



Effect of Curing Cycle on Fatigue Life of Cracked AA7075-T6 Aircraft Sheet Repaired with a Boron/Epoxy Composite Patch

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Abstract

Fatigue failure cracking under fluctuating stressing has been a major concern of engineering for many years. Fatigue failures have become more common and thereby significantly influencing engineering design as sources of vibration and dynamic loading of components have increased. The repairs of damaged metallic structures using bonded composite patch have been proved as an economical method to increase the fatigue life of damaged structures. This paper deals with an experimental study on the fatigue life of AA7075-T6 aluminium alloy plates reinforced with single-sided boron/epoxy patches. The study focuses on the effect of different curing cycles on the fatigue life of reinforced/repared structure. Bonds were tested with four different curing cycles. Distortion behaviour of bonded repair is studied. The influence of the curing cycle conditions on fatigue life is presented. Also, the repair patch effectiveness on the stress state at the crack tip in increasing the fatigue life of the unrepaired structure is shown. The obtained results show that repair of the cracked plate can be done effectively by using bonded composite patches. It is shown to decrease the crack growth rate to enhance the fatigue life of the structure by 70%. Elevated curing temperatures enhance the adhesion between the metal surface and the composite patch and overall durability. However, elevated cure temperature may result in distortion with bending curvature. The optimum combination of temperature and duration for the curing cycle is of importance to reduce the distortion.

Keywords Cure behaviour · Fatigue life · Boron/epoxy · AA7075-T6 · Crack · Failure

1 Introduction

The technology of advanced composites has been utilised for a long time in repair/reinforcement procedures of metallic aero-structures. The bonded patch composite is an example of these advances in aviation and other disciplines such as civil engineering. The repair reliability remains an engineering concern from a technology and design evaluation perspective. It can be dangerous if the repair is applied and implemented badly in comparison to an unrepaired structure (Gwo-Chung and Shyan 2004). A commonly used repair/reinforcement method is mechanical fastening, by attaching the metallic patch with bolting or riveting over the

damaged part to rearrange the load distribution and minimise the stress concentration. This mechanical method is less appealing, due to non-uniformity of the load transfer and the formation of stress concentration areas nearby the fasteners. The main advantages of composite materials for repair applications are their high strength, relatively low weight and good corrosion resistance (Niedernhuber et al. 2016). Bonded patch repair to metallic structures has been used before by the Aeronautical Research Laboratory (ARL) in Australia as another option to bolted repairs in their development programs. The application of the bonded composite repair in some of the ARL fleet repair programs exhibited a significant improvement of the durability of aerostructures in-service environments (Baker and Jones (1988). The Composite Patch Bonded Repair [CPBR method consists of the following steps: (1) preparation of the metal surface, (2) the composite patch application, (3) Curing of the repair materials, and (4) a cooling step (Heatcon Abaris Training 2008)].

Structural fatigue cracking is inevitable in all ageing aircraft because of usage. High strength aluminium alloys such as AA7075-T6 are particularly susceptible to fatigue

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cracking. AA7075-T6 is our material of choice for this study. This alloy is used due to its high strength, but this compromises crack initiation/growth performance and corrosion resistance, especially susceptibility to stress corrosion cracking. Structural profiles also notably affect the fatigue existence. For examples, square holes or sharp corners lead to high local stresses where fatigue cracks can initiate (Chakherlou and Vogwell 2003). Aircraft parts may also be exposed to multiaxial stress states of fluctuating amplitude, which can lead to fatigue crack initiation and growth and catastrophic failure (Bureau of Accident Investigation 1988). When subjected to cyclic loading, the presence of defects reduces the operating life of structural components. This can cause damage to the interior structure of laminated composites, which are barely visible in many instances. Hence, fatigue and reliability analyses and prediction of structural performance are important to investigate, especially with the development of composite materials. Such studies have been done in various cases by using different techniques (Khashaba et al. 2017; Biscaia et al. 2017; Heshmati et al. 2017).

Bonded Repair technology is becoming more advanced and accepted, but additional research is needed to help predict how well bonded composite patches perform. There are many parameters that influence the success of the repair of damaged metal components, such as the mechanical and geometrical properties of the repair materials (composite and adhesive) (Benyahia et al. 2014; Baker et al. 2009; Chung 2003; Bouanani et al. 2013). Important characteristics are the shape of the patch, the fibre orientation, number of plies in the composite, as well as the thickness of the adhesive (Mall and Schubbe 2009; Riccio et al. 2014; Brighenti et al. 2006; Zouambi et al. 2019).

The repair's performance is influenced by some other parameters too, including the thermal residual stresses caused by the adhesive during curing, debonding of the adhesive due to fatigue loading and bending curvature due to curing.

