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DEVELOPMENT OF A COMPOSITE REPAIR AND THE ASSOCIATED INSPECTION INTERVALS FOR THE F111C STIFFENER RUNOUT REGION

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SUMMARY

This paper presents an overview of the structural aspects of the design and development of a local reinforcement designed to lower the stresses in a region of the F-111C wing pivot fitting which is prone to cracking. The stress analysis, with particular emphasis on the use of a unified constitutive model for the cyclic inelastic response of the structure, representative specimen testing, thermal analysis and full scale static testing of this design are summarised.

INTRODUCTION

F-111C aircraft in service with the Royal Australian Airforce (RAAF) are known to experience cracking in the Wing Pivot Fitting (WPF), and this has resulted in failure during Cold Proof Load Testing (CPLT) [1]. CPLT was originally introduced to ensure the continued airworthiness of the structure and involves the application of loads to the aircraft, at -40^{o} C, where the load cycle is defined by the current Structural Integrity Program (SIP). Currently SIP III requires in a -2.4g, 7.33g -3.0g, 7.33g load applied to the aircraft. In the F-111C aircraft in service with the RAAF the critical area is Stiffener Runout No. 2 (SRO #2) on the upper surface of the WPF. The cause of these cracks and subsequent failure in the CPLT is discussed in [2]. In summary, local bending of the stiffener results in compressive yielding in the stiffener runout under high positive *g* loads. As a result, subsequent negative *g* loads, and to a limited extent positive *g* loads, produce very high tensile strains. The structure of the wing is comprised of a 2024-T856 wing skin spliced onto a D6AC steel WPF by an array of flush stainless steel fasteners. Cracking is occurring inboard from the splice in the runout region of the top surface integral stiffeners. The general geometry of the critical region is shown in Figure 1.

This paper presents an overview of the design and development of a boron epoxy reinforcement (doubler) designed to lower the stresses in the critical region [3,4]. This design is currently being installed progressively on RAAF fleet aircraft prior to them undergoing their next CPLT.

In order to determine the associated inspection intervals 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. Since during CPLT SRO #2 undergoes gross plastic yielding, to obtain this information requires a detailed elastic-plastic analysis. However, classical techniques for modelling this cyclic behaviour have inherent difficulties in representing the response to large cyclic inelastic strain excursions. Indeed, the use of classical analysis techniques resulted in an inspection interval, for the modified structure, of under 500 hours. This contrasts with service experience which has shown that there is little further cracking. Indeed, for modified aircraft there has been no further cracking since 1985. To overcome this shortcoming the use of a "unified constitutive" model, see [5,6], was investigated.

REPAIR PROCEDURES

The reinforcement (doubler) consists of two discrete boron/epoxy doublers bonded, using an elevated temperature cure cycle, to the upper surface of the F-111 WPF. The doublers are located over those stiffeners known to be subject to the highest strains, see Figure 2. One large doubler covers stiffeners number 4 and 5, and a smaller doubler covers stiffener number 2. The doublers were fabricated from uni-directional boron/epoxy composite and are co-cured with an adhesive at elevated temperatures. Each doubler consists of two segments. The lower segment is bonded to the D6AC steel to provide a bridge level with the aluminium skin step, and the upper segment is bonded to the aluminium skin and over the lower segment. Each segment of the preferred design consists of approximately 60 plies of boron in the middle, suitably tailored to achieve single plies at the extremities of the doubler. The lower segment contains a cutout region adjacent to the aluminium skin step, designed to reduce the stress concentration produced by the discontinuity of the load path, see Figure 3.

FINITE ELEMENT MODELS

Two Dimensional Models

Initially a series of two and three dimensional finite element models were created to represent the unreinforced critical region, see [3,4], and the models were calibrated using strain data from a survey conducted in the US [7] in support of the failure investigation of a WPF. For the reinforced model, the left hand wing was deemed to be the design case, as from all available data at the time, this wing recorded the highest strain levels and thus the most severe problem. In order for the model to adequately represent the local detail of the structure in the wings, spring elements (or equivalent plate elements) were used to account for the restraint provided by the remainder of the structure. One spring was attached to the extreme left hand side of the model and the other to the vertical web. Both springs only possessed stiffness along their major axes. The bolts which join the WPF to the upper and lower aluminium splice were included and were modelled as described in [3].

