THOUGHTS ON ASSESSING AND MAINTAINING STRUCTURAL INTEGRITY

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ABSTRACT

This paper addresses recent Australian activities in two primary areas; viz: wide spread fatigue damage and repairs to primary structures. The problem of WSFD is discussed first. The paper then proceeds to discuss policy and procedures for the use of composite repairs to principal structural elements (PSE) and structurally significant items (SSI) and a modified form of the current FAA two stage evaluation and approval process is proposed.

KEYWORDS

Wide spread fatigue damage and repairs to primary structures

INTRODUCTION

The end of the cold war coupled with the down turn in the global economy and the high acquisition costs associated with the purchase of modern military and civilian aircraft has resulted in greater utilisation of existing aircraft fleets. This, in turn, has been reflected in an increasing number of structurally significant defects and increases the possibility of a reduction, or loss, of structural integrity due to fatigue. In the Australian scene the importance of maintaining continued airworthiness was highlighted by the November 1990 failure of a Royal Australian Air Force (RAAF) Macchi aircraft, Young [1], which suffered a port wing failure whilst in an estimated 6g manoeuvre. A tear down inspection program involving two fuselages, two fins and five horizontal tail planes was subsequently undertaken [1]. Six of the wings showed significant cracking indications and of approximately 1000 holes which were examined 100 revealed fatigue cracks, including major cracking in the D series rivet holes. This failure event subsequently led to a major change in the RAAF approach to Air Structural Integrity Management. These events have highlighted the need to develop both new methodologies and analysis tools for the assessment of structural integrity, particularly when the critical component contains large numbers of interacting flaws, as well as fresh approaches to maintenance and repair techniques. To this end the present paper focuses on Australian activities in two primary areas; viz: wide spread fatigue damage, and repairs to primary structural components.

CRACKING IN RAAF MIRAGE IIIO AIRCRAFT

During full scale fatigue testing of Mirage IIIO fighter aircraft wings at the Swiss Federal Aircraft Factory (F+W) fatigue cracks were discovered at the innermost bolt holes along the rear flanges of the main spars, with failure associated with cracking at bolt hole number 5. The test was then continued with a starboard RAAF Mirage IIIO wing (2190 hours service) and a port Swiss Mirage IIIS wing (510 hours service). After a relatively short test life cracks were then found at the inner most bolt holes. Cracking was also found in a number of other locations including Frame 26. Cracking was also experienced in the lower (tension) wing skin both at the fairing hole (nearest the main spar) and at the fuel decant hole in the lower wing skin, see Jones [2] for more details.

Following this test crack indications were confirmed at identical locations, both in the spar and in the lower wing skin, in wings of the RAAF Mirage IIIO fleet. At bolt hole number one the existence of two single leg anchor nut (SLAN) rivet holes meant that the cracking developed at the SLAN holes as well as at the main bolt holes. We were thus faced with the problem of distributed and interacting (three dimensional) flaws. This phenomenon prompted the development of a life enhancement scheme for the main spar using interference fit steel bushes. This essentially stopped cracking at the main bolt-holes and made the SLAN hole the next critical item. There were also instances when poor manufacturing processes meant that the SLAN rivet holes were not correctly aligned, i.e. straight. This resulted in a significantly greater level of crack interaction. A potential remedy for this problem was subsequently produced. This involved the bonding of close fit rivets into the SLAN holes, see [2].

To overcome the skin cracking in the (nearby) lower wing skin two boron epoxy repairs were developed, one for drain hole cracks and one for fairing hole cracks. These were implemented throughout the RAAF fleet and were present, with no evidence of failure, until retirement of the fleet.

The F+W fatigue test undertaken in Switzerland also resulted in a major failure, together with a number of nearby cracks, in the fuselage Frame 26. The major crack occurred at hole 18 on the left side of Frame 26A and extended across the entire flange and well into the web. Other cracks occurred in the region between holes 1 and 23 and included; a 9 mm crack in the inner strap plate and small (less than 3 mm long) cracks in holes 4, 7, 8, 18, 20 and 22. Prior to the major crack being detected significant cracking had also occurred in the outer strap plate in the bottom of the frame. At the time of the failure this region contained cracks with lengths of 40 mm, 31mm, 20mm, 18mm, and 12mm.