Numerous studies have been conducted on the repair or reinforcement repairs by CPBR for various applications of damaged components (Klemczyk and Belason 1994; Ouinas et al. 2007; Wang et al. 2006; Alderliesten 2009; Datla et al. 2011; Mall and Conley 2009; Günther and Maier 2010; Benyahia and Bouiadjra 2014; Benyahia et al. 2015; Sohail et al. 2017; Albedah et al. 2018). The most suitable materials have been identified as boron/epoxy and carbon/epoxy composites. The thermal expansion coefficient of boron/epoxy is advantageous, because it is closer to the thermal expansion coefficient of aluminium than carbon/epoxy (Gwo-Chung and Shyan 2004). Klemczyk and Belason (1994) showed that the rigidity of aircraft metallic structures was improved by using boron/epoxy patch repair or reinforcement. Ouinas et al. (2007) used boron/epoxy, as well as graphite/epoxy

for the repair of aircraft structures. The boron/epoxy patch resulted in a more significant reduction in the stress intensity factor at the tip of the crack tip. The thermal residual stresses were also found to be less for the boron/epoxy due to the adhesive curing. Various researchers Wang et al. (2006), Alderliesten (2009) and Datla et al. (2011) have performed several other studies experimentally and numerically. Mall and Conley (2009) investigated repaired cracks in thin and thick panels and observed significant bending. Asymmetric repair also resulted in non-uniform crack growth between the unpatched and patched faces of the thick panels. Analysis of metallic aircraft structures development and qualification showed secondary bending structures curvature of single-sided repairs in the repair area. This secondary bending structures curvature influenced the mechanical behaviour and further detailed studies were recommended (Günther and Maier 2010).

Expanding upon bonded repair, especially the evaluation of the fatigue behaviour of aluminium alloy sheets repaired with composite patches, the results indicated that the aircraft structures fatigue life could be improved with the patch repair. For example, Benyahia and Bouiadjra (2014) and Benyahia et al. (2015) conducted experimental fatigue tests on bonded composite repairs of aluminium alloy AA2024-T3 cracked sheet repaired with carbon/epoxy patch. The obtained results show the fatigue life of the damaged structure was increased, especially if the patch repair is performed at relatively small crack sizes. In the paper of Sohail et al. (2017), the behaviour of fatigue cracks in AA2024-T3 and AA7075-T6 under two levels of applied fatigue stresses plates repaired with the bonded composite patch was analysed. It was reported that the patches improved the fatigue properties, but this improvement was less when the applied load was increased. The AA2024-T3 had better fatigue properties in both repaired and unrepaired cases. Another study by Albedah et al. (2018) performed fatigue tests on repaired specimens bonded with elevated temperature curing. They evaluated the effects of thermal residual stresses and showed that fatigue life was greatly reduced by an initial adhesive debond. Their results also showed that the stress intensity factor increased with an increase in the debond width and due to thermal residual stresses.

There are two phenomena that decrease the performance of this technique; the adhesive debond by fatigue loading and the thermal residual stresses due to the adhesive curing. Many different curing effects of composites and adhesive curing have been reported. Studies showed that adhesive bonds experience a decrease in volume while they cure, causing either an accumulation of residual stresses inside the adhesive joint or a distortion of the bonding substrates (Wisnom et al. 2006; Taylor et al. 2005). Residual stresses in composites occur during the curing stage due to resin shrinkage (Olivier and Sawi 2010; White and Hahn 1992).

Timoshenko (1925) predicted the curvature and displacement of a one-dimensional bi-material beam using an analytical solution that is influenced by the coefficients of thermal expansion and by the material stiffness. However, the approach given by Timoshenko is limited to material properties that are uniform and constant. Also, it does not account for the entire curing process of composite and other important factors such as polymer shrinkage. To use a composite repair, it must be considered that, like all materials, adhesives and materials have an operating temperature range. Outside these limits, the repair could be unsuccessful. Other researchers have only used a single set of curing parameters, mostly those that have been recommended by the composite patch manufacturers. The novelty of our research is that we have tested different curing parameters to determine differences in terms of bond strength, distortion effects and fatigue life of repaired AA7075-T6 sheets. There are limited investigations available on the influence of curing parameters of structural epoxies on fatigue life of repaired metallic panels considering different curing regimes in a single study. Our study gives an overview on the fatigue life of bonded repair difference using four different curing process parameters. This will assist with applications of adhesive bonding by allowing implementation of smart cure cycles that give a large process window for successful bond repair.

In this investigation, the performance of composite patch repairs on AA7075-T6 sheets was compared to unpatched AA7075-T6 sheets. The fatigue life behaviour of aluminium sheets repaired with a composite patch with four configurations of curing cycles was studied. Through this study, we also aimed to provide more understanding of the composite

bonding repair process for future finite element analysis (FEA) to investigate the failure modes and more in-depth the mechanics of materials in the repair field. Experimental data will support more in-depth FEA analysis such as the effect of different curing conditions to find a model that gives the optimum combination of temperature and duration for suitable cure cycles to reduce secondary bending and formation of distortion of repaired structures after curing. FEA will allow numerous designs to be simulated before testing, which would otherwise be time-consuming and expensive to carry out experimentally.

