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On the USAF 'risk of failure' approach and its applicability to composite repairs to metal airframes

W. Hu [a,*], S. Barter [b], J. Wang [b], R. Jones [a], A.J. Kinloch [c]

[a] Centre of Expertise for Structural Mechanics, Department of Mechanical and Aerospace Engineering, Monash University, Clayton, Victoria 3800, Australia
[b] Aerospace Division, Defence Science and Technology Group, 506 Lorimer Street, Fishermans Bend, Victoria 3207, Australia
[c] Department of Mechanical Engineering, Imperial College London, Exhibition Road, London SW7 2AZ, UK

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ABSTRACT

The USAF report on the risk analysis of aging aircraft fleets notes that the operational life of individual airframes is seldom equal to the design life of the fleet and that the life of an aircraft fleet tends to be determined more by its inherent operational capability and maintenance costs rather than by the number of flight hours specified at the design stage. As such this paper focuses on whether the USAF approach to risk assessment can be used for airframes repaired with a composite patch/doubler. To this end the present paper describes a test program designed to study the effect of adhesively-bonded composite repairs to fatigue cracks that, prior to repair, have grown from small naturally-occurring materials discontinuities. This study reveals that crack growth in composite repairs conforms to the exponential growth equation used in the USAF approach to assessing the risk of failure. Furthermore, the exponent, ω, in the exponential growth law can be determined from the crack growth history associated with the unrepaired specimens and the simple reduction in the stress due to the application of the composite patch/doubler, using the 'cubic rule' that was previously used to assess crack growth in the RAAF F/A-18 (Hornet) fleet.

1. Introduction

The 1988 Aloha Boeing 737 accident was one of the first instances to draw the attention of the general public to the problem of aging aircraft [1]. The subsequent USAF report on the risk analysis of aging aircraft fleets [2] stated that "the operational life of individual airframes is seldom equal to the design life of the fleet and that the life of an aircraft fleet tends to be determined more by its inherent operational capability and maintenance costs rather than by the number of flight hours specified at the design stage". This conclusion, i.e. the difference in the tools and methodologies needed for *ab initio* design and aircraft sustainment, was also highlighted in [3]. Implicit in these findings is the fact that, as explained in ASTM E647-13a Annex X3 [4], crack growth data obtained using ASTM E647-13a like specimens, where the cracks have been grown from long artificial notches, are inappropriate for assessing aircraft sustainment related issues. This led [5,6] to conclude that tests that use ASTM E647-13a like specimens, where prior to patching the cracks have been grown from long artificial notches, are inappropriate for assessing the effect of composite repairs to operational aircraft.

In this context, it should be noted that to date the vast majority of studies into the effect of composite repairs to cracked structures have used ASTM E647-13a like specimens where prior to patching the crack was grown from large artificial notches [7,8]. However, as explained in ASTM E647-13a, and discussed above, such specimens do not reflect how a crack in an operational structure will grow. This is because the fatigue threshold associated with such ASTM E647-13a specimens is much greater than that seen by cracks that have grown from small naturally-occurring material discontinuities typical of those found in operational aircraft [3]. This is aptly illustrated in Fig. 1 which presents the da/dN versus ΔK curves associated with the growth of small cracks in 7050-T7451 and also a corresponding da/dN versus ΔK curve that was determined using the ASTM E647-13a K_{max} test procedure where the crack was grown from a large artificial notch. Here we see that, in contrast, to the sigmoidal shape seen for da/dN versus ΔK curves determined in accordance with the fatigue test standard ASTM E647-13a, i.e. the curve DBC, the da/dN versus ΔK curves associated with naturally-occurring cracks, i.e. curve ABC, has a much lower threshold. Further, curve ABC more closely follows a Paris-like equation to this apparent threshold which, in the mid-ΔK and below region, has little (if any) R-ratio dependency. A good example of the need use the curve ABC, rather than the curve DBC, is given in [9] which discusses

* Corresponding author.
E-mail address: wenchen.hu@monash.edu (W. Hu).

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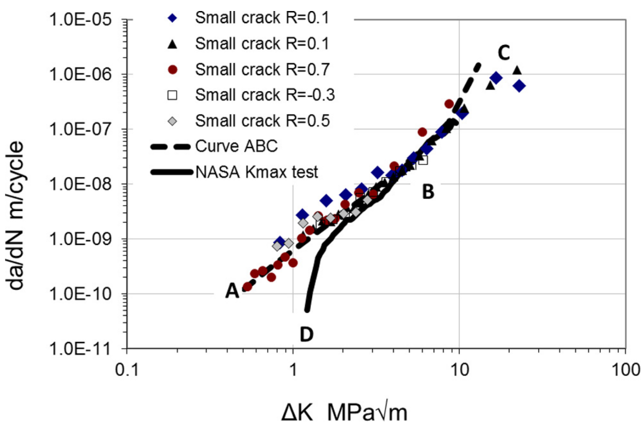

Fig. 1. Comparison of da/dN versus ΔK curves associated with naturally-occurring cracks and ASTM K_{max} test cracks for 7050-T7451, from [3].

how to assess the operational life of critical locations in the Lockheed Martin F-22.

