FINITE ELEMENT ANALYSIS FOR THROUGH THICKNESS REINFORCEMENT REPAIR OF DELAMINATED CARBON-EPOXY PANELS

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Keywords: *Finite element analysis, delamination, composite repair, through-thickness reinforcement*

Abstract

This paper outlines and examines a Through-Thickness Reinforcement Repair (TTRR) technique for delaminated polymer matrix composite (PMC) laminates. This involves drilling fine holes through the delaminated region and bonding pins into these holes. A rapid finite element technique was developed to rank different configurations as to their effectiveness as a repair technique. A previous experimental study found the TTRR technique to be highly effective, with 8 pins bonded in a ring within a 50 mm circular delamination. This repair restored the open hole compression strength to 93% that of the undamaged laminate. A further increase in pin density was found to provide no additional improvement in the repair efficacy. These results were confirmed by the finite element analysis.

1 Introduction

Carbon/epoxy composite materials are steadily displacing metal alloys in the construction of aerospace structures due to their superior specific strength and stiffness. These materials are also used for small-volume replacements of ageing aircraft components that are no longer available [1]. However, due to the lack of through-thickness reinforcement, composite laminates are susceptible to delamination damage. Even damage that is not visually detectable can degrade significantly the strength of the component.

Delaminations are most commonly generated by impact (typically runway debris, hailstones or tool drops during maintenance) or the over-tightening of fasteners. This can lead to a significant degradation in compression strength because the delaminated sub-laminates can buckle more easily than the intact laminate. One repair technique involves injection of epoxy resin into the delamination region [2-6]. The aim of this approach is to re-bond the delaminated sub-laminates. This technique may not be effective because the surfaces of the delaminations can be contaminated by the environment, making any bonding ineffective, and it is impossible to assess the bond quality by non-destructive inspection [5,7].

It is proposed that a more effective alternative is Through-Thickness Reinforcement Repair (TTRR). Here, holes are drilled through the delaminated region and pins bonded into these holes. It is hypothesised that the pins will effectively shorten the length of the sub-laminates and increase the load at which they buckle. Also, it will be much easier to produce a reliable bond between freshly drilled holes and pretreated pins than between contaminated delamination surfaces.

A previous experimental investigation [8] confirmed that such repairs were highly effective in restoring the open hole compressive strength of such through thickness reinforcement repairs in panels in the delaminated region surrounding the hole.

A rapid finite element technique has been developed to rank different configurations as to their effectiveness as a repair technique.

This paper describes the TTRR repair technique and an investigation of the effects of delamination and TTRR on the open-hole compression strength of quasi-isotropic carbon/epoxy laminates. A summary of the experimental study is included as is a description of the finite element techniques and analyses used to determine the strength of undamaged panel and the relative effectiveness of the TTRR technique using different pinning patterns. The practicality of the repair technique for use on real composite aircraft structures is also discussed.

2 Repair Technique

The repairs were applied to delaminations in 4 mm thick quasi-isotropic carbon epoxy panel. These 50 mm diameter delaminations surround a 6.35 mm holes in the panel.

A set of 1.0 mm holes were drilled through the panel. They were located within the delaminations region and they formed a pattern of concentric rings about the 6.35 mm hole. Stainless steel M1 x 5 machine screws were used as through-thickness reinforcement. These were bonded in the holes using Hysol 9320NA epoxy adhesive. The adhesive was first smeared over the surface of the specimen until it had run through the holes and emerged out the back face. The stainless steel screws, that had been previously degreased using methyl ethyl ketone, were then pushed through the adhesive-filled holes. The holes were sufficiently large so that the screws could be pushed through with little resistance. This allowed sufficient adhesive to remain between the screw thread and the hole wall to ensure adequate bonding.

3 Experimental Study

Open hole compression testing [8] has shown that such repairs have the potential to produce a high degree of strength recovery.

3.1 Specimen Manufacture

The test specimens were manufactured using T300/914C unidirectional tape with a lay-up sequence of $[45\ 0\ {-45}\ 90]_{4S}$. The specimen dimensions were 150 mm x 100 mm x approximately 4 mm. A 6.35 mm diameter hole was drilled in the centre of each specimen. A total of 28 specimens were manufactured for this study.