2 Materials and Methods

2.1 Materials

Three panels were made for each cure condition. The layers per panel were: an aluminium AA7075-T6 plate, an adhesive layer and a structural composite layer. The dimension of the AA7075-T6 aluminium plate was 400 mm × 160 mm × 2 mm and contained a 6 mm size rivet hole with a wire-cut notch of 1 mm length perpendicular to the length of the plate emanating from the hole's edge. Figure 1 illustrates the geometry of the plate and composite repair patch. Cytec/Solvay FM73 (Cytec Engineered Materials 2011) knitted epoxy film of 0.25 mm thick was used as an adhesive between the aluminium and the boron/epoxy patch. The structural composite layer was composed of six plies, 5.2-mil thick, of type B4/5521 unidirectional 4-mil Boron/epoxy prepreg tape from Specialty Materials, Inc. (Boron 5521 Prepreg Tape

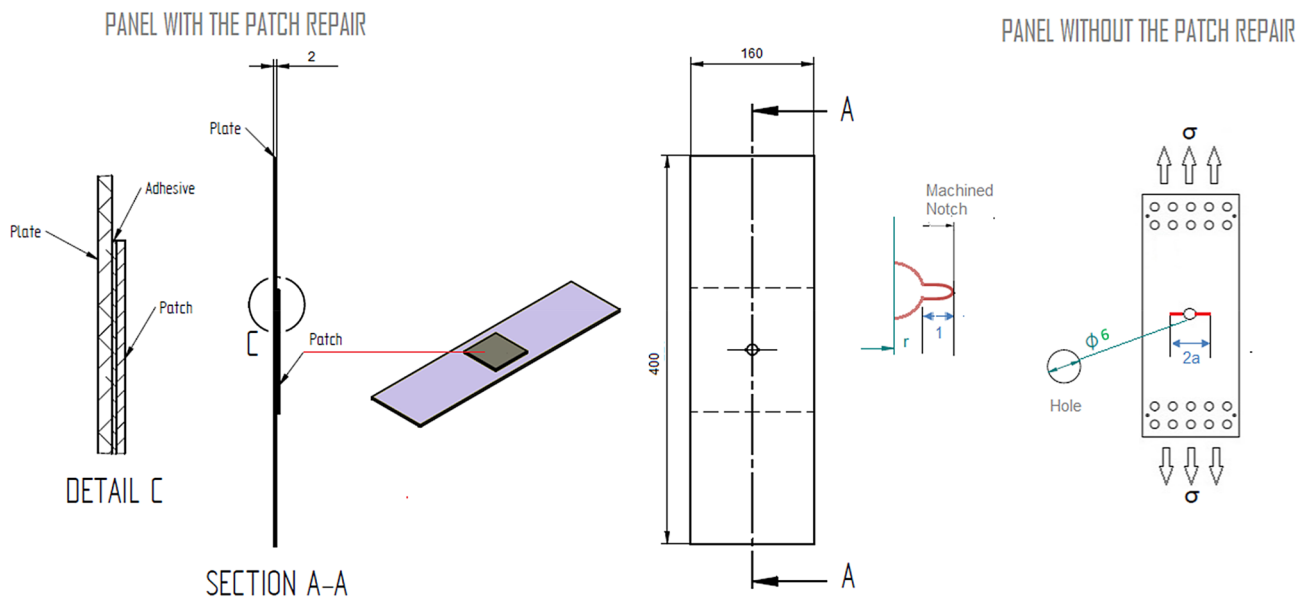


Fig. 1 Geometry of the plate and composite repair patch

2013). The materials properties from the supplier materials datasheet utilised in this study for the bonded repair of structures are summarised in Table 1. The chemical composition (in weight %) given for the AA7075-T6 sheet by the manufacturer was 5.6%Zn, 2.6%Mg, 1.4%Cu, 0.19%Cr, 0.13%Fe, 0.07%Si, and aluminium the balance.

2.2 Surface Treatment

An important factor influencing the strength of the bonded joint is the surface preparation of the metal. Many studies have been done on the influence of surface treatment on the mechanical behaviour of bonded assemblies (Kinloch 1987; Adams and Wake 1984; Emami Mehr et al. 2019). To prepare the bonded area, the metal surface was subjected to mechanical abrasion (grit blasting) with uniform alumina grit. The process employed 210 μm alumina grit at a pressure of 300 kPa and a working distance of 10–15 cm to obtain the required surface roughness without inducing compressive stresses. The surface was then cleaned with trichloroethylene to remove impurities. Figure 2 shows the bare prepared aluminium panel.

2.3 Composite Panel Manufacturing

The lay-up is shown schematically in Fig. 3 and it shows the sequence that was employed. The AA7075-T6 aluminium was the base plate; it was bonded with the FM73 adhesive to six layers, Type B4/5521 unidirectional prepreg tape. In the case when the crack position is perpendicular to the loading, the best stacking layup is with the fibres in the unidirectional

orientation, as the fibres will inhibit the opening of the notch in fatigue (Zouambi et al. (2019)). An orientation of the fibres for the composites layups in the perpendicular direction to the crack direction is used to obtain optimum results, as also noted by other authors (Pastor et al. 2008; Hosseini-Toudeshky et al. 2007). Plies were all stacked with the unidirectional orientation running perpendicular to the crack and parallel to the fatigue loading direction, the rivet hole being at the centre of the patch. The following conventional layers were applied over the prepreg tape: peel ply, release film, breather fabric cloth, and then the vacuum bag. The vacuum was sealed off at the edge of the vacuum bag with Cytec sealant tape. The full panel with the layup was put in an air convection furnace for curing while maintaining a vacuum of 80 mbar drawn over the layup. An overview of the final samples is shown in Fig. 4 and the steps to the final patched test panel. The patch and adhesive had dimensions of 100 mm \times 100 mm \times 1.03 mm (thickness including six prepreg tape layers and the adhesive layer).

