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Effective Block Approach for Aircraft Damage Tolerance Analyses

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A damage tolerance analysis method for aircraft structural integrity, designated as the effective block approach, has been developed and subsequently applied to predict the fatigue crack growth life of aircraft structures subjected to flight spectrum loading. The predictions by the method were made not based on constant-amplitude crack growth data, but instead relied on spectrum crack growth data obtained from previous full-scale fatigue tests or representative coupon fatigue tests. The spectrum crack growth data were normally measured from observations of the fracture surfaces by quantitative fractography. Verification and consistency studies of the method were performed against fatigue test results under different flight spectra. The predicted fatigue lives and inspection intervals for fracture-critical locations in aircraft structures were found to be in close agreement with representative coupon and full-scale fatigue tests. This study has demonstrated that the method provided significant advantages in damage tolerance analysis over conventional fatigue lifing approaches for military aircraft structures and components. It was also found that the work has improved the value of structural integrity advice provided for some Royal Australian Air Force air platforms and supports the safe operation of the fleets through to their planned withdrawal date.

Nomenclature

a = crack length or depth
C = coefficient constant (for Paris-based empirical crack growth model)
$C_{\mathrm{A-TS}}$ = Paris constant obtained from analyzed fatigue crack growth data for tested spectrum
$C_{\mathrm{A-UTS}}$ = Paris constant obtained from analyzed fatigue crack growth data for untested spectrum
$C_{\mathrm{T-TS}}$ = Paris constant obtained from tested fatigue crack growth data for tested spectrum
$C_{\mathrm{T-UTS}}$ = Paris constant obtained from tested fatigue crack growth data for untested spectrum
$\mathrm{d}a/\mathrm{d}N$ = rate of crack growth (change in crack length per constant-amplitude cycle)
$\mathrm{d}a/\mathrm{d}t$ = rate of crack growth (change in crack length per spectrum flight hour)
K = stress intensity factor at a crack tip
K_c = critical stress intensity factor
K_{Ic} = critical stress intensity factor under plane strain conditions
K_{ref} = reference stress intensity factor at a crack tip
m = exponent constant (for Paris-based empirical crack growth model)
$m_{\mathrm{A-TS}}$ = exponent constant obtained from analyzed fatigue crack growth data for tested spectrum
$m_{\mathrm{A-UTS}}$ = exponent constant obtained from analyzed fatigue crack growth data for untested spectrum
$m_{\mathrm{T-TS}}$ = exponent constant obtained from tested fatigue crack growth data for tested spectrum
$m_{\mathrm{T-UTS}}$ = exponent constant obtained from tested fatigue crack growth data for untested spectrum
β = beta (geometry and load condition) factor
ΔK = stress intensity factor range at a crack tip (for constant-amplitude loading)
σ_{ref} = reference stress

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I. Introduction

MILITARY aircraft often encounter highly variable environmental conditions and operate under different mission profiles that can result in fatigue failure in aircraft structures. To help maintain aircraft structural integrity until the planned withdrawal date, the Defence Science and Technology Organisation (DSTO) has conducted a range of scientific studies, including full-scale fatigue tests on aircraft structures and relevant component/coupon fatigue tests, under representative flight spectra. The objectives of these activities, amongst others, were to substantiate the service life and/or to predict inspection intervals for the fracture-critical structures and components. To achieve these goals, robust analytical fatigue lifing tools are required and DSTO is developing new approaches for fatigue crack growth (FCG) prediction under flight spectrum loading [1–5].

Fatigue crack propagation in aircraft structures under flight spectrum loading or variable-amplitude (VA) loading is traditionally predicted based on crack growth rates obtained from constant-amplitude (CA) fatigue testing using cycle-by-cycle approaches [6–11]. In contrast, this paper presents a damage tolerance analysis method for aircraft structures based on the crack growth information obtained under aircraft flight spectrum loading. This approach, named the effective block approach (EBA), predicts the FCG life of aircraft structures subjected to flight spectrum loading based on FCG rates obtained from full-scale fatigue tests or relevant coupon fatigue tests [2–5]. The flight spectrum FCG data were measured from observations of the fracture surfaces by the quantitative fractography (QF) process. The EBA method developed recently is able to account for some of the complex crack growth behaviors observed under different flight spectra [5,12]. To demonstrate its robustness, verification and consistency studies were performed using fatigue test results under different operational spectra. The predicted fatigue lives and inspection intervals for a number of fracture-critical locations in some aircraft structures were compared with representative components and coupon fatigue tests.

