# Hybrid Repair Methods for Aircraft Primary Structure Repairs

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## *ABSTRACT*

*Currently bonded repairs can only be permitted on those aircraft primary structures suffering cracks/damages having a residual strength well exceeding the design limit load prior to application of the bonded repairs. This paper focuses on the approaches to meet the certification requirement by combining a bonded patch with other methods that enhance residual strength of the damaged structures. An overview of the recent research in this area conducted by Defence Science and Technology Group and its research partner organisations is presented. The outcomes from six individual research programs indicated the hybrid repair methods are promising for primary structure repair applications. Significant residual strength increases were achieved through optimum damage removal and/or inclusion of alternative load paths. These methods also provide significant additional fatigue life post premature bond failure that would allow any possible bond line defect/damage to be identified by NDI means long before a catastrophic failure. The adhesive bond in the hybrid repairs was proven to provide significant benefit in enhancing the static strength and fatigue resistance. Key issues for the application of these hybrid repair methods and future research directions are also discussed.*

## 1.0 INTRODUCTION

Adhesively bonded patch repairs for aircraft components have been successfully applied in Australia and other countries, saving hundreds of millions of dollars and significantly enhancing aircraft availability [1-2]. Although it is generally appreciated that these repairs have many advantages over repair procedures based on mechanical fastening, and have in the past been applied to several metallic primary aircraft components, difficulties in meeting some of the current certification requirements have limited applications of bonded repairs on aircraft primary structures [2-3]. The main obstacles are: i) quality control procedures that can feasibly be applied to bonded repairs undertaken in non-ideal situations cannot demonstrate initial and ongoing bond integrity with sufficiently high probability of reliability; and ii) current NDI techniques are not yet available, or sufficiently reliable, to detect weak bonds in a practical situation. Thus, currently bonded repairs can only be permitted to be applied to those primary structures suffering cracks / damage having a residual strength well exceeding design limit load prior to application of the bonded repair.

Thus further focused research on bonded repairs aimed at overcoming these certification difficulties should have a huge potential benefit for application of bonded repairs to both military and civil aircraft.

Defence Science and Technology Group (DST) has developed a research roadmap for bonded repairs on primary structures. Key issues contained in this systematic approach to meet the certification requirements include: development of reliable procedures to detect weak bonds through advances in NDI (including a novel adhesive bond proof tester) and bond-line structural health monitoring methods [4-9], enhancement of residual strength through optimum damage removal [10-13] and adding alternative load paths [14-19], and improvement of design methodology and application of quality assurance to bonded repairs. The repair design must properly consider all the factors required by certification of a bonded repair including defect and damage tolerance [12, 20], fatigue life assessment, service environment factors, etc.

This paper focuses on a subset of the above research strategy, that is, to meet the certification requirement for applications of bonded repairs on primary structures by combining a bonded repair with other methods that enhance residual strength of the damaged structures, such as optimum damage removal and alternative load path addition. In the following sections, an overview of the recent research in this area conducted by DST and its research partner organisations will be presented. The outcomes from these research programs indicated that hybrid repair methods are promising for primary structure repair applications. Key issues for the application of these methods and future research directions will also be discussed.

## 2.0 HYBRID REPAIR – OPTIMUM DAMAGE REMOVAL/BONDED PATCH

The first step in the decision chart for repairs to flight-critical composite structure [2] is to define an optimum damage cut-out. In this section three repair scenarios will be presented, where significant residual strength increases were achieved by applying shape optimization for crack/damage removal for both composite and metallic components, and bonded patch repairs with distinctly different design concepts were applied that provide significant benefit in enhancing the static strength and fatigue resistance.

### 2.1 Patch Covering the Damaged Area

Several aircraft wing spars were observed to experience in-service cracks growing from the electrical grounding hole and adjacent satellite holes.

The spar is made of high strength aluminium 7175-T73652 alloy. It is 51 mm (2 inches) wide and has a thickness of 2.03 mm at the area around the holes. Under load, this area of the spar is in a stress state of near pure-shear.

A finite element (FE) analysis was conducted to compare repair options, including i) standard cut-out, ii) optimum cut-out, and iii) a bonded patch repair combined with the optimum cut-out (Figures 1, 2 and 3).

Boron/epoxy prepreg was chosen as the patch material. A layup of $[45/-45/45_3/-45/45]$ was specified. It matched the stiffness of the spar in the $45^o$ direction, which is perpendicular to the crack direction. EA9394 paste adhesive was chosen for its relatively low temperature cure and high strength.

The analysis of the bonded patch was undertaken for three environmental conditions, namely room temperature/ambient, cold temperature (-54ºC)/dry and elevated temperature (105ºC)/wet conditions to ensure that structural integrity and repair effectiveness were properly assessed.

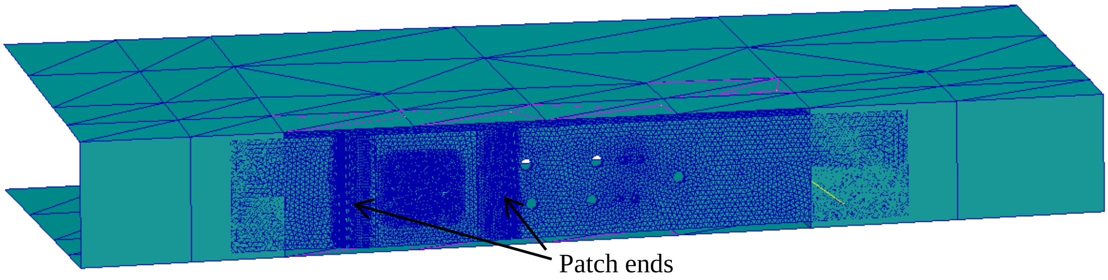



**Figure 1: The FE model of the wing spar (with patch).**

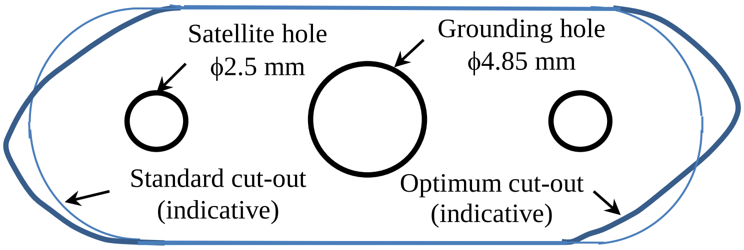



**Figure 2: Comparison of the various configurations of the grounding hole location analysed. The standard cut-out profile and optimum cut-out profile [21] shown are only indicative.**

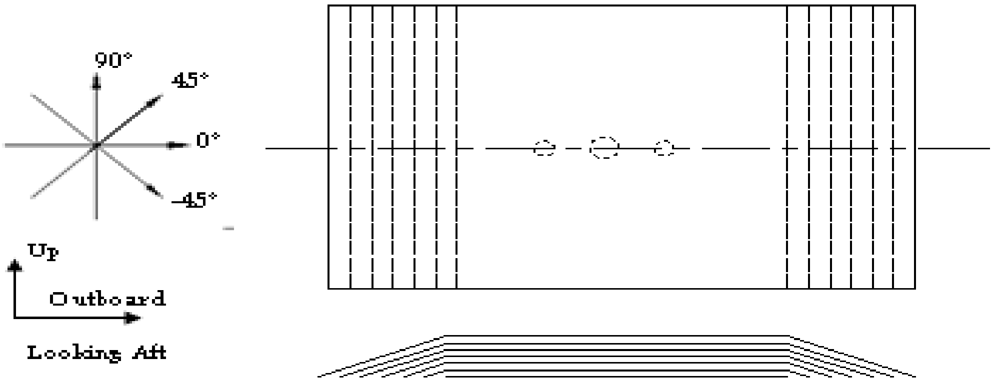



**Figure 3: Boron/epoxy patch (patch size 82.5 mm x 51 mm (3.25 in x 2 in), layup [45/-45/45$_3$/-45/45], 3 mm ply drop at each end, dotted lines indicating the grounding and satellite hole locations)**

The FE analysis results showed that:

- The original configuration (without cracks) exhibited peak stresses at the grounding and satellite holes that were rather high, which facilitated formation of the fatigue cracks.
- The cut-out was effective at removing the high stress concentration at the crack tips. However, compared to the original configuration, the peak stress with the cut-out was higher (by around 10%).
- Compared with the original configuration, the optimum cut-out reduced the peak stress by around 12%.
- Bonded patch repair combined with optimum cut-out reduced the peak stress by around 50% compared with the original configuration.