2.4 Curing Parameters

The tests were performed with four types of curing conditions. The first (A) was subjected to a curing cycle as per the manufacturer's datasheet—heating up in 30 min and held at 120 °C for 60 min (Cytec Engineered Materials 2011). The second (B) was subjected to a combination of low temperature of 80 °C over a long cycle time of 480 min and post cured at 120 °C for 60 min as the recommended cycle as per the datasheet to ensure adhesion at elevated temperatures. A reduction in the cure temperature may affect important

Table 1 Mechanical properties of different repair materials

Properties	Materials		
	AA7075-T6	Boron patch	Adhesive (FM73K)
Longitudinal young modulus (GPa)	68.5	200	2.778
Transverse young modulus (GPa)	68.3	18	2.778
Poisson ratio	0.33	0.23	0.38
Hardness (HV20)	186	–	–
Elongation at break (%)	12	–	–
Shear modulus (GPa)	–	5.59	1.006
Thickness (mm)	2	0.13/layer	0.25

Fig. 2 Surface preparation by grit blasting



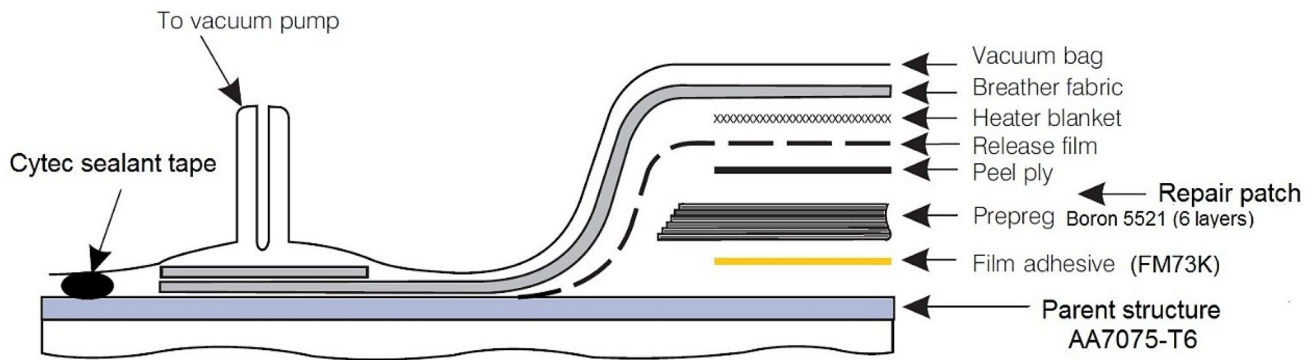
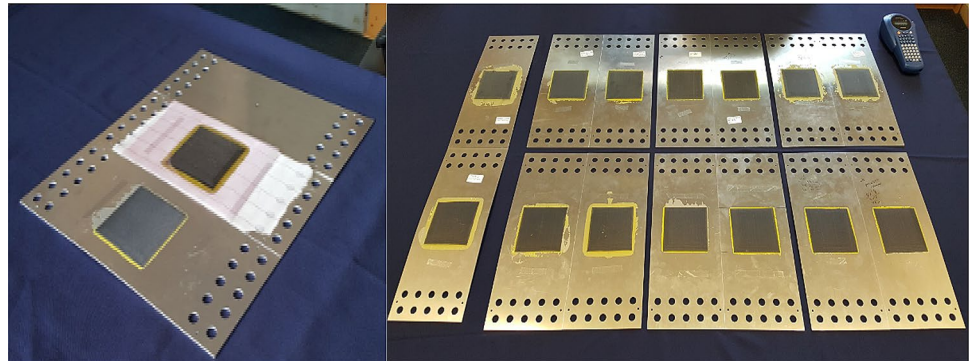


Fig. 3 Detail of vacuum bag lay-up

Fig. 4 The steps to the final patched test panel



properties of the bonded repair integrity. Other researchers (Baker and Jones 1988) favoured a lower curing temperature because of the differences in coefficients of thermal expansion between the parent and the bond material. The third (C) was subjected to a lower temperature of 80 °C, held for a longer time of 480 min. At elevated temperatures, the curing temperature influences residual stress build-up and shape distortion. Long curing cycles at elevated temperatures minimise residual stress levels during moulding (Kim and Hahn 1989; Weitsman 1980). This is due to the viscoelastic nature of thermosets at elevated temperatures (especially in the rubbery state) where large stress relaxation can occur (Montgomery and MacKnight 2005). Therefore, the fourth (D) was subjected to a prolonged cure cycle, heating to 120 °C for 105 min, to evaluate the prolonged curing time.

