DESIGN ANALYSIS AND VALIDATION FOR A BONDED COMPOSITE REPAIR TO PRIMARY AIRCRAFT STRUCTURE

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ABSTRACT

This paper presents an overview of the design procedure and of the repair-substantiation program undertaken for a safety-critical repair to an F-111 lower wing skin. The design methodology relies on identifying an appropriate bonded joint, representative of the load transfer across the crack. This can be used both to simplify the stress analysis required for designing the repair, and, more importantly, to generate the relevant design allowables. A 3D finite-element model, validated against a strain survey on a full-scale test wing, is used to quantify the stress concentration factor and secondary-bending effects due to geometrical details around the site of cracking.

The repair substantiation involved both detailed finite element (FE) stress analyses and structural testing at three levels, ranging from coupon-size specimens (representative bonded joints) to quasi- full-scale specimens representing a spar-stiffened wing-box structure. The close coupling of analysis and testing is shown to lead to a time- and cost-efficient certification package, with a confidence level comparable with full-scale structural testing. Some important similarities and differences between repaired and part-through cracks are briefly discussed.

KEYWORDS

Bonded repair, Fatigue cracking, Secondary bending, Stress analysis

1. INTRODUCTION

Bonded composite repairs have been successfully applied to both military and civilian aircraft over the past twenty years (Baker and Jones, 1988; Baker, 1994). Although this repair technology is widely acknowledged to have several advantages relative to traditional, mechanically fastened, repairs, its practical implementation in critical applications is hindered by certification requirements. This arises because, from a certification viewpoint, the use of adhesive bonding and of fibre-composite patches represents a relatively new technology, so that every prospective application needs to be rigorously scrutinised to ensure compliance with airworthiness requirements. Consequently, aircraft operators may face unacceptably long delays in obtaining approval for implementing bonded-repair options, whereas traditional repairs can be implemented immediately, in accordance with the Structural Repair Manual approved as part of the original certification.

This paper outlines the design methodology, and summarises the results of a comprehensive repair-substantiation program undertaken for a safety-critical bonded repair to an F-111C aircraft (A8-145) in RAAF (Royal Australian Air Force) service. This repair followed the discovery of a crack on the lower wing skin of aircraft A8-145 during a routine visual inspection. The visual indication was associated with fuel seepage around the crack site. A more detailed inspection revealed a through-thickness crack, 48 mm long tip-to-tip. Fracture mechanics calculations (using a handbook value of 46 MPa $\sqrt{m}$ (42 ksi $\sqrt{in}$) as the effective fracture toughness) indicated that the residual strength had been degraded to 168 MPa (24.4 ksi), which is considerably less than the Design Ultimate Stress of 358 MPa (51.9 ksi) specified for this region of the wing. A mechanically fastened metallic doubler was initially considered, but this repair option was discarded because of undesirable aerodynamic implications and because the underlying structure would be un-inspectable; further limitations of traditional repairs are discussed in Baker and Jones (1988). Thus, the only viable alternative to a bonded repair would have been scrapping the wing.

2. CRACK LOCATION AND RESIDUAL STRENGTH

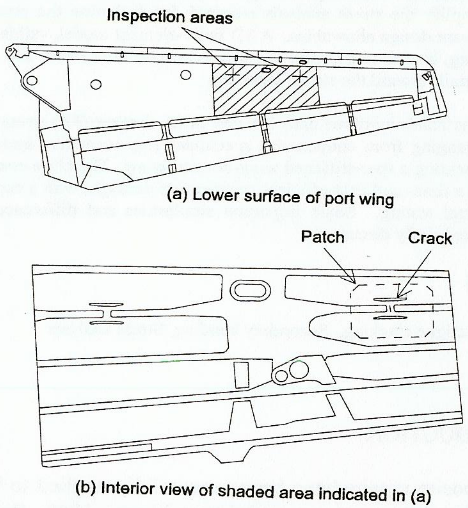

Figure 1 Location of cracking on the F-111 lower wing-skin, at FASS 281, showing also the outline of the repair patch.

Figure 1 indicates the location of cracking on the lower wing-skin. The shaded areas indicate raised portions which serve as integral stiffeners and as attachment lands for spars. The cracking occurred in an area lying below the forward auxiliary spar, at the

spar station FASS 281.28, where the thickness of the integral stiffener is reduced from approximately 8 mm to the nominal skin thickness (at this location) of approximately 4 mm (with the minimum acceptable thickness being 3.6 mm).

