
7) whilst the critical crack size is easily calculated, it has been observed that for typical fighter aircraft metallic

materials the critical crack depth is typically about 10 mm (see for example [3][10]) for highly stressed areas[3].

Exponential growth from small discontinuities of about 0.01mm in depth can be found to be a common behaviour for numerous materials used in different aircraft types [3]. Exponential CG is also observed for a range of loading spectra types (e.g. tension dominated, compression dominated, fighter aircraft, transport aircraft etc), various geometries e.g. [13], including fuselage lap-joint splices [4], and for crack sizes from a few micrometers through to many millimetres. An example is provided in Fig. 1 where CG data was obtained using quantitative fractography (QF) from cracks found in a lower wing skin of an F-111 test article [14] at the end of the test. This test article had also experienced the equivalent of 4500 hours of service life before testing. From this data it could be determined that:

- in most cases the cracks had grown from very early in the life of the wing;
- that cracks had grown at numerous locations covering two-thirds of the wing span showing that despite variations in geometrical detail and span-wise location that the CG rate was generally similar (a trend that has been observed for other aircraft [3] where stressing is similar); and
- that the data when presented on a log crack depth versus linear life plot was generally exponential.

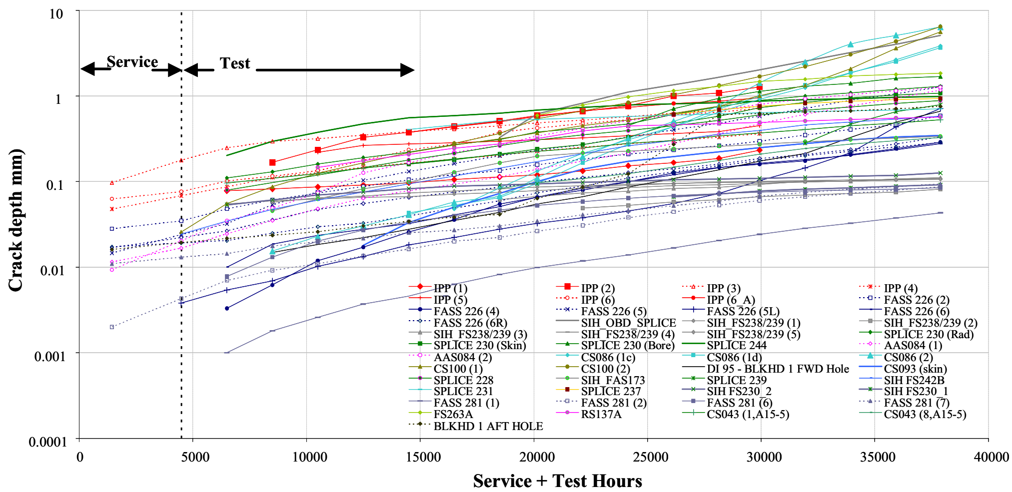

Fig. 1. A sample of crack growth curves from the AA2024-T8 lower wing skin of an F-111 test article, described in [14]. Note each point represents the crack growth increment for one block. (CS = central spar, BLKHD = bulkhead, FASS = Forward Auxiliary Spar Station inches, IPP = Inner Pivoting Pylon, RS = Rear Spar).

There are advantages in such a method of presenting data. Firstly, the growth of the cracks while small is clearer than a linear/linear presentation. Secondly, given the similar CG rates it can be seen that the major source of scatter is the effectiveness of the initiating discontinuities at establishing a crack-like defect.

Thus with the appropriate use of initiating defect size (see Section 5.1) and a knowledge of the final crack size, a simplified CG curve can be derived, based on the expectation of immediate initiation and exponential growth. From this, a crack size at some other life may be determined by extrapolation. As will be shown, the framework supporting this method is considered to provide a conservative life estimate [5].

3. Crack initiation

---

[3] While many cracks in highly stressed components may exceed this value at the time of failure it is usually observed that significant tearing has occurred prior to this.

Some of the practical aspects of fatigue crack initiation in service aircraft will be briefly discussed in this section. Compared to the carefully prepared surfaces typical of laboratory fatigue specimens, production aircraft structures have many more inherent surface discontinuities. When these discontinuities are sufficiently crack-like and the local cyclic stresses generated by the service loads applied are sufficiently high, initiation will be very early in an aircraft's service life. These discontinues include:

- machining damage in the form of grooves, small surface tears and nicks, scratches from the removal of tools (drills, reamers etc.) and handling during assembly, burrs and poorly de-burred holes;
- broken inter-metallic particles at and just below the surface of machined or abraded surfaces, porosity within the material; and
- in the case where the surface has been treated either by; chemical treatments - etch pitting or by mechanical blasting – folds laps and embedded particles.

While generally these defects are very small, of the order of 0.01mm deep they can initiate cracks very quickly after the entry of the aircraft into service. For structures with many critical features (e.g. bolt holes) or large uniformly loaded areas of high stress many cracks may initiate, and some of these will be larger than the others and become the critical cracks that determine the life of the structure (i.e. the lead cracks). Some examples of these features are shown in Fig. 2. The near immediate growth of cracks from discontinuities of the order of 0.01mm when first subjected to cyclic loading has also been noted by others e.g. [15].

In the general literature there is also the view (e.g. [16]) that a significant period of initiation is required before CG occurs and that this period is taken up with a process of slip formation which at a surface results in slip bands that intrude and extrude from the surface with the intrusions eventually forming into cracks. Formation of cracking at grain boundaries and in or about intact second-phase particles of carefully polished laboratory specimens has also been noted, but in reality this mechanism is not relevant to many aircraft structures. Indeed, a insignificant period of crack initiation has been found for the crack that is critical to the airframe structure, as found during many full-scale fatigue tests and several aircraft accidents where fatigue cracking was the cause [3]. Notwithstanding this, for carefully finished parts typically in engines or for those components that are stressed at low levels, such as in the many parts of helicopters or secondary parts of an airframe or control system, there may be a significant delay in the formation of cracking while either the above degradation occurs or some service induced, fretting, corrosion or maintenance damage etc. takes place.