3.2 Delamination Creation

Delaminations were created by controlled lateral loading of the specimens, supported on a plate over a 50 mm diameter hole, concentric with the central hole. The specimens were fully clamped to the support plate and a bolt was inserted through the central hole. The bolt was then loaded in a test machine in displacement control (1 mm min^{-1}) until a load drop of 30% was detected. The load at which this occurred was approximately 9 kN for all specimens. This set-up produced relatively consistent quasi-circular delaminations of approximately 50 mm in diameter. The C-scan image of a delaminated panel is shown in Figure 1. The specimens were water-coupled during C-scanning which also contaminated the delaminated surfaces, an effect similar to exposure to the in-service environment.

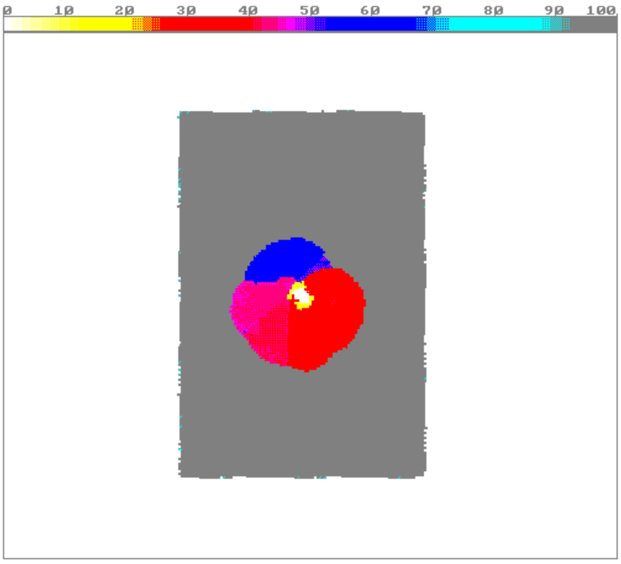
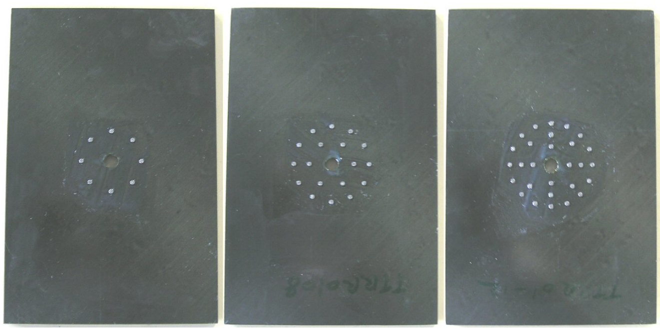


Fig. 1. C-scan of delaminated panel showing quasi-circular delaminations approximately 50 mm in diameter near the laminate mid-plane. The scale indicates the percentage depth of the reflected signal

3.3 Repair

Some of the delaminated specimens were repaired with TTRR as described above. Three different pin configurations were used and the respective pinning patterns are shown Figure 2 which presents photographs of the three repaired specimen types.



8 pin repair 18 pin repair 28 pin repair

Fig. 2. Photograph of pinned specimens

3.4 Repair

Four specimens were prepared for each repair configuration and six specimens were prepared each for the undamaged and delaminated (no repair) conditions. The test matrix is presented in Table 1.

Table 1: Test matrix and results summary

Condition	No. Specimens	Compression Strength (KN)	% Undamaged Strength
No Damage	6	130.6 ± 1.4	100
Delaminated	6	88.5 ± 1.0	68.2
Delaminated and Repaired 8 Pins	4	121.2 ± 4.0	93.3
Delaminated and Repaired 18 Pins	4	117.7 ± 5.2	90.6
Delaminated and Repaired 28 Pins	4	119.5 ± 1.9	92.0

3.5 Compression Testing

The specimens were tested in a SACMA SRM 2R compression after impact test rig with anti-buckling edge constraints. A V-groove and pin rocker arrangement was used between the upper specimen clamp and loading platen to produce a uniform load across the width of the specimen. Spring loaded mounts were installed on the test fixture to press MTS 632 series extensometers against the front and rear specimen face. The extensometers were positioned back-to-back and approximately 25 mm from the edge and bottom of the specimen. This was assumed to provide far-field strains.