The purpose of this depression in the stiffener is to allow full fuel flow and drainage between adjacent bays of the wing-box fuel tank. Two side-stiffeners are used to compensate for the loss of spanwise stiffness due to this fuel-flow passage, as indicated in Fig. 1 These geometrical features lead to a significant stress concentration factor and to out-of-plane secondary bending at the location of cracking, as discussed in Rose *et al.* (1995).

3. REPAIR DESIGN PROCEDURE

The detailed design calculations undertaken by the RAAF for the bonded repair to A8-145 have been presented by Davis *et al.* (1995). The present section outlines the underlying strategy of the design procedure (Rose, 1988).

The essential concept is to clearly distinguish two load-transfer zones, and consequently to divide the design analysis into two stages. One load transfer occurs at the outer boundary of the repair patch, and the other occurs in the immediate vicinity of the crack. It is a sound design practice to ensure that these load-transfer zones are well separated from each other, so as to provide an intermediate region of low adhesive stresses which acts as an anchor against creep and also ensures damage tolerance relative to debonding.

The proper implementation of this design strategy in a specific case requires careful consideration of the relevant structural features from the viewpoint of load transfer.

For the *first stage* of the analysis, it is important to determine whether the load flow to the region being repaired is solely through the skin (and if so, what boundary conditions could most appropriately represent the interaction with the surrounding structure) or whether there is an essential interaction with other structural features, such as the forward auxiliary spar in the present case. This possible interaction with the sub-structure could be particularly significant when assessing the thermal residual stress induced by differential thermal expansion due to curing of the bonded patch at a higher temperature than ambient e.g. curing at 80°C for the present repair.

The *second stage* of the stress analysis deals with the load transfer to the repair patch across the crack. The key step here is to identify correctly an appropriate "representative bonded joint". For the present repair, the two significant features which should be accounted for are the stress-concentration factor and the secondary bending associated with the stiffener run-out. The latter will lead to positive peel stresses, which are known to be highly detrimental for strength and durability of adhesively bonded joints. However, there are no adequately validated failure criteria which can be used with confidence to quantify the effects of peel stress on failure. It is therefore essential to obtain the design strength-allowables directly from experimental tests on a *representative bonded joint*. This joint should simulate the actual repair to the extent of exhibiting a similar ratio of peel stress to shear stress, at least within the elastic range.

An attractive feature of the design procedure outlined above is that a stress analysis of the representative joint leads to an estimate for the maximum value of the stress intensity factor following a repair, and provides estimates for the maximum strain in the adhesive and the maximum fibre stress in the patch. This permits a comprehensive assessment of the feasibility and effectiveness of a repair, which makes this approach eminently suitable as a basis for a design manual such as the RAAF Engineering Standard C5033 (Davis, 1994).

4. REPAIR SUBSTANTIATION REQUIREMENTS

The repair substantiation program undertaken by AMRL for the repair to A8-145 followed closely the strategy and requirements of a certification plan formulated by the RAAF for bonded composite repairs. The principal requirements were:

(i) to validate the analytical formulae used for the repair design by an *independent method*;

(ii) to verify, through *representative article testing*, that the structural integrity of the repaired structure has been restored to the level specified (or implicit) for the original certification of the aircraft, with respect to (a) static residual strength, (b) durability and damage tolerance, with proper accounting for environmental effects and for possible detrimental effects of the repair on the structural integrity of the surrounding structure.

4.1 *Design Load Cases*

Three load cases were identified for demonstration of adequate static strength. The *principal load case* corresponds to a balanced symmetric manoeuvre at high speed (Mach 1.4, wing sweep angle 50°), for which the design ultimate stress is 358 MPa (51.9 ksi) at a skin temperature of 75°C (167°F).

Because the mechanical properties of adhesives and of polymer-matrix composites vary significantly with temperature, and because of the significant mismatch in the thermal coefficients of expansion between the composite patch and the underlying metallic skin, two further load cases were specified:

(i) A *high temperature* of 110°C (230°F), derived from computational fluid dynamics modelling, for which the associated design limit stress was specified to be 234 MPa (34 ksi) using the manufacturer's stress information. A deficiency in the original stress equation was subsequently identified which, when corrected, resulted in a more modest value of 143 MPa (20.8 ksi) for the limit stress at this high temperature.