An example that shows the size of typical initiating discontinuities and the lack of a delay in crack formation is provided here. The results of a fatigue test program [17] where up to five low K_t AA7050-T7451 specimen sets were tested at four stress levels, and where the CG was collected by QF on a spectrum block-by-block basis is summarised here. The CG results are shown in Fig 3 where all the curves appear to start shortly after the first application of the loading, as indicated by the blocks being detected down to close to 0 hours in most cases. The exceptions being the data for specimens KS1G3 and KS1G66 in which an apparently significant period of initiation is seen. However, fractographic examination of these failure surfaces indicated that the initiating discontinuity was sub-surface. It is well known that in this material the CG for the early period would be expected to be slower than for surface breaking cracks due to the absence of the laboratory air environment. Once the crack front breached the surface of the material the CG rate soon became very similar to those coupons (tested at the same net section stress) where the cracks had initiated from surface breaking discontinuities. Early crack retardation (over the lead cracks) may also be noted when the applied stress level decreases due to several other causes. These may be:

- poorly oriented grains in the location of the initiating discontinuity and/or slightly harder grains;
- off-plane crack initiation about the discontinuity producing a tortuous initial crack front, non-optimum shaped discontinuities; and
- cracks initiating at grain boundaries [7] which experience difficulty in transitioning into the grains, as well as others reasons outside the scope of this paper.

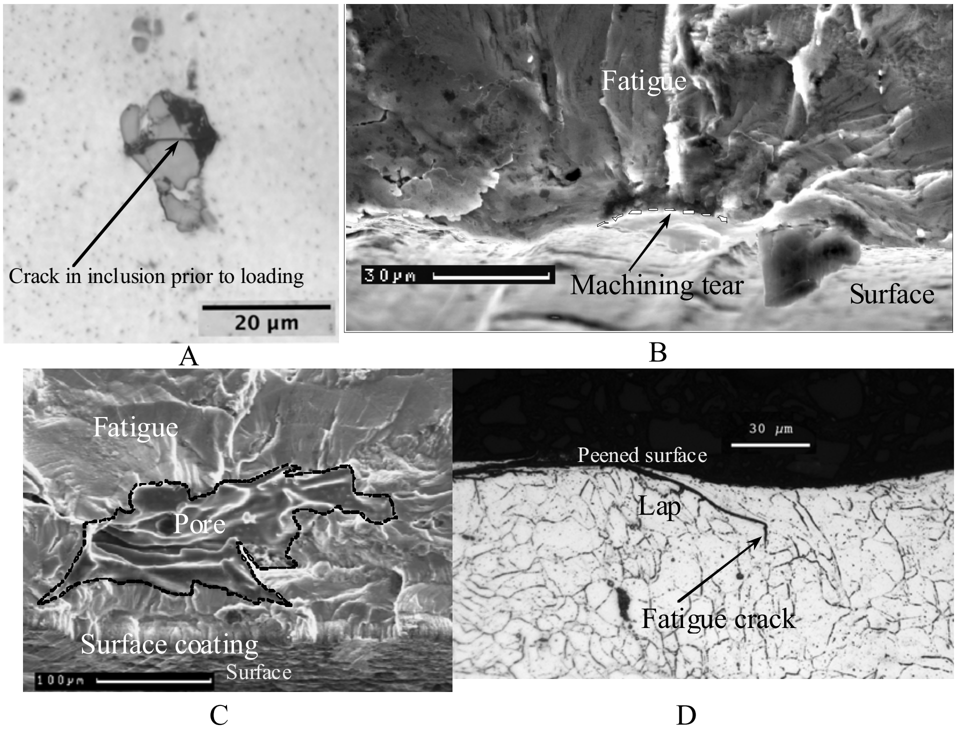

Fig. 2. Some examples of discontinuities that may exist in a metallic airframe at the time it enters service. Each of these discontinuities will act as an effective crack starter reducing initiation to a few effective load cycles.

The lead crack methodology conservatively ignores this behaviour (i.e. the period of nucleation, slow growth and increase in acceleration rate until the lead crack acceleration rate is approached or achieved) since the lead crack from (say, for the 324.1 MPa stress level) KS1G36 would have already failed the aircraft in the example given in Figure 3.

4. Fatigue crack growth

4.1. Approximately Exponential Crack Growth

The lead crack is said to grow in an approximately exponential manner given the assumptions and simplifications stated previously. But what is meant by approximately exponential CG? This statement is intended to indicate that, for engineering purposes measured CG data appears to be well represented by a straight line when presented on a log crack depth versus linear life graph. However, it is acknowledged that this is a simplification. For example, when the data shown in Fig. 2 are plotted in rate terms; the increment of CG per block of loading against the corresponding crack size (a), it can be seen that there is scatter within each stress level. Some of this scatter may be the result of measurement difficulties, and some may be also due to the local material variability influencing CG. There are usually small changes in the growth rate which are due to local changes in the material such as grain orientations, grain boundary crossings, local second phase particles, local closure effects due to changes in the local surface roughness etc. Thus when the crack is growing at its slowest and sampling fewer grains, these variations are at their largest and the CG will vary most while the crack is small, i.e. 1mm or less in depth in the aluminium alloy data shown in Fig 3. Additionally, since the data in Fig 3 are for multiple stress levels a stress dependency can also be noted [13]. However, the trend-lines produced by fitting power curves to the individual stress level data sets indicate an approximate slope of one, i.e. the growth rate is approximately proportional to the crack depth, again indicating approximately exponential CG over two orders of magnitude (e.g. Fig. 1 and Fig 3) in crack size.