- For cut-out only options, should the cracks re-occur, the stress intensity factor would exceed the material fracture toughness value.

From the above it is clear that the cut-out may effectively increase the residual strength over the cracked structure and the bonded patch can further incease the strength very significantly. The analysis was validated by a full scale static test at room temperature/ambient environment.

Estimated using the simple cubic rule [58], the optimum cut-out and the hybrid repairs would result in a fatigue life of 1.5 and 8 times, respectively, for the spar compared with the original configuration. For relatively short fatigue life requirement, the optimum cut-out alone would be an acceptable repair option. For relatively long fatigue life requirement, and also in a longer crack case, addition of a bonded patch would become necessary to provide sufficient fatigue life and eliminate the possibility of crack reoccurence[1].

The method described represents a typical hybrid approach for application of repair to thin metallic structure, combining optimum damage removal with conventional bonding of a thin overlapping boron patch, which allows non-destructive eddy current inspection of the parent structure performed through the patch.

## 2.2 Patch around the Damaged Area

As part of the aircraft battle damage repair (BDR) research program, DST was tasked by the Australian Army to develop a repair solution for a helicopter main rotor blade subject to ballistic damage at the main spar section.

The repair requirements were to:

- restore stiffness / strength
- maintain aerodynamics performance
- maintain dynamic balance / natural frequency (not discussed in this paper)
- be applicable within a short timeframe, with limited requirements for tools, materials and technician skills [22]

BDR applications generally only require restoring a short fatigue life capability (50 to 100 flight hours). However, for this dynamic component (due to the high frequency of loading) restoration of high fatigue resistance is required from the repair.

The dimensions of a 3D specimen are shown in Figure 4 and Figure 11. The tube specimen has a nominal thickness of 4.31 mm, made using 25 plies of carbon-epoxy biaxial and uniaxial fabric prepreg materials [23]. The specimen is representative of a typical main spar section (spar cell 2 in Figure 5). A circular hole was cut through the specimen at the centre to represent damage due to ballistic impact and conventional subsequent damage removal.

The main load conditions include a centrifugal force and lift force (a tensile force in x-direction and distributed load in z-direction, respectively, Figure 5). The lift force results in a bending moment (around y-axis, Figure 5), which creates a tensile force and compressive force on the spar lower panel and upper panel, respectively. Thus, due to the combined effect of the centrifugal and lift forces, the lower panel is the critically loaded structure.

A test fixture was designed to apply the loads for the 3D spar specimen (Figure 4). The combination of centrifugal and lift forces was experimentally approximated by applying uneven tension loading along each spar cap. The fixture featured a universal joint mechanism to introduce load path eccentricity and was compatible for use with a standard axial mechanical load frame.

[1] Further discussion will be provided in Section 5 about inspection interval and how the credit of fatigue life enhancement can be given to the hybrid repair

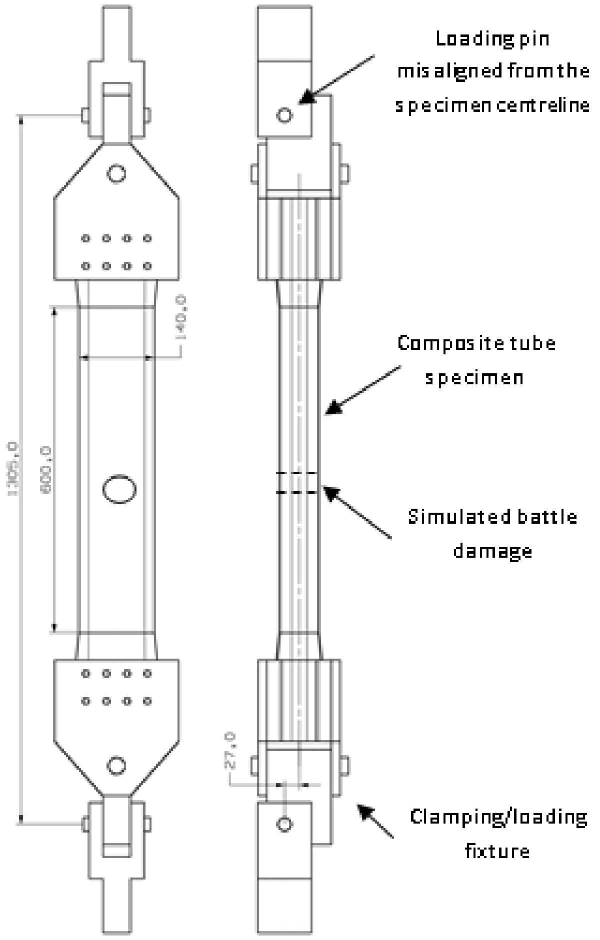



**Figure 4: 3D spar specimen and loading fixture (dimension in mm)**

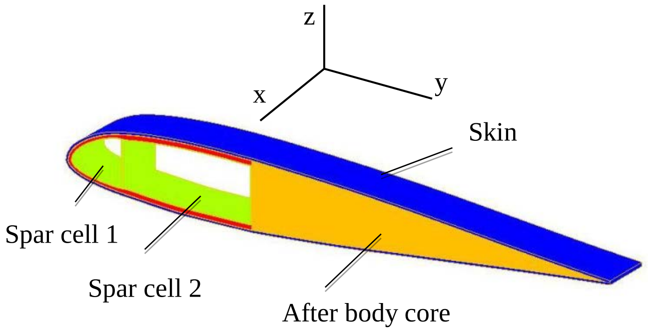



**Figure 5: Section view of a typical helicopter main rotor blade**

In this study, design and testing of both 3D and 2D specimens were conducted interactively to progress the repair design-validation process efficiently and cost-effectively. An initial 3D specimen test was conducted to validate and calibrate a full 3D FE simulation model and also to determine the failure onset load. 2D specimens and an anti-bending constraint mechanism (Figure 6), representative of the critical lower panel location, were then designed based on the FE model to test and compare the proposed repair options. A final 3D test was conducted to validate the selected repair design.

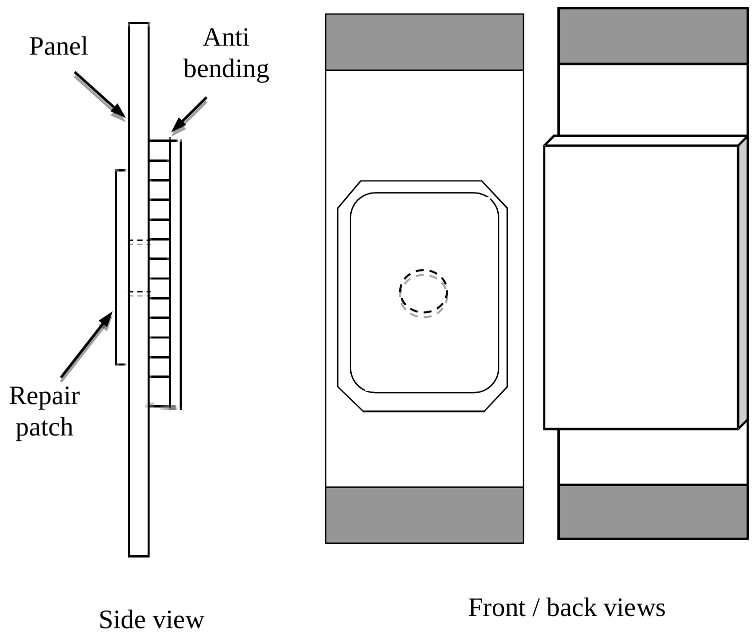



**Figure 6: Illustration of a 2D specimen with a patch and anti-bending mechanism (15 mm thick Nomex honeycomb and a laminate panel bonded to back side of the specimen)**

For a single (external) side repair of a 4.3 mm thick laminate panel, conventional repair options for laminates of this thickness would be scarf or step repairs [24]. These methods require removal of material across large undamaged areas, which significantly reduce the residual strength of the structure prior to application of bonded repairs. As discussed earlier, the lower residual strength in the absence of the repair increases the difficulty to meet the current bonded repair certification requirements for primary structure applications. Scarf and step repairs generally have a high demand on technician skills, repair facilities and repair time, which are also unsuitable for BDR applications.

An alternative approach was considered: to increase the residual strength prior to application of bond repairs by optimum damage cut-out. A FEM simulation indicated that the optimum hole shape (Figure 7) designed using the shape optimisation approach described in Reference [25] resulted in over 30% increase of the residual strength compared with the circular hole damage removal, which in turn has higher residual strength than an untrimmed damage case with sharp cracks present.