(ii) A *low temperature* of -40°C (-40°F) corresponding to conditions for the Cold Proof-Load Test used for structural integrity management of the F-111.

4.2 *Fatigue Loading*

The specified fatigue loading was based on a detailed cycle-by-cycle load spectrum available from multi-channel recorders fitted to RAAF F-111 aircraft. This spectrum is representative of RAAF usage and is currently used for structural management of the F-111 fleet. The loads expressed in terms of the wing-pivot bending moment were converted to nominal stresses at FASS 281 using a stress equation which was verified as part of the development of a detailed 3D FE stress analysis (Callinan et al. 1996).

The full load-spectrum contains over 1.7 x 10^6 cycles per 500 flight-hour block (200 flights), so that spectrum truncation was essential to reduce the testing time to an acceptable level. Two truncated spectra were used to generate the results reported below:

(i) a *blocked spectrum* in the form of an exceedance diagram, blocked into discrete load levels, and

(ii) a *cycle-by-cycle spectrum*, which is more realistic in that the highest peak loads are not necessarily matched to the lowest valleys, as is the case with the blocked spectrum.

Both of these spectra were truncated to approximately 40 x 10^3 cycles per 500 flight-hour block. The truncation procedure is based on that described by Broek (1989), and has been verified as part of the present testing.

4.3 *Validation Strategy*

The strategy employed to address the repair-substantiation requirements involved a closely integrated program of testing and of detailed stress analysis. The stress analysis was used not only to verify the analytical design formulae, as required above, but also to guide specimen design, and to provide a basis for correlating crack (or damage) growth rates and failure modes between the various levels of testing, thereby providing a high level of confidence in the interpretation of test results and their applicability to the actual in-service repair.

The RAAF requirement with regard to environmental effects and damage tolerance is that the testing conditions should be severe relative to anticipated service conditions, but realistic. The present substantiation package involves three levels of testing, as described below in Section 6, and it will also seek to derive the maximum benefit from the extensive previous service and laboratory experience with bonded repairs, as summarised by Baker (1994), particularly in assessing the durability at high temperature and the susceptibility to impact damage . It is also noted that the testing reported below uses axial (spanwise) loading only, but the effect of torsional loads representative of those expected for the present repair has been investigated experimentally by Belason *et al.* (1994).

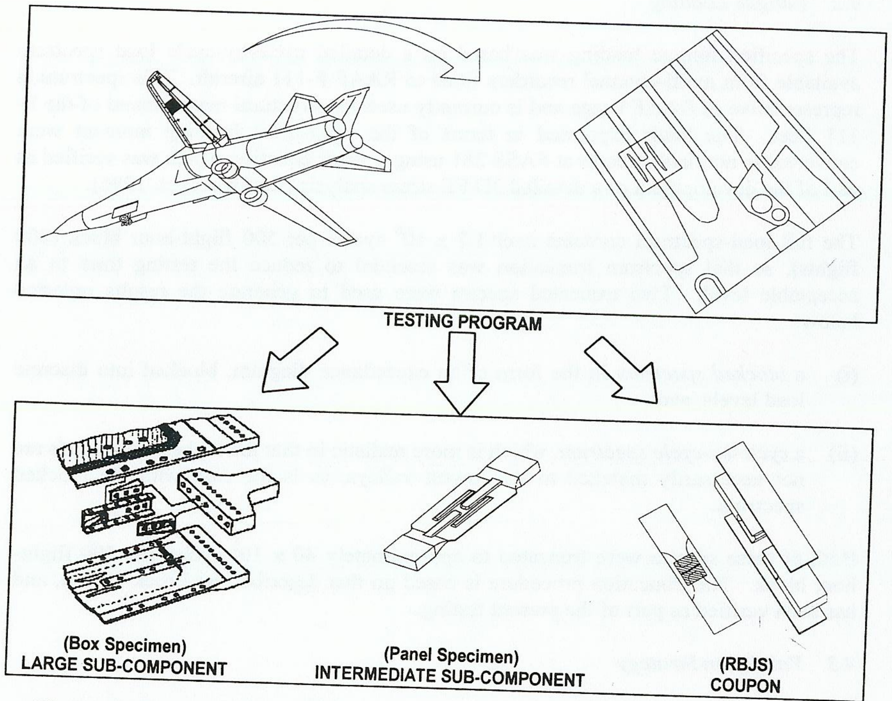

Figure 2 Three levels of structural testing undertaken for repair substantiation of a bonded composite repair to primary aircraft structure

5. STRESS ANALYSIS

The formulation and successive refinement of a three-dimensional finite-element (FE) model of the cracked wing skin and of the supporting spar (Fig. 2) are reported in detail by Callinan *et al.* (1996). The main results and conclusions can be summarised as follows.