The full plate with patch was heated in a convection furnace. The ramp rate to all the curing temperatures was 2 °C/min. Plates were taken out of the furnace and left to cool at ambient conditions after curing. To ensure repeatability, the curing process was controlled by a computer while the temperature of the workpiece (composite panel) was continuously monitored by a thermocouple fixed to the aluminium base plate. The four curing conditions that were used are summarised as follows:

- A. 120 °C for 60 min (as per manufacturer specification).
- B. 80 °C for 480 min with post-curing ramped to reach 120 °C in 60 min.
- C. 80 °C for 480 min.
- D. 120 °C for 105 min (60 + 45).

2.5 Distortion Measurement

Distortion measurements were done accurately with metrology equipment controlled by computer at NMISA (National Metrology Institute of South Africa, (www.nmisa.org)). Distortion was measured by a laser metrology scanning technique using a laser scanner. The distortion measurement was through scanning the length of the plate (400 mm) and took place over 83 intervals. Composite plates were placed convex-side down with the bare aluminium surface facing up for measurement.

2.6 Fatigue Testing

Fatigue crack growth tests of repaired and unrepaired structures were performed on an Instron 1342-H1314 servo-hydraulic system with a dynamic capacity of 25 kN following the ASTM E647 standard (ASTM International 2013). Samples dimensions were chosen to closely match dimensions where the width of the panels is wider to comply with the spacing in stiffened structures of aeroplane skin. The test panel setup is illustrated in Fig. 5. All fatigue tests were conducted at cyclic constant-load amplitude, frequency of 10 Hz and stress ratio of 0.1. Crack length data and number of cycles were logged on a computer. Panels were fatigued until crack lengths approached the edge of the patch.



Fig. 5 AA7075-T6 Aluminium Panel fitted with COD “Clip-Gauge”

3 Results and Discussion

3.1 General Observations

All plates showed distortion after curing. Distortion culminated in warping along the length of the plate. Figure 6 shows an example of a warped plate and it can be seen that the warp curvature is towards the aluminium’s side.

3.2 Plate Distortion

An inverted plot of the profile from the data for curing condition A, as an example, is shown in Fig. 7. It can be seen from Fig. 7 that the profile is symmetrically around the apex, which is the centre of the patch by way of the lay-up geometry. The distortion is significant but it should be noted that the distortion between the apex and any baseline is exaggerated due to the aspect ratio scaling between the physical plate and the graph. The aluminium ligaments up to both sides of the patch baseline are straight as expected, because they were unconstrained during cooling and their contraction must be linear. The two linear-fit trend lines reinforce this fact. The distortion of the plate is given as the distance from the ligament baseline to the apex. Table 2 represents the measured distortion of bonded repaired samples in different curing conditions.

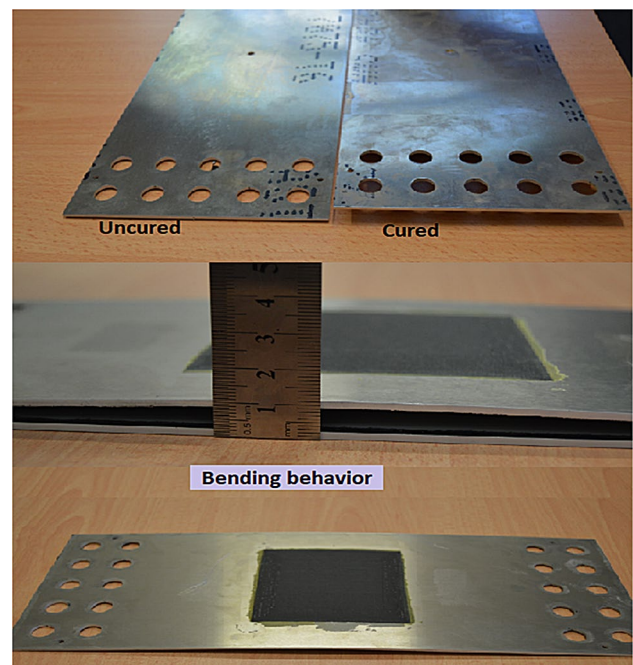
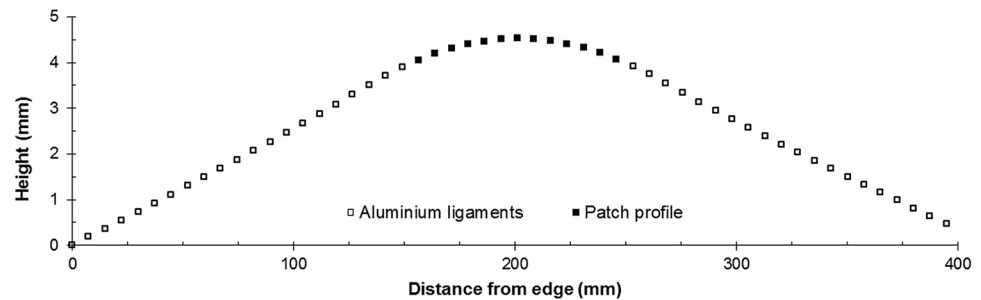