CRACKING IN RAAF MACCHI MB326H AIRCRAFT

The Macchi MB326H is an excellent example of the importance of understanding and managing WSFD. In this case Aermacchi conducted a series of fatigue tests on the MB 326 structure in order to meet the requirements of the Italian Air Force Specifications and AvP 970. Failure occurred in: i) centre section lower spar; ii) WAF and main lower spar cap. However, whilst fatigue tests on the fin revealed cracking at the main fin attachment no major failures occurred.

In the late 1970's fatigue cracking was discovered, in RAAF service aircraft, in the 04A centre section and this led to a change in the management philosophy for this component. In the early 1980's a Life-of-Type Extension Program (LOTEX) was carried out by the Commonwealth Aircraft Corporation (CAC) and Hawker de Havilland (HDHV) to extend the operational life of

the MB326H. In November 1990 Macchi aircraft A7-076 suffered a port wing failure whilst in an estimated 6g manoeuvre. It was subsequently found that failure was caused by fatigue cracking originating from the "D17" rivet hole in the lower spar cap. As a result of this event a Macchi Recovery Program was initiated to determine the structural condition of the fleet and to reassess the fatigue lives and management philosophies of the main structural components. As recommended in DEF STAN 00-970 a tear down inspection program was established. In this program ten post LOTEX wings, two fuselages, two fins and five horizontal tail planes were destructively inspected, see [1] for details. Six of the wings showed significant cracking indications. Of the, approximately, 1000 holes which were examined 100 revealed fatigue cracks, including major cracking in the D series rivet holes.

This program revealed the fatigue critical locations in the centre section lower spar boom to be bolt holes 3-6 and 17-20. The flaws were highly three dimensional in nature and, from the failure investigation of aircraft A7-076, the failure process had involved a number of interacting cracks. In this case cracking had progressed from a Web attachment Fastener hole through the flange as well as from the nearby Wing Attachment Fastener (rivet) Hole, see Fig. 1. Fractographic evidence also indicated multiple crack origins at the root of the rivet hole, see Fig. 1. A more detailed description of this program, its underlying philosophy and the Macchi Aircraft Structural Integrity Management Plan (ASIMP) is given in [1].

CRACKING IN RAAF F111C AIRCRAFT

F-111C aircraft in service with the Royal Australian Air Force (RAAF) have been found to experience cracking in a number of locations. Cracks have been found in Stiffener Runout Number 2 (SRO #2) and in Mouse Hole number 13 (MH#13) in the Wing Pivot Fitting (WPF). Cracking in SRO #2 has, in some instances led to failure, during Cold Proof Load Testing (CPLT) in the USA, Bland [3]. Interestingly the concept of performing CPLT, at -40^{o} C, was first introduced to ensure continued structural integrity of the USAF F111 fleet, see Jones et al [4]. The current load cycle applied during CPLT, which is referred to as SIP III, involves loading to -2.4g, 7.33g -3.0g, 7.33g . Stiffener Runout No. 2 (SRO #2), on the upper surface of the WPF, is the most critical location. Here the local bending field results in compressive yielding under high positive *g* loads. The negative *g* loads then produce very high tensile strains. It is these tensile strain that were responsible for failure in CPLT. To overcome this problem it was necessary to: 1) Determine critical crack sizes and inspection intervals, both for unmodified and modified aircraft, 2) Change the geometry of the local region, thereby reducing the K_t, and 3) Provide an alternate load path so as to partially by pass the critical region. To achieve the first two requirements a series of detailed finite element analyses were performed and the results used in conjunction with fractographic observations and fatigue meter histories. To meet the third objective a boron epoxy doubler (reinforcement) was developed [4].

To meet the requirements for continued airworthiness it was necessary to determine the associated inspection intervals. To this end it was necessary to obtain the residual stress, after CPLT, and the stress "per g" both with and without doubler and with various grind out configurations. To obtain this information required a detailed elastic-plastic analysis since during CPLT SRO #2 undergoes gross plastic yielding. This analysis program was supported by series of full scale structural tests. (In this test program the strains in the vicinity of SRO#2 were found to be in excess 18,000 $\mu\varepsilon$.)