A thin repair patch (9 plies) was applied to maintain aerodynamic profile for a rotor blade. To minimize shear and peel stresses the patch was tapered at its loaded edges by stepping plies from a single ply up to the full thickness at an optimum transfer length (Figure 8) determined from a detailed FEM analysis. To limit adhesive stresses above the hole edge (another critical location), a novel design was adopted by cutting though the majority of plies in the patch with a same shaped hole as that in the parent structure (Figure 8b, right). A thin, softer cover skin was used to provide a seal and limited load transfer over the hole (Figure 8b, left). With this patch design concept, the strength increase from the patch was achieved through increasing the load and loading capacity in the load bypass region of the specimen, plus an adequate, additional load through the softer skin cover over the hole. When properly designed, this repair configuration can effectively eliminate failure initiation in the adhesive layer (and the resultant first composite ply failure due to high adhesive stress). A FE analysis indicated that the patch repair can further increase the loading capacity significantly compared with the specimen with the optimum damage removal only. A conventional patch design without a cut-out in the middle [26] was also used for comparison (Figure 8a). Due to the high stress in adhesive layer above the hole edge, initial failure in the adhesive is expected with this patch design. As indicated in Reference [27], this initial failure would occur in the “damage tolerant” region. For a practical application of this design, the disbond growth rate as a function of fatigue loading needs to be known. Since for a parent/patch made of composite materials (in contrast to metallic materials) failure may occur in the first ply of the parent/patch materials rather than in the adhesive layer in this region, this may make the damage tolerance assessment more complicated.

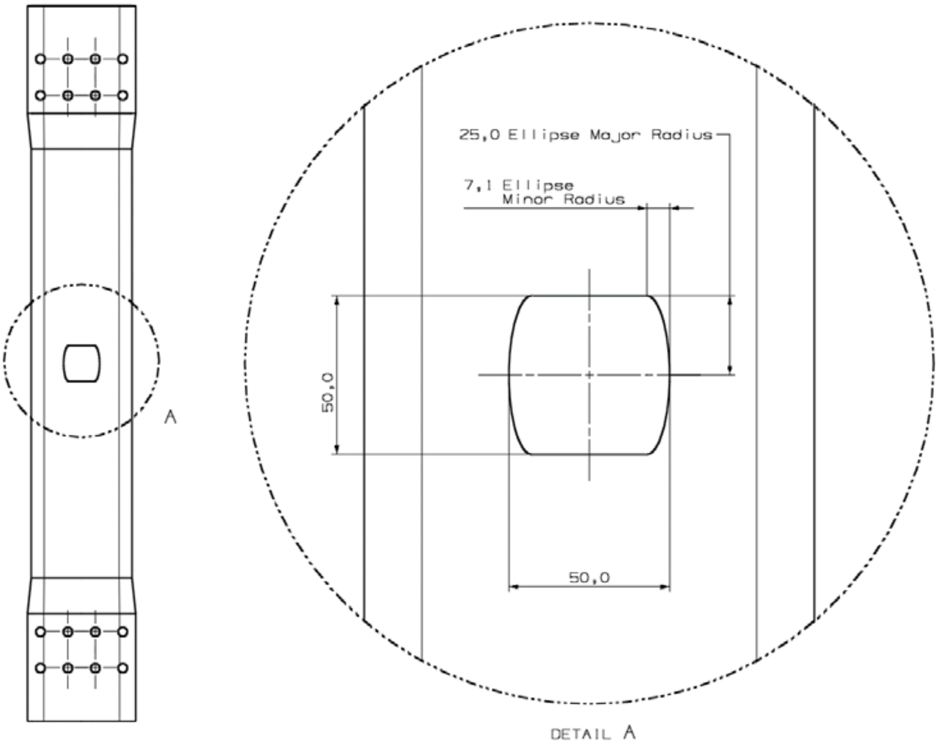
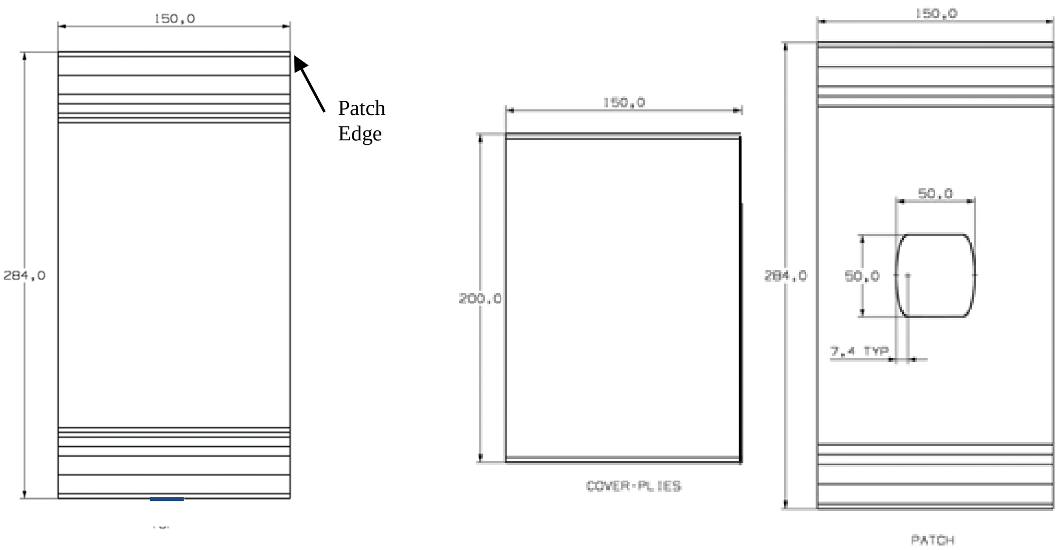



**Figure 7: 3D spar specimen with optimum shaped damage removal**




**(a) "Conventional" patch**

**(b) New concept patch**

**Figure 8: Repair patch configurations**

The 2D test results are summarised in Table 1. Clearly the damage reduced the strength of the pristine specimen very significantly (by 68% and 52% for damage onset load and ultimate load, respectively). Optimum damage removal alone increased the loading capacity by 44% and 21% respectively in terms of damage onset and ultimate loads, respectively. In terms of damage onset load, the conventional repair patch design (Configuration 1) and the novel repair design (Configuration 2) achieved 28% and 88% increase, respectively. Note that, as indicated from strain gauge readings, the onset of failure with the conventional repair patch design (Configuration 1) was the adhesive failure at the upper and lower edges of the hole, whilst the onset of failure with the novel repair patch design (Configuration 2) was the laminate failure at the hole side edges (Figure 8), which was expected (refer to the earlier discussion in this section). Similar improvement in terms of the ultimate load was achieved from the two patch repair configurations. It was over 60% increase compared with that of the damaged specimen without repair, and was 78% of the pristine specimen's ultimate load.

**Table 1: Summary of 2D specimen test results**

| Specimen type | Damage onset load (kN) | Ultimate load (kN) |
|---|---|---|
| Pristine | - | 560 |
| Damage with 50mm hole | 180 | 270 |
| Damage with optimum cut-out | 260 | 325 |
| Repair configuration 1 | 230 | 440 |
| Repair configuration 2 | 340 | 435 |

In the further 3D test, the initial damage on the hole edge of the 3D specimen was trimmed off (and the hole diameter was slightly enlarged by 5 mm). Repair Configuration 2 was implemented. The specimen was tested on the 2 MN Instron tensile test machine (Figure 9) with a loading rate of 1 mm/min on displacement control. The results between a 3D FEM simulation and strain survey again agreed well (Figures 10 and 11). The failure onset load was around 410 KN, which indicated a strength increase of around 80% due to the repair. The failure was initiated in the laminate as anticipated, rather than in the adhesive layer

For a battle damage scenario this strength increase is highly significant and would probably allow a capability for a continuing mission or at least for a ferry flight to a more capable repair depot for component replacement, which would not be possible without the repair.