Fig. 6 Curvature of panels results in bending due to the curing process

Fig. 7 Bending profile from the data for curing condition A**Table 2** Distortion variation of bonded repaired samples in different curing conditions

Curing condition	Cure label	Distortion	
		Plate (mm)	Patch (μm)
120 °C/60 min	A	4.912	617.75
80 °C/480 min // 120 °C/60 min	B	4.129	484.25
80 °C/480 min	C	3.735	374.75
120 °C/60+45 min	D	4.923	625.50

The observed distortion can be explained by the following reasoning. Before the composite patch is bonded to the component to be repaired, each part, plate and patch, is stress-free. The adhesive and the boron/epoxy prepreg tape do not cure at room temperature when it is installed on the plate. They also do not cure during the heating stages, meaning that there is no constraint up to the curing temperature; but during the curing process of the bonding, each part must be heated to an elevated cure temperature in the furnace; all the materials expand according to their individual coefficient of thermal expansion (CTE). The mismatch in CTE results in the plate being in tension and the patch being in compression after cooling down to ambient conditions. The aluminium is constrained by the stiffer patch that results in the composite area bending on cooling towards the material with the highest coefficient of thermal expansion, i.e. the aluminium. The contraction is related to the expansion of the materials according to the cure temperature range chosen. As seen cure condition A and D have higher distortion levels with the use of higher curing temperatures (120 °C) compared to B and C (80 °C). The variation in distortion levels in relation to the temperature and time of the different cure cycles can be seen in Table 2. We can conclude that the effect of different coefficient thermal expansion on the distortion levels is more pronounced as a change in temperature ΔT increases during curing or in the cooldown. Zhu et al. (2013) also concluded that curing-induced distortion in adhesive bonding of AA6061 and steel was due to differences in CTE of the two materials, as well as curing temperature. They suggested that differences in CTE intrinsically induced the distortion,

while the high curing temperature extrinsically generated the distortion. Finally, the cured adhesive then captured the distortion. It was found that the peak curing temperature was the main contributor to the distortion.

3.3 Fatigue Life

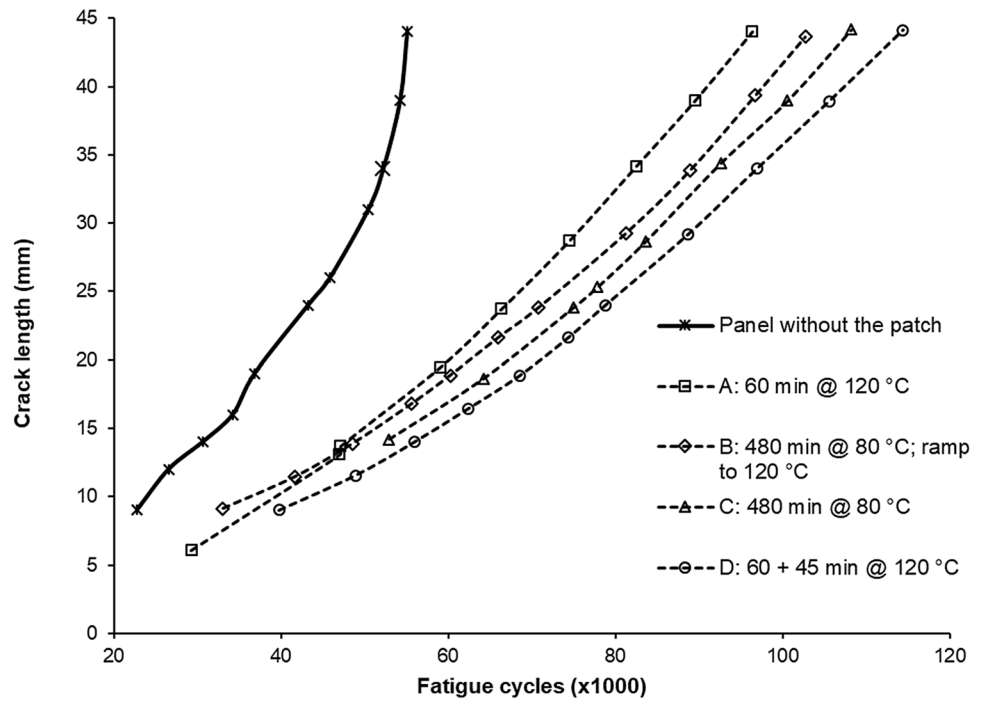
As discussed, during the curing process, adhesively bonded composite/metal laminate structures that are held at elevated temperatures for material processing, develop high residual stresses because of the difference in coefficients of thermal expansion (CTE) for different materials. This thermal mismatch can result in delamination or debonding of materials, which facilitates fatigue crack growth at the polymer/metal interface. Further, the influence of different curing temperatures and times investigation was carried out on the fatigue life of the repaired structure. Fatigue crack growth tests were performed on unrepaired and repaired panels.