The specimens were loaded to failure in a 250 kN MTS hydraulic test machine under displacement control of 0.5 mm min^{-1}. Load, displacement and strain were recorded at a sampling rate of 1 Hz. A photograph of the experimental set-up is shown in Figure 3.

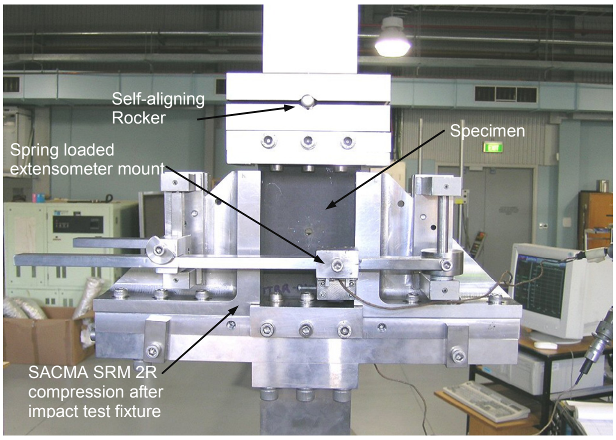


Fig. 3. Compression test set-up

4 Experimental Results and Discussion

The results are summarised in Table 1 and in Figure 4 where the error bars represent one standard deviation.

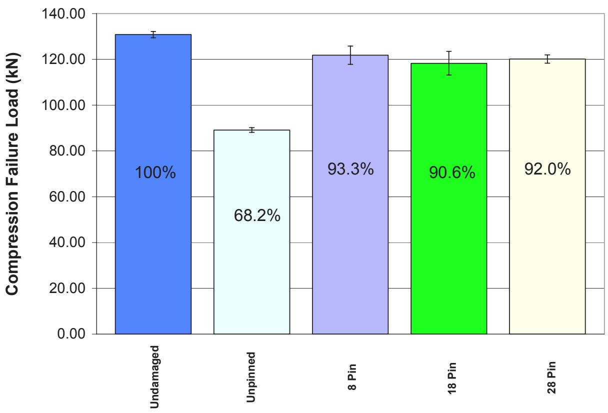


Fig. 4. Comparison of compression failure load of undamaged, delaminated, and repaired specimens

The undamaged specimens failed at an average load of 131 kN, corresponding to a far-field strain of -7,500 με. There was very little scatter in the results of all the specimens tested, with a coefficient of variance of 1.0%. Fibre splitting in the 45^{o} plies at the surface was observed prior to failure. Specimens exhibited the classic characteristics of compression failure, a flat line failure originating at the edge of the hole and propagating perpendicular to the applied load out to the edge of the specimen. The cross-sections displayed interleaved brooming characteristic of a well aligned and restrained test specimen.

The delaminated specimens failed at a significantly smaller compressive load than the undamaged specimens, with an average reduction in peak load of 32%. The scatter in the results was small, with a coefficient of variance of 1.1%. Failure of the specimens occurred through local buckling of the delaminated region and subsequent growth of the delamination. Damage propagation occurred laterally, normal to the direction of the applied load as predicted by the finite element model. The average far-field strain at failure was −5200 με. It observed that failure occurred soon after the onset of local buckling. The compression stiffness of the specimen decreased slightly as a result of this buckling.

As may be seen in Table 1 and Figure 4, the 8-pin repair configuration was able to restore the compression failure load of the delaminated panel to 93% of that for the undamaged specimen. Further increases in pin density had no significant effect. The far-field strain at failure of all repaired specimens was slightly lower than that of the undamaged configuration, at around -7,000 με. The scatter in the results of the repaired specimens was greater than the undamaged and delaminated panels, with standard deviations of 4.0, 5.2 and 1.8 kN for the 8, 18 and 28 pin configurations respectively. However the coefficient of variance for all these configurations was 4.4 % or less.