**Figure 9: Testing of a repaired 3D specimen**

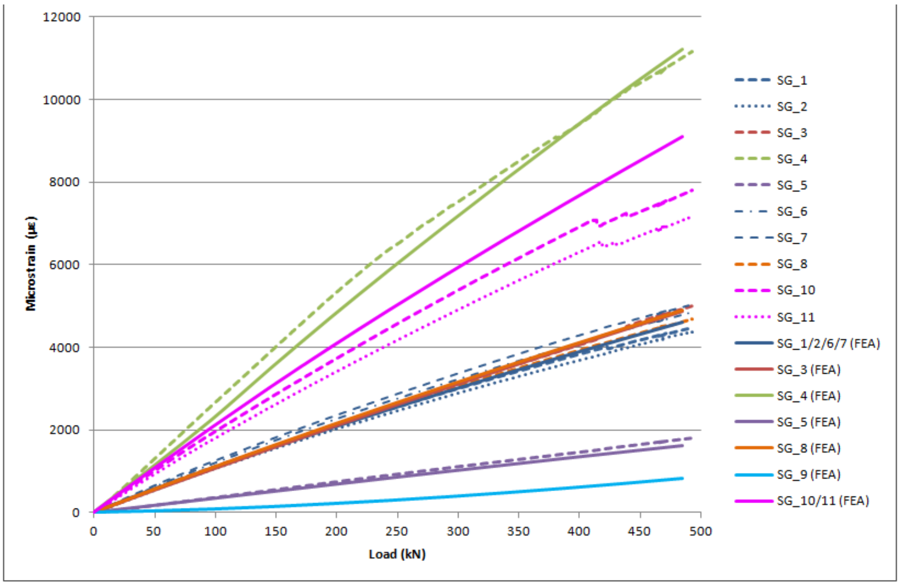



**Figure 10: Figure 10: Strain-load results, repair Configuration 2**

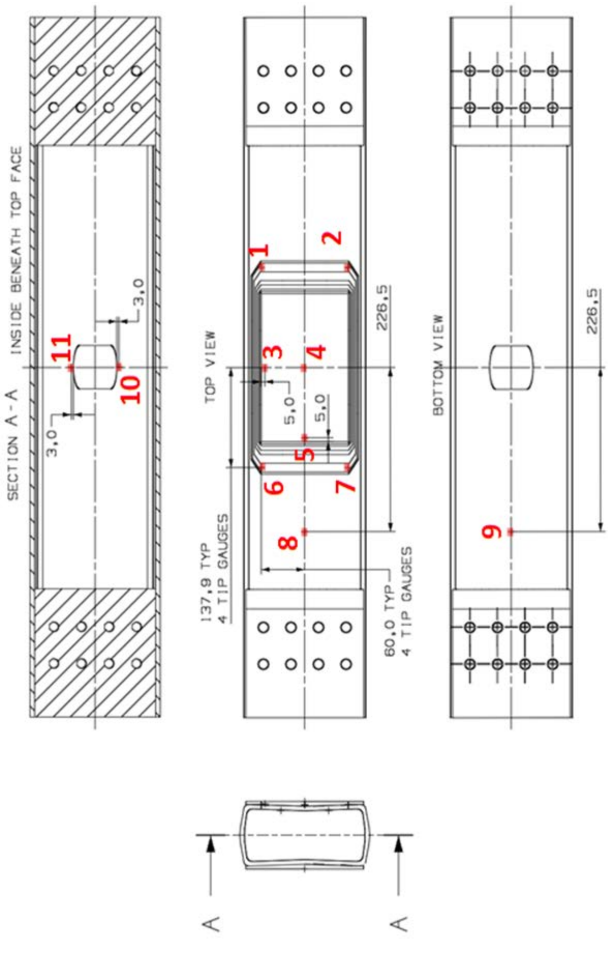



**Figure 11: Strain gauge positions for repair Configuration 2 (dimension in mm)**

## 2.3 Distance Patch Repair

A stop hole is often used as a temporary measure to resist fatigue crack propagation in metallic structures. The effect of the stop hole is generally rather limited. A crack often soon re-occurs due to the relatively high stress at the stop hole. However, a much more effective option is to cold expand the hole (for example with an oversized fastener) to develop compressive stresses around the hole and at the tip of any residual crack. Again however, crack growth will be very rapid if re-initiation occurs.

A stress optimised stop hole achieved through hole shape optimisation may also significantly further reduce the stress [28] especially if cold working of the cut-out can also be achieved and thus extend the fatigue life of cracked structures. However, the fatigue life enhancement from a stress optimised stop hole alone even with cold working could still be limited. A hybrid repair solution that combines a stress optimised stop hole and a bonded patch would provide significant further fatigue life enhancement.

Unlike the patches described above in Section 2.1 (over the damage cut-out) and Section 2.2 (around the damage cut-out), the patch in this hybrid repair is placed at a distance away from the optimised stop hole.

Similar to the approach used in Section 2.2, this reinforcement has several features: i) the bonded patch helps reduce the stress at the damage/high stress region through load bypass; ii) the patch rigidity does not have to match the parent structure and the level of stress reduction can be controlled by varying the patch thickness, and iii) adhesive failure as a critical failure mode can be eliminated. In addition, potential crack occurrence/growth in the parent structure can be monitored through visual inspection.

The idea to attach the bonded patches away from the damage cut-out or potential crack development area has been adopted in many applications [29-32]. Figure 12 shows a bonded patch design described in Reference [30], where two patches with shapes generated from a shape optimisation algorithm redirect the load away from the high stressed hole edge. DST and its research partners adopted a similar but significantly simplified design for repair of a composite helicopter spar by bonding two rectangular patches at each side of the hole in a repair design [32].

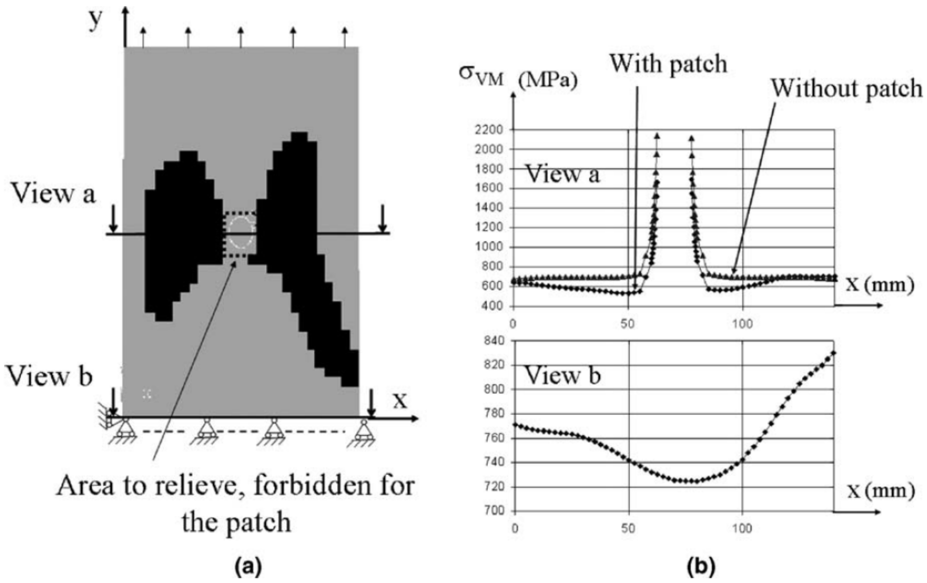



**Figure 12: Optimal patch geometry for the tensile test specimen reported in Reference [30]. (a) Optimized shape and (b) Von Mises stress profile.**

Four different typed specimens were considered. As showed in Figure 13, these included a specimen with a crack, a specimen with a crack and a standard (round) stop hole at the crack tip, a specimen with a crack and an optimum stop hole, and a specimen with a crack and an optimum stop hole plus two patches bonded on both sides. The specimens are made of 7075 aluminium alloy, with a dimension of 380 mm x 100 mm x 6.35

mm. A uniaxial spectrum tensile fatigue load was applied [33].

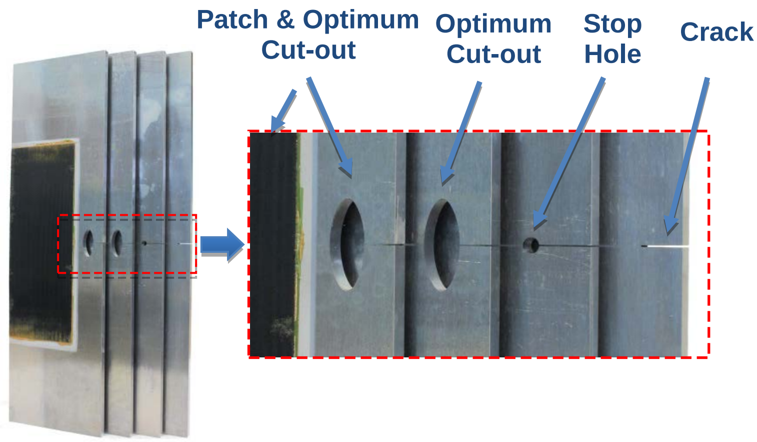



**Figure 13: Four specimen configurations- cracked and different repair configurations.**

Compared with the specimen with the standard stop hole, the stress intensity factor of the specimens with optimum stop hole and hybrid repair were reduced by 63% and 73%, respectively. By applying a simple cubic law, the fatigue life for crack re-initiation can be estimated to be 20 and 52 times for the specimens with optimum stop hole and hybrid repair, respectively, compared to the specimen with the standard stop hole. As shown in Table 2, the measured results indicated more significant fatigue life enhancement from the optimum stop hole and hybrid repair than estimated from the simple cubic rule. Full details of this research work will be reported in a journal publication currently being prepared [13].