The panel is symmetrical along its length with respect to the hole and notch. The crack length starts at 4 mm, which is the 3 mm radius of the hole plus 1 mm for the machined notch. The crack length measurement is calibrated at the start of each test and measurement data were automatically logged. The fatigue propagation curves terminate at 44 mm which was chosen and defines fatigue life, in the case of the repaired panels to prevent the crack from growing past the edge of the half-patch at 50 mm. The fatigue crack growth curves for unrepaired and repaired panels are presented in Fig. 8.

Results from fatigue tests to determine the effect of different cure parameters on the repair efficiency and fatigue life of the structure showed that the fatigue life of the unrepaired structure according is decreased compared to the repaired structure. The general trend in Fig. 8 is uneventful with a gradual stabilisation of the crack growth rate. The final order of fatigue life cycles for cure condition is $A < B < C < D$.

The repairing of the cracked plate using bonded composite patches is effective in slowing down the crack growth rate and thereby enhancing the fatigue life. The panels with the bonded composite repair show an increase in the number of cycles in an average of 70% when compared to the unrepaired panel, which improves the fatigue life considerably.

Fig. 8 Effect of curing condition on crack propagation in the repaired panels



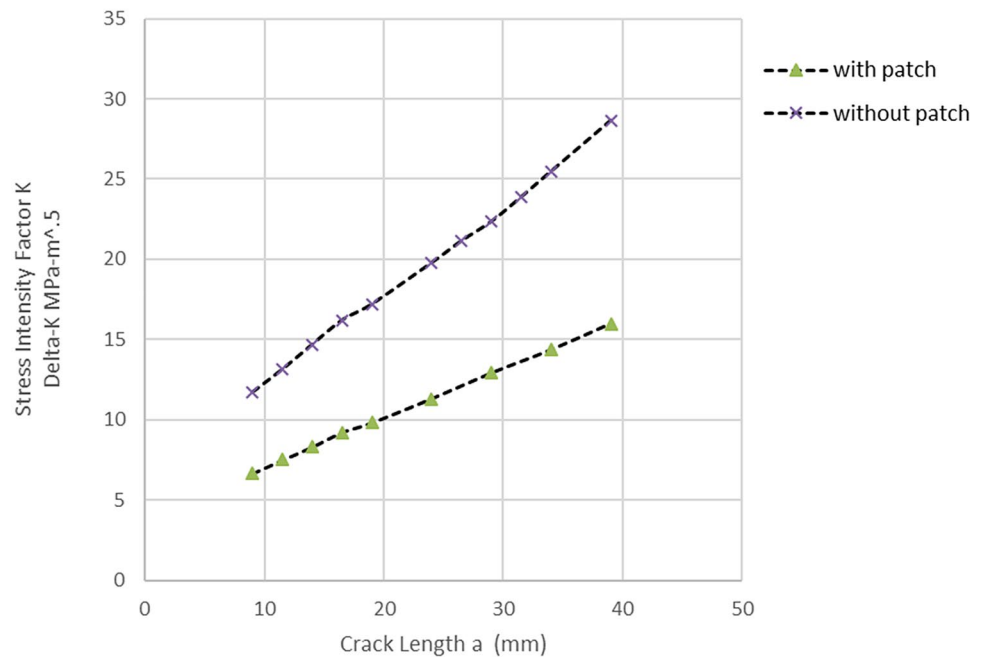
The use of a composite patch offers advantages for repaired cracks to increase the service life as shown in Fig. 8.

The increase in the fatigue life performance from unrepaired to repaired is due to the decrease in stress intensity factor due to the boron/epoxy patch bridging the crack. This is confirmed in Fig. 9 representing the reduction in

stress intensity factor (SIF) between panels with and without patch. Figure 9 shows that the stress intensity factor increases as the crack length increases both in panels with and without patch, this due to higher stress concentration at the crack tip. The stress intensity factor was determined experimentally during the fatigue test from the data by

Fig. 9 Reduction in stress intensity factor with repair (Curing condition A)

Reduction in Stress Intensity Factor with Repair



using Eq. (1) as per ASTM E647 (ASTM International 2013):

$$\Delta k = \frac{\Delta P}{B} \sqrt{\frac{\pi a}{2W} \sec \frac{\pi a}{2}} \quad (1)$$

where P is the load in MPa, B and W are the thickness and width of the sample in mm, respectively, a is the crack length; and $\alpha = 2a/W$.

3.4 Fatigue–Distortion Relationship

Figure 10 shows the relationship between plate and patch distortion and fatigue life. It is expected that increasing distortion would reduce fatigue life, presumably due to induced bending stresses that could contribute to the crack growth. It can be seen that fatigue life decreases as the plate distortion increases, that is, for cure conditions A, B, and C. The exception is cure condition D, which has the largest plate distortions, but the longest fatigue life.