(i) The stress distribution prior to cracking depends sensitively on the precise thickness variation around the fuel-flow passage. Considerable mesh refinement and attention to details of the local geometry was required to obtain a good

correlation between FE results and the results of a strain survey undertaken on a full-scale (uncracked) wing available at AMRL.

(ii) Calibration of the FE results against the full-scale strain survey provided a refined stress equation for that location (*i.e.* a more accurate relation between the nominal local stress and the wing-pivot bending moment than was available from the original, approximate, aircraft design calculations). This refined stress equation was used in setting the stress levels for the structural testing.

(iii) The FE results indicated the presence of significant stress concentration and secondary bending, both prior to cracking as well as in the cracked and repaired structure. These effects were not accounted for in the original repair-design analysis (Davis 1994, Davis *et al.* 1995). There are consequently significant differences between the adhesive shear stress distribution obtained from this design analysis and the FE results. Nevertheless, it will be seen that the final repair designed on the basis of the RAAF Engineering Standard C5033 (Davis 1994) proves to have adequate strength and durability.

6. VALIDATION TESTING: SPECIMENS

As mentioned above, three levels of structural testing were undertaken (in addition to the strain survey on a full-scale test wing which was used to calibrate the FE analysis). Ranked in order of increasing complexity (and cost), these are as follows.

6.1 Representative Bonded Joints

These are *coupon-sized specimens* designed primarily to characterise the strength and durability of the adhesive and the composite patch under conditions of high peel stress, simulating those in the actual repair. The intention with these specimens is to explore more fully than would be cost effective for the more complex specimens, the effects of (i) temperature and moisture conditioning, (ii) simulated damage and (iii) various approaches to load-spectrum truncation. The results provided guidance in formulating the detailed specifications for representative article testing.

6.2 Panel Specimens

These specimens (working area approx. 300 mm x 190 mm) are intended to simulate the wing skin including the geometrical features shown earlier in Fig. 1, *i.e.* an integral stiffener, gouged out to provide a fuel-flow passage, with side stiffeners to restore the overall spanwise stiffness. The panels are wide enough that crack growth beyond the side stiffeners would not be significantly affected by edge effects, should crack growth occur after the application of patches to the (pre-cracked) specimens. Consistent with this width requirement, the panel specimens are also intended to be small enough to fit within a purpose-built environmental chamber providing a controlled temperature and moisture level. Thus, these specimens represent the *structural-detail level* of testing.

6.3 Box Specimens

These specimens consist of two flat panels attached by three spars to form a box-like structure (approx. 900 mm x 430 mm x 65 mm) simulating the actual wing. This structure allows the full compressive loads of the load spectrum to be applied, whereas these loads are truncated for the panel specimens. In addition, the spacing between the spars, the fastener spacing, the location of the patch edge relative to the closest fastener, and the distribution of thermal residual stress due to curing are all representative of the full-scale structure. Thus, these specimens provide a *quasi- full-scale level* of testing. Only one box specimen has been tested so far and it is envisaged that only one more box specimen may need to be tested, to provide confidence in transferring the results derived from panel specimens to the actual full-scale structure.

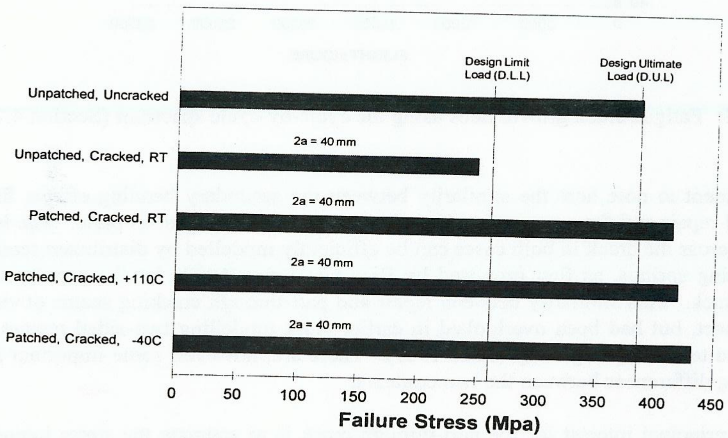

Figure 3: Results of static strength tests on panel specimens.