**Table 2: Measured fatigue life of four typed specimens**

| Specimen type | Normalised fatigue life* | |
|---|---|---|
| | Crack re-initiation | Ultimate failure |
| Cracked | N/A | 1.03 |
| Standard stop hole | 1.00 | 2.19 |
| Optimum stop hole | 28.1 | 29.1 |
| Hybrid repair | >115** | >115** |

* Normalised against crack re-initiation fatigue life of the specimen with standard stop hole
** No crack re-initiation was observed before fatigue test stopped for most specimens tested

## 3.0 HYBRID REPAIR – FASTENED/BONDED PATCH REPAIR

Several papers [34-43] have investigated the behaviour of fastened/bonded hybrid joints. Hart-Smith [38] provided a non-linear analysis of bonded and bolted joints and concluded that the hybrid joint configurations cannot achieve any significant advantage over adhesive bonding in well-designed intact structures, however Hart-Smith suggested these may prevent bond defect/damage propagation. The combination of mechanical fastening and bonding has been employed to safeguard against defects within the adhesive layer which may cause premature or catastrophic failure [35]. Due to the significant stiffness difference between the adhesive bonding and mechanical fastening, usually only after the bond has failed would the fasteners begin to carry the major load in the hybrid joint. It is the safety factor from the fasteners that has allowed the certification of these joints in some aircraft structures.

Significant experimental and computational work has recently been jointly conducted at DST and Monash University to examine the feasibility to utilise this hybrid repair for primary structure repair applications. The research has so far been focused on testing of composite structure repairs at the coupon specimen level.

## 3.1 Experiment

Fastened, bonded and fastened-bonded hybrid joint specimens were manufactured and tested under static and fatigue loadings. The composite adherends were manufactured using HexPly M18/1/G939 carbon fabric/epoxy prepreg with a nominal ply thickness of 0.225 mm [23]. The adhesive used was FM300-2K film adhesive.

### 3.1.1 Thin Laminate Specimens

The laminate specimens were in a double lap joint configuration (Figure 14). Specimens comprising of rivets (Cherry MaxiBolt CR7621U-05-04) had either a square array configuration (6 rivets) or a staggered array configuration (3 rivets) commonly found in aircraft structures/repairs. All specimens had the outer adherends of 5 plies thick with a stacking sequence of [(0/90)/(±45)/(0/90)/(±45)/(0/90)] and the inner adherend of 10 plies thick with a stacking sequence of [(0/90)/(±45)/(0/90)/(±45)/(0/90)]s. The outer adherends were manufactured with a 5 mm long end taper in the form of a ply drop as shown in Figure 14.

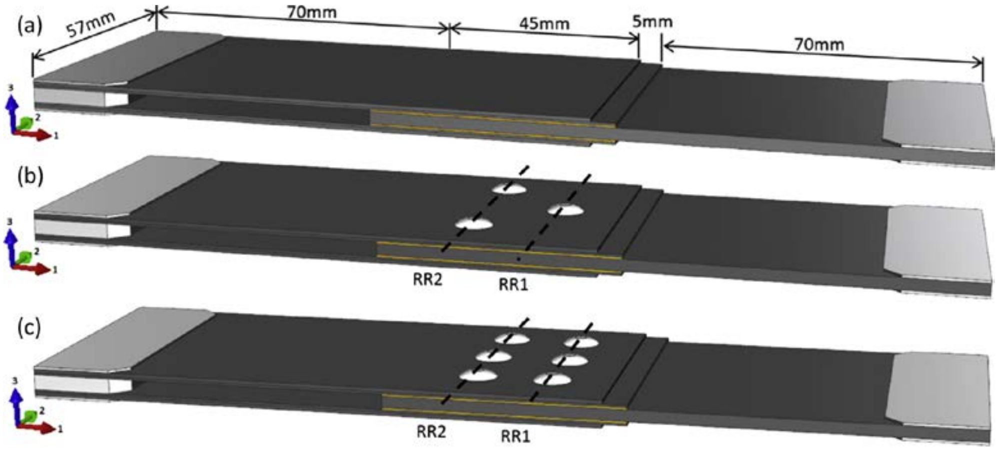



**Figure 14: Thin overlap joint specimens. (a)Bonded specimen; (b)3-rivets hybrid specimen; (c) 6-rivets hybrid specimen. The riveted specimens are identical to the hybrid specimens shown in (b) and (c) except that there is no adhesive bond in these specimens.**

To examine the defect tolerance performance, two kinds of manufactured defects were introduced to some of the specimens, namely, i) a disbond of 2 mm in length was created at the end of bonded and hybrid joint specimens by placing a Teflon tape; and ii) a weak bond was created to the bonded and hybrid joint specimens by under-curing the adhesive (93 Cº for 60 min which gave 46% of nominal strength).

The static strength test results are summarised in Table 3. The results showed that:

- Static strengths of the bonded and hybrid specimens without bond-line defect are significantly higher than that of riveted specimens (by over 60%).
- Static strengths of the bonded and hybrid specimens without bond-line defects are close to each other.
- A short disbond has insignificant effect on the static strength of bonded and hybrid specimens.
- The bonded specimen with a weak bond-line has significantly lower strength
- The hybrid specimen with a weak bond-line has a similar strength to the riveted specimen, which exceeds 5,000 $\mu\varepsilon$ (load converted to strain of pristine specimen).

The last point is particularly important. Considering the knockdown factor between average and B-base strength values (typically by 20%), the residual strength provided by the rivets would be sufficient to meet the requirement for bonded repair applications in most applications.

**Table 3: Measured static strength of thin laminate joint specimens**

| Bond-line condition | Joint type* | Average joint strength, kN (strain, με) |
|---|---|---|
| Pristine | Riveted | 36.8 (5,750) |
| | Bonded | 60.9 (9,516) |
| | Hybrid | 61.6 (9,625) |
| Defective -2mm disbond | Bonded | 60.1 (9,391) |
| | Hybrid | 58.4 (9,125) |
| Defective – low temperature cure | Bonded | 16.9 (2,641) |
| | Hybrid | 35.3 (5,516) |

* For hybrid specimens, only the results from those specimens with the square array rivet pattern were included

Fatigue tests were conducted in load control mode at a frequency of 5 Hz of constant amplitude tensile loading with a stress ratio ($R = \sigma_{min}/\sigma_{max}$) of 0.1. The peak load was increased after each 100,000 cycles until the specimen was failed. This incremental loading approach is effective in comparing the fatigue resistance among different specimen types in a relative scale, as reported in [44, 45]. The block loading regime is described in Table 4.

**Table 4: Block loading regime for thin laminate joint specimens**

| Cycles (N) | Strain (με) | Upper limit (kN) | Lower limit (kN) |
|---|---|---|---|
| 100,000 | 3,000* | 19.2 | 1.92 |
| 200,000 | 4,000 | 25.6 | 2.56 |
| 300,000 | 4,500 | 28.8 | 2.88 |
| 400,000 | 5,000 | 32.0 | 3.20 |
| 500,000 | 5,500 | 35.2 | 3.52 |
| 600,000 | 6,000 | 38.4 | 3.84 |

* For the two specimen configurations with weak bond, initial loading began at 2,000 με.

The test results are summarised in Table 5. The results showed that:

- For the pristine specimens, the fatigue resistance of the bonded specimens is significantly higher than the riveted specimens and the fatigue resistance of the hybrid specimens is in turn significantly higher than bonded specimens.
- A short disbond has insignificant effect on the fatigue resistance of bonded and hybrid specimens.
- The bonded specimen with a weak bond-line has virtually no fatigue resistance.
- The hybrid specimen with a weak bond-line has a similar fatigue resistance to the riveted specimen.