It appears that the repaired specimens failed by the same mechanism as the delaminated specimens, local buckling ("pop-out") of the delaminations around the central hole leading to microbuckling of the 0^{o} plies. The microbuckle propagated from the edge of the central hole, along the line of maximum compression stress, to the edges of the specimen. During testing at least some of the early pop-out buckles were clearly arrested by the pins. Additional load on the specimen, relative to the unrepaired specimen, was required to drive the pop-out buckle beyond these pins and this is the reason for the restored strength. There was no indication of any reduction in the compression stiffness during loading.

It is uncertain whether the second phase of local pop-out buckling caused disbonding between the pins and laminate, and whether increasing adhesion and/or adhesive strength would provide greater resistance to this pop-out. Certainly some of the bonded pins near the horizontal centre line of the repaired specimens were pulled through the panel. If this hypothesis is correct then even more specimen strength may be restored through improved bonding and adhesive selection.

5 Finite Element Modelling

A finite element model was used to predict the failure load for the undamaged specimen.

In order to study the efficacy of various repair configurations it is necessary to predict the delamination propagation load. This would require the application of a comprehensive fracture mechanics approach (e.g. the virtual crack closure technique (VCCT) [9]). This complex methodology is labour and computationally intensive.

A simplified technique, based on observed failure modes, was developed in order to rank candidate configurations. This does not predict the delamination propagation load but it does give a measure of the relative propensity of a particular configuration towards delamination growth.

5.1 Model Description

A quasi-isotropic carbon/epoxy panel with a stacking sequence of $[45\ 0\ -45\ 90]_{4s}$ was modelled in MSC NASTRAN using the 2D composite shell formulation with CQUAD4 elements. The dimensions of the panel were 150 mm long by 100 mm wide. A 6.35 mm diameter hole was incorporated at the centre of the panel to represent a fastener hole. The material properties used in the model corresponded to T300/914C carbon epoxy unidirectional prepreg and are given in Table 2.

Table 2. Material Properties of T300/914C UD prepreg

Property	Value	Unit
E_1	130	GPa
E_2	9.5	GPa
ν_{12}	0.32	-
G_{12}	4.3	GPa
ε_{1t}	0.015	-
ε_{2t}	0.006	-
ε_{1c}	0.011	-
ε_{2c}	0.02	-
γ_{12}	0.01	-
Ply Thickness	0.125	mm

A maximum strain failure criterion was used to determine the strength of the undamaged panel. However, for the delaminated panel, failure was expected to occur by delamination growth and hence this failure criterion was deemed inappropriate.

A 50 mm diameter mid-plane delamination was modelled as a bifurcation of the shell representing the panel into two separate subshells in the delaminated region. These subshells were separated by a distance of 0.2 mm at the centre of the delamination with the separation distance tapering linearly to zero at the edge of the delamination. This geometric configuration allowed the delaminated region of the panel to displace outwards as compressive loading was applied, and thus simulating the behaviour in real life.

The through-thickness reinforcement pins were modelled using 1D bar elements with a diameter of 1 mm and the properties of steel. Three pinning patterns were investigated corresponding to the repair configuration of the test specimens described above. These were: a one-ring, a two-ring and a three-ring configuration, all with equal-distant spacing between the rings. The rings were set between the circumference of the centre hole and the edge of the assumed 50 mm circular delamination. Eight pins were incorporated for the one-ring configuration, 18 pins were incorporated for the two-ring configuration, and 28 pins were incorporated for the three-ring configuration, as shown in Figure 5.

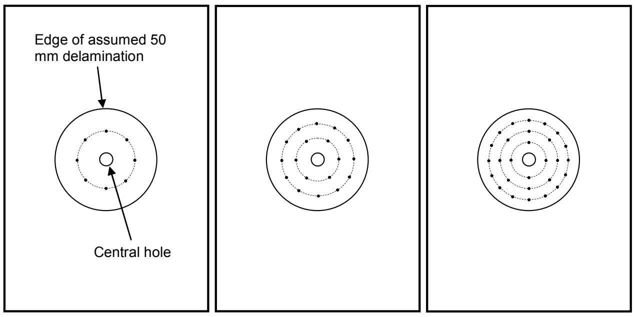


Fig. 5. 8, 18 and 28 pin repair configurations

To properly predict the delamination propagation load, a comprehensive fracture mechanics approach (e.g. the virtual crack closure technique (VCCT) [9]) would be required. However, due to the complexity of the method, the authors have opted to use a simplified approach to determine the relative effectiveness of the various repair configurations. This does not predict the delamination propagation load; however, it allows the ranking of the various configurations as to their relative propensity towards delamination growth.

