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On the Growth of Fatigue Cracks from Material and Manufacturing Discontinuities Under Variable Amplitude Loading

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This paper focuses on problems associated with aircraft sustainment-related issues and illustrates how cracks, that grow from small naturally occurring material and manufacturing discontinuities in operational aircraft, behave. It also explains how, in accordance with the US Damage Tolerant Design Handbook, the size of the initiating flaw is mandated, e.g. a 1.27-mm-deep semi-circular surface crack for a crack emanating from a cut out in a thick structure, a 3.175-mm-deep semi-circular surface crack in thick structure, etc. It is subsequently shown that, for cracks in (two) full-scale aircraft tests that arose from either small manufacturing defects or etch pits, the use of da/dN versus ΔK data obtained from ASTM E647 tests on long cracks to determine the number of cycles to failure from the mandated initial crack size can lead to the life being significantly under-estimated and therefore to an unnecessarily significant increase in the number of inspections, and, hence, a significant cost burden and an unnecessary reduction in aircraft availability. In contrast it is shown that, for the examples analysed, the use of the Hartman–Schijve crack growth equation representation of the small crack da/dN versus ΔK data results in computed crack depth versus flight loads histories that are in good agreement with measured data. It is also shown that, for the examples considered, crack growth from corrosion pits and the associated scatter can also be captured by the Hartman–Schijve crack growth equation.

INTRODUCTION

The importance of fatigue to engineering structures has been recognised since the mid-nineteenth century and has been further highlighted as a result of recent problems associated with aging aircraft,[1–3] rail and civil infrastructure*[4–6]

as well as in the ASTM E647-13a test standard. Nevertheless, despite its practical significance, the question of how to predict the fatigue life of a structure is still not fully understood. Indeed, whilst fatigue crack growth has been the subject of a large number of books, papers and articles, some of which are included as Refs. 7–28, the question of which crack growth curve to use when assessing the remaining life of operational aircraft or the effect of structural repairs (or modifications) on operational life is still an active research topic. In this context, the recent review paper[7] described:

i. The difference between the analytical damage tolerance assessment methodologies need-

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*In Ref. 4, it is commented that, in the USA, there are more trips per day over structurally deficient bridges than there are McDonald's hamburgers eaten in the entire USA.

ed for aircraft design and aircraft sustainment**;

ii. The growth of cracks in a number of operational aircraft and full-scale fatigue tests;

iii. The 2011 FAA ruling on the limit of validity (LOV) for civil transport aircraft above 75,000 pounds (34,000 kg) takeoff weight, its requirement for full-scale fatigue tests and its implications with respect to damage tolerance assessment;

iv. The effect of a corrosive environment on the growth of widespread fatigue damage in civil transport aircraft above 75,000 pounds takeoff weight;

v. How to account for the variability in crack growth seen in tests on specimens containing both long cracks and cracks that initiate from small naturally occurring material discontinuities in complex geometries under realistic variable amplitude load spectra.

vi. The effect of crack closure and other forms of crack tip shielding[†] on cracks that grow from small naturally occurring material discontinuities;

vii. The ability of the ASTM Adjusted Compliance Ratio (ACR) technique[33,34] to determine the intrinsic (minimal crack tip shielding) crack growth behaviour of a material;

viii. How to use the Hartman–Schijve variant[35] of the NASGRO equation,[36] see Ref. 7 and the "Appendix" section, which is embedded in both the commercially available crack growth computer programs NASGRO[37] and AFGROW,[38] in conjunction with long crack growth data to determine the growth of cracks that initiate from small naturally occurring material discontinuities in complex geometries under realistic variable amplitude load spectra.

In this context, it should be noted that Appendix X3 of ASTM E647-13a[39] comments that:

"predictions of small-crack growth in engineering structures based on laboratory large-crack (near threshold) data may be extremely nonconservative."