From the test result it is clear that indeed the rivets would be able to provide a safeguard for the hybrid repairs and the adhesive bond can provide enhanced static strength and fatigue resistance.

**Table 5: Fatigue test results of thin laminate joint specimens**

| Bondline condition | Joint type* | Failure load (kN) / Strain (με) | Average cycles to failure |
|---|---|---|---|
| Pristine | Riveted | 32.0/5000 | 303,768 |
| | Bonded | 32.0/5000 | 442,175 |
| | Hybrid | 38.4/6000 | 556,699 |
| Defective -2mm Crack | Bonded | 35.2/5500 | 415,097 |
| | Hybrid | 38.4/6000 | 532,282 |
| Defective – low temperature cure | Bonded** | 12.8/2000 | 2,096 |
| | Hybrid** | 28.8/4500 | 321,994 |

* For hybrid specimens, only the results from specimens with the square array rivet pattern were included
** Fatigue test started at 2,000 με loading regime.

### 3.1.2 Thick Laminate Specimens

The thick laminate specimens were in a step lap joint configuration (Figure 15). The adherends were 24 plies thick with a stacking sequence of $[(0/90)/(\pm45)_2/(0/90)]_6$. All the steps had the same thickness. The fasteners selected for the specimen's configurations were aerospace grade Aviaquip NAS1581C4T4 countersunk bolts which were ¼ inch in diameter [46] with suited MS21044 self-locking nuts. The bolted and hybrid configurations have a bolt at the centre of each step (Figure 15).

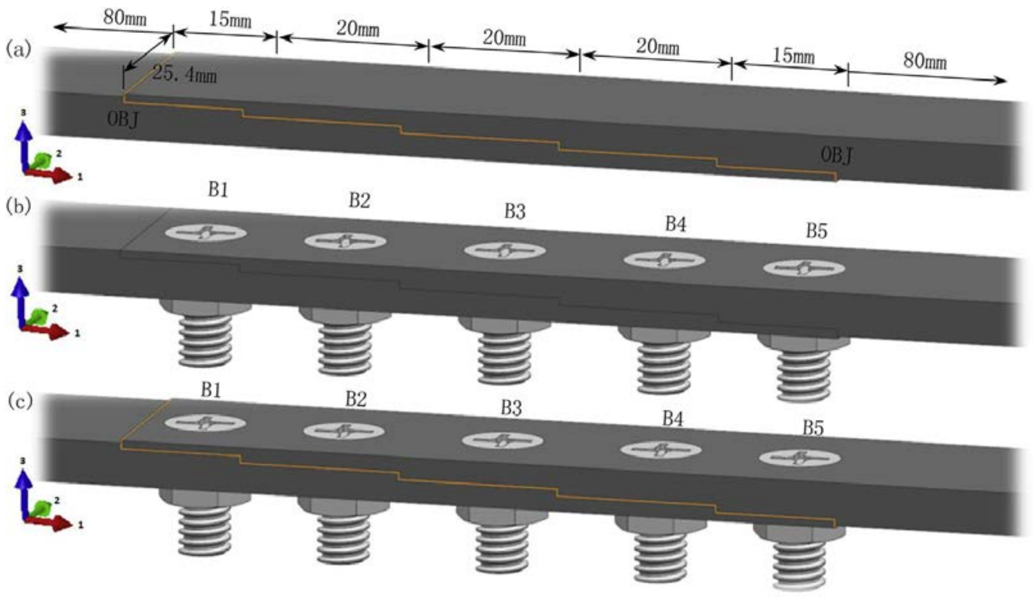



**Figure 15: Three major geometry configurations; (a) bonded step lap Joint; (b) bolted step lap joint; (c) hybrid step lap joint.**

The static strength test results are summarised in Table 6. The results showed that:

- In the cases where there was no bond-line defect, the static strength of the bonded specimen was higher than that of the bolted specimen (by 21%). The static strength of the hybrid specimen was in turn higher than that of the bonded specimen (by 10%).
- A short disbond has insignificant effect on the static strength of bonded and hybrid specimens.
- The bonded specimen with a weak bond-line has substantially low strength.
- The hybrid specimen with a weak bond-line has a similar strength to the bolted specimen, which exceeds 4,000 με.

Considering the knockdown factor between average and B-base strength values (typically by 20%), the residual strength provided by the rivets would still be sufficient to meet the requirement for bonded repair applications in many cases.

**Table 6: Measured static strength of thick laminate joint specimens**

| Bondline condition | Joint type | Average joint strength, kN (strain, με) |
|---|---|---|
| Pristine | Bolted | 27.8 (4,413) |
| | Bonded | 33.6 (5,333) |
| | Hybrid | 36.8 (5,841) |
| Defective -2mm Crack | Bonded | 33.1 (5,254) |
| | Hybrid | 35.5 (5,635) |
| Defective – low temperature cure | Bonded | 9.07 (1,440) |
| | Hybrid | 25.3 (4,016) |

Fatigue testing was conducted in load control at a frequency of 5 Hz of constant amplitude tensile loading with a stress ratio ($R = \sigma_{min}/\sigma_{max}$) of 0.1. The peak load was increased after each 100,000 cycles until the specimen was failed. The block loading regime is described in Table 7.

**Table 7: Block loading regime for thick laminate joint specimens**

| Cycles (N) | Strain (με) | Upper limit (kN) | Lower limit (kN) |
|---|---|---|---|
| 100,000 | 1,000 | 6.30 | 0.63 |
| 200,000 | 2,000 | 12.6 | 1.26 |
| 300,000 | 3,000 | 18.9 | 1.89 |
| 400,000 | 4,000 | 25.2 | 2.52 |
| 500,000 | 4,500 | 28.4 | 2.84 |
| 600,000 | 5,000 | 31.5 | 3.15 |

The fatigue test results are summarised in Table 8. The results showed that:

- For the pristine specimens, the fatigue resistance of the hybrid specimens is significantly higher than the bolted specimens and bonded specimens.
- Unlike in the case of the thin specimens, the fatigue resistance of the bonded specimens is significantly lower than that of the bolted specimens.
- A short disbond has insignificant effect on the fatigue resistance of bonded and hybrid specimens.
- The bonded specimen with a weak bond-line has substantially low fatigue resistance
- The hybrid specimen with a weak bond-line has a fatigue resistance close to the riveted specimen.

**Table 8: Fatigue test results of thick laminate joint specimens**

| Bondline condition | Joint type | Failure load (kN) / Strain (με) | Average cycles to failure |
|---|---|---|---|
| Pristine | Riveted | 25.2/4,000 | 309,604 |
| | Bonded | 18.9/3,000 | 214,973 |
| | Hybrid | 28.4/4,500 | 400,103 |
| Defective -2mm Crack | Bonded | 18.9/3,000 | 232,873 |
| | Hybrid | 28.4/4,500 | 406,147 |
| Defective – low temperature cure | Bonded | 6.30/1,000 | 8,139 |
| | Hybrid | 18.9/3,000 | 208,012 |

The significant difference of the fatigue resistance between bonded and hybrid joint specimens suggested more clearly the benefit to use the hybrid specimen instead of the bonded specimen. On the other hand, this result also triggered further (successful) investigation to optimise the bonded step repair without compromising the bolted repair part should a faulty bond line exist [18]. Effective methods to increase bonded strength include adding a thin overlap at the edge of the step joint and/or having more steps/increasing step length.

### 3.1.3 Detection of Disbond Initiation and Growth during Fatigue Tests

Visual, ultrasonic A-scan and thermographic inspections were conducted to detect damage during the fatigue tests. The specimens were inspected at the beginning, after 1,000 cycles, 3,000 cycles and 10,000 cycles of each block loading regime when the specimens were unloaded. Thermal imaging data was also captured throughout the test duration until failure.

Under the right lighting conditions initial cracking at the edge of the joint is detectable by visual inspection. In an actual aircraft repair scenario, the repair patch may not have a visible bond-line region except at the patch edges. Thus visual detection in an aircraft repair is beneficial up to the point of detecting crack initiation in bond-lines at the edges.

Ultrasonic A-Scanning was found to be able to detect a crack even when it was shorter than 2 mm. When crack length exceeded 5 mm in length they were readily detected using Ultrasonic A-Scanning. Thermographic inspection could detect 3~4 mm long cracks and was found particularly useful for detecting sudden crack propagation.