II. Effective Block Approach

The EBA predicts the FCG life of aircraft structures subjected to flight spectrum loading based on FCG rates obtained from previous full-scale fatigue tests and/or relevant coupon fatigue tests. The earliest version of this approach was proposed by Gallagher in the mid-1970s [13] with a cycle-by-cycle life calculation algorithm. The approach was then not further pursued, mainly due to the limitation of fatigue experimental technologies under spectrum loading around that time, such as long testing times and significant costs [3]. The EBA was used initially by DSTO during the assessment of fatigue crack growth in F/A-18 aircraft structures [1,4]. The method was further developed based on a substantial FCG database for either representative specimens or real airframe structures under flight loading conditions. The database has been established by the QF process that has been under constant development at DSTO in Melbourne since the 1970s [14–17]. Using this database, some general trends have been observed that have proven to be useful rules-of-thumb; for example, it was found that if the QF FCG data were available for an applied spectrum at a given stress level, it was possible to predict crack growth curves for the same spectrum but at a different stress level using the cubic-stress rule [18,19] or flight-by-flight approach [3].

For flight spectra under which there were no FCG data available, the Paris-based EBA [1,4] was developed to predict crack growth curves at critical locations in F/A-18 airframes by using a relative spectrum analysis method. Consider the Paris-based EBA Eqs. (1) and (2):

$$\left(\frac{\mathrm{d}a}{\mathrm{d}B}\right)_i = [C_{\mathrm{VA}}(K_{\mathrm{ref}})^{m_{\mathrm{VA}}}]_i \tag{1}$$

where C_{VA} and m_{VA} are experimentally derived Paris-like parameters that are unique to the material and also the VA loading spectrum, B represents one spectrum block, and the reference stress intensity factor K_{ref} can be expressed by the following equation:

$$K_{\mathrm{ref}} = \sigma_{\mathrm{ref}}\beta\sqrt{\pi a} \tag{2}$$

where σ_{ref} is an arbitrary reference stress from the spectrum (e.g., peak load), and β is the geometry and loading factor that is usually expressed as a function of crack length a for a particular cracked geometry.

It was observed that, for a range of different applied loading spectra, the empirical exponent remained reasonably constant ($m_{\mathrm{VA}} \approx 2$),[†] whereas the coefficient C_{VA} varied significantly. Thus, a relative scaling method was developed to adjust the coefficient C_{VA} based on the results of separate traditional fatigue analyses (Appendix 1 in [4]). The fundamental advantage of this approach is that any inaccuracy of the traditional fatigue analysis methods is limited to only affecting the relative spectra severity scaling. The EBA model, which characterizes the baseline crack growth rate behavior, remains unaffected.

A recent comprehensive representative coupon fatigue test program of a bomber aircraft wing was undertaken to explore the effect of different spectra on the FCG rates. The FCG data were measured by QF and then converted to the curves of crack length verses simulated flight hour, so that the FCG rates could be calculated based on incremental polynomial method according to American Society for Testing and Materials Standard E 647-08 [20]. The material of the coupons was 2024-T851 aluminum alloy. Figure 2 shows the FCG behavior at a representative wing location (BWingLoc 1) for two different spectra, BSpec 1 and BSpec 2, respectively [21]. The reference stress intensity factor K_{ref} was calculated using Eq. (2), with an appropriate geometry factor distribution β for the locations. The reference stress was used to determine the empirical Paris-based FCG constants C_{VA} and m_{VA} for the given material and for the applied spectrum. The geometry factor distribution β for these simple geometric coupons was estimated from a linear superposition of 2-D finite element analysis and handbook solutions.

