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ROUTES TO IMPROVED DAMAGE TOLERANCE; PREDICTION OF DAMAGE TOLERANT PERFORMANCE OF HIGH STRENGTH AND HYBRID STRUCTURES

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Abstract. *While damage tolerant design is now the norm throughout fixed wing aircraft structure, landing gear and engines remain certified via safe life. In common with many helicopter parts such components are made of high strength materials. It is shown that under damge tolerant design much of the strength of high strength materials cannot be exploited. There are similar constraints encountered when attempts are made to use high strength titanium and aluminium alloys to reduce aircraft weight via increased design stress. The benefit of using the crack retarder approach in damage tolerant designs to effectively exploit greater fractions of material strength is described. Results are presented which allow the increases in design stress to achieve specified crack growth lives to be quantified.*

Keywords: *Damage tolerance; Hybrid materials; material strength effects; compliance ratio*

1 INTRODUCTION

The past 50 years have seen a dramatic shift in the techniques used to design civil fixed wing aircraft against fatigue and other forms of mechanical damage. All structurally significant parts apart from the engines and landing gear, have moved to a damage tolerance approach; the former remaining on safe life. The situation with helicopters is significantly different, with many structurally significant items remaining of safe life design[1].

The contribution which the damage tolerant approach has made to aircraft safety may be assessed by noting that airworthiness accident rates for Europe and North America are now less than 10% of all aircraft accidents[2]. The alternative approach – safe life is treated in all civil regulatory material as being a technique of last resort, in that its use is permissible providing the damage tolerant approaches are not demonstrable. The safe life philosophy, even when backed up by inspection relies on a calculation of service life where the probability of cracking is extremely remote[3]. Thus this is an approach relying on damage initiation, rather than damage growth.

The classical damage tolerant approach has proved either vulnerable or impracticable in two important categories; the effects of multiple site damage in high lifetime aircraft[4], and secondly the difficulties of its application to components designed as requiring selection of high strength materials to satisfy design requirements. The difficulties arising in both situations are a consequence of inability of current levels of technology to reliably detect low levels of local damage. Damage tolerant aircraft pre Aloha airlines accident[5] were designed against single instances of worst case anomalous defects, rather than distributions of small defects of marginal detectablity, collectively acting to reduce aircraft strength and life.

2 HIGH STRENGTH MATERIALS

The common factor linking the engines and landing gear in fixed wing aircraft to helicopters is of course the use of high strength materials such as high strength aluminium alloys, steels and titanium alloys which are inevitably operating at significantly greater operating stresses, than would be the case with the traditional aluminium alloys used in fixed wing aircraft. Examination of the simple integrated form of the Paris law[6] such as:

$$N = \frac{1}{C(\Delta\sigma)^m} \int_{a_1}^{a_2} \frac{da}{a^{m/2}}$$

shows that the life N to grow a fatigue crack between starting and finishing crack lengths a_1 and a_2 is inversely proportional to the stress range $\Delta\sigma$ raised to the exponent in the Paris law m. Thus as stress range increases crack growth lives reduce. Materials of increased strength do not particularly increase their resistance to fatigue crack growth for a given ΔK; attempts to fully exploit increased strength via increased design stresses, cause the crack growth rate to increase and crack growth lives (and failure crack lengths) to reduce.

The consequences are reduced service hours in which to detect cracks. This renders achievement of damage tolerance design in the traditional way difficult if not impossible for these components. For safe life design, effectively based on crack initiation approaches, high static strength is a benefit, but the strength cannot be exploited in the fatigue crack growth dominated damage tolerance designs.