The fatigue test of the hybrid specimens with weak bonds clearly demonstrated that the fasteners in a hybrid joint configuration could act as failsafe mechanisms that provide sufficient service life for the bond-line defect to be detected during normal service inspections.

## 3.2 Computational Simulation

Finite element (FE) analyses were performed to verify the static and fatigue strength of mechanically fastened, bonded and hybrid joints. Most configurations used in the test programs were modelled. Adhesive non-linear material properties, surface contacts and frictional forces were included in the three-dimensional FE models (an example is shown in Figure 16). The Multi Continuum Theory [47, 48] was used to simulate the progressive failure process and the stress state for all specimens, whilst the strain energy release rate (SERR) as a function of crack length for bonded and hybrid specimens were also compared. Results showed that the FE models could accurately predict the bonded, fastened and hybrid joint strengths (typical examples shown in Figures 17 and 18). The SERR assessment revealed that the fasteners enhance the joint fatigue resistance through two mechanisms. Namely, the clamping forces supress Mode 1 crack initiation/propagation and the fasteners arrest Mode 2 cracks. The position of the first row of fasteners is critical in determining the crack growth rate. As the crack enters the fasteners' clamping zone there is a significant drop in SERR resulting in a much slower crack growth rate, therefore increasing the fatigue resistance of the hybrid joint configuration. For the hybrid specimens with weak bond-line, the simulation confirms as anticipated that the joint essentially switches to fastened joint after the initial bondline failure (Figure 18). The ‘defective semi-cured’ adhesive is quite detrimental for a bonded joint configuration. By using a ‘hybrid’ configuration the likelihood of sudden catastrophic failure is avoided through the added residual strength of the fasteners.

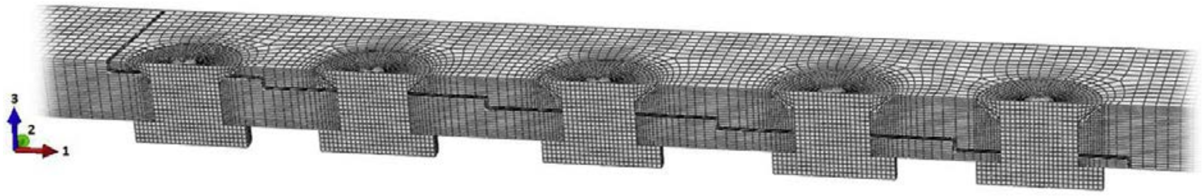


Figure 16: A typical FEM model - Meshed assembly of a hybrid step lap joint configuration.

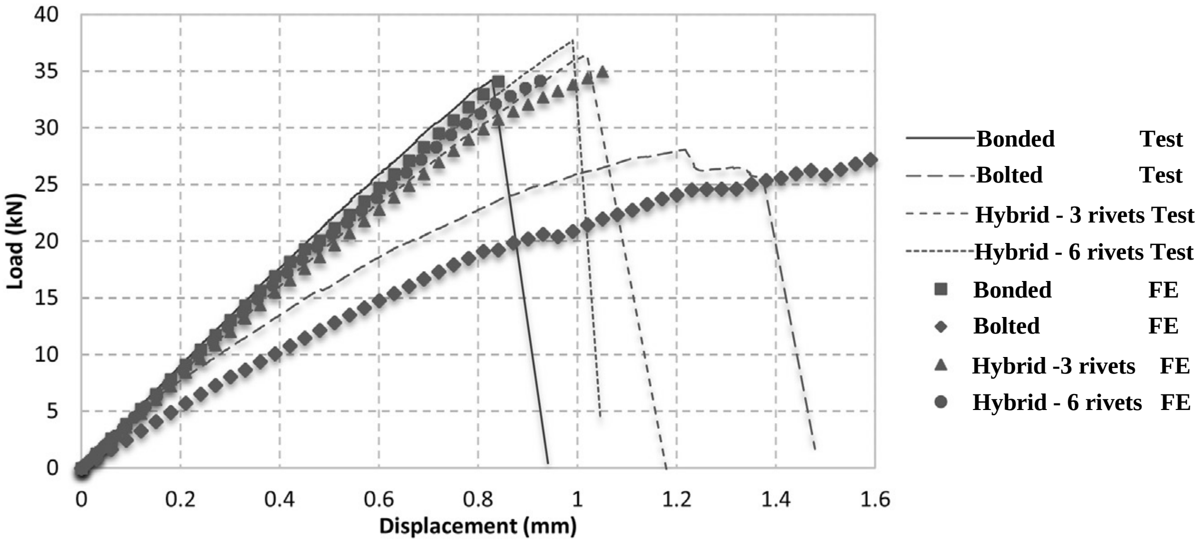



Figure 17: Load-displacement comparison between experimental and FEA for thick pristine specimen configurations; displacement represents the specimen's total elongation for the given load.

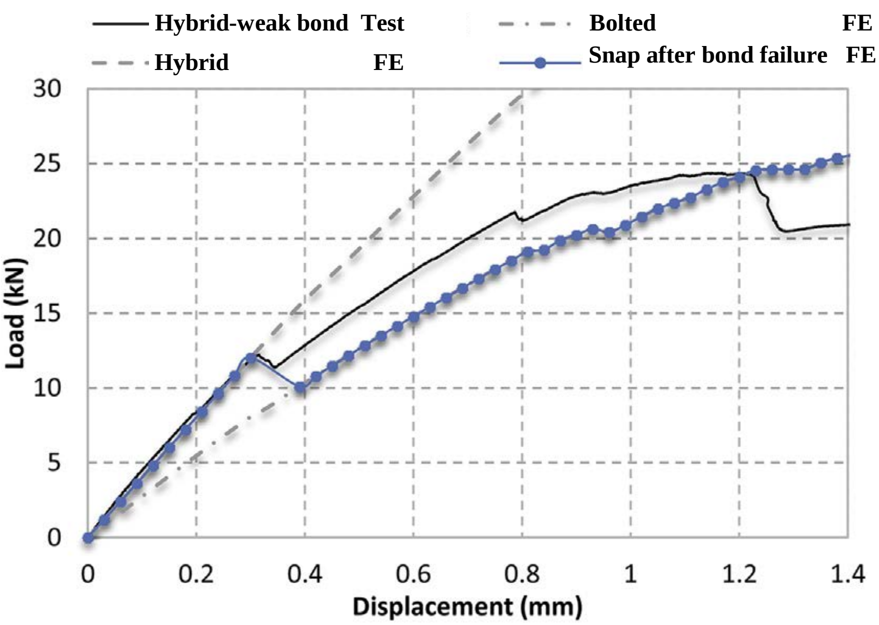



Figure 18: Load-displacement comparison between experimental and FEA for thick hybrid specimen with weak bond line.

# 4.0 HYBRID REPAIR – Z-PINNING/BONDED PATCH REPAIR

Preliminary research was also conducted to examine the feasibility of using micro-pins as a method for repairing aircraft structure. When used with the hybrid repair, the pins serve as a mechanical fastening system in addition to the adhesive, much in the same way as rivets/bolts in a hybrid repair described above. That is, this functional mechanical fastening method may help advance the use of adhesive bonding on composite aircraft structure by providing a means to ensure the repair can sustain the ultimate load case, whilst the bond-line could provide the fatigue durability required. Potentially, the benefit of using such a large number of small diameter pins is that the bearing stress around the pin hole can be lowered to avoid bearing failure.

The single-lap-joint specimen dimensions, Figure 19, were designed to be similar to standard test coupon size with the overlap length designed to suit the smaller array of pins. To avoid pin pull-out [49] a combined adherend thickness of 8mm was used at the overlap to provide sufficient vertical stabilisation against pin rotation [50], Figure 19.

