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The lead crack concept applied to defect growth in aircraft composite structures

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ABSTRACT

Currently, the cyclic fatigue growth and residual strength of damaged aircraft composite structures under operational loads is not fully understood. This leads to structures generally being designed to a no damage growth criterion with many knock down factors included to cover unknown/untested effects. Thus, full optimisation of composite aircraft structures is unlikely to be achieved under the no damage growth criterion. In 2009 the US Federal Aviation Administration (FAA) introduced a slow growth approach to certifying composite, adhesively bonded structures and bonded repairs which could improve the situation and is worthy of further investigation. In this paper the growth of some (limited) damage types available in the literature are reviewed and a framework proposed to address the damage tolerance assessment of these structures.

1. Introduction

The cyclic fatigue lifing of aircraft composites or adhesively bonded aircraft structures under operational loads is an area which is not well understood and requires development. A key issue is that there are no widely accepted tools or metrics to assess the durability of composite components of a service aircraft once defects or damage are detected. Technically, if damage is outside Original Equipment Manufacturers' (OEM) "go or no go" or repair limits the aircraft should be grounded until repaired or replaced (likely at Depot). Often these repairs affect large areas of the composite component for even small instances of damage. This can lead to significant reduction in aircraft availability. To ease this impact, the airworthiness manager requires a tool set that can provide an estimate of the remaining safe operational period after damage is detected (given the size detected was below critical), rather than the current Find-and-Fix approach. Alternatively, when a bonded repair is designed a safe operational period is required from a typical inherent defect.

Traditionally the certification of composite or adhesively bonded aircraft structures was based on a "no growth" (or no damage progression) design philosophy. Not only does this approach require that the composite structures were designed such that any fatigue loads would be well below the endurance limit [1], it also requires considerable knock-down factors to ensure compliance and to cover potential strength reductions due to unknowns such as environmental degradation, impact damage etc. This approach can result in structures that are overly conservative and thicker than required if (say) a slow damage growth methodology was applied, and thus currently may provide little weight cost benefit over traditional metal structures (e.g. [2–4] etc).

In 2009 the US Federal Aviation Administration (FAA) introduced a slow growth approach to certifying composite and adhesively bonded structures and bonded repairs [5]. The precise wording given in FAA Advisory Circular 120-107B [5] is:

> *The traditional slow growth approach may be appropriate for certain damage types found in composites if the growth rate can be shown to be slow, stable and predictable. Slow growth characterization should yield conservative and reliable results. As part of the slow growth approach, an inspection program should be developed consisting of the frequency, extent, and methods of inspection for inclusion in the maintenance plan.*

In short, a damage tolerance approach can be applied to composites and adhesively bonded structures if the growth of defects or damage is systematic and thus can be predicted; and there is a reasonable operational life between the damage being detected and complete failure of the component (i.e. loss in residual strength).

Unfortunately a lack of understanding of, and an inability to predict, the damage growth that arises from material discontinuities or in-service induced damage, is an obstacle that hampers the establishment of this approach.

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The FAA additionally gives design consideration advice that:

If significant damage growth should occur, growth characteristics of each defect/damage type should be assessed under repeated loads expected in service. Between initial detectability and the extent of damage established for residual strength determinations. The statistical variability in damage growth should be such that the structure will provide at least the same level of safety as currently required of metallic structure [6]".

Whilst the requirement for validation testing to include such considerations as clearly visible impact damage for composite aircraft is included in some standards (e.g. [5,7] etc) no specific guidance is currently provided (i.e. which locations, size, detectability definitions etc). No other damage types are explicated noted in the guidance.

In this paper the growth of some (limited) damage types are reviewed and a framework is proposed to address the damage tolerance assessment of damaged aircraft composite structures based upon a lead crack growth framework [8] for metallic structures.

2. Assessment of available damage growth data

A limited review of publically available data for the growth of discontinuities in composites or adhesively bonded joints was performed; however, accurate and well documented data are relatively rare, particularly when one considers the wide range of material types, applications, loading conditions, and failure mechanisms that should be considered. Despite the limited data, the available data provides some insight. For illustrative purposes in this paper, data obtained from six sources covering a range of composite and bonded applications are highlighted:

a. The aluminium alloy (AA) 7075 base plate with a square cut-out over which an AA 7075 patch was bonded and tested under constant amplitude loading [9]. The adhesive's (FM73M.06) end delamination growth (measured using digital image correlation) per cycles data for two stress levels is shown in Fig. 1;
b. The growth (measured using Moire fringe technique) shown in Fig. 2 is from two initial impact damage sizes induced in 7 mm thick XAS/914C carbon fibre/epoxy panels tested under a fighter aircraft spectrum [10];
c. The edge bondline delamination (measured visually) in the aluminium double-lap shear joint specimen investigated experimentally [11], are shown in Fig. 3. The central adherend and bonded patches were AA 2024-T3 and the film adhesive was FM73 adhesive from Cytec. The specimen width was 20 mm, and the specimens were cycled at two stress levels;
d. The damage growth of several selected open-hole compression AS4/E7K8 PW -10/80/10 fatigue specimens was monitored by through-transmission ultrasonic (TTU) C-scanning [12]. A few examples are shown in Figs. 4 and 5 where specimen A16 was tested at a level of 63%, A17 and A18 of 75%, A19 and A20 of 70% of design limit stress.
e. The delamination growth measured using ultrasonic C-scan as a function of constant amplitude $R = -1$ cycles in an AS/3501/6 laminate $[(\pm45_2/22.5/67.5)_6/\pm22.5/90/0_2]_2$ subjected to 46 J impact (visible) damage and tested at −54 °C (95% RH) as provided in [6] is shown in Fig. 6.
f. The impact damage (25.4 mm diameter steel rod; 1.5 J) growth measured by C-scan in three 24 ply 67% T300/5208 76 × 356 mm laminates each containing a central damaged hole (9.5 mm diameter) and tested under constant amplitude loading of 283 MPa at $R = -1$ is shown in Fig. 7 (i.e. Fig. 68, a selection of the many data sets from [13]).

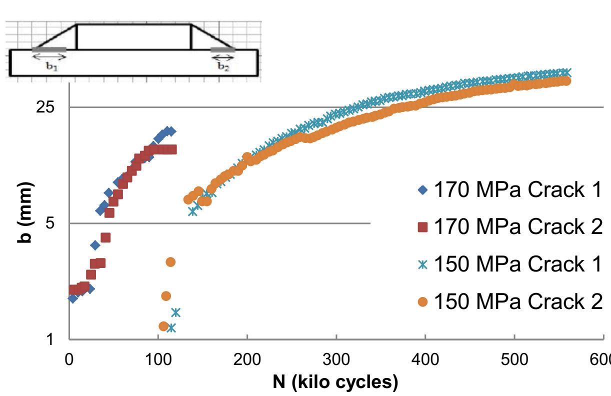

Fig. 1. Disbond length (b) measurements for the type I asymmetric specimens, adapted from [9].

It should be noted that the various authors above chose different metrics to describe the damage (e.g. crack length, delamination width etc) and no further comment on these are made here.

From these data it would appear that as a first approximation (despite the scatter seen for nominally the same conditions) that the lead crack concept [8] for damage tolerance of metals may apply to the growth of damage in composites and bonded structures.

Some of the lead crack characteristics include:

1. The damage appears to commence growing from the first application of loading; and
2. There is an initial near-exponential period of stable growth that appears relatively independent of the initial damage size and the slope is proportional to the applied stress level.

Thus a conservative estimate of the time to failure can be made by knowing the initial damage size, the worst-case slope and a critical growth size (i.e. the lead crack concept). For example see FG1 or FH5 in Fig. 2.

3. Predicting damage growth via modelling techniques

A review of the current approaches for predicting crack (disbond/delamination) growth in both composite and adhesively bonded structures was presented in the recent paper by Pascoe et al. [9,14]. Ref. [9] noted that it is seems accepted that the strain energy release rate (SERR – G) (or $\sqrt{G}$) has a strong correlation with delamination/disbond growth and that, as a result, several authors have presented variants of the Paris crack growth equation to represent delamination/disbond growth. Initial formulations tended to express the growth rate (da/dN) as a function of either G_{max} or ΔG, e.g. [15]. Martin and Murri [15] stated that for composites, the exponents relating the growth rate (da/dN) to G_{max} or ΔG are relatively high (e.g. >2) and, as a result, concluded:

For composites, the exponents for relating propagation rate to strain energy release rate have been shown to be high especially in Mode I. With large exponents, small uncertainties in the applied loads will lead to large uncertainties (at least one order of magnitude) in the predicted delamination growth rate. This makes the derived power law relationships unsuitable for design purposes.[1]

[1] Hence also for the purpose of certifying composite and adhesively bonded structures and bonded repairs to either metallic or composite structures.