The effect of strength on damage tolerant (crack growth resistance) capability can be quantified by calculating the fraction of material strength which can be usefully exploited in a crack growth design situation. For a structural element to be damage tolerant, the maximum cyclic stress level which can be applied when a fatigue crack is grown from some start point a_s to a final failure point a_f in a defined number of cycles or flight hours requires calculation. Stresses greater than this result in the required life not being achieved; stresses smaller than this value will result in the required life being exceeded, but at the expense of increased weight of the component. For variable amplitude loading, the relevant stress is the maximum stress achieved in the load spectrum. This can be expressed in terms of the fraction of static proof strength which it represents. The damage tolerance capability of different materials can be compared by calculating the maximum stress to achieve the desired service life and expressing it as a fraction of static proof strength. This procedure was followed for a range of aluminium alloys and steels, using the TWIST transport wing spectrum[7], and a single edge notch sample geometry. For each selected material, the number of flights of TWIST to grow the crack from 1 mm to 13 mm was calculated as a function of peak stress in the spectrum. The largest stress which produced a total life of 10,000 flights was expressed as a fraction of the static proof strength of the alloy in question. The calculations were performed using the AFGROW[8] package for crack growth calculations. The results of such a series of calculations expressed as a function of the proof strength itself, are shown in figure 1.

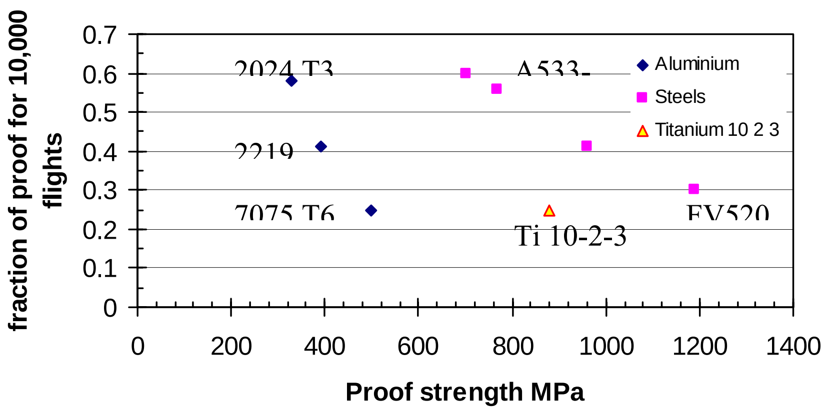

Figure 1 The influence of static strength and alloy type on the largest peak stress in the TWIST spectrum at which 10,000 flights can be achieved. The stress is expressed as a fraction of the proof strength of the alloy.

It can be seen that 2024, the lowest strength of the aluminium alloys can be used at a maximum stress value of around 60% of its static proof strength; 7075, can be used only at a maximum of 25% of the static proof, if 10,000 flights are to be achieved. The rest of the static strength of this high strength alloy cannot be exploited. Similar results are found for steels and titanium alloys.

In contrast the effect of static strength on crack initiation behaviour – of importance in the calculation of safe lives – is almost entirely beneficial. As static strength increases, high cycle fatigue strength also increases. This is more marked for steels and titanium alloys with fatigue endurance limits of around 0.5 of the tensile strength, than it is for aluminium alloys which can achieve only 0.3 of the static ultimate strength. Thus for crack initiation based safe life designs, a much greater proportion of static strength increases can be exploited.

Increasing material strength results in lives which are increasingly dominated by crack initiation and early crack growth processes with reduced fractions of life spent in propagating macroscopic cracks. Thus the high strength materials are increasingly resistant to crack initiation, but crack growth resistance does not improve sufficiently to counter the increased crack driving force arising from the increased service stresses.

Design situations thus may be divided into two categories

(a) Those which are manufactured of relatively low strength materials such as aluminium alloys, structural steels etc, and are relatively lightly stressed, examples including aircraft, bridges and other civil structures, pressure vessels, ships and offshore structures.

(b) Higher strength materials which are used to construct machinery such as gas turbines, landing gear, reciprocating engines, and which additionally are subjected to large contact stresses and sliding wear.

In category (a) damage tolerance approaches to life calculation can be applied relatively easily and with good accuracy. In category b, damage tolerance is difficult to apply as the macroscopic crack or damage growth stage is so short, and crack initiation is calculated via damage summation from S-N curves.