This statement reinforces the conclusion, given in Ref. 7, that for sustainment-related problems any crack growth analysis should use the associated

short crack da/dN versus ΔK data. This conclusion was first reported in the USAF study performed in support of the USAF F-15 program.[20]

It has also been shown[7,26] that various independent findings[7,26,30,31,40,41] support the hypothesis presented in Refs. 7,26 that, for combat and civil transport aircraft,[41] true corrosion–fatigue interaction may not occur and that the effect of the environment on the growth of fatigue cracks from small naturally occurring corrosion sites essentially decouples with the environment:

(i) Creating material discontinuities of various sizes, which depend on the level and nature of the corrosion damage, such that cracks subsequently grow from these discontinuities;
(ii) Growing existing cracks during extended periods of inactivity.[32]

As such, if we are to assess the remaining life of operational aircraft or the effect of structural repairs (or modifications), intergranular cracking[42] on operational life, it is essential that we understand how to compute crack growth from the mandated initial flaw sizes documented in the US Damage Tolerance Design Handbook.[9] To this end, the examples given in the present paper illustrate that, whereas the ASTM standard E647-13a states that the use of long crack da/dN versus ΔK data to determine the number of cycles to failure can lead to non-conservative lives, the use of long crack da/dN versus ΔK data to determine the number of cycles from the mandated initial sizes can also yield inspection intervals that are much too short. This would significantly reduce aircraft availability and also significantly increase both manpower requirements and maintenance costs. As such, the use of long crack da/dN versus ΔK data for aircraft sustainment is inappropriate.

WHICH CRACK GROWTH CURVE?

As discussed above, different analyses are needed to assess design- and sustainment-related crack growth problems. Indeed, as first documented in Ref. 20, the assessment of cracks in aircraft that arise from small naturally occurring material discontinuities in complex geometries under realistic variable amplitude load spectra requires the use of small crack da/dN versus ΔK data. In this context, the paper by Jones and Tamboli[27] was one of the first to show that, when assessing the effect of repairs to aircraft, the use of ASTM-like (long crack) test specimens was inappropriate in that they had the potential to yield non-conservative estimates of the effect on the repair on the operational life of the aircraft. (This mirrors a related statement in Appendix X3 of ASTM standard E647-13a on the non-conservative results that can be obtained when long crack da/dN versus ΔK data is used to determine the operational life of a structure.)

**The ab initio design of aerospace vehicles requires that all structures be designed in accordance with damage tolerance design principles which, for military aircraft, are detailed in the Joint Services Structural Guidelines JSSG2006 and in the USAF Damage Tolerant Design Handbook[9]. However, as explained in Refs. 7,9, whereas the initial (starting) crack sizes mandated in Ref. 9 for ab initio design are relatively large, in contrast cracks in operational aircraft, i.e. in aircraft sustainment-related problems, often initiate and grow from small naturally occurring material discontinuities.[29–32] In such cases, as explained in ASTM E647-13a and,[7] the crack growth analysis should use the appropriate short crack da/dN versus ΔK curve.

[†]Crack tip shielding is defined in Refs. 11,12 as any process that results in the effective value of ΔK seen during a load cycle being less than ΔK.

In this section, we will show that, when assessing the remaining life of aircraft from the USAF Damage Tolerant Design Handbook's mandated initial flaw sizes,[9] the use of ASTM (long crack) da/dN versus ΔK data can also be inappropriate since it can yield inspection intervals that are too short and hence can significantly reduce aircraft availability and increase both manpower requirements and maintenance costs. To illustrate this, consider crack growth in:

(i) The 1969 General Dynamics, now Lockheed Martin, F-111 wing fatigue tested under a representative F-111 usage spectrum, see Ref. 43 for details. (This example was chosen since an early F-111 in-flight failure was largely responsible for the USAF adopting a damage tolerance approach.[1])

(ii) Cracking from a small etch pit in an F/A-18 Y488 bulkhead tested as part of the DSTO[‡] Flaw Identification through the Application of Loads (FINAL) test program, see Refs. 44,45.