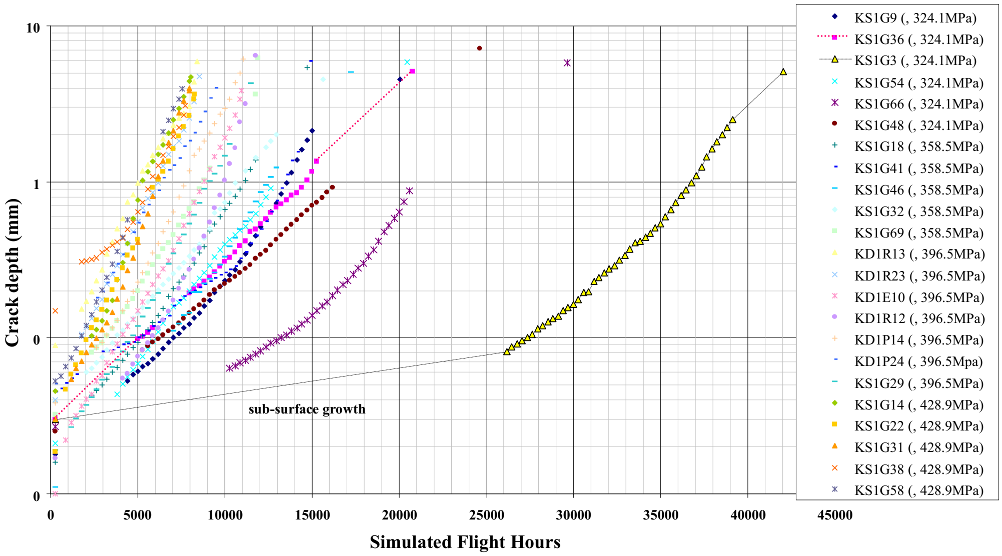

Fig 3. AA7050 T7451 crack growth data for fighter spectrum for four stress levels. Note each point represents the crack growth increment for one block. One block represented approximately 300 airframe hours (AFHRS) of a modern fighter loading sequence.

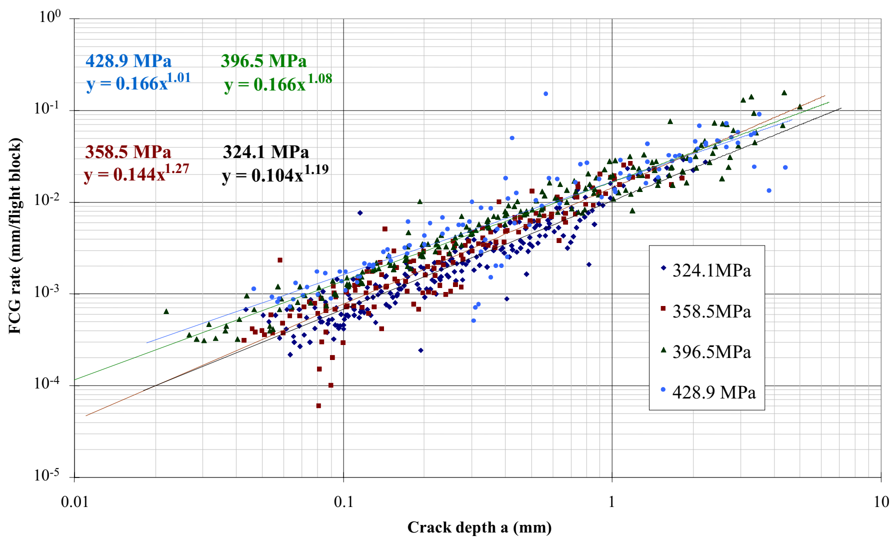

Fig. 4. Crack growth rate data from Fig 3.

5. Exceptions to exponential crack growth

This section will shown that the lead crack methodology is applicable for cases where approximately exponential CG may not be apparent.

5.1. Early crack growth

Because of the small sizes of typical crack initiating discontinuities and the sometimes irregular shape of these (as noted in Section 3), it can be difficult to measure the depth of a discontinuity and estimate how crack-like it was. Murakami and Endo [18] to [21] suggested that the surface area of the discontinuity may be used to characterise the discontinuity. In some materials, this may be satisfactory, but due to other considerations, noted in Section 3, this will not always be applicable. The use of the CG about the discontinuity for the lead cracks, by back-projection to zero hours of the data from the established exponential growth is one way to estimate the crack-like effectiveness of the discontinuity. Such a method is referred to as the Equivalent Pre-crack Size (EPS), see [10]. An example of this method is given below.

Normally the fractographic measurement commences from an apparent boundary on the discontinuity/fatigue interface and advances along the crack surface away from the origin in an approximately straight line through a section where the fatigue progression marks are most apparent (e.g. [11]) towards the deepest point of the cracking. When the data are plotted, in this case for seven cracks, a graph like that shown Fig. 5A can result. Here a departure from overall approximate exponential CG is seen as a higher early CG rate. However, when the best estimate of the depth of initiating discontinuity is added to the measured crack depth an approximate exponential growth is usually produced as shown in Fig. 5B. For the lead cracks, this approach is very effective, but as can be seen in the long life cracks (KSIG3 and KSIG66) shown in Fig 3, this method will not always be useful at correcting early CG for non-lead cracks.