Classical techniques for modelling this cyclic (plastic) behaviour had inherent difficulties in representing the response to large cyclic inelastic strain excursions. Indeed, the use of classical analysis techniques resulted in a predicted inspection interval, for the modified structure, of under 500 hours. This contrasted with service experience which has shown that with both the doubler and the change in the runout geometry there had been little further cracking. Indeed, for modified aircraft there has been no further cracking since 1985, see [4] for more details. To overcome this shortcoming a "unified constitutive" model, see [4], was used. With this approach the stress per g and the residual stress thus calculated were consistent with fleet experience and resulted in an extension of the inspection interval from under 500 hours to more than 1400 hours.

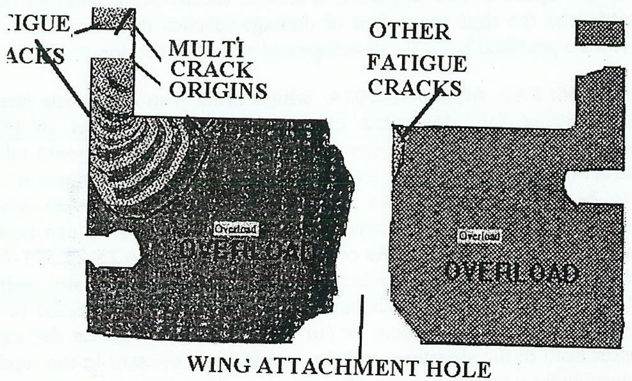

Fig. 1. Cross section view of failure surfaces in the Macchi spar.

CRACKING IN RAAF F/A-18 TEST ARTICLE

The various fatigue test(s) undertaken by McDonnell Douglas in support of the F/A-18 have indicated a large number of potential hot spots, including the FS488 aft bulkhead flange, mold line and wing attachment lug. To further assess the fatigue performance of the FS488 aft bulkhead a full scale fatigue test was performed in Australia on a stand alone FS488 bulkhead. The test was performed to primarily address the region of the wing attachment lug. In this test failure resulted from a fatigue crack approximately 6 mm deep. However, post-failure inspection of the test article revealed the presence of several hundreds of cracks within the critical region, see Barter et al [5]. This test program again highlighted the need to develop both advanced NDI

techniques as well as new analysis tools for the assessment of structural integrity, particularly when the critical component contains large numbers of interacting three dimensional flaws.

IMPLICATIONS FOR DESIGN AND ANALYSIS METHODOLOGIES

In previous sections we have seen that the trend in operating existing aircraft approaching or beyond their intended design life is often reflected in an increasing number of structurally significant defects. Indeed, the particular problems discussed in this paper have been chosen to highlight importance of understanding and managing ageing structures and to highlight the need for simple (to use) and yet accurate damage assessment tools . From these case studies it is apparent that a method of rapidly analysing cracks, including interacting cracks, in complex geometries and under realistic loading conditions, together with a capability to account for complex crack interaction is required. To achieve this goal the finite element alternating method Nishioka et al [6,7,&8] and Jones et al [9] has been applied and is under development both in Australia and in the United States. The finite element alternating technique (FEAT) makes extensive use of the analytical solution for a 3-D elliptical flaw subject to arbitrary crack face loading. The key to implementing this solution in the finite element alternating technique was the development, by Nishioka and Atluri [6], of a general procedure for evaluating the necessary elliptic integrals. To date this technique has been successfully applied to solve a range of three dimensional problems; viz. thick plates [6], pressure vessels [7], aircraft attachment lugs [8] and was recently extended [9] to include arbitrary interacting cracks.