Fatigue Crack Growth in the 1969 F111 Wing Test

In this test, a crack initiated at a cut-out location, designated as fuel flow hole[§] number 58, on the lower (tension) surface of the D6ac steel wing pivot fitting (see Figs. 1, 2, 3). In accordance with,[9] the mandated initial flaw size is 1.27 mm. The test crack growth data and the crack growth predictions obtained in Ref. 43 using AFGROW[38] together with long crack da/dN versus ΔK data are reproduced in Fig. 4 which shows the number of flight hours for the crack to have grown from the mandated 1.27 mm depth to failure.

Jones[7] also presented an analysis of this problem. In this instance, the analysis used the Hartman–Schijve representation of the small crack da/dN versus ΔK data for D6ac steel, viz:

$$\mathrm{d}a/\mathrm{d}N = 2.05 10^{-10}\left[(\Delta K)^2/(1-K_{\max}/220)\right] \quad (1)$$

Equation (1) is essentially a simple Paris crack growth equation that has been modified to reflect the increase in growth rate that occurs when $K_{\max}$ approaches the fracture toughness of the material. Appendix X3 of ASTM test standard E647-13a states that: "It is not clear if a measurable threshold exists for the growth of small fatigue cracks". As a result, in this analysis, the small crack growth equation was obtained, as recommended in Ref. 7, from the long crack data for this material[7] by setting the threshold term (in the Hartman–Schijve variant of the NASGRO crack growth equation[7]) to
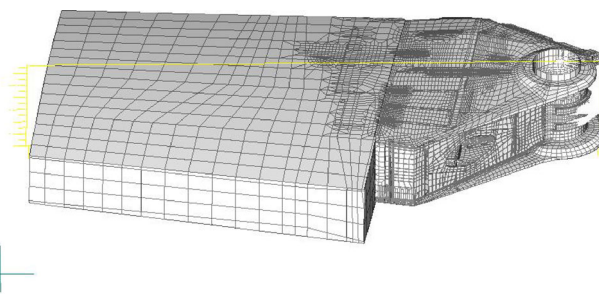

Fig. 1. Full 3D F-111 model, from Ref. 45.

a small value, in this instance to zero. The stress intensity factors were computed using the stress field determined from the finite element analysis of the full three dimensional model of the wing shown in Figs. 1, 2, 3, from Ref. 7. In the Lockheed test, the size of the initiating crack was approximately 0.204 mm, which is similar to the depth of corrosion pits seen in aircraft,[26,46] and the computed crack growth history, post-1.27 mm, is also shown in Fig. 4.

Comparing the various predictions shown in Fig. 4 with the measured test data, we see that, unlike the analysis given in Ref. 7, the AFGROW analysis,[43] which used the long crack da/dN versus ΔK relationship, predicted failure after approximately 859 flight hours which is significantly shorter than the (approximately) 2724 flight hours seen in the test.

Fatigue Crack Growth in an F/A-18 Aircraft Bulkhead

The next problem considered involved cracking that initiated from an etch pit in an F/A-18 Y488 bulkhead tested as part of the DSTO FINAL test program (see Ref. 44). This test program utilised ex-service Canadian Forces (CF) and U.S. Navy (USN) wing attachment centre barrel (CB) sections, which are fabricated from AA7050-T7451, and which were subjected to a modified mini-FALSTAFF spectrum which is representative of flight loads seen by fighter aircraft (see Ref. 44 for more details). Since cracking in the bulkhead was three-dimensional, a three-dimensional FE model was required (see Figs. 5, 6 from Ref. 45). As explained in Ref. 26 in the case of the F/A-18 Hornet, it is known that a significant proportion of cracking initiates from pits from the chemical etching of the AA7050-T7451 which is conducted as a precursor to the IVD aluminium corrosion preventative scheme. In such cases, the equivalent pre-crack size (EPS), which[26] shows to strongly correlate to the pit depth, has a typical (mean) depth of 10 μm.[7,26,47] As such, this analysis assumed an EPS of 10 μm. The approximate location of the centre of the crack analysed in Ref. 45 is shown in Fig. 6 (from Ref. 45), where node 4390 represents the centre of the initial semi-elliptical surface flaw. Furthermore, given

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[‡]The Australian Defence Science and Technology Organisation (DSTO).