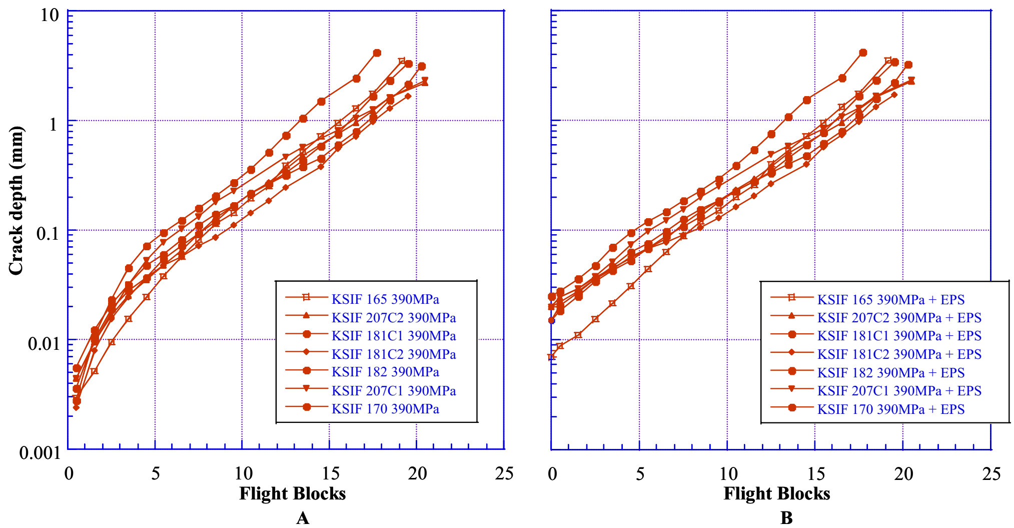

Fig. 5. Crack growth curve plots for AA7050-T7451 coupons with log depth versus the number of applied fighter spectrum blocks without correction to the crack depth 'A'. This shows the fall-off in crack growth near the origin, and 'B' shows the same cracks with the estimated depth of the initial flaws added. There is now no fall-off in the growth curves [22].

5.2. Geometry Effects

As noted above, as a crack travels through a component, changes in geometry can cause changes in the rate of acceleration of the CG. A change from a thick section to a thin section may increase the rate of growth and, conversely, from a thin section to a thick section, decrease the rate of CG. In many cases such section changes are not important since they occur when the crack is already growing very fast and is near the end of its life. In such cases the change from a single initial exponential rate can be conservatively ignored. Where the change occurs earlier and produces slower CG after the change, the earlier faster rate of growth is generally used to ensure that the life calculations are conservative, although such a decision should be taken on a case-by-case basis.

A second geometric effect may come from load shedding to adjacent structure. This may or may not produce cracking in this secondary structure, but the stiffness change will influence the primary crack leading to deviation from exponential CG. An example of this can be seen from the CG results for cracking in the wing splice area of an early F-111 fatigue test. The wing splice consisted of a D6ac steel plate fastened to a AA2024-T8 plate. As the AA2024-T8 plate cracked from one hole, fastenings at adjacent holes restricted the compliance at the cracking hole resulting in the departure from exponential CG seen in Fig 6. Exponential CG can still be used for lifing purposes for the lead crack, with the faster growth from the early part of the lead crack giving a conservative estimate of the life.

Load shedding due to local bending may also be a cause of a slowing of the CG rate as the crack gets deeper. Again for conservatism the earlier rate of growth is usually projected forward to the established critical crack size to give a predicted total life.

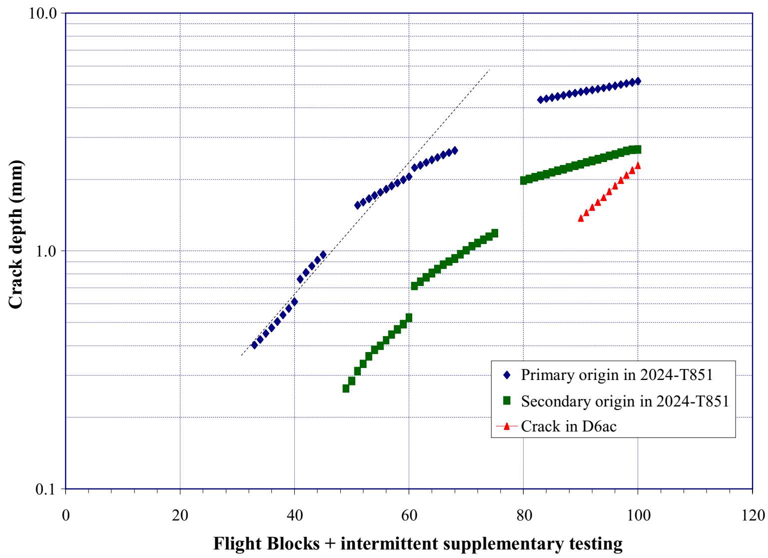

Fig 6. Crack growth from the wing pivot splice during the F-111 aircraft full-scale fatigue test, adapted from [23]. Note proof load cycling at 40 and 60 blocks results in tearing which produces step changes in the crack growth history. One block was approximately equivalent to 400 AFHRS.

5.3. Residual stresses

One example of residual stress affecting CG is that produced by shot peening. The resulting compressive residual stresses retard CG when the crack is small and thus enhance the life of a peened area. Examples of CG through compressive residual stress layers produced by the glass bead peening of AA7050-T7451 coupons is shown in

Figure 7. The CG curves show a significant reduction in the CG rate up to a crack depth of approximately 0.25mm. The subsequent or latter CG rate was consistent with the entire growth rate of unpeened coupons of the same configuration tested with the same spectrum and loads [11]. This indicates that the peened coupons maintained a lower exponential CG rate prior to exiting the residual stress layer, followed by a transition to a higher exponential rate similar to the non-peened rate i.e. a piece-wise exponential CG rate, see Figure 7. This example shows that approximately exponential CG behaviour can apply within a residual stress field. With the knowledge of the depth at which peening becomes ineffective, a piece-wise exponential approach can be used to estimate the time taken for a crack to reach critical depth.