MATERIAL PROPERTIES AND FAILURE CRITERION

The material properties used for the boron/epoxy system were E_{11}=208.1 GPa, E_{11}/E_{22}=8.18, $G_{12}=G_{13}$=7.24 GPa, G_{23} =4.94 GPa, $E_{33}=E_{22}$, $\nu_{12}=\nu_{13}$ =0.1677, ν_{13} = 0.035 and the adhesive has G= 750 MPa and ν = 0.35. The Young's modulus and Poisson's ratio for the aluminium were taken to be 71×10^3 MPa and 0.33, respectively. Young's modulus, Poisson's ratio and yield stress (compressive) for the steel were taken to be 207 GPa, 0.3 and 1600 MPa respectively. The doubler design criteria were to achieve a stress reduction sufficient to prevent yielding of the steel, and to withstand an applied loading of 7.3*g* or -2.4*g** . The design allowables for the adhesive shear stress $\tau <$ 40 MPa, peel stress $\sigma_p <$ 40 MPa, and fibre stress $\sigma_f <$ 1000 MPa for the boron/epoxy system were assumed. The interlaminar stresses in the doubler must also be below their critical values. The manufacturer's value for the boron/resin system used are : $\tau_{xy} <$ 90 MPa and $\sigma_y <$ 70 MPa

ELASTIC-PLASTIC ANALYSIS

Unified Constitutive Model

Classical techniques of modelling plasticity have a high degree of accuracy provided the inelastic strains are kept small. However, they become increasingly inaccurate when the material exhibits high inelastic strains and undergoes cyclic loading that causes gross plastic yielding. The current problem in the F-111 SRO #2 undergoes the two previously mentioned phenomena and the use of classical techniques was found to lead to large errors. To overcome this shortcoming the Unified Constitutive model, originally developed by Ramaswarmy [5], was used. (This model is an extension of the generic back stress and drag stress model proposed by Bodner and Stouffer [6].) Indeed, the authors have extensively used, and developed [8,9], these constitutive equations to model non-linear material behaviour in aluminium, adhesives and steels. In order to represent the material D6ac steel correctly in the high strain, cyclic loading regime, experimental stress/strain data was obtained from coupon tests [10], at room temperature. The experimental stress/strain curve under a loading of III SIP was compared with one generated using the Unified Constitutive model (see Figure 4) and one generated using classical plasticity finite element (see Figure 5). From these two Figures it is clear that classical plasticity does not adequately represent the material behaviour under cyclic loading and as the strain level is increased so too is the error.

Finite Element Configuration

Considerable wing-to-wing geometry variation exists in the fleet. Consequently, although the basic finite element models used in this analysis were as described in [3,4], it was necessary to perform a parametric study to quantify the effect of this variation on structural integrity. A total of 36 models, incorporating the various geometric configurations including plate thickness variation for each of the 7 main configurations, grind out depths and kink angles, i.e. the angle between the upper plate and the aluminium wing skin, were examined. The parameters in the analysis matrix analysed were directly based on the fleet inspection data collected by the RAAF.

Model Calibration

The unreinforced models were calibrated from the USAF strain survey data [7]. The finite element mesh used and gauge locations 6L and 9L used in the calibration process are shown in Figure 6. The reinforced models were calibrated from the recent CPLT of the A8-113 aircraft [16] and the 'kink' angle, i.e the angle between the upper plate and the aluminium skin, was obtained from the RAAF doubler application reports for the A8-113 wing. In general good agreement, i.e to within 7%, was obtained with the experimental strain gauge data. The unreinforced finite element analysis was calibrated with the USAF strain survey. The plastic analysis results are shown in Figure 7.

Residual Stresses

This analysis revealed that, for the nominal wing geometry, the peak residual stress following CPLT was approximately 1000 MPa; this contrasts with a value of approximately 1950 MPa obtained using classical plasticity, and that following reworking of the stiffener, see below, and doubler installation this reduces to less than 50 MPa. It was also found that following doubler installation the stress per g, at the critical location, is reduced by approximately 30%.