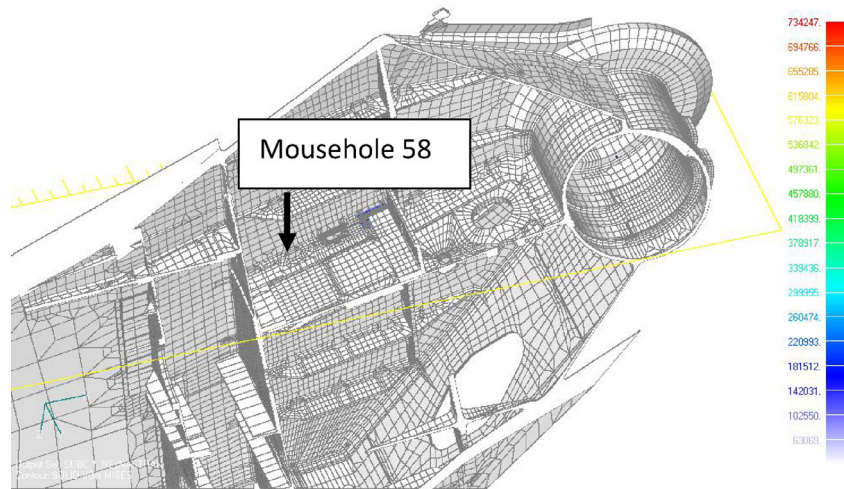[§]This feature is also termed a mousehole or a weep hole.

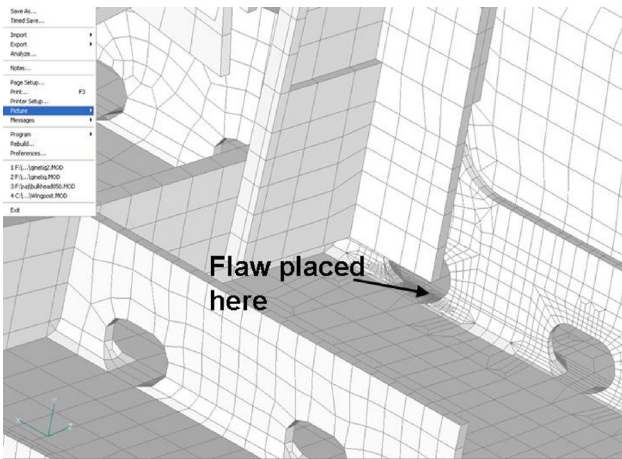Fig. 2. Interior of the DSTO 3D F111 model, from Ref. 45.

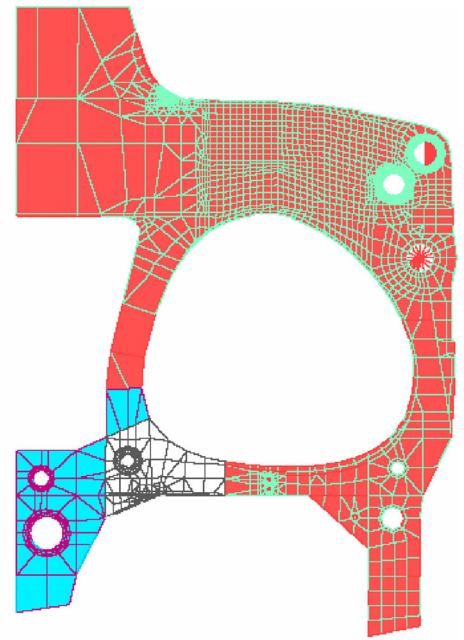Fig. 3. Interior of the DSTO 3D F111 model, from Ref. 45.