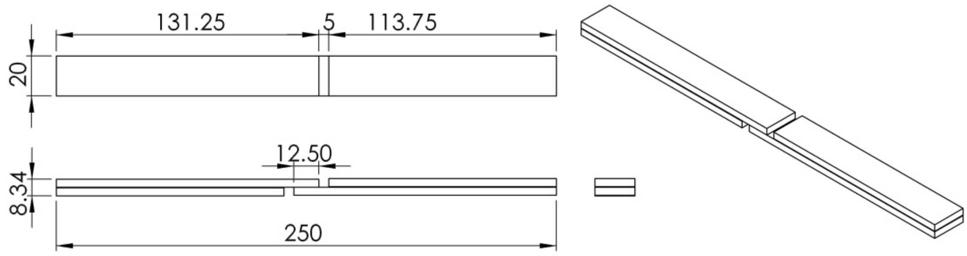



**Figure 19: Dimensions of the lap joint specimen for pinning assessment**

The laminate was made of IM-7 carbon fibre and Cycom 5250-4 bismaleimide (BMI) resin prepreg. The adhesive used was FM-32 film adhesive. The pin material was 0.5 mm diameter stainless steel, similar to other z-pinning studies [51]. The cured laminate was drilled to create a hole array using a CNC router before stainless steel wire was threaded through with an interference fit. The wire was cut and ground down to create "pins". Based on previous work, a square pinning array with 4-diameter spacing was used [52], which equated to a 5 percent areal density, higher than that used in earlier z-pinning research [49]. Two ply stacking configurations were examined; $[0]_{30}$ and $[45/0_2/-45/90]_{3s}$. A typical joint overlap of 12 mm and a specimen width of 20 mm provided a square array of 45 pins (5 rows by 9 columns). Mechanical testing used 1 mm/min crosshead displacement. Joints were conditioned to 1.2% by weight of moisture uptake and tested at 177°C for the hot/wet condition. Fatigue cycling used R-ratio of 0.5 and 5 Hz cycling.

The room temperature static test results showed that the load carried by the pinned joints in the unidirectional laminate and the stiff orthotropic laminate were very similar, both at 7.4 kN, which is 10% higher than the load carried by the bonded joint, 6.7 kN. The test results also showed that the loading capacity of the hybrid joint is similar to that of the pinned joint.

The elevated temperature/wet strength of the hybrid joint was 15% lower than the room temperature strength, and was 65% higher than that of the bonded joint at hot/wet (data from [53] was used for this comparison).

A fatigue test conducted at 0.5 ultimate load on a pinned joint made from orthotropic laminate and without adhesive ran out after one million cycles. The same joint when fatigued at 0.8 ultimate load, failed at only 400 cycles. The fatigue performance of the pinned joint was well below that expected for an equivalent bonded joint at room temperature, limited by the mechanical characteristics of the pinning method.

FE modelling using StressCheck [54] examined the bearing and bypass stresses in the laminate. The model geometry (Figure 20) matched the coupon specimens used in the experimental investigation, Figure 19.

Elements in the model were a mixture of 3D solid pentahedral (6 noded) and hexahedral (8 noded). The load transfer between adherends was achieved by defining contact zones on (i) outer surface of each pin, (ii) inside surface of each hole and (iii) mating adherend surfaces. A 7kN load was applied to the end of the specimen. The modelling was principally to examine the stress concentrations of the pinned joints. Due to the dense small holes compared to the sparse large holes in the bolted joint case, the stress concentration was expected to be lower. The results indicated the stress concentration was indeed rather low (Figure 20).

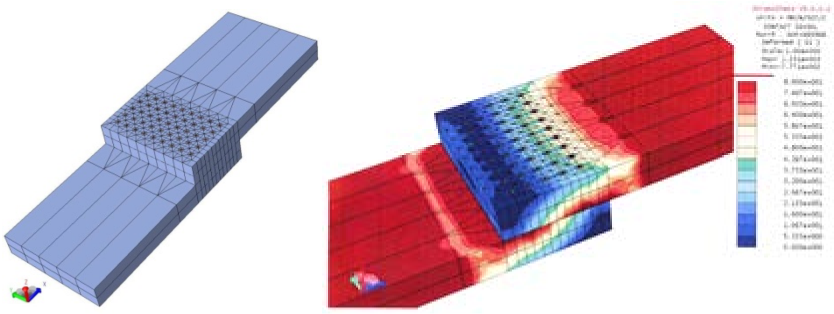


**Figure 20: FEM model and typical stress plot**

Although preliminary, the above results demonstrated this hybrid repair approach for primary structures is promising and warrants further research effort.

# 5.0 DISCUSSION

## 5.1 Significance of Hybrid Repair in Terms of Repair Certification

The key requirement to gain certification for the bonded repair applications on primary structures, where the residual strength of the damaged structures is insufficient prior to application of the bonded repair, is to ensure initial and through-life strength and structural integrity of the patch system and specifically the adhesive bond. Before quality control of bonded repairs can be sufficiently advanced, the possible solution to this requirement would be to ensure the initial bond strength through a proof test and to ensure the through-life bond strength by conducting periodic proof tests / applying structure health monitoring [55].

However, proof tests that apply a load directly to an aircraft component, such as the cold proof test applied periodically on the wing structure of F-111 [56], can be very costly and often infeasible. Further there is also a considerable risk of damaging the parent structure as well as degrading the repair. Alternative proof test approaches, such as Laser shock [57] and Bonded Repair Coupon [6, 7], are under development.

In the hybrid repair configurations discussed above, the enhanced residual strength and fatigue life of the repaired structures in the absence of the adhesive bond can be routinely determined. If the inspection interval is specified in relation to this fatigue life, then the risk of catastrophic failure due to weak adhesive bond would be eliminated.

## 5.2 Credit to the Adhesive Bond

After each inspection, if no bond line damage is founded, then the credit can be given to the adhesive bonded patch alone for fatigue life extension of the parent structure through its stress reduction function over the past flight hours [55]. Thus, through properly specified inspection intervals, including patch replacement should disbond be found, the expected full life extension can be achieved through bonded repairs with the hybrid approach providing a safety factor.

### 5.3 Further Research

Although significant work has been conducted and the hybrid repair concepts show promise for primary structure repair applications, significantly more work needs to be carried out.

Direct extension of the repair research work presented in Sections 2-4 above include:

- For hybrid repairs combining optimum damage removal and a bonded patch, the optimisation of repair patch shape, lay-up and ply-drop (thickness profile) is to be explored
- For hybrid repairs combining mechanical fasteners and a bonded patch, i) at coupon level, different parent structure/hybrid patching systems need to be explored; ii) at the full sized specimen level with a full hybrid patch, design and validation are needed
- For Z-pinning, i) at coupon level, the investigation needs to be extended from the short overlap specimens to long overlap/step joint specimens; ii) at the full sized specimen level with a full hybrid patch, design and validation are needed
- To explore the applications where the residual strength could be enhanced by using both optimum damage removal and addition of alternative load path
- For all of the above, various representative loading and environmental conditions need to be considered.

In general, further research must focus on more realistic repair applications. The individual tasks described would be part of this overall strategy.

## 6.0 SUMMARY

- Currently bonded repairs can only be permitted on those primary structures suffering cracks/damages having a residual strength well exceeding design limit load prior to application of the bonded repair.
- One approach to meet the certification requirement is to combine a bonded patch with other approaches that could enhance residual strength of the damaged structures. DST and its research partner organisations have recently conducted significant research in this area. The outcomes from six individual research programs indicated the hybrid repair methods are promising for primary structure repair applications.
- The computational and experimental work indicated very significant residual strength increases can be achieved by applying shape optimization for crack/damage removal for both composite and metallic components. The applications of bonded patch repairs following the optimum cut-out, with three different patch design concepts that suit different application scenarios, were shown to provide significant benefit in enhancing the static strength and fatigue resistance.
- In fastened-bonded hybrid patch repairs, the required residual strength can be achieved, in the absence of an adhesive bond, through the alternative load path from the mechanical fasteners. It was shown experimentally that this hybrid repair has significantly higher fatigue life than that of a fastened patch repair for both thin and thick composite structures. The computational modelling indicated that mechanical fastening assists in preventing Mode I and arresting Mode II crack propagation. Similarly, z-pinning patch repair has also been demonstrated to have capability to provide high static strength to a bonded joint through providing alternative load paths. At elevated temperatures, the reduction of joint strength in a mechanical fastened /pinned joint is negligible or at least much less significant than that of an adhesive bonded joint, which is also a significant factor when residual strength enhancement is considered.
- In the hybrid repair configurations, the residual strength and fatigue life of the repaired structures without presence of the adhesive bond can be routinely determined. When the inspection time is specified in relation to this fatigue life, the risk of catastrophic failure due to weak adhesive bond could be eliminated.

- If no bond line damage is founded during an inspection, then the credit can be given to the adhesive bond in the hybrid repairs for fatigue life extension of the parent structure retrospectively through the stress reduction during the past flight hours. Thus, through properly specified inspection intervals, replacing the patch should disbond be found, the expected full life extension can be achieved through the hybrid repairs.
- Significant more work is required in this area of research, including direct extension of the repair research work presented in the paper. Ideally and importantly, the further research should focus on a real repair application or demonstration for a real application.

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