7. VALIDATION TESTING: RESULTS

7.1 Static Strength

Figure 3 summarizes the static results on panel specimens. The notable features are the following:

(i) The residual strength in the presence of a 40 mm crack is degraded to below the Design Limit Load, confirming the fracture mechanics prediction mentioned earlier (Section 1).

(ii) Application of a bonded patch restores the residual strength to above the Design Ultimate Load capability. Indeed, it can be seen that the repaired strength exceeds the strength of an uncracked panel, even at the extremes of test temperature.

(iii) The failure mode at room temperature (RT) and at low temperature (-40°C) involved simultaneous crack growth and fibre failure across the crack. This is consistent with the stress analysis results indicating a through-thickness stress concentration in the patch immediately over the crack.

It is recalled from Section 6.3 that the construction of the box specimen closely simulates that of the actual wing box. Thus, the consistency between the crack growth rates, confirmed by FE stress analysis showing similar stress distributions in the critical area, establishes the panel specimens as *representative test articles*.

7.2 Fatigue Durability

Room-temperature fatigue tests conducted so far, on both panel specimens and on one box specimen, have indicated continued crack growth after repair, but at a considerably slower growth rate.

Figures 4 and 5 show the results obtained using the two load spectra described in Section 4.2. The important features are:

(i) the residual life without a repair is very short;

(ii) the crack growth rates after repair are very consistent, both between specimens of the same type as well as between the two types of specimen, taking account of the different initial crack lengths.

Static strength tests undertaken after varying amounts of fatigue crack growth always indicated an adequate level of residual strength. In the worst case, after the crack had grown to a length of 134 mm, the residual strength was measured as 374 MPa (54.3 ksi), *i.e.* 97% of Design Ultimate Stress (for the specimen thickness used).

Given that boron/epoxy composite patches allow easy detection and measurement of crack length by non-destructive eddy-current techniques (Baker and Jones 1988), and that the repair location is readily accessible, these results provide a sound basis for a *safety-by-inspection* approach to structural integrity management, as the inspection interval can be chosen to coincide with current maintenance schedules with an ample safety factor.

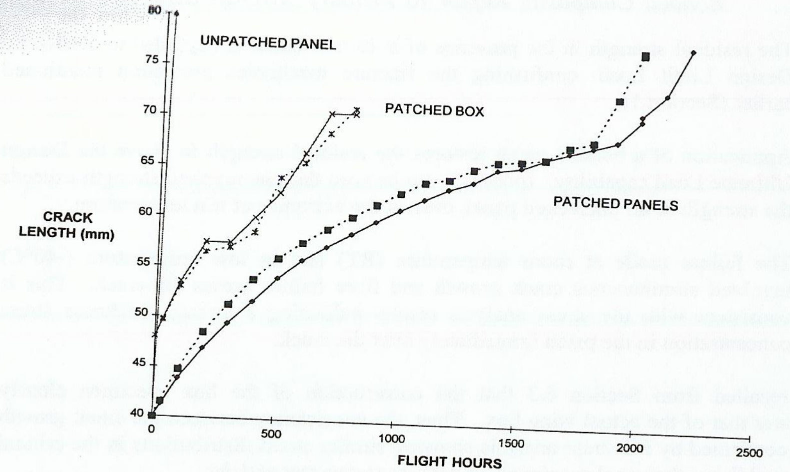

Figure 4: Fatigue crack growth rates using the blocked spectrum (Section 4.2).

8. RECENT DEVELOPMENTS

The results reported above indicate that the design strategy used in the RAAF Engineering Standard is sound, but the key concept of identifying a representative bonded joint requires careful consideration. In the case of the repair to A8-145, the appropriate joint should simulate the significant stress concentration and secondary bending effects at the location of cracking.