Fig. 5. A plan view of the finite element model of the F/A-18 bulkhead, from Ref. 45.
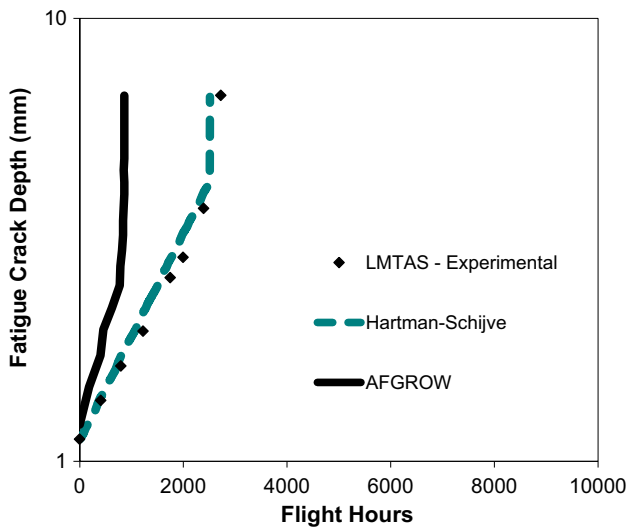

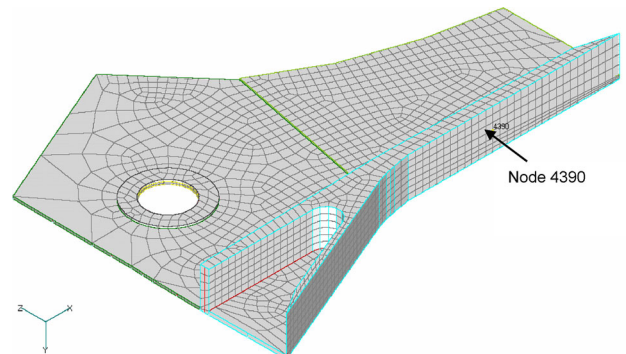Fig. 4. Measured and computed crack growth in the F-111 wing test.

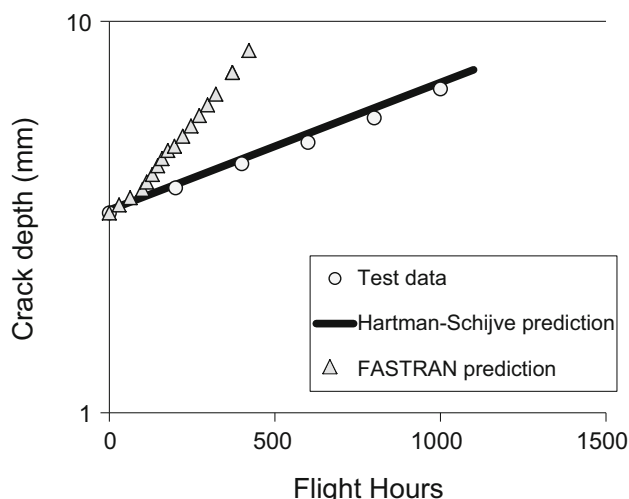Fig. 6. The local mesh, from Ref. 45.

Fig. 7. Measured and predicted crack growth histories.

Fig. 8. Measured and computed crack growth curves for 7075-T651, from Ref. 49.

that the crack is a surface crack in a thick section, the mandated initial flaw size to be assumed in any damage tolerance analysis is a 3.175-mm-deep semicircular crack.

As in the previous analysis, the crack growth analysis performed in Ref. 45 used a weight function technique together with the stress field, as determined from the FE model of the bulkhead, to compute the associated stress intensity factors. Reference 45 presented the measured crack growth history and the computed crack growth histories obtained using the Hartman–Schijve crack growth equation representation of the small crack da/dN versus ΔK data for this material as well as that computed using FASTRAN, together with the long crack da/dN versus ΔK relationship for this material (see Fig. 7).