3 DAMAGE TOLERANCE OF POLYMER MATRIX COMPOSITES

The damage tolerance capability of polymer matrix composites is in some ways similar to that of high strength metallic materials. Polymer matrix composites with static strength values in the quasi isotropic lay-up in which they are used on aircraft of 700-1,000 MPa count as high strength materials. The in plane fatigue strength is exceptionally good approaching 80-90% of the static strength. Crack growth resistance under both static and fatigue loading for cracks perpendicular to the in plane fibres is very great. Famously, the out of plane toughness (G_{IC}) in response to tensile stresses in the through thickness direction is significantly lower, as is the in plane shear toughness (G_{IIC}). This behaviour extends from static fracture into fatigue delamination growth under tensile mode I and shear (mode II). Polymer composite fatigue delamination crack growth rates can be characterised in terms of ΔG the range of strain energy release rate of the fatigue cycle, in the same way as metals da/dN is characterised using ΔK. However, growth rates are fast and the window for inspection is small.

It has been demonstrated[9,10] that for impact damaged cfrp, stresses to grow fatigue delaminations from the impact damage are 60% of the ultimate compression strength for 10^6 cycles, whereas damage at the BVID level will cause reduction to 75% of the ultimate compression strength. Polymer composite materials resist damage growth until high fractions of the ultimate compression strength are imposed, but once damage growth occurs the remaining life is short.This behaviour is reminiscent of the crack initiation and growth behaviour of high strength metallic materials, and in many ways the problems of ensuring damage tolerance in the two materials are very similar.

4 ROUTES AHEAD – HYBRID MATERIALS

Hybrid material concepts take many forms and have been explored over many years, but until recently have not been implemented for the majority of aircraft construction. These concepts include fibre – metal laminates such as GLARE, which of course has great resistance to the propagation of fatigue cracks[11-13]. GLARE laminates do not have the static strength and stiffness implicit in the carbon fibre polymer composites. They are relatively expensive to manufacture, and there are a number of other less expensive routes to reduction of fatigue crack growth rates. These involve use of combinations of bonded fibres and laminates working together with ductile aluminium substrates. One of these – the bonding of straps or retarders to aluminium substrates has been the subject of considerable recent research. When a strap of one material is bonded to a substrate of a different material, there are a number of separate effects which together can inhibit fatigue crack growth in the substrate. These are illustrated in figure (2).

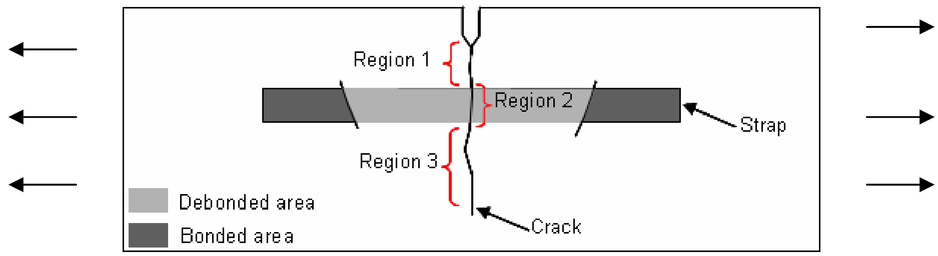

Figure 2 Illustration of the regions of influence of a bonded strap on a crack propagating in a substrate

Figure 2 shows a simple single edge notch cracked sample with a bonded strap in the path of the crack in the substrate. The sample is subjected to tensile loading. The bonded strap interacts with the crack and substrate by:

(a) Changing the local stress distribution- the strap is an additional load path, taking load away from the substrate material in the surface region. Increases and decreases in strap stiffness from the substrate value will change the shear stresses operating in the interface. This is an effect that will operate largely in region 2, but extending a little way into regions 1 and 3.
(b) Acting as a bridge across the substrate crack surfaces, reducing crack opening and therefore the value of the maximum stress intensity Kmax. This bridging effect continues to operate after the crack tip has passed the region of the strap (region 3).
(c) As Schjive[14] has pointed out, the effect of a strap on load transfer and on crack bridging will be a function of the strap stiffness and geometry, and the compliance of the substrate- also a function of its stiffness and geometry. The effect can be encapsulated by the compliance ratio μ[14].

$$\mu = \frac{\Sigma(E_{strap}A_{strap})}{(E_{subst}A_{subst}) + \Sigma(E_{strap}A_{strap})}$$

Where E_{strap}, E_{subst} A_{strap} A_{subst} correspond respectively to the elastic modulus and cross sectional area of the strap and the substrate.