Examination of the peened specimens revealed that the discontinuities initiating fatigue cracking were the result of the peening process [11]; and included laps, folds, cuts and embedded fractured glass beads at the surface of the specimens. An example of one of these types of discontinuities initiating a small fatigue crack is shown in Fig. 2D. A larger initial crack size in the case where residual stresses are retarding CG, based on an analysis of coupon data such as that presented in Figure 7 will need to be considered since using a smaller EPS typical of non-retarded CG (i.e. approximately 0.01mm) would extend the period that the crack stays within the residual stresses influence, and therefore extend the estimated life in a non-conservative manner. Contrary to this observation for peened surfaces, the selection of a small EPS as a preferred starting point in cases where no residual stress field exists, which is the case for most lead exponentially growing cracks, is conservative since it increases the calculated exponential growth rate [5].

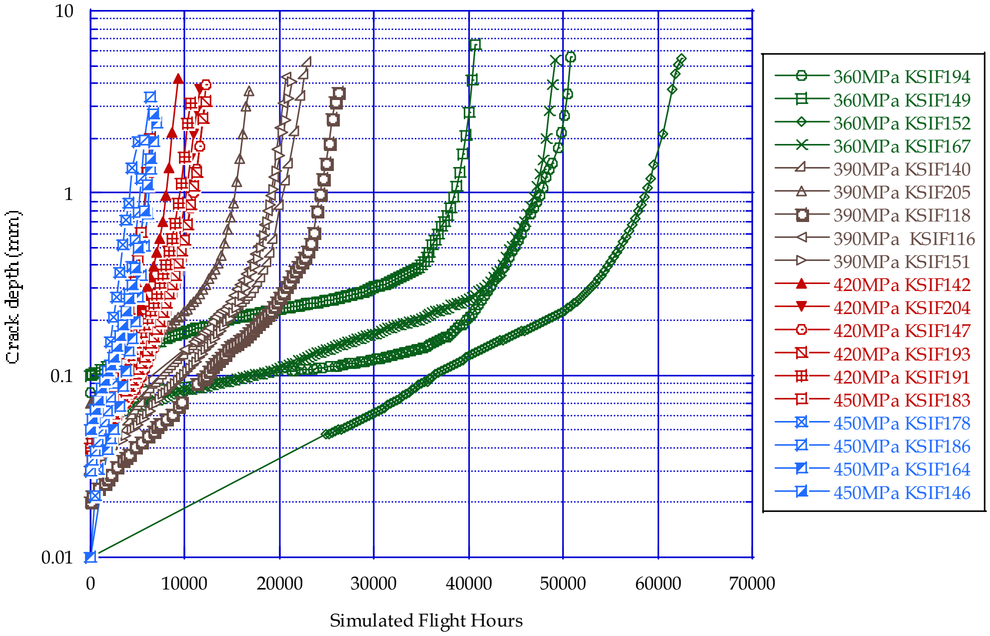

Fig. 7. Crack growth history data for shot-peened AA7050 coupons tested under a fighter spectrum at several stress levels [11]. Note each point represents the crack growth increment for one spectrum block.

5.4. Retardation from high loads

The retardation of the CG rate after the application of infrequent high loads is well known. It is also known that the general transport aircraft spectrum TWIST leads to significant retardation at certain levels of peak stress in thin sheets [24]. Wang and Blom [25] reported the CG data for 4mm thick AA2024-T3 centre crack panels shown in Fig.

8. The centre slots were not cyclically pre-cracked before testing. It is evident that some of the data does not appear exponential. The CG curve for C4B9 (260 MPa) meets the lead crack criterion; while retardation affects the CG at lower stresses (130 and 195 MPa). In the latter cases, trend-lines for each stress level can still provide a reasonable description of the over-all CG behaviour (see Fig. 8). Alternatively, separate exponential trend-line equations can be used for the non-retarded and retarded rates, to form the CG curve upon which a lifing calculation may be made. This retardation is an interesting problem in itself, since there is no guarantee that such an isolated high load will in fact ever be applied to a particular fleet aircraft. For this reason it is often prudent to clip such high loads in a test spectrum so that a false estimate of the life of the airframe is not developed due to retardation effects.

In contrast to the apparent retardation seen in some of the data in Fig. 8, no such discrete retardation is seen in Fig. 1. The test wing from which this data was obtained was subjected to a 1.2DLL of approximately 7.33g (where the peak spectrum load was approximately 6g) at each block as discussed in [14]. The reason for the lack of apparent retardation was that these 1.2DLLs were applied regularly at reasonably short (2000Hr) intervals in both the test and the fleet, the wing skin was thick and the critical crack size was about the thickness of the skin. To investigate the effect of the regular application of high loads CG experiments were conducted using spectra with and without the over-load. The CG results clearly demonstrated an increased life in the case of the data that included the 1.2DLL [26]. It would appear that the CG plotted on a block-by-block basis, where the block was defined as all the loads up to a repeat in the 1.2DLL, produced an average CG per block over the life of each crack that indicated exponential growth. Since the fleet received the same 1.2DLL at the same interval, these CG curves were considered appropriate for fleet lifing purposes.

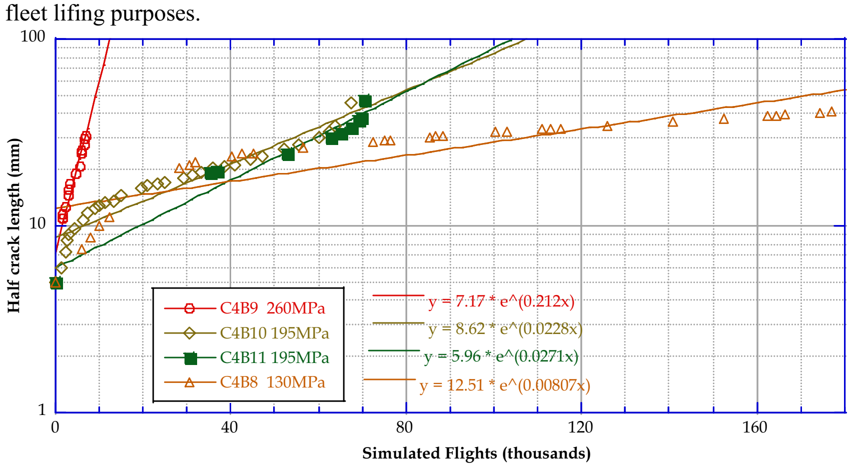

Fig. 8. Crack growth from 4mm thick CCT panels subjected to the TWIST spectrum, adapted from [25]. (TWIST = 4000 flights/block).