As mentioned above, the crack depth history computed in Ref. 7, allowing for changes in the aspect ratio of the flaw as the crack grows, used the Hartman–Schijve small crack growth equation given in Ref. 48 for AA7050-T7451, viz:

$$\mathrm{d}a/\mathrm{d}N = 7.0 10^{-10}(\Delta K - 0.1)^2/(1 - K_{\max}/47) \quad (2)$$

Examining Fig. 7 we see that, as was the case for crack growth in the F-111 wing test, using the Hartman–Schijve equation representation of the short crack da/dN versus ΔK data yields a computed crack depth history that is in good agreement with the measured data. We also see that using FASTRAN, together with the long crack da/dN versus ΔK data, results in a predicted time to failure of approximately 470 flight hours whereas failure in the test occurred at approximately 1000 flight hours.

Fatigue Crack Growth from Corrosion Pits

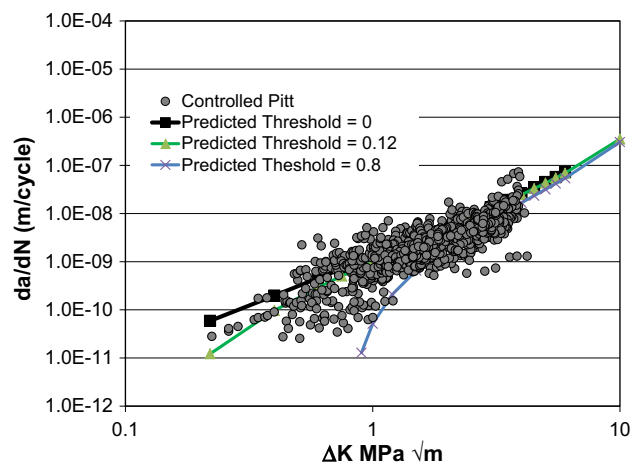The previous section dealt with crack growth from both etch pits and manufacturing defects, in this section, we will show that crack growth from corrosion pits also conforms to the Hartman–Schijve variant of the NASGRO equation. Burns et al.[49] examined crack growth at $R = 0.5$ in water vapour saturated N_2 from controlled corrosion pits in 7075-T651 that were approximately 230 μm deep and had a surface width of approximately 230 μm. The complex shapes associated with the various corrosion pits, when taken in concert with the effect that these shapes have both on the local stress field and on the shape of the subsequent cracks, meant that it was not possible to precisely determine the stress intensity factor (K) for cracks that emanated from these corrosion pits. Nevertheless, by assuming, as a first estimate, a semi-circular crack[49] obtained reasonable first estimates for the value K and plots of the da/dN versus ΔK associated with cracks that grew from these corrosion pits were presented, as seen in Fig. 8 (from Ref. 49).

Tests were also reported by Kim et al.[50] on crack growth in 7075-T651 tested at $R = 0.1$ from corrosion pits which were created using two different corrosion protocols (procedures), i.e. (1) EXCO exposure according to ASTM G34 standard,[51] which produced pits with a typical depth of 0.01 mm, and (2) the Alcoa developed Aluminum-Nitrate-Chloride-Immersion Test (ANCIT) exfoliation test standard[52] which led to both intergranular cracking and corrosion pits of a similar depth to those resulting from the use of EXCO (see Ref. 50 for more details). The da/dN versus ΔK data associated with these tests is shown in Fig. 9.

The Hartman–Schijve equation for crack growth in this material as given in Refs. 27,53 is:

$$\mathrm{d}a/\mathrm{d}N = 1.86 10^{-9}(\Delta K - \Delta K_{\mathrm{thr}})^2/(1 - K_{\max}/111)$$
$$(3)$$

where ΔK_{thr} is a threshold term, which, as explained in Appendix X3 of ASTM E647-13a, will be

Fig. 9. Measured and computed crack growth curves for 7075-T651, from Ref. 50.