Figure 2 clearly indicates that there is a considerable difference in Paris-type exponent m_{VA} for the two different spectra. As a consequence, it was necessary to consider the empirical Paris exponent m_{VA} as a spectrum-specific constant. Therefore, Eqs. (3) and (4) were developed to predict the unknown constants for the untested spectrum (UTS) using the derived constants for the tested spectrum (TS) as follows:

[†]The exponent $m_{\mathrm{VA}} \approx 2$ implies exponential crack growth (i.e., a straight line when log crack depth is plotted against linear life). This has been shown to be the case for a wide range of aircraft structural problems [19] including that shown in Figure 1 where the material of the coupons was 7050-T7451 aluminum alloy.

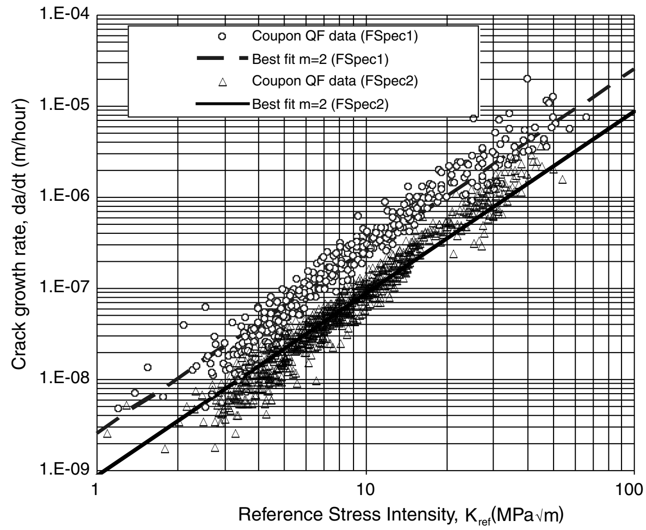


Fig. 1 Effect of fighter flight spectra on empirical Paris exponent constant [4].

$$m_{\mathrm{UTS}} = m_{\mathrm{T_{TS}}} + (m_{\mathrm{A_{UTS}}} - m_{\mathrm{A_{TS}}}) \tag{3}$$

$$C_{\mathrm{UTS}} = C_{\mathrm{T_{TS}}}\left(\frac{C_{\mathrm{A_{UTS}}}}{C_{\mathrm{A_{TS}}}}\right) \tag{4}$$

where the prescripts T and A refer to constants derived by tested FCG data or analysed FCG data, respectively. Refer to [5] for further detail on the mathematical derivation based on the physical observations. Here, the analysis constants were derived from the output of a traditional FCG fatigue analysis code, in this case, FASTRAN [10]. The EBA relative spectra method relies on the ability of the analysis code to predict the *relative difference* in FCG rate between the two spectra. However, it effectively disregards any *absolute* inaccuracy of the analysis code. The general outcome is that the predicted constants for the untested spectra under this approach are significantly more accurate compared to the raw output of a traditional FCG fatigue analysis. This will be demonstrated in the next section.

III. Verification and Consistency Studies of Effective Block Approach

To validate the aforementioned EBA approach and to test its robustness, the following verification and consistency studies were carried out using benchmark FCG data obtained from full-scale fatigue tests of bomber aircraft wings [22,23] and complementary coupon spectrum fatigue tests [21]. For a bomber aircraft wing location BWingLoc 1, the FCG rate under bomber spectrum BSpec 1 was predicted based on measured FCG data at a different wing location BWingLoc 2 under a different spectrum BSpec 3. The two locations consisted of very similar geometry but subjected to different spectra. The relative difference in the FCG rates between the two spectra were estimated based on output from the FASTRAN analysis code using the EBA Eqs. (3) and (4). Figure 3 compares the prediction against measured fractographic FCG data of BWingLoc 1 under BSpec 1 (including a best fit of this data). The predictions by the EBA with the variable Paris exponent correlate well with the experimental data. Note that the current EBA predicted $m_{\mathrm{VA}} = 3.3$ is close to the best-fit test result ($m_{\mathrm{VA}} = 3.0$). Figure 4 shows the same data presented on a crack length versus time plot. The EBA-predicted crack growth curve matches the test data well, but is slightly more conservative in the fast crack growth region.