The application of a high load regularly in a fatigue test, leading to an overall single exponential rate as indicated above was consistent with the observations of an experiment on AA7050-T7451 using a round hourglass type specimen. A very high tensile load (in comparison to other loads in the sequence) was applied every 200 constant amplitude (CA) loads of R=-1. These high loads had a profound effect (Fig. 9) on the local CG, although, when the crack was measured over its entire crack length, a much more uniform effect was evident. The CG directly after the application of the overload, was found to have increased over the growth rate prior to the overload. This increase in growth then decayed rapidly until the growth rate had dropped below the CA growth rate prior to the overload. The effect of the retardation then decayed until the growth rate had stabilised at a new acceleration rate depending on the crack depth (crack tip K), prior to the next overload. Although the growth rates varied markedly between overloads, measuring the crack depths of each of the overloads by QF revealed that the overall CG was exponential until the overload started to grow by tearing (at about 1mm in crack depth), Fig. 10.

5.5. Quasi-static fracture

While the exponential relationship appears to be a reasonable approximation over most of the life of many typical fatigue cracks in primary aircraft structure, growth rate acceleration towards the end of life can cause an upward trend in the CG rate. From the fractographic observation of many of these cracks this appears to be accounted for by the onset of static modes of fracture [27] such as inclusion fracture, local tearing and ultimately tearing along the entire crack front. More rapid crack acceleration may also be the result of changes in the geometry of the section through which the crack is traveling and the coalescence of other cracks growing nearby the main crack to form a larger crack. All of these influences are most strongly evident towards the end of the life of the crack, since this upward deviation from a single exponential growth rate generally occurs near the end of CG. Whilst this behaviour may occupy a considerable length of the total growth, it only accounts for a small fraction of the total life of the crack and thus this period of growth may be ignored under the lead crack method. This is demonstrated by the data in Fig. 11, where Wanhill [28] evaluated the fatigue performance of 160mm wide centre notched 2048-T851, 2024-T3 and 7075-T6 aluminium alloy sheets. The thicknesses were 3.3 mm, 2.9 mm, and 4.0 mm respectively, and they were tested under the FALSTAFF spectrum loading with maximum stresses of 196 and 245 MPa. The acceleration over a small fraction of the total growth at the end of life is clearly seen.

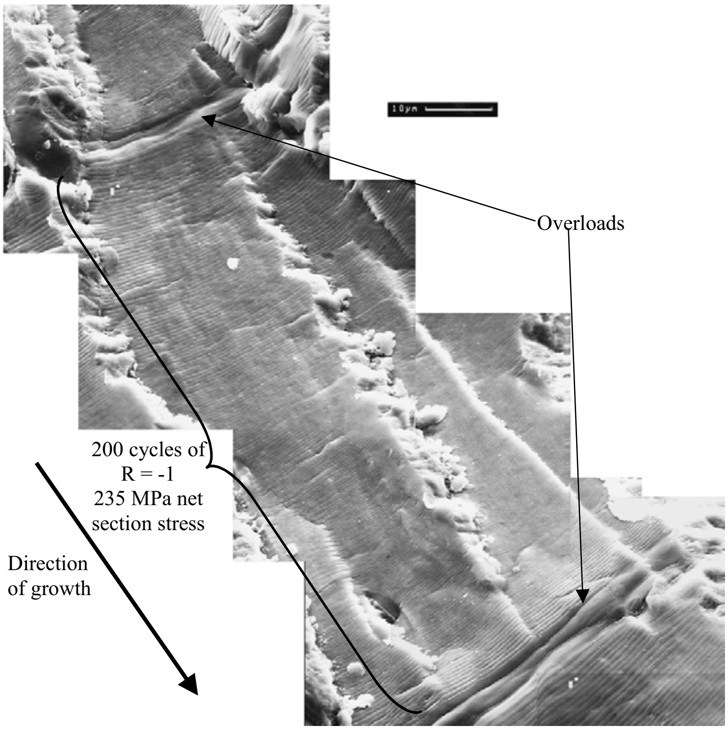

Fig. 9. A view of a region of crack growth produced in a AA7050-T7451 round specimen loaded with blocks of 200 cycles of constant amplitude followed by one overload cycle of 20,000με.

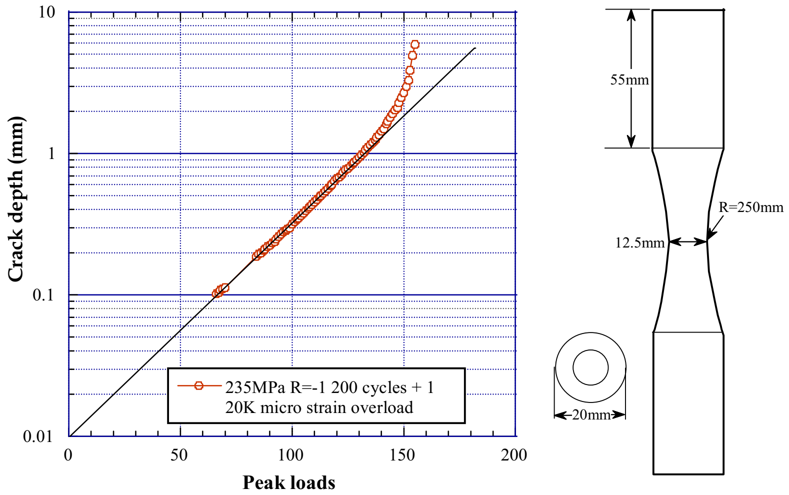

Fig. 10. A schematic of the specimen used and a plot of the crack growth in a 7050-T7451 round specimen loaded with blocks of 200 cycles of constant amplitude followed by one overload cycle of 20,000με.