Based on the same assumption as the VCCT, that the stiffness of the crack front remains constant during loading until propagation occurs, the energy release rate may be deduced to be proportional to the square of the applied forces. Consequently, these forces at the crack front alone, which may be determined using multi-point constraints (MPC), may be used as an indication of the relative propensity, for the various configurations, towards delamination growth. A higher MPC force would signify greater instability at the crack front and hence a lower failure load. Although this approach would not provide an absolute prediction of the failure load, a relative comparison of the efficacy of the different repair configurations could be easily obtained.

A total of 8 equally-spaced MPCs were incorporated in the delaminated model around the edge of the delamination to extract the forces. It was further assumed that Mode I loading was the dominant failure mode, and hence only the Mode I (through-thickness) component of the force was considered. The finite element mesh is shown in Figure 6 indicating the locations of the MPCs.

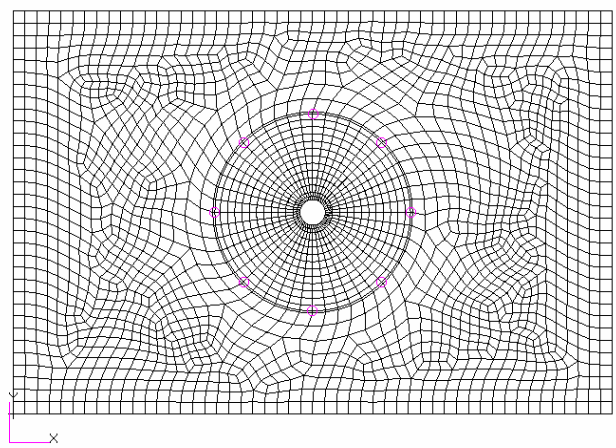


Fig. 6. Finite element mesh of the delaminated panel showing 8 MPC around the edge of the delamination

The models were loaded with a distributed compressive force applied the lengthwise (X) direction. As in the experimental rig, anti-buckling side constraints were incorporated to suppress global buckling. This was achieved by applying out of plane constraints to all elements within 6 mm of the panel edge. A linear static analysis was used for the undamaged open-hole model while non-linear static analyses were conducted for the delaminated and repaired models.

5.2 FE Results

Results of the undamaged panel showed that failure initiation around the hole in the 45^{o} plies occurred at a load of 105 kN. However, because the principal load bearing members were the 0^{o} plies, this did not signify final failure of the composite. At a higher load of 133 kN, failure initiation in the 0^{o} plies was observed in the FE model. This was deemed the predicted failure load for the undamaged open-hole composite panel. This value compares well with the experimentally determined value, 130.6 ±1.4 kN.

For the delaminated panels, the MPC forces at the edge of the delamination were calculated and compared. Figure 3 shows the relative MPC forces for the different repair configurations normalised against the unrepaired panel.

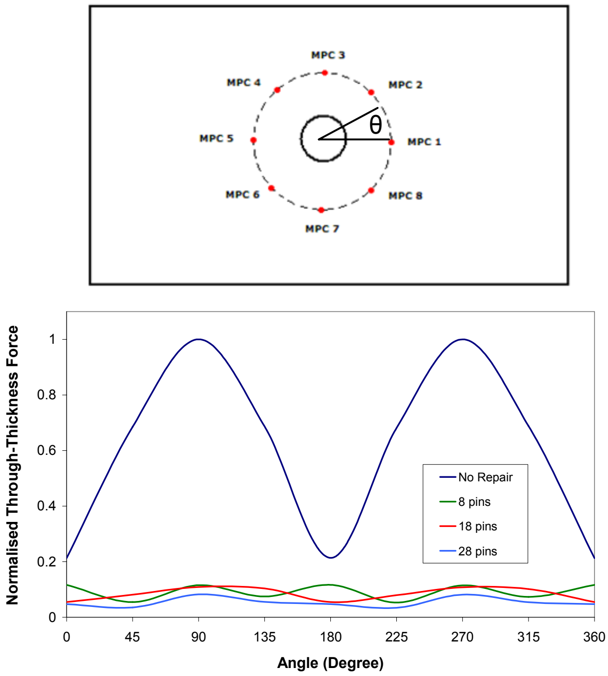


Fig. 7. Normalised MPC forces at the edge of the delamination for the various repair configurations