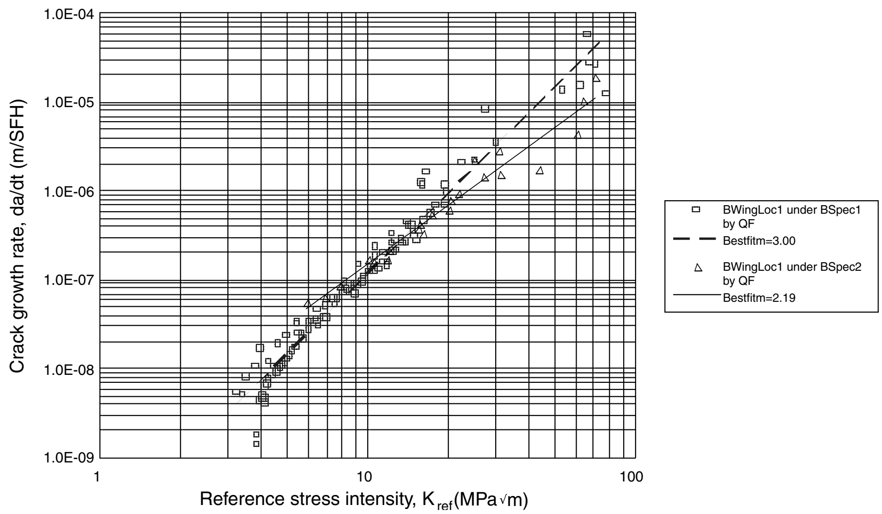


Fig. 2 Change in empirical Paris exponent constant for different spectra.

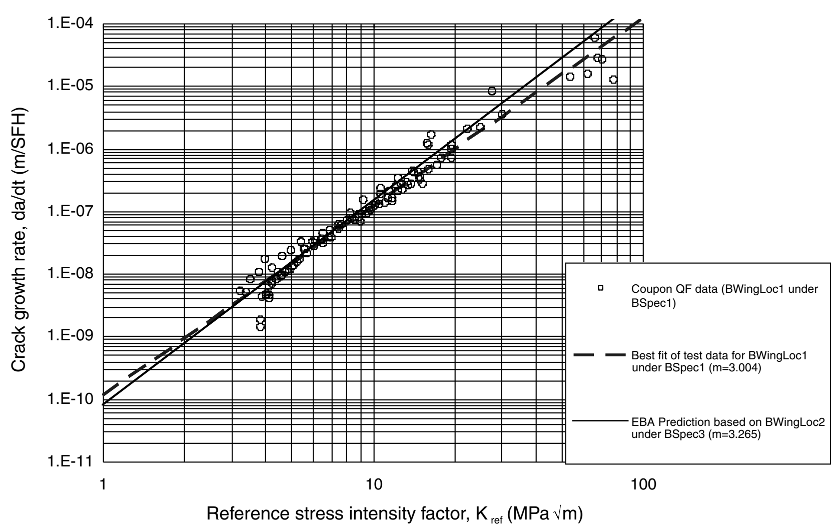


Fig. 3 Predicted FCG rate of BWingLoc 1 under BSpec 1 based on BWingLoc 2 under BSpec 3.