Some recent developments regarding the influence of secondary bonding on bonded repairs are worth highlighting here. Arendt and Sun (1994) have reported an analysis of crack-patching efficiency for a one-sided repair to an *un-supported* cracked plate, in which it was found that the energy release rate (and hence the stress intensity factor K) failed to reach a limit, with increasing crack length. This cast doubt on the adequacy of the analytical design procedure developed by Rose (1988). However, a more detailed examination of this repair configuration, using a three-dimensional finite-element analysis (Callinan *et al.* 1997), has confirmed that K does in fact approach a limiting value, which can furthermore be accurately estimated by a one-dimensional analysis of the relevant representative bonded joint (Wang et al. 1996). Thus, analytical formulae, which are particularly convenient for parametric studies of patch design, can also be used for one-sided and un-supported repairs, although the asymptotic value of K is significantly higher than for the corresponding two-sided repair, or for a one-sided repair to a fully supported plate.

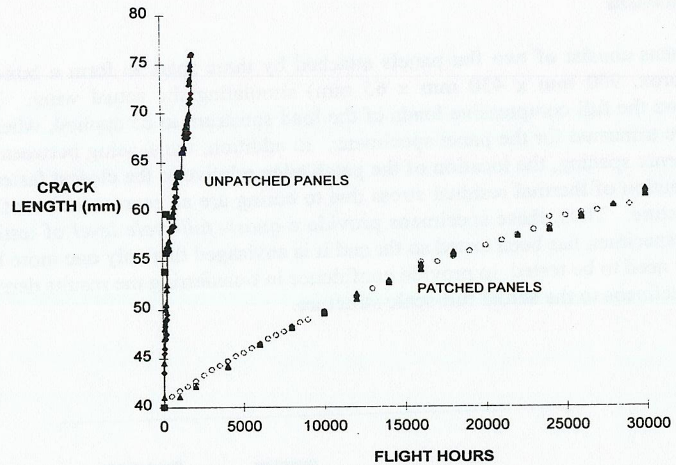

Figure 5: Fatigue crack growth rates using the cycle-by-cycle spectrum (Section 4.2).

It is pertinent to note here the similarity between the secondary bending effects for a one-sided repair and for a part-through surface crack in a homogenous plate. The load transfer across the crack in both cases can be efficiently modelled by distributed tension and bending springs, as first proposed by Rice and Levy (1972) for the part-through surface crack. This similarity between repair and part-through cracking seems obvious in retrospect, but had been overlooked in earlier work modelling two-sided repairs by distributed tension springs only (Rose 1987). There are, however, some important and interesting differences between the two contexts:

(i) the principal interest for the part-through crack is to estimate the stress intensity factor (and hence the crack growth rate) at the deepest point along the crack front, rather than at the ends where the crack front intersects the free surface, whereas the focus of interest is the reverse for a bonded repair, due to the presence of an adhesive layer which inhibits through-thickness crack growth.

(ii) determining the appropriate compliance matrix for the distributed springs requires the solution of a two-dimensional elasticity problem for an edge-cracked strip, for the part-through crack, and the compliances vary with position along the crack, corresponding to the varying crack depth, whereas for the repair case, the compliance matrix can be estimated from a one-dimensional strength-of-materials analysis for the representative bonded joint and these compliances remain constant along the crack. Thus, the resulting coupled integral equations have constant coefficients for the case of bonded repairs, which may provide more scope for analytical techniques, as in the case of tension springs only (Rose, 1987).

(iii) for the part-through crack, the distributed springs model is considered to be least reliable near the ends, whereas that limitation would not seem to apply in the repair case for self-similar crack growth, assuming a nominally straight crack front through the thickness of the repaired plate.

While an appropriate theoretical framework exists for characterising the efficiency of one-sided repairs, it must be emphasized that there is a dearth of reliable material property data, obtained under well controlled and reproducible conditions. It is particularly important to generate experimental data, at a high level of confidence, which shows the effects of peel stress on the strength and durability of the relevant representative joint. It is also important to characterise fatigue crack growth rates in plates under the action of both membrane and bending forces, to simulate the effects of secondary bending in repairs.

9. CONCLUSION

The close coupling of the FE analysis (Section 5) with the three levels of validation testing (Section 6) has provided a cost and time-efficient repair substantiation package. The results provide a high level of confidence that the static residual strength has been fully restored to the original design ultimate stress and a sound basis for structural integrity management of the repaired structure on the basis of safety by inspection. This successful repair-substantiation also confirms the validity of the RAAF Engineering Standard C5033 as a basis for the design and application of bonded composite repairs to thin-skin metallic structure.

9. ACKNOWLEDGMENT

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