REDESIGN OF THE STIFFENER RUNOUT REGION

Finite Element Analyses

When bonded to the upper surface of the WPF the doubler reinforcement was found to considerably decrease the stress magnitude in the stiffener runout, approximately 30%, see [3]. Even so if the geometry of the SRO is left unmodified the stress level may be sufficient to cause yielding. This is a direct result of the stress concentration produced by the sharp re-entrant corner at the runout. To suppress local yielding various grind radius configurations (4mm, 11mm, and 22mm) at the corner as well as complete removal of the stiffener runout were investigated [3]. From the results of the stress analyses, using Von Mises' criterion, for the 4mm, 11mm, 22mm and the total grind out cases, the 22mm case was chosen as the optimum rework. Indeed, it was found that the combination of a 120 ply doubler and a grind out radius of approximately 22mm will achieve the desired stress reduction and prevent yield under CPLT loading. This prediction was confirmed via a thermal emission experiment [12].

In the case of the "complete" stiffener grindout (see [3]), the maximum value of the stress in the critical region is less than for the basic reinforced structure. However in this case the problem of crack initiation in the plate is a major concern.

DOUBLER STRESSES

The locations of the most critical points in the doubler were also investigated [3]. The design of the reinforcement was developed to minimise the stresses at these locations. These 'design points' are shown in Figure 8 and are labelled A to I. (The adhesive filler is present in the doubler cut out to provide it with stability.) For the adhesive the combined stress state elevates the permissible shear stress, such that the computed values are below critical (see [3]). The combined stress state in the boron/epoxy is also beneath critical values. This can be evaluated either using the critical energy or using a stress polynomial failure criteria, see [12]. The fibre stresses are below the critical design value. The adhesive filler has the effect of slightly reducing the peak values of the stress at points 'C' and 'F'. It also decreases the peak bending at point 'H' on the doubler.

For extreme kink angles in conjunction with geometric imperfections the interlaminar stresses (strain energy) in the upper doubler exceeded allowables. This was reflected by early failures in CPLT. To overcome this the local matrix material thickness was increased in the critical region of the doubler. This also has the effect of increasing the local load transfer region. As such interlaminar failure considerations, rather than the adhesive allowables, drove the final design concept. This is in marked contrast to the PABST design philosophy, see [13], whereby joints and composite repairs are designed on the basis of the maximum load transfer capability of the adhesive. In this case adopting the PABST design approach would have resulted in an unconservative design and catastrophic failure of the repair. Indeed, this is illustrated by the early failures in CPLT for doublers which were initially designed on the basis of adhesive allowables.

FULL SCALE STRUCTURAL TESTS

In the final stages of the proof-of-concept phase, a detailed strain survey was conducted on an F-111C wing both with and without doublers [14]. The results of the reinforced case survey were compared to those for the unreinforced case [15]. A damaged and subsequently repaired wing and wing carry through box (WCTB) were made available for use in the ARL strain survey program. A rig was fabricated to support the wing and WCTB, which was capable of applying the proof load conditions of 7.3*g* and -2.4*g*. Load was applied to the wing by hydraulic jacks located at the four pylon points, the number 2 flap track attachment point on the rear spar, and at the rib nearest the wing tip.

The wing loading arrangement reproduced the loads experienced by the wing during certification testing of the F-111 in the CPLT, at a sweep angle of 26^{o}.

Cold Proof Load Test

The final test in the 'proof of concept' phase of this design was under CPLT conditions at -40^{o}C [16]. Due to the scale and economical restraints it was decided that, for the purposes of ascertaining the reinforcement integrity, only the area in the immediate vicinity of the reinforcement needed to be cooled. For this to be achieved an insulated chamber constructed from polystyrene foam was erected around the

WPF. The construction extended two metres outboard from the wing root, two metres chordwise and was one and a half metres deep.

With the temperature of the upper wing surface approximately steady at -40°C, the wing was successfully loaded to -100% proof load. Following this the load jacks were reconfigured, whilst holding the temperature steady, and uploading was commenced. A strain reading was taken at +95%, but then the wing suddenly failed approaching +100% proof load.