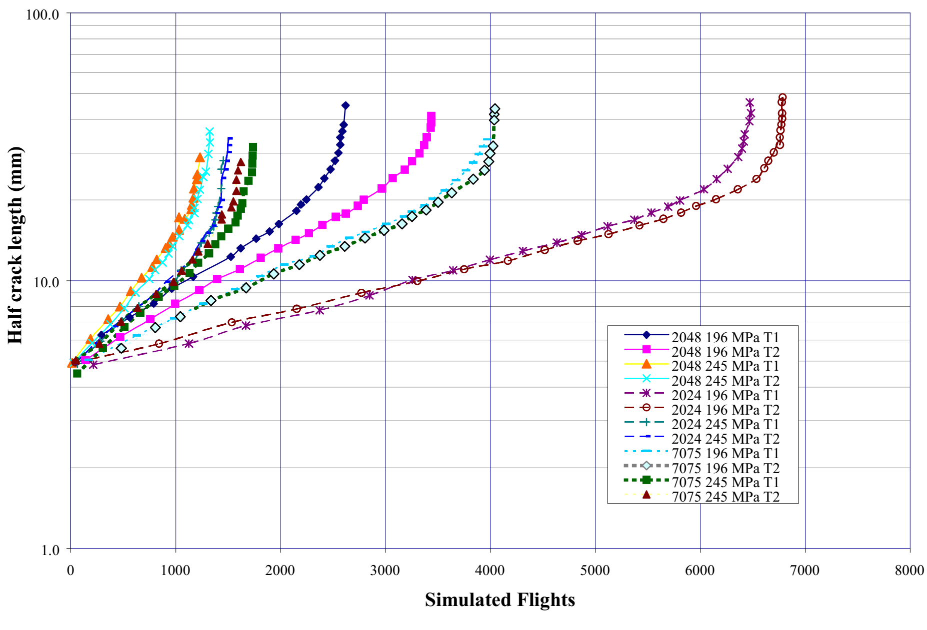

Fig. 11. Measured crack growth histories for AA2048-T851, AA2024-T3 and AA7075-T6 under FALSTAFF loading, adapted from [28].

6. Lifing methodology

The lead crack concept of CG has been developed into a methodology that can be used to calculate virtual test points from full-scale and large component tests, fleet cracking or from service-induced cracks found in teardown articles taken from the fleet and in some cases from cracking in highly representative coupons. The methodology may be separated into three distinct assessments: Crack initiation and the discontinuity's crack-like effective size (EPS here), the nature of the CG and the RST crack size under 1.2DLL in the area being considered.

Putting these three aspects together with the assumptions of CG from entry into service, exponential growth or piece-wise exponential growth and either a demonstrated (preferred) or analytical (K_{Ic}) RST crack size for the 1.2DLL case, virtual test points can be established for locations that do not have test demonstrated life by RST failure available. By way of an example, the benefit of the lead crack methodology can be seen in Fig. 12, where only a simple extrapolation was required from the point at which test cycling was effectively stopped in this structural location due to the implementation of a repair, up to the point at which it would have failed the 1.2DLL criterion had the repair not been applied. In this case, the CG data are consistent with the cracking having started immediately at the beginning of the test and that the CG was well modelled by an exponential equation. The a_{crit} was conservatively chosen as the thickness of the web since it was thought that the CG rate would increase after becoming a through crack and the new rate was unknown.

It is considered that the lead crack methodology can be used in the cases of non-overall exponential CG considered (Section 5) by choosing an appropriate exponential CG rate. Methods of adapting the observations of exponential CG for conducting predictions of yet-to-be-tested fatigue specimens under VA loading are summarised in [30].

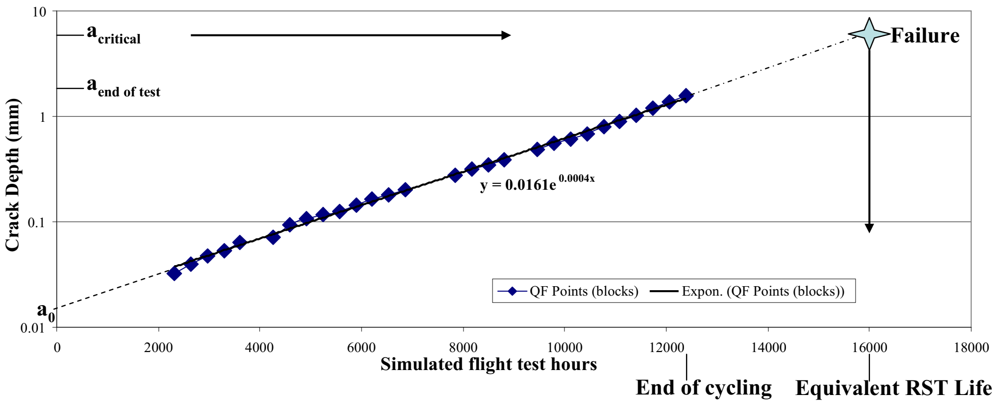

Fig. 12. Schematic of the growth of a typical crack (data from [29]) from an F/A-18 full scale fatigue test showing the crack depth versus time history for blocks of loading as determined through QF and then extrapolated to the estimated critical crack depth for the residual strength load.

7. Conclusions

This paper has considered fatigue crack growth in several airframe materials tested under several variable amplitude spectra to demonstrate that lead cracks grow in an approximately exponential behaviour. A framework was articulated to summarise the conditions and assumptions for which exponential crack growth may be applied to the lifing of airframe components. Several groups of example data were presented for which exponential cracking was not immediately apparent, however, it was considered that the lead crack framework was still useful in these cases. Finally, a simple means of utilising the lead crack framework for the interpretation of fatigue test results was presented.