One major advantage of this technique is that by combining the finite element method with the analytical solution we enable accurate results to be obtained using only a relatively coarse mesh. Furthermore, since cracks are not modelled explicitly this means that the crack configuration can be changed without complex re-meshing and as the crack geometry changes it also removes the need for tedious re-meshing. Consequently, for problems associated with WSFD and MSD the finite element alternating method has been shown to be a very efficient and cost effective method of analysis [9]. This is aptly illustrated in Figure 2, which presents the results for a single semi-elliptical surface flaw in an infinite body obtained using a 6x6x6 mesh, see [9] for more details. In this figure the mean difference between the finite element alternating technique values and those published in the open literature, Rooke and Cartwright [10], was less than 5%. Furthermore, the values obtained also tended to lie between other published data.

COMPOSITE REPAIRS TO PRIMARY STRUCTURES FOR MILITARY AIRCRAFT

To address the problem of continued airworthiness Australia has pioneered the development of adhesively bonded repairs, Baker and Jones [11]. However, before this technology can be routinely used for repairs to primary structural components a certification methodology must be developed. In line with current FAA, USAF procedures and RAAF practice all structural repairs carried out to aircraft should be approved by a competent airworthiness authority. The repair may: a) already be in the structural repair manual (SRM); or b) have previously been approved by the associated airworthiness authority, in which case the repair may be incorporated by following unit procedures. In those cases where prior approval does not exist the action taken will depend upon whether or not the damage affects a PSE/ SSI or whether the repair will have an effect on other major repair criteria. If the damage affects a PSE/ SSI, or the repair impacts on other major repair criteria, the repair would require approval by the appropriate airworthiness directorate. It is recommended that following the principle proposed by Jones and Smith [12] that, without guidance, repairs should not be attempted to damaged structures where the residual strength has been reduced beneath 1.2 times limit load.

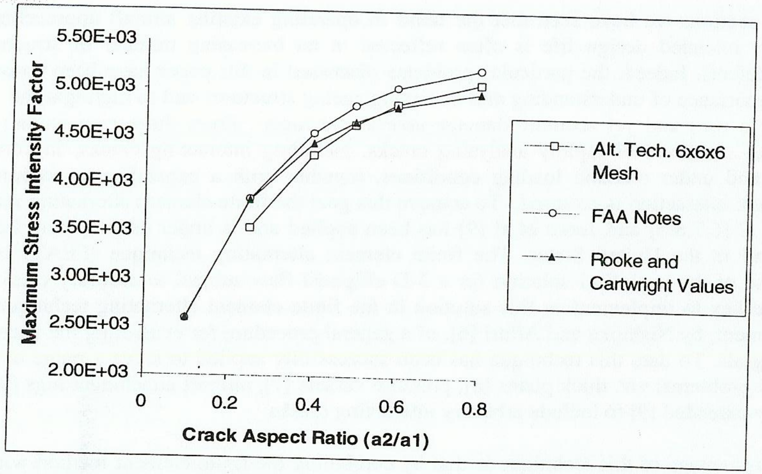

Fig. 2. Comparison of the maximum stress intensity factor versus crack aspect ratio, from [9]

For composite repairs to PSE/SSI reference [12] proposed a two stage repair approval, based on FAA AC No: 25.1529-1. A typical two stage evaluation would be:

a) A static structural strength evaluation is made prior to release of the aircraft into service with a stated time for completion of the damage tolerance evaluation.

In accordance with FAA AC No: 20-107A 'static strength should be demonstrated through a program of ultimate load tests in the appropriate environment, unless experience with similar designs, material systems and loading is available to demonstrate the adequacy of the analysis supported by subcomponent tests, or limit load tests.'

b) A damage-tolerance evaluation of the repair is made within a prescribed time period after this interim release. The final evaluation must reflect any changes in the related inspection program, including threshold, interval, and inspection procedure.

In accordance with FAA AC No: 25.1529-1, final repair approval should not be made until a damage-tolerance evaluation has been completed and has shown that the repair is adequate to assure continued airworthiness. In accordance with FAA AC's No: 25.1529-1 and AC No: 25.571-1A the damage tolerance evaluation of the repair is intended to ensure that should serious fatigue, corrosion, environmental degradation, impact damage, disbonding,

delamination or accidental damage, occur to the repair then the remaining structure can withstand reasonable loads, without failure or excessive structural deformation, until the damage is detected.