(d) When the crack tip first encounters the edge of the strap as it grows into region 2, most commonly, a disbond develops at the interface between substrate and strap. Under the influence of the alternating shear stress at the interface, the disbond propagates with an elliptical crack front, and will gradually separate the strap from the substrate, reducing the effective stiffness of the strap as it does so and consequently reducing the crack bridging effects on growth rate. Rates of delamination crack growth will depend on the delamination crack growth rates for the adhesive used.

(e) Because the majority of adhesives used in aerospace applications require elevated temperature cure, and the coefficient of thermal expansion of aluminium is greater than most strap materials, tensile residual stress fields are developed in the substrate in the region surrounding the strap. These stresses will counter the retardation effects of crack bridging, by increasing the mean stress intensity at the crack tip and reducing crack closure. The resultant crack growth rates will depend on the balance between reduction in crack opening arising from crack bridging, and reduced crack closure arising from residual stresses.

A number of recent studies have demonstrated the benefits of the bonded strap approach. Heinimann et al[15, 16] show significant improvements of between 2 and 4 on crack growth life for a range of bonded retarder concepts. Detailed studies of the influence of strap stiffness and geometry have been made by Zhang, Figueroa Gordon,[17,18] with modelling work by Zhang, Boscolo, Allegri et al[19].

In these tests the influence on fatigue lives and fatigue crack growth rates of strap material stiffness, strap geometry, cure temperature (whether there are residual stresses or not) have been systematically investigated. The straps were bonded to SEN T samples of 7085 T 7651 aluminium alloy, 140 mm wide and either 5 or 10 mm thick. The crack retarder straps were bonded to one side of the samples. Three strap materials were studied. These were the fibre-metal laminate GLARE 1 (3/2), a high strength titanium alloy Ti 6 Al-4V and a unidirectional glass fibre (GFRP) Hexcel 913. Strap width was 20 mm, and the strap edge was located 20 mm from the notch tip (37 mm from the SEN sample edge). Straps were bonded either using FM94 film adhesive, cured at 121°C for 60 minutes, or by using Redux 810 adhesive cured at room temperature. Samples were subjected to fatigue loading at 10 Hz with an R ratio of 0.1. The loads were progressively reduced so that as the crack tip encountered the edge of the strap the nominal ΔK was 8 MPa $m^{1/2}$. Further details of samples, bonding and experimental procedures are described in 17-19.

Figure (3) shows the effect of strap stiffness expressed as substrate/strap compliance ratio, on fatigue crack growth rates at various crack lengths in relation to the strap position. All of the tests had the same load conditions and nominal stress intensity values. The bonding was performed at room temperature so there were no residual stresses. It can be seen that maximum retardation effect is found at the first strap edge, where there is a reduction in growth rate of a factor of 10 for μ= 0.16, produced by a titanium strap on a 5 mm thick substrate, and a factor of 2 for μ=0.016 produced by a gfrp strap on a 10 mm substrate. These ratios are broadly maintained while the crack tips are under the strap; but are progressively reduced as distance from the strap increases. As distances from the strap of 20-30 mm the growth rates of all the strapped samples converge to 1.5 X 10^{-7} m/cycle- a factor of 2 less than that of the substrate alone.

Figure (4) shows sample results from figure 3 plotted as crack length vs cycles. This representation integrates all the individual growth rate differences into the total life. The