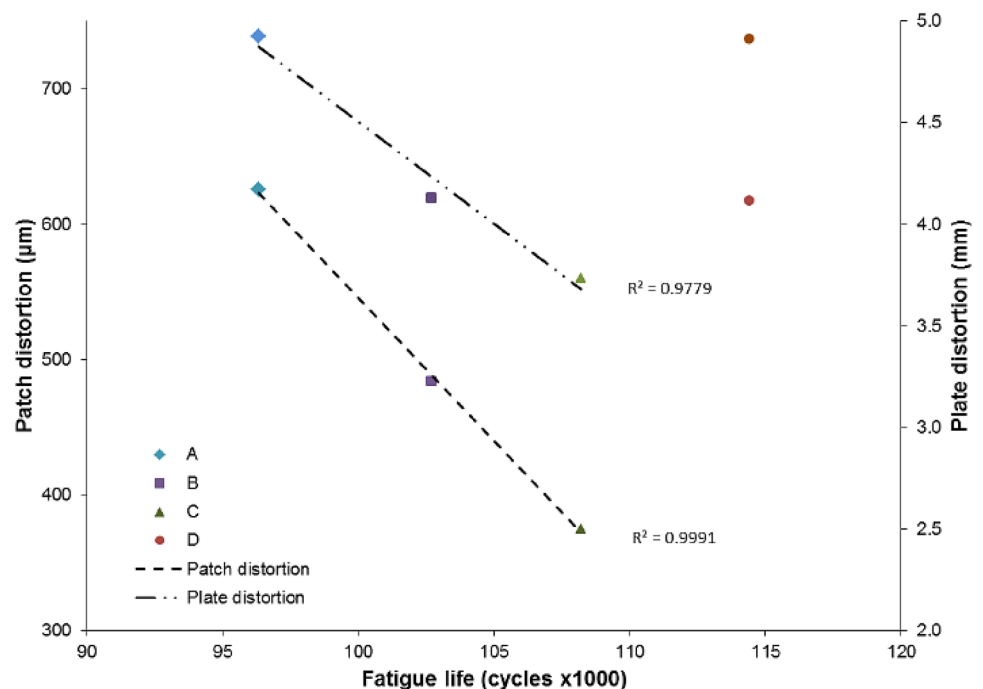
The fatigue life follows a straight-line relationship with the plate distortion (for cure conditions A, B and C), and the curve fitting coefficient has an $R^2 = 0.9779$, which indicates a strong correlation. The fatigue life in the case of the patch distortion also follows a linear relationship with an $R^2 = 0.9991$, which indicates even a stronger relationship. Cure condition D does not follow the straight-line relationship in Fig. 10, and the possible reason for its unique behaviour is the use of the elevated curing temperatures for

a longer time. This enhanced the adhesion between the metal surface and the composite patch and overall durability.

3.5 Failure Mechanism

Adhesive bonding can be described as a procedure of joining parts with a non-metallic material (adhesive) which go through a hardening reaction causing the parts to bond with surface adherence (adhesion) and internal strength (cohesion). Under overstress loads, the interfaces between different adhering materials can suffer adhesive failure. Ideally, the joining surfaces should be clean and generally require proper preparation of the aluminium surface. It is recommended by the adhesive supplier that the adhesive be cured at $120 \text{ }^\circ\text{C} \pm 3 \text{ }^\circ\text{C}$. A curing process at $80 \text{ }^\circ\text{C}$ would probably result in an insufficiently cured adhesive layer. There are also questions related to the interface properties of the adhesive and prepreg tape epoxy. It could be expected that the interface does not reach its optimum properties if the curing process deviates from the recommendation, especially the cure temperature. However, curing time is also the other important factor. The longer time curing of lower temperatures as in curing process (C) and (B) gave longer fatigue life than condition (A). The use of cure condition (A) parameters for an extended curing time as in cure condition (D) also showed the effect of curing time on the fatigue life. In addition, the metal surface treatment could have influenced the effects observed, as the surface preparation of the metal is an important factor influencing the strength of the bonded joint. Different types of adhesives, pre-treatment

Fig. 10 The relationships between fatigue life and plate distortion



methods and processing techniques are available, but there is still a need for further improvements (Duncan et al. 2003; Broughton and Lodeiro 2002). The best solution involves close consideration of multiple criteria and seeking advice from the respective suppliers is recommended. In some cases, extensive experimental testing may be required.

During the fatigue test of the patched plates, the crack growth exceeds the patch edge. The crack continues growing towards the end of the aluminium plate as shown in Fig. 11; in other words, the crack had propagated across the entire width of the specimen without any sign of detachment of the patches from the plate after long time cycling. While kept under continuous cycling loading

after the crack passed the edge of the plate, a debonding line between the patch and plate was observed as seen in Fig. 12. The fatigue test was first completed by growing the crack inside the patch zone, and then the panels were subsequently pulled apart. Specimens, namely unpatched and patched with boron composite patch in cure condition (A, B, C and D) were loaded to failure in tension in order to observe the failure mechanism of the damaged repair. For the purpose of comparison, it was done at the same crack length growth (70 mm) in order to analyse the effect of the strength of the assembly on the repair efficiency. Results observed explains that the strength of the assembly is highly increased with the composite patch

Fig. 11 Crack growth behaviour in the repaired plate