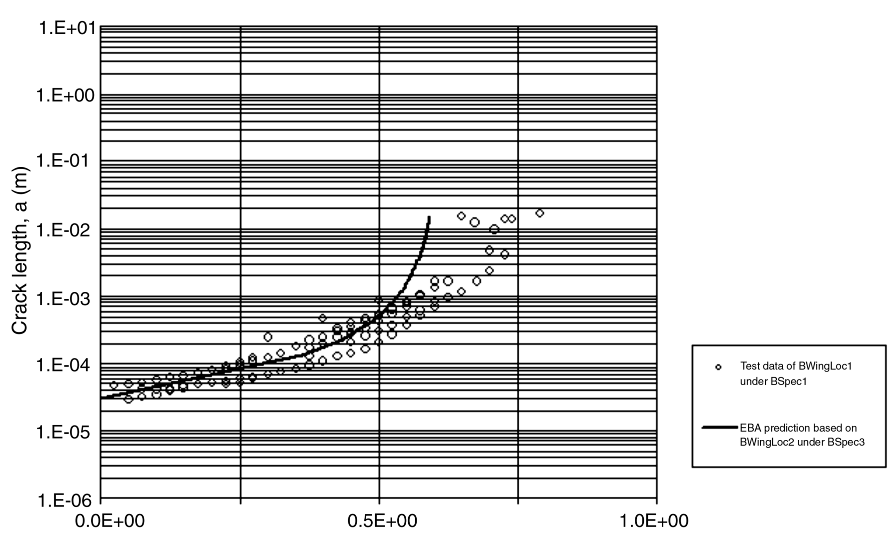


Fig. 4 Predicted FCG life of BWingLoc 1 under BSpec 1 based on BWingLoc 2 under BSpec 3.

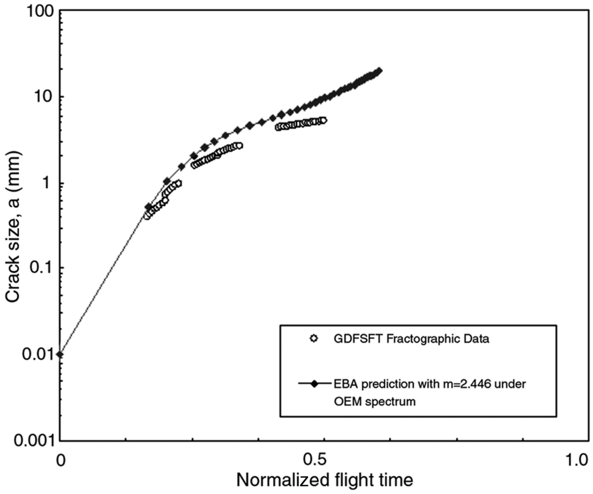


Fig. 5 Verification of the EBA prediction against the original equipment manufacturer's (OEM's) full-scale fatigue test data.

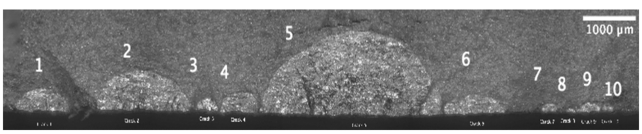


Fig. 6 Multi-initiation of fatigue cracks at the FTG [23].

The EBA prediction for a specific wing location requires the input of a geometry factor distribution $\beta(a)$ for the location of interest. In the case of complicated geometry around a wing splice fastener hole that involved an interference-fit joint, a reverse-engineering method was used to determine the β distribution full-scale fatigue test fractographic data at this location. The equation for the reverse-engineering method [2] can be obtained with a rearrangement of Eqs. (1) and (2) as follows:

$$\beta(a) = \frac{1}{\sigma_{\text{ref}}\sqrt{\pi a}}\left(\frac{1}{C_{\text{VA}}}\,\frac{\mathrm{d}a}{\mathrm{d}B}\right)^{1/m_{\text{VA}}} \tag{5}$$

where C_{VA} and m_{VA} are the Paris parameters that were determined from coupon tests under the same spectrum as that applied to the full-scale fatigue test.

The initial crack size of 0.01 mm shown in Fig. 5 was derived based on the wing full-scale fatigue test fractographic data [22] and the coupon fatigue test results [12] under the same spectrum (the coupons were machined from the 2024-T851 component of the wing splice plate). Figure 5 shows that the EBA is able to provide quite reasonable and conservative predictions for the wing splice fastener hole under the design spectrum (see [12] for details). This reverse-engineering method provides an alternative, and is anticipated to be a more accurate, means of deriving complicated geometry factors than, say, detailed 3-D finite element methods. All potential complexities associated with the full-scale representative test article are included, such as interference-fit loads, residual assembly stresses, nonlinear contact-surface effects, localized plastic deformations, and joint relaxation with load cycling. These are all effectively included in the reverse-engineered β distribution, which now encompasses effects well beyond the simple linear-elastic geometry factor that engineers are usually familiar with.