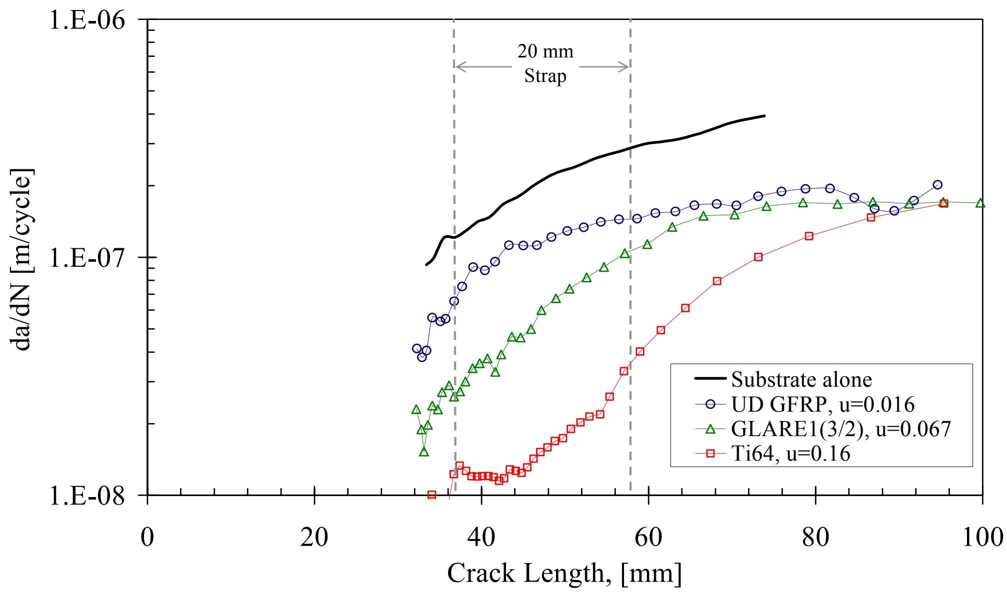

Figure 3 Fatigue crack growth rates plotted against crack length for different strap materials with different μ values. The behaviour of the substrate alone is also shown. All samples with room temperature cure

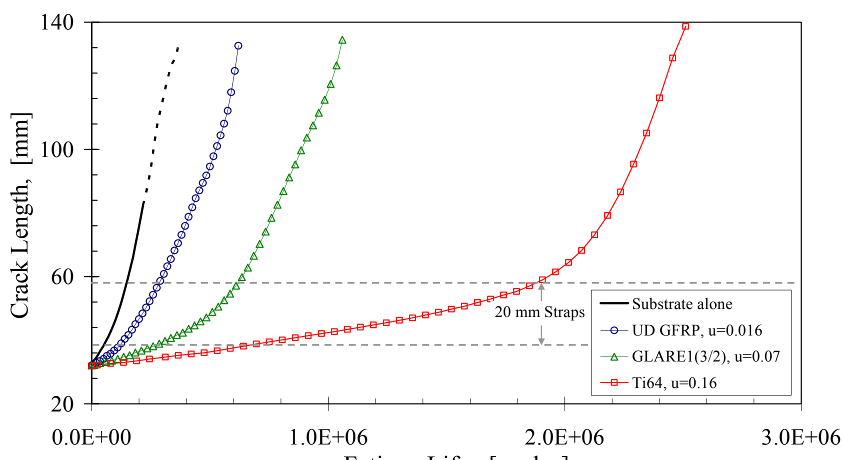

Figure 4 Crack length vs. cycles for the test data shown in figure 3, showing the effect of μ value on the total lives to failure

figure shows that the differences in life in this instance are similar to the differences in growth rate- approximately a factor of 10 increase in life for the titanium strap and a factor of 2 for the grp strap.

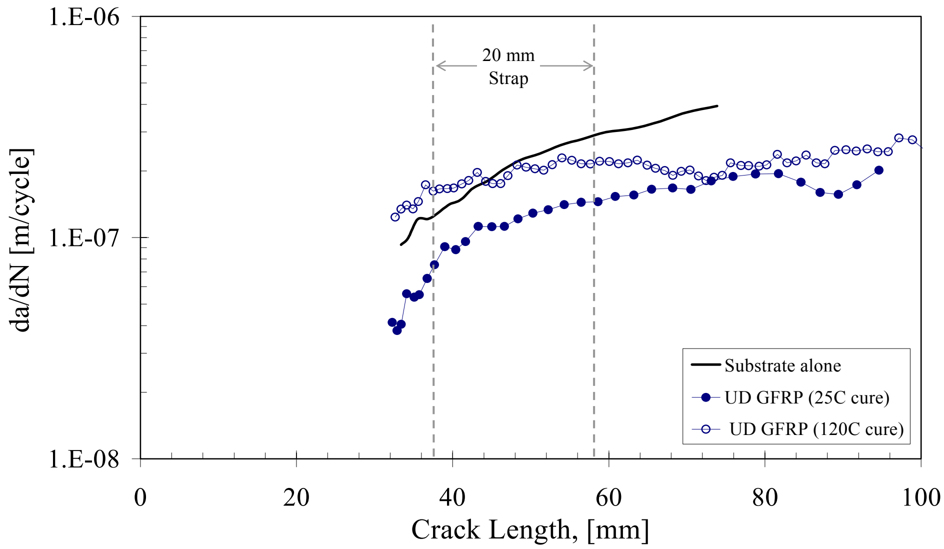

Figure 5 Crack growth rate plotted against crack length for grp strap; μ=0.016, showing influence of residual stress