Following the current FAA procedures for damage tolerant assessment, as given in FAA AC No 25-25.571-1A, the damage tolerant assessment of both the structure and the composite repairs should be determined by analysis, supported by test evidence in the appropriate environment, unless (as stated in FAA AC No 25-25.571-1A) 'it has been determined that the normal operating stresses are of such a low order that serious damage growth is extremely improbable'. In this process it should be shown that: (a) The repaired structure, with the extent of damage established for residual strength evaluation, can withstand the specified design limit loads (considered as ultimate loads); (b) The damage growth rate both in the structure, the adhesive and the composite repair, allowing for impact damage, interply delamination and adhesive debonding under the repeated loads expected in service (between the time the damage becomes initially detectable and the time the extent of damage reaches the value for residual strength evaluation) provides a practical basis for development of the inspection program.

Furthermore in accordance with FAA AC No 20-107A, which deals with composite structure, the repeated loads should allow for load-time effects, i.e. dwell time as in [3] and environmental i.e. temperature and humidity, spectra. The loading conditions should take into account the effects of structural flexibility, load dwells and rate of loading, unless it can be shown experimentally to be insignificant. This is particularly important when assessing residual strength of the repaired structure. The damage tolerance characteristics can be shown analytically by reliable or conservative methods. As outlined in FAA AC No 25-25.571-1A this could involve: (i) By demonstrating quantitative relationships with structure already verified as damage tolerant; (ii) By demonstrating that each damage event would be detected before it reaches the value for residual strength evaluation; or (iii) By demonstrating that the repeated loads and limit load stresses both in the structure, and the interlaminar stresses in the repair etc, do not exceed those of previously verified designs of similar configuration, materials, and that inspectability is not compromised. The maximum extent of immediately obvious damage from discrete sources should also be determined and the remaining structure shown to have static strength for the maximum load (considered as ultimate load) expected during the completion of the flight. This may be an analytical assessment.

The static strength, allowing for fibre failure, interlaminar failure and disbonding, of the composite repair should be demonstrated through a program of component ultimate load tests, including any associated load dwells as in [3], unless experience with similar designs, material systems and loadings is available to demonstrate the adequacy of the analysis. As outlined in FAA AC No 20-107A: (a) When carried out, the component static test may be performed in an ambient atmosphere if the effects of the environment are reliably predictable, via a series of subcomponent or detail test specimens, and are accounted for in the static test or in the analysis of the results of the static tests. (b) The static test articles should be fabricated and assembled in accordance with production specifications and processes so that the test articles are representative of production structure. (c) When the material and processing variability of the composite structure is greater than the variability of current metallic structures, the difference should be considered in the static strength substantiation either by deriving proper allowables or design values for use in the analysis, and the analysis of the results of supporting tests, or by accounting for it in the static test when static proof of structure is accomplished by coupon test. (d) Composite repairs to structures that have high static margins of safety may be substantiated by analysis supported by subcomponent, element, and/or coupon testing in the appropriate environment.

The nature and extent of analysis or tests will depend upon experience with applicable previous repairs, see [12]. In general in the absence of experience with similar repairs substantiation tests, approved by relevant airworthiness authority, should be performed. When selecting the damage/degradation to be considered in the repair substantiation, the following failure modes, unique to composite materials, should also be included; viz: (a) Failure of doubler/repair by fibre failure; (b) Failure in the adhesive (cohesion failure); (c) Failure at the adhesive/substrate interface(s) (adhesion failure); (d) Failure of the doubler/repair by interlaminar failure.

The environment used for testing should also be appropriate to the expected service usage. The repeated loading should be representative of anticipated service usage. The repeated load testing should include damage levels typical of those that may occur during fabrication, assembly, and in-service, consistent with the inspection techniques employed. The damage tolerance test articles should be fabricated and assembled in accordance with production specifications and processes so that the test articles are representative of the in-service repair.