The test article wing had been salvaged from a crashed aircraft, and was repaired to a state capable of withstanding 'significant' test loads. The repairs to this wing, as described in [16], were extensive and of 'boiler plate' construction as distinct from aircraft quality. Thus, from the inception of the strain survey program, higher loads were approached cautiously.

During loading of the wing at high levels, the deflected shape of the wing revealed a sharp change in slope at the outboard end of this mechanically repaired region. It was in this region that failure occurred during this CPLT. The areas of failure, in this vicinity (see [16]) were the rear and forward spars, lower wing skin, a section of the upper wing skin and all internal spars. Note, the failure occurred through the aluminium wing structure and thus was not affected by the applied temperature. No structural damage was detectable on the WPF.

Whilst there was a concern over the non-linear behaviour of a few doubler gauges (see [16]), sufficient data were available to demonstrate that the doublers behaved satisfactorily during CPLT. The delamination and debond of the small doubler was considered to have been caused by the 'shock' loading experienced immediately upon failure of the wing. This load was significant as can be deduced from the resultant damage to the test rig in the immediate vicinity below the doubler. In view of the observed non-linearity in a few strain gauge readings the corners of the production doublers were 'softened' to reduce their stiffness. This modification results in the corner of the doublers having a chordwise, as well as a spanwise, taper thus reducing the load concentration at these locations.

IN-SERVICE PERFORMANCE

Currently fourteen aircraft have been reinforced and there were problems associated with three aircraft reinforced early in the program. A brief description of the early prototype doublers and the current production doubler system is given. The early prototypes of the doublers were installed on Aircraft A8-148, followed by aircraft A8-144. In summary, failures, at high load levels during CPLT, occurred in both doubler sets. Both sets of wings were considered to vary significantly (geometrically) from the specification drawings. Post-failure investigations revealed some deficiencies in the doubler application process, and these were subsequently addressed.

Following the early doubler failures modifications were made to the application process resulting in the current production models. Specifically, as the failures occurred in the boron/adhesive interface, the changes mainly addressed this area. In summary, (a) The matrix thickness near the top of the lower doubler was increased to increase the damage tolerance of this region, (b) Micro-cracking induced into the boron matrix due to the grit-blasting procedure was eliminated, (c) All adhesive thicknesses were doubled.

All these changes were certified through laboratory coupon testing and Aircraft A8-142 was the first to receive these changes. In addition, an attempt was made to apply higher preloads during the application process. Unfortunately, due to a problem with the system used to preload the wing, the doubler was formed on the aircraft wing at a preload of 98 kN (22,000 lbs), but was cured to the wing at 129 kN (29,000 lbs). This resulted in non-conformity of shape at the higher preload. Following the doubler application process on unloading the wing, at RAAF Base Amberly, doubler failure was experienced, at zero load, during a 10 minute pause to collect strain data. This failure did not occur on commencement of the strain survey but took a considerable time to occur.

Two doublers on A8-142 that also failed during CPLT Post-failure investigation revealed large void areas. This failure, which also occurred during a 10 minute pause to collect strain data, was also attributed to the non-conformity of the doublers to the wing curvature during the higher preload curing procedure.

Subsequent to this failure the preload was fixed at 89 kN (20,000 lbs), for both the forming and application of the doublers. A final change to the application procedure was the incorporation of a humidity control system, used to minimise the environmental humidity during the curing of the doublers. Aircraft A8-126 was the first to undergo CPLT, with the doublers incorporating all the above changes, and did so successfully. To date a further nine aircraft have successfully completed CPLT.

These two failure events highlighted the time dependent nature of the failure process for composite bonded repairs and illustrated the need to account for load and strain holds in the design process. Failure was often interlaminar failure in the composite doubler. As such interlaminar failure considerations, rather than the adhesive allowables, drove the final design concept. This is in marked contrast to the PABST design philosophy, see [13], whereby joints and composite repairs are designed on the basis of the maximum load transfer capability of the adhesive. In this case adopting the PABST design approach would have resulted in an unconservative design and catastrophic failure of the repair. Consequently, for composite doublers attention must be paid to assuring that the interlaminar stresses are beneath the (rate dependent) design allowables. To this end an extensive study of the effects of rate dependency, strain holds, creep and time on the mechanical performance of adhesives and composite joints was performed. This study has resulted in a detailed understanding of these processes together with an experimentally validated design philosophy for designing damage tolerant repairs, see [9,18].