These effects are greatly modified by the introduction of adhesives with elevated temperature cure and residual stresses. An example is shown in figure 5 for the grp strap with room temperature and elevated temperature cures. In the region before entering the strap growth rates are increased by a factor of 3; leaving the strap they are increased by a factor of about 30% and tend towards the same final value at long crack lengths where crack bridging is the retardation mechanism. In the region at the start of the strap, growth rates in the presence of residual stress are faster even than that of the unstrapped substrate, but this behaviour turns to retardation as the crack tip grows away from the region of largest residual stress. Beyond 70 mm crack length growth rates with and without residual stress are similar, there being little effect of residual stress on the crack bridging retardation. The net effect on overall crack life is to increase it, but less than for the room temperature cure.

5 SIMILITUDE AND DAMAGE TOLERANCE IN HYBRID STRUCTURES

The incorporation of a strap into a structure with its attendant changes in compliance and addition of local residual stresses together with changes in the crack closure and crack bridging arrangements imply a loss of similitude in crack growth analysis. The value of ΔK in conventional structures is of course given by the expression

$$\Delta K = \beta\sigma\sqrt{\pi a}$$

where σ is the remote stress, a the crack length and β a factor dependent on structure geometry and crack length. The local approach and similitude permits crack growth rates obtained on small laboratory coupons to be used to predict crack behaviour in structures of different scale and geometries, if the ΔK is the same. This will not be true in general for hybrid structures with bonded straps, as the straps will introduce additional parameters influencing crack growth rates, the location and size of which are independent of crack length and structure geometry. Use of identical ΔK does not imply identical growth rates.

The earlier part of this paper showed how the capability of different materials to exploit their static strength in damage tolerant design situations could be compared. Hybrid materials with straps clearly demonstrate increases in crack growth life compared with the unstrapped aluminium,.

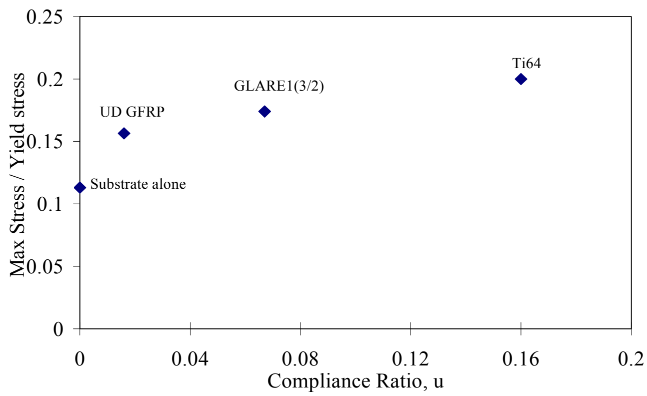

Figure 6 Increases in applied stress to achieve a crack growth life of 10^5 cycles with increasing μ value, expressed as a fraction of yield strength. Room temperature cure adhesive.

To assess damage tolerancein the same way as in figure 1, a constant amplitude loading comparison can be attempted for the specific geometry used in the experimental tests reported above. Crack growth rates can be represented as a function of the nominal (geometry dependent) ΔK. These data can be used to calculate crack growth lives at a range of different stress values for the SENT geometry at R = 0.1 The stress value required to achieve a life of 10^5 cycles can be identified and expressed as a fraction of the yield strength of the substrate. This is shown in figure 6. Figure 6 shows substantial increases at first for grp straps. As μ is increased further to the value of 0.16, the stress fraction continues to increase, to almost double that of the substrate 7085 material alone.

6 CONCLUDING REMARKS

The concept of bonded retarders offers real benefits to creation of damage tolerance in higher strength materials. The topic is still in its infancy and the full extent of the benefits of application of optimised strap materials bonded to skin stringer panels remains to be explored.

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