It can be seen from Figure 7 that the greatest forces occurred at the lateral ends (MPC 3 and MPC 7), 90^{o} to the applied load. It may therefore be inferred that the delaminated panel would fail through damage propagation in the lateral direction. This agrees with the experimental results, where the failure mode was observed to be delamination progression in the lateral direction.

The 8-pin repair configuration can be seen to reduce the peak MPC forces significantly over the unrepaired panel. However, it is interesting to observe that further increases in pin density yielded greatly diminished returns. Evidently, the 18 and 28-pin configurations showed no significant improvement in the repair efficacy compared to the 8-pin design. This again confirms the experimental results which exhibit similar behaviour as may be seen from Table 1 and Figure 4. The FE analysis indicates that all the repairs would suppress delamination up to a load of about 10 times the delamination progression load for the unrepaired panel; this corresponds to a load well in excess of the failure load for the undamaged plate. In the experimental study, delamination progression was observed in the repaired specimens at loads approximately 90 % of the failure load for the unrepaired specimens. This was however associated with pull-out of the pins, which was not modelled in the numeric study.

6 Practicality of the Repair Technique

The Through-Thickness Reinforcement Repair technique could be used to repair delaminated composite aircraft panels at a depot level, where a dedicated drilling machine is available. However, it may also be possible to use this technique in the field as a rapid repair method without the need to remove the damaged component from the aircraft. The challenge of the later proposition lies in the ability to drill small diameter holes in PMC composite panels in-situ. Due to the small size of the hole, it is extremely difficult to manually drill the hole with a conventional hand tool. Thus, a technique or a specialised tool (portable drill press) must be developed.

An alternate solution is to use 2 mm diameter pins. The effectiveness of such repairs is the subject of our further research.

7 Conclusion

A Through-Thickness Reinforcement Repair technique has been developed to repair delaminated CFRP composite laminates. This involves drilling 1 mm holes within the delaminated region in concentric rings, into which stainless steel M1 machine screws are adhesively bonded.

A simplified finite element technique, based on observed failure modes, was developed in order to rank candidate configurations. This does not predict the delamination propagation load but it does give a measure of the relative propensity of a particular configuration towards delamination growth.

Experimental and finite element studies were conducted to assess the effectiveness of the technique to restore the compression strength of delaminated panels. The configurations examined included an 8-pin 1-ring, an 18-pin 2-ring and a 28-pin 3-ring configuration with equal-distant spacing between the rings. It was found that the controlled 50 mm circular delamination reduced the compression strength of the panels to 68% that of the undamaged specimens. The TTRR technique was found to be highly effective in restoring compression strength by providing buckling resistance to the delaminated region. The 8-pin 1-ring repair configuration restored the compression strength of the delaminated panels to 93% that of the undamaged specimens. The higher density pinning configurations were shown, both theoretically and experimentally, not to provide any additional benefits.

There was good qualitative agreement between the experimental and numerical results, indicating that the simplified finite element model may be used to optimise the number and location of the through-thickness repair pins.

The TTRR could be used as a highly effective depot-level repair technique for composite aircraft components. However, its utility as a rapid on-the-spot field repair technique requires development of methods and tooling for the manual in-situ drilling of small diameter holes in composite aircraft components. Alternately the pin diameters must be increased. The efficacy of repairs with larger diameter pins is the subject of further research.

Acknowledgements

The authors gratefully acknowledge the following for their valuable assistance with the indicated activities; K. Houghton (CRC-ACS) - specimen manufacture, H. Morton (DSTO) - C-scanning and D. Parslow (DSTO) - mechanical testing.

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