CPLT of A8-113

Finally, as part of a wider data gathering program, aircraft A8-113 was strain surveyed during routine CPLT at Sacramento Air Logistics Command, USA, during April 1990. Prior to this survey, a preliminary strain survey was conducted on this aircraft at Amberley, Australia, to certify the strain gauges and acquisition system, and to gather baseline data [19].

This aircraft was extensively strain gauged, including the doublers and the critical stiffener areas. The results of the survey [20] indicated that no permanent set was evident in the stiffener runouts, as predicted in [3], indicating that the boron/epoxy doublers were effective in preventing further yielding at these locations. All data gathered from gauges on the doublers were linear, indicating no deterioration of the adhesive bond.

CONCLUSIONS

From the results of this analysis and the testing program summarised in this work, an all boron/epoxy doubler reinforcement should achieve the initial design requirements, that is WPF survival in the CPLT, when applied to a fleet aircraft. This was confirmed during routine CPLT testing of an F-111 aircraft. From the results of numerous strain surveys of a boron/epoxy reinforced WPF, it was been shown that substantial strain reductions were achieved. In the critical region, the relative strain reductions were approximately 40% and 20% for the positive and negative load cases respectively.

The reinforcement was loaded to ± 100% proof load under ambient and -40^{o}C conditions with no visible signs of deterioration. Further yielding of the WPF had not occurred at these load levels. The measured strain reductions were in good agreement with those predicted by finite element analysis. Good agreement was also shown between finite element analysis and test specimens designed to reproduce the adhesive stresses in the doublers.

The parametric study on the F111C Stiffener Run Out Number 2 has shown that the local geometry, i.e plate thickness, kink angle, etc, has a significant effect on the stress and the residual stress distribution through the stiffener. The most significant parameter affecting the stress and residual stress is the radius of the grind, with the plate thickness also having a lesser effect. The analysis performed on the doublered aircraft show that the local angle of the plate to the skin has a significant effect on the bending of the local section. This was confirmed in the calibration of aircraft A8-113.

The results of the elastic/plastic analysis, and the residual stresses, obtained in the parametric study on the F111C Stiffener Run Out Number 2 have subsequently been used as the stress input to the RAAF DADTA analysis. Here one of the main objectives of the DADTA analysis was to determine the inspection intervals for RAAF F111C aircraft fitted with the boron doubler applied. Using this stressing information for a aircraft with no doubler, and worst case scenario of geometry, the inspection interval predicted was 87 hours. However, for the same configuration, but with a doubler, the inspection interval was calculated as 2,000 hours. At the other extreme, the inspection interval has been predicted to be 446 hours with no doubler and 6,000 hours with a doubler. This result now enables the RAAF to reconsider the current maintenance schedules with a potential dramatic cost saving and increased aircraft availability.

From the technical point of view two major points were highlighted; viz:

(i) Interlaminar failure considerations, rather than the adhesive allowables, drove the final design concept. This is in marked contrast to the PABST design philosophy, see [13], whereby joints and composite repairs are designed on the basis of the maximum load transfer capability of the adhesive. In this case adopting the PABST design approach would have resulted in an unconservative design and catastrophic failure of the repair. Consequently, for composite doublers attention must be paid to assuring that the interlaminar stresses are beneath the (rate dependent) design allowables.

(ii) Classical techniques for modelling the cyclic behaviour had inherent difficulties in representing the response to large cyclic inelastic strain excursions. Indeed, the use of classical analysis techniques resulted in an inspection interval, for the modified structure, of under 500 hours. To overcome this shortcoming the use of a unified constitutive model was necessary.

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The analysis was performed using the PAFEC finite element program in conjunction with the PUPPIES (PAFEC), alpha release, suite of interactive programs, and the non-linear adhesive joint analysis was performed using the BJ4C (Bonded Joints For Composites) program marketed by MONSAFE at Monash University.

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