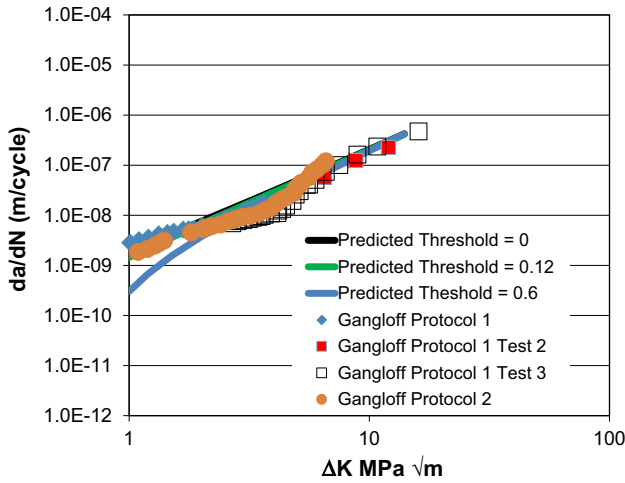
Fig. 10. Measured and computed crack growth curves for 6013-T6, from Ref. 54.

small for naturally occurring small cracks. As explained in Ref. 7, the scatter in the experimental data can be captured by allowing for small changes in the value of the term ΔK_{thr}. As a result, estimates of the scatter in the measured $\mathrm{d}a/\mathrm{d}N$ versus ΔK curves obtained in Ref. 49 were found using Eq. 3 with values of $\Delta K_{th} = 0.0$, 0.12 and 0.8 MPa $\sqrt{m}$, and the corresponding computed curves are shown in Fig. 8.

Estimates of the scatter in the measured $\mathrm{d}a/\mathrm{d}N$ versus ΔK curves obtained in Ref. 50 were found using Eq. 3 with ΔK_{th} values of 0.0 MPa, 0.12 MPa and 0.6 MPa $\sqrt{m}$, and the corresponding computed curves are shown in Fig. 9. Here, we see that, in both cases, allowing for experimental error and errors in the values of K estimated in Ref. 49, there is good agreement between the scatter in the measured and the predicted scatter.

Let us next address crack growth in the 6013-T6 specimens which were first corroded in ASTM G69 to produce corrosion pits[54] to produce pits with a maximum depth of approximately 1.1 mm. Tests were subsequently performed at $R = 0.05$ in a corrosive ASTM G69 environment as well as in laboratory air at both $= 0.05$ and 0.33,[54] and the resultant $\mathrm{d}a/\mathrm{d}N$ versus ΔK curves are shown in Fig. 10. In each case, we see that the various datasets can be represented by the Hartman–Schijve variant of the NASGRO equation, viz:

$$\mathrm{d}a/\mathrm{d}N = 1.210^{-9}(\Delta K - \Delta K_{thr})^2/(1 - K_{max}/31.5) \quad (4)$$

with $\Delta K_{thr} = 0$ MPa $\sqrt{m}$ for the $R = 0.05$ tests in G69, 1.6 MPa $\sqrt{m}$ for the $R = 0.05$ tests in laboratory air and 0.6 MPa $\sqrt{m}$ for the for the $R = 0.33$ tests in laboratory air.