References

[1] Schijve J. Fatigue of structures and materials in the 20th century and the state of the art. Fatigue, 2003; 25: 679-702.
[2] Barter SA, Molent L, Goldsmith N, Jones R. An experimental evaluation of fatigue crack growth. Engineering Failure Analysis, 2005, 12/1: 99-128.

[3] Molent L, Barter SA. A comparison of crack growth behaviour in several full-scale airframe fatigue tests, Fatigue 2007; 29: 1090-1099.
[4] Jones R, Molent L, Pitt S. Understanding crack growth in fuselage lap joints. Theoretical and Applied Fracture Mechanics, 2008; 49: 38-50.
[5] Molent L, Singh R, Woolsey J. A method for evaluation of in-service fatigue cracks, Engineering Failure Analysis, 2004; 12/1: 13-24,
[6] Underhill PR, DuQuesnay DL. The effect of dynamic loading on the fatigue scatter factor for Al 7050, Fatigue, 2008; 30: 614–622.
[7] Liao M, Benak T, Renaud G, Yanishevsky M, Bellinger N, Mills T, Prost-Domasky S, Honeycutt K. The development of short/small crack model for airframe materials. Proc. Aging Aircraft Conference, Phoenix, 21-24 Apr 2008.
[8] Macheret Y, Koehn P. Effect of improving load monitoring on aircraft probability of failure. IEEE, Big Sky, Montana, 2007.
[9] Mohanty JR, Verma BB, Ray PK. Prediction of fatigue life with interspersed mode-I and mixed-mode (I and II) overloads by an exponential model: Extensions and improvements, Engineering Fracture Mechanics, 2009, 76: 454–468.
[10] Molent L, Sun Q, Green A. Charcterisation of equivalent initial flaw sizes in 7050 aluminium alloy, Fatigue and Fracture of Engineering Materials and Structures, 2006, 29: 916-937.
[11] Barter SA. Fatigue crack growth in 7050T7451 aluminium alloy thick section plate with a glass bead peened surface simulating some regions of the F/A-18 structure, DSTO-TR-1477, Defence Science and Technology Organisation, Melb., Australia, 2003.
[12] Walker K, Weller S, Walker J. F-111 wing pivot fitting upper plate critical features fatigue assessment - pre wing optimisation modification, DSTO-TR-1682, Defence Science and Technology Organisation, Melbourne, Australia, Feb 2005.
[13] Huynh J, Molent L, Barter S. Experimentally derived crack growth models for different stress concentration factors; Fatigue, 2008; 30/10-11: 1766-1786.
[14] Boykett R, Walker K, Molent L. Sole operator support for the RAAF F-111 fleet. 11th Joint NASA/FAA/DOD Conference on Aging Aircraft, Phoenix, 21-24 Mar 2008.
[15] Murakami Y, Miller KJ. What is fatigue damage? A view point from the observation of low cycle fatigue process. Fatigue 2005; 27: 991–1005.
[16] Suresh S. Fatigue of Materials. Cambridge University Press, USA, 2001.
[17] Pell RA, Mazeika PJ, Molent L. The comparison of complex load sequences tested at several stress levels by fractographic examination, Engineering Failure Analysis, 2005; 12/4: 586-603.
[18] Murakami Y, Endo M. Effects of defects, inclusions and inhomogeneities on fatigue strength. Fatigue, 1993. 16(3), pp163-182.
[19] Murakami Y, Endo M. The behaviour of short fatigue cracks. Editors K. J. Miller and ER. de los Rios (EGF,1986) , 1986: 275-293.ESIS.
[20] Murakami Y, Endo M. Effects of hardness and crack geometries on □Kth of small cracks emanating from small defects. The behaviour of short fatigue cracks, EGF1, (Edited by KJ. Miller, and ER. de los Rios), Mechanical Engineering Publications. 1986: 275-293.
[21] Murakami, Y. Effects of small defects and nonmetallic inclusions on the fatigue strength of metals, JSME International Journal, JSME, 1989, 32/2: 67-180.
[22] Barter SA. The fatigue of a high strength aluminium alloy in air – quantitative fractographic methods and life prediction, PhD dissertation, Monash University, Australia, 2008.
[23] Anon, F-111 A-4 left hand wing fatigue test, Part I – basic test phase blocks 1 through 60, Part II – supplementary test phase, and Part III – Block 61 and on, Dec 71 to Apr 72. FZS-12-327, Lockheed Martin, USA, Dec, 1971.
[24] Wanhill RJH. Gust severity effects on fatigue crack propagation in aluminium alloy sheet materials. NLR-MP-78041, The Netherlands, 1978.
[25] Wang GS, Blom AF. A strip model for fatigue crack growth predictions under load conditions. Engineering Fracture Mechanics, 1991; 40, 3: 507-533.
[26] Diab H, Goldsmith N. Fractography results of F-111 loads interpretation and truncation validation (LITV) coupon test program. DSTO-TR-2000, Defence Science and Technology Organisation, Melb., 2007.
[27] Ritchie RO, Knott JF. Mechanisms of fatigue crack growth in low alloy steel. Acta Metallurgica, 1973; 21:639-648.
[28] Wanhill RJH. Damage tolerance engineering property evaluations of aerospace aluminium alloys with emphasis on fatigue crack growth, NLR TP 94177, The Netherlands, 1995.
[29] Barter SA. Fractographic investigation report FFT55 0030, Defence Science and Technology Organisation, Melb., Australia, Feb 2004.
[30] Molent L, McDonald M, Barter S, Jones R. Evaluation of spectrum fatigue crack growth using variable amplitude data, Journal of Fatigue 2007; 30/1: 119-137.