After a recent full-scale fatigue test of a bomber wing, the tear down revealed multiple fatigue crack initiations at the fuel transfer groove (FTG) at the lower wing skin. The cracks appeared to have initiated from the inner surface and propagated inwards under the spectrum loading (BSpec3) as shown in Fig. 6. The largest crack (Crack 5) had a depth of 1.27 mm (0.05 in). The open circles in Fig. 7 are the FCG data of Crack 5 measured by the QF process. With the initial crack size of 0.0116 mm measured at the FTG from the wing fatigue test fractographic data [23], Fig. 7 further demonstrated that the EBA prediction based on the FCG data of BWingLoc 2 under BSpec 1 is quite close to the results of the wing full-scale fatigue test with slight nonconservatism at the fast crack growth region.

IV. Applications to Aircraft Structures

Application of the EBA to predict FCG in a horizontal stabilator spindle in the F/A-18 aft fuselage structure is first presented. The location of the spindle and the associated finite element model is shown in Fig. 8 [24]. Coupon tests were conducted on the spindle material, AF1410 steel, under representative flight spectrum loading for three different stress scale levels. The ability to model the different stress scale levels was considered fundamental to confidently model the FCG rates at arbitrary stress levels in the actual spindle component. Figure 9 shows the crack growth curves, measured by QF, along with the Paris-type EBA model fitted to these data points.

Before an EBA model was considered, a significant effort was made to model the coupon data using a traditional crack growth analysis tool, AFGROW [8], using all available crack growth retardation models and exploring the full range of adjustable parameter values. It was found possible to match any of the FCG curves from one stress scale level, but impossible to provide a reasonable match to all three stress levels using a single set of

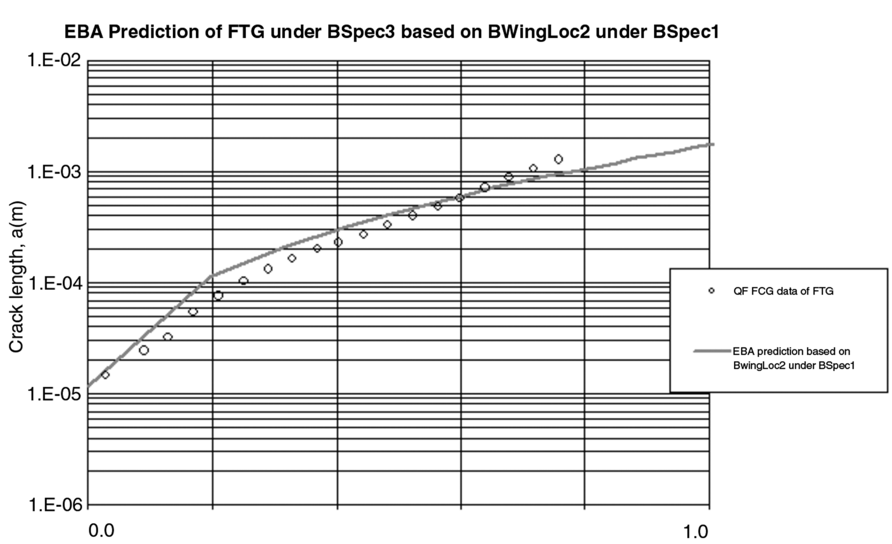
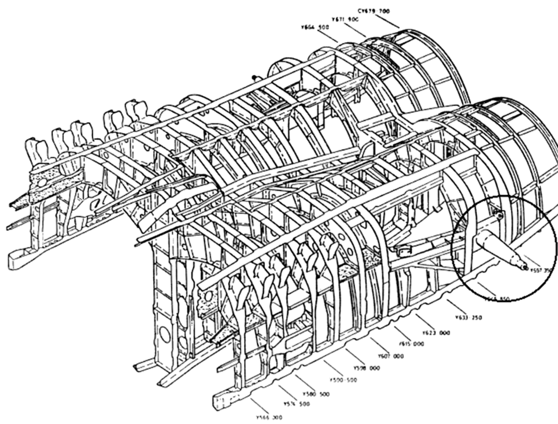
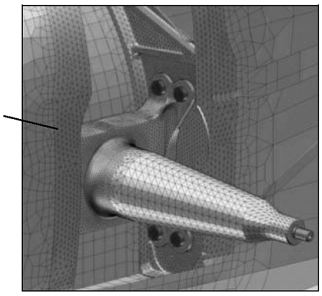