It would appear that, in each case, crack growth from corrosion pits can be represented as per the Harman–Schijve crack growth equation.
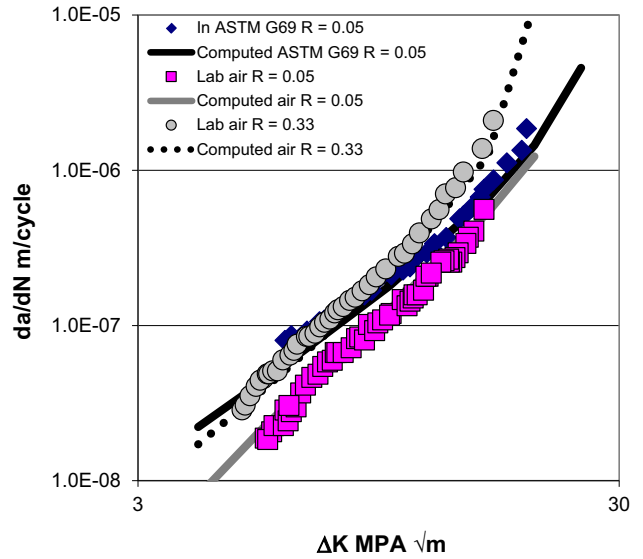
CONCLUSION

The examples presented in this paper substantiate the conclusions given in Appendix X3 of the ASTM test standard E64713a that, for aircraft sustainment-related problems, the use of long crack $\mathrm{d}a/\mathrm{d}N$ versus ΔK data to compute the life to failure is inappropriate. Whereas in Appendix X3 it is stated that this approach can give non-conservative lives, the examples presented in this paper show that using long crack $\mathrm{d}a/\mathrm{d}N$ versus ΔK data to compute the life from the mandatory initial crack sizes stated in the USAF Damage Tolerance Design Handbook can also yield remaining lives, and hence inspection intervals, that are much too short. This would result in significantly reduced aircraft availability and a significant increase in both manpower requirements and maintenance costs.

We have also seen how crack growth from corrosion pits can also be captured by the Hartman–Schijve variant of the NASGRO crack growth equation.

APPENDIX: THE HARTMAN–SCHIJVE VARIANT OF THE NASGRO CRACK GROWTH EQUATION

The NASGRO equation[36] can be written in the form:

$$\mathrm{d}a/\mathrm{d}N = D\left[(1-f)/(1-R)\right]^m \Delta K^{(m-p)}(\Delta K - \Delta K_{thr})^p$$
$$/(1 - K_{max}/A)^q = D\,\Delta K_{eff}^{(m-p)}\left(\Delta K_{eff} - \Delta K_{eff,thr}\right)^p$$
$$/(1 - K_{max}/A)^q \quad (5)$$

where D, m, p and q are constants, $f = K_{op}/K_{max}$, K_{op} is the value of the stress intensity factor at which

the crack first opens, $\Delta K_{\text{eff}} = K_{\max} - K_{\text{op}}$ and the terms ΔK_{thr} and A are best interpreted as parameters chosen so as to fit the measured $\text{d}a/\text{d}N$ versus ΔK data. The NASGRO equation[36] contains the Hartman–Schijve crack growth equation,[7] viz:

$$\text{d}a/\text{d}N = D\left(\Delta K_{\text{eff}} - \Delta K_{\text{eff,thr}}\right)^p / (1-K_{\max}/A)^{p/2} \quad (6)$$

as a special case, i.e. $m = p$ and $q = p/2$. (Here, $\Delta K_{\text{eff,thr}}$ is the associated effective threshold value of ΔK_{thr}.)

As explained in Ref. 7 and Appendix X3 of ASTM E647-13a, there is little R ratio and hence little crack closure effects associated with cracks that grow from small naturally occurring material discontinuities. In such cases for cracks that grow from small naturally occurring material discontinuities, the function f asymptotes to R and ΔK_{eff} can be approximated by ΔK so that Eqs. 5 and 6 become

$$\text{d}a/\text{d}N = D\Delta K^{(m-p)}(\Delta K - \Delta K_{\text{thr}})^p / (1-K_{\max}/A)^q \quad (7)$$

and

$$\text{d}a/\text{d}N = D\left(\Delta K - \Delta K_{\text{thr}}\right)^p / (1-K_{\max}/A)^{p/2} \quad (8)$$

respectively.

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