The extent of initially detectable damage should be established and be consistent with the inspection techniques employed during the repair and in-service. Flaw/damage growth data should be obtained by repeated load cycling of intrinsic flaws or mechanically introduced damage. The number of cycles applied to validate a no-growth concept should be statistically significant, and may be determined by load and/or life considerations. The growth or no growth evaluation should be performed by analysis supported by test evidence or by coupon tests. The extent of damage for residual strength assessments should be established and residual strength evaluation by component testing or by analysis supported by test evidence should be performed considering that damage. The evaluation should demonstrate that the residual strength of the structure is equal to or greater than the strength required for the specified design loads (considered as ultimate). It should be shown that stiffness properties have not changed beyond acceptable levels. For the no-growth concept residual strength testing should be performed after repeated load cycling.

In the process recommended in [12] it is essential that an inspection program, together with the associated SID, should be developed consisting of frequency, extent, and methods of inspection for inclusion in the inspection schedule. Inspection intervals should be established, allowing for the structure, the adhesive and the composite repair, such that the damage will be detected between the time it initially becomes detectable and the time at which the extent of damage reaches the limits for required residual strength capability. For the case of no-growth design concept, inspection intervals should be established as part of the maintenance program. In selecting such intervals the residual strength level associated with the assumed damage(s) states it should be considered that : (i) The structure should be able to withstand static loads (considered as ultimate loads) which are reasonably expected during a completion of the flight. The extent of damage should be based on a rational assessment of service mission and potential damage relating to each discrete source. (ii) The effects of temperature, humidity, and other environmental factors which may result in material property degradation should be addressed

in the damage tolerance evaluation. These requirements directly reflect current FAA approaches for Composite Aircraft Structures, see FAA AC No 20-107A.

EXTENSION OF CLASSICAL LAP JOINT THEORY

There are several methods available for designing composite repairs to cracks in thin metallic skins, i.e. typical thickness less than 3mm, see Baker and Jones [11]. The finite element method was the first to be developed, and has been used to design several complex repair schemes such as the repair of fatigue cracks in the lower skin of Mirage aircraft [17] and cracks on the upper surface of the wing pivot fitting of F111C aircraft in service with the Royal Australian Air Force (RAAF), see [4]. Following the development of this approach the work presented by Rose [13] revealed that the stress intensity factor for a crack repaired with a bonded composite doubler approached a constant value as the crack length increased, thus significantly simplifying the design process. Simple approximate formulae for this asymptotic value were then derived for the case of a 'repaired' crack in a thin panel under a remote uniform stress field, [13, 2, 14]. These formulae were based on the premise that, for a sufficiently long crack in a structure subjected to a remote uniform stress field, the central region of the doubler (repair), over the crack, behaves like an overlap joint.

As a result of this analogy the problem of a bonded symmetric lap joint, see Hart-Smith [15] and Jones et al [16], can be used, as a first approximation, to determine the peak adhesive stresses and strains in composite repairs to cracks in thin skins.. Unfortunately, although structural adhesives and composites are often strongly load history and rate dependent the analysis presented in [15] did not allow for this dependency. Consequently this analysis was extended in [16] to allow for this observed dependency. Indeed, it was shown in [16] that the peak adhesive shear stresses and strains, developed in both a joint and a composite repair, could be strongly dependent on the loading rate. Indeed, reference [16] showed that the use of the (rate independent) analysis procedures outlined in [15] tended to give results which lay between those obtained for high and low loading rates. Unfortunately this meant that the (rate independent) solutions for the adhesive shear stresses and shear strains and the interlaminar shear stresses in the composite could not be said to be generally conservative. (This formulation was subsequently found to be necessary to explain the failure mechanism occuring in an adhesively bonded pipeline.)

As an illustration let us consider the boron epoxy repair to cracks in the lower wing skin of Mirage III aircraft described by Baker et al [17]. In this case we take the lower adherend, i.e. the aluminium wing skin, to have E = 70000 MPa and a thickness of 3 mm, and the upper adherend, i.e. the boron epoxy repair, to have a primary modulus E_{11} of 209000 MPa. and a thickness of 1.0 mm. The adhesive is considered to be FM73 and is taken to be 0.2 mm thick and the overlap length was considered to be 80mm, see [17] for more details of this repair.