Fig. 7 Verification of the EBA prediction against the wing full-scale fatigue test data.



a) Spindle location in aft fuselage



b) Finite element model of spindle

Fig. 8 F/A-18 aft fuselage horizontal stabilator spindle [24].

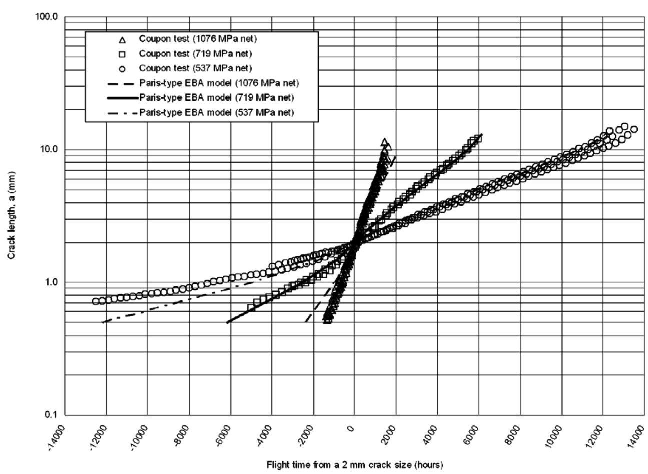


Fig. 9 FCG curves measured by QF from the AF1410 steel coupon test specimens, showing the Paris-type EBA model fitted to all three stress scale levels [24].

parameters. However, the Paris-type EBA model provided a good fit to the data for all three stress levels using a single parameter set (m_{VA} and C_{VA}).

The EBA model was validated using the FCG data measured by QF at a critical location in the spindle from a full-scale fatigue test. An electrical discharge machine notch was manufactured at a high-stress location to provide FCG data for validation purposes. The FCG was predicted here using the Paris-type EBA model, using the parameters derived from the coupon tests, Fig. 9, and detailed stress distributions obtained from the spindle finite element model. This validation provided good confidence in fatigue life predictions for the horizontal stabilator spindle component, Fig. 10.

The critical locations in the F-111 wing were identified during full-scale fatigue tests, as shown in Fig. 11. For a wing location BWingLoc 3 under BSpec 1, the EBA predictions were made based on two sets of experimental FCG data; one is BWingLoc 3 under

Fig. 11 F-111 wing in full-scale fatigue testing [26].