It has long been known that the growth of lead cracks under service loading types of load spectra, i.e. the fastest growing cracks as defined in [10], in operational aircraft is generally (approximately) exponential [10–17] so that the corresponding da/dN versus ΔK relationship is essentially a simple Paris law [3,5,17,18]. As a result the USAF approach to assessing the risk of failure[1] by fracture, which is a certification requirement in the US Joint Structural Guidelines JSSG2006, incorporates an exponential crack growth model [2,10]. An example of this approach is given in the ASIP 2013 Lincoln lecture [20].

The experimental data analysed in [21] suggested that for composite repairs to cracks in operational structures, i.e. cracks that have arisen and subsequently grown from small naturally-occurring material discontinuities, the effect of the fibres bridging the crack would be a second-order effect. From this it follows that the growth of small naturally-occurring cracks repaired with a composite patch should be near exponential albeit with a reduced growth rate due to the reduction in stress produced by the addition of the repair. The results of the present study support this hypothesis. This is an important finding since given that the experimental data presented in [21] revealed that the growth of cracks, with initial lengths 5 mm or greater, repaired with a bonded composite patch is also exponential it means that the growth of both long and small cracks repaired with a composite patch exhibit (near) exponential crack growth. This, in turn, means that the risk assessment tools and the associated computer code (PROF) developed by the USAF [9,20] are equally applicable to composite repairs as to aging airframes.

2. The test program

2.1. The metallic test coupon

Cracks in operational aircraft generally initiate and subsequently grow from small material discontinuities. Consequently, to investigate the effect of a composite repair to cracks that have initiated and subsequently grown from a small material discontinuity a number of tests were performed on 6.35 mm thick 7050-T7451 aluminium alloy dog-bone test coupons, see Fig. 2. The metal coupons were cut from the 7050-T7451 plate so that the long axis of the coupons was transverse to the rolling plane and the plane of the coupons was in the rolling plane. These coupons had a working area that was 25 mm wide and 74 mm long. Both sides of the coupons contained 29 rows (in the length direction) of small laser induced notches. The number of notches in any given row alternated between 5 and 6. In each row the notches were 4 mm apart and the distance between each row was 1 mm. This array of notches was located in the centre of the working section, see Fig. 3. To minimize any interaction of any resultant delaminations/disbonds the array was staggered as per Fig. 3. Each laser induced notch has a typical dimension of 0.2–0.3 mm deep and 0.04–0.07 mm wide. A view of a typical laser notch is shown in Fig. 4.

2.2. The repaired patch test specimen

Three specimens, which we will term Specimens 1, 2 and 3, were tested. Each had a 0.65 mm thick boron epoxy patch (consisting of five layers of unidirectional boron fibres with 3 mm drop off on each ply) on both[2] sides. The patch was rectangular and had the dimension of 100 mm long and 25 mm wide which can fully cover the narrowest work area, see Fig. 5. The patch was bonded to the aluminium alloy using the hot-cured epoxy-film adhesive, FM300-2K (from Cytec-Solvay, Australia). The bonding conditions employed were 120 min at 121 °C. Prior to patching an additional layer of FM300-2K adhesive film was also B-staged co-cured with the unidirectional boron/epoxy 5521/4 prepreg (from Textron, USA). The standard surface treatment [7] used by the Australian Defence Science and Technology Group (DST Group) was used, viz:

a) An initial abrasion using 'Scotch Brite' (from 3M, Australia) pads;
b) Solvent clean with MEK;
c) Grit blast;
d) Application of the coupling agent (a solution of the silane 'Silquest A-187').

2.3. Test schedule

Each specimen was subjected to cyclic loading with a repeated block that marked the fracture surface with a repeating pattern. The loading block had 300 cycles at R = 0.1 and 10,000 cycles at R = 0.8 and the peak load applied was 50 kN (i.e. every cycle had the same peak load applied). This enabled the crack growth history to be measured using quantitative fractography [22]. Prior to patching the specimens were subjected to fifty-eight load blocks. This was done so as to establish fatigue crack growth from the laser induced discontinuities prior to patching. To help distinguish the end of the fifty-eight loads blocks, and prior to patching, the specimen was subjected to an additional 900 cycles at R = 0.1 at this point. After the additional load block the lead crack in Specimens 1, 2 and 3 had grown to a depth length of approximately 1.5, 1.1 and 1.5 mm, respectively.

2.4. Experimental observations of the failed repair patched specimens

Photographs of the failed repair patched specimens after fatigue testing are shown in Fig. 6. Clearly, failure of the Specimens revealed not only fracture though the aluminium alloy but also both disbonding of the adhesive and first-ply delamination in the repair composite patch.

Examining the failed section of Specimen 1 it was observed that, although 10 or 12 laser induced notches were exposed on the frac-

[1] In the context of risk assessment it should be noted that it is now known that for a given initial flaw size the variability in the crack growth histories can often be capture by allowing for small changes in the threshold term in the Nasgro crack growth equation [3,19].

[2] This test configuration was used since when evaluating the effect of a repair to a wing skin of thickness "t" it is common to test a specimen with a thickness "2t" with a patch on both sides [7]. In this way unwanted bending effects are eliminated.