In this case [16] solved the governing, load history dependent, equations for the cases when a remote stress of 490 MPa was applied to the wing skin. This value was chosen to approximate the ultimate (material) load carrying capacity of the aluminium wing skin. In an attempt to represent realistic in flight loading rates the load was assumed to be applied to the skin in 0.1 and in 3.5 seconds. In this analysis the overlap length of 80 mm was divided into 100 equally spaced grid points. The resultant maximum values for the adhesive stresses, and (engineering) strains, are given in Figure 9 together with the solution obtained as per [15] using the manufacturers (rate independent) stress-strain curve.

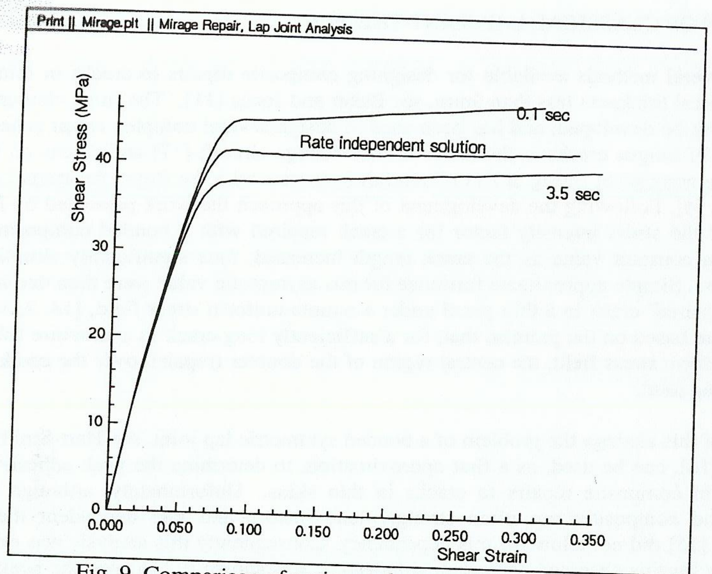

Fig. 9. Comparison of various solutions using a rate dependent analysis

This example reveals that, for the specific repair considered, the peak adhesive stresses and strains, developed in the joint/repair, are strongly dependent on the loading rate. In this case use of the (rate independent) analysis gave results which lay between those obtained for the high and low loading rates, see [16] for more details. Unfortunately, this meant that the (rate independent) solutions for the adhesive, and therefore the composite, shear stresses and shear strains could not be said to be generally conservative.

For repairs to more complex 3D flaws in thick structural components the simple design approaches outlined above are no longer applicable and a detailed finite element analysis is generally required. In this case the need to perform a rate dependent analysis means that a consistent Jacobean, see Trippit et al [18], is often necessary. The development of a methodology for the derivation of a consistent Jacobean for generally anisotropic materials is currently underway as part of a collaborative industry research program between Monash University and AMCOR Pty Ltd.

At this point it should be noted that bonded structures are becoming increasingly used in non-aerospace applications, i.e. in the oil and gas industry and for civil infra-structure. In these industries recent events have highlighted the load history dependent nature of the failure

processes. This is aptly illustrated by a recent (Monash) test program on bonded pipelines in which, during certification testing, the degradation of the joint took more than four hours to first become apparent. This observation highlighted the need to allow for load history in the initial design process.

CONCLUSION

The past decade has seen a number of very significant changes both within Universities and in the global engineering industry and the problem of aging structures, both civilian and military, has become even more acute and visible The problems discussed in this paper have been chosen to illustrate particular aspects of this phenomenon as well as to highlight perceived trends and future needs, both in the fields of certification and in damage assessment and repair technology.

To assist in the extension of composite repair technology to primary structural components this paper has also presented a possible methodology for the damage tolerant assessment of composite repairs to primary structural elements. The proposed methodology is based on existing FAA, RAAF and USAF procedures. The central point is a two stage repair approval. In the first stage it is sufficient to show that the repair will meet intermediate and short term strength requirements (ultimate strength). A more extensive investigation is then required to show long term strength requirements. However, the latter approach must be scheduled for completion before the aircraft is put at risk from a structural fatigue failure.

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