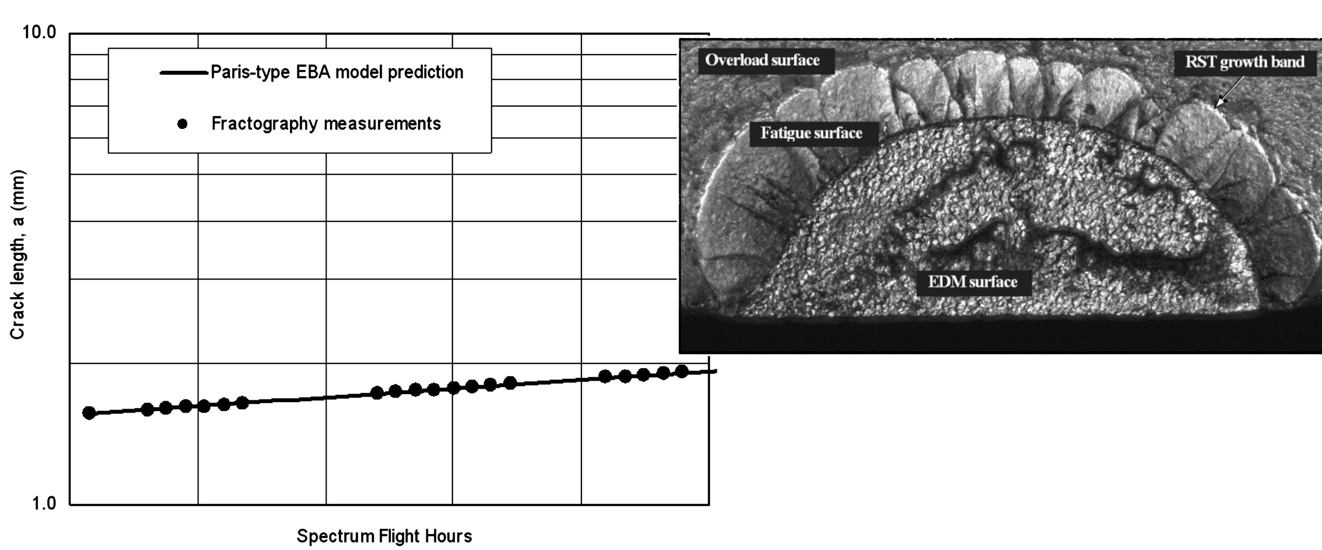


Fig. 10 FCG curve measured by QF at a critical location in the full-scale spindle fatigue test article, providing validation of the Paris-type EBA model prediction [24] [electrical discharge machine (EDM), residual strength test (RST)].

Table 1 EBA-predicted severity factors, fatigue life, and inspection interval at BWingLoc 3 based on two sets of coupon FCG data

	Prediction for BWingLoc 3 under BSpec 1		Relative difference, %
	Based on BWingLoc 3 under BSpect 3	Based on WingLoc 2 under BSpec 1	
Severity factors between	0.345	0.374	−8
Fatigue life (spectrum flight hours)	21,123	22,153	−5
Inspection (spectrum flight hours)	5680	6248	−9

BSpect 3 fractographic data, and the other is WingLoc 2 under BSpec 1. Table 1 shows the EBA-predicted severity factors, fatigue life, and inspection interval at BWingLoc 3. The severity factor is a measure of how severe BSpec 1 is in comparison with that of BSpec 3, both at the residual strength requirement of $1.2 \times$ DLL (design limit load) [25], for the current location BWingLoc 3. The life for a fracture-critical structure is defined as the mean fatigue life divided by a scatter factor (plus an additional factor if the structure is not monitored). Although the scatter factor should be derived from a safe S–N curve for a risk of in-service failure of 1 in 1000, an upper bound of 4.0 has been used here, as recommended in [25]. For demonstration purposes for the EBA methodology development, the predicted life in this study is defined as the time for a crack to grow until failure under the residual strength requirement of $1.2 \times$ DLL and then factored by 4. The results demonstrated that the EBA can provide consistent and reliable predictions.

V. Conclusions

The underlying principal of the EBA developed to predict the fatigue crack growth lives of aircraft structures subjected to flight spectrum loading was presented. The capability of the EBA has been demonstrated in several verification and consistency studies. The applications of the EBA to the structural damage tolerance analysis for two military aircraft components were conducted and assessed. Based on the results from this study, the following conclusions can be drawn.

1) This work has shown that the EBA predictive capability is quite robust. Verification studies have found that the current version of the EBA can accommodate a variable Paris-like exponent parameter to reflect spectrum-specific variables for different spectra. It revealed that the EBA has the potential to provide reliable predictions for the FCG in aircraft structures under flight spectrum loading.

2) It has been found that the EBA capability and accuracy for spectrum FCG prediction are dependent on the ability of selected FCG predictive tools to generate a reasonable FCG relative severity pattern, so that FCG for untested spectra can be predicted based on the tested spectrum FCG data.

3) The EBA parameters were derived from FCG data measured by QF that are normally on a block-by-block basis, in other words, crack growth increments per each spectrum block. Therefore, the number of repeating blocks in the spectrum should be large enough for the parameters to accurately represent the average nature of FCG rate under the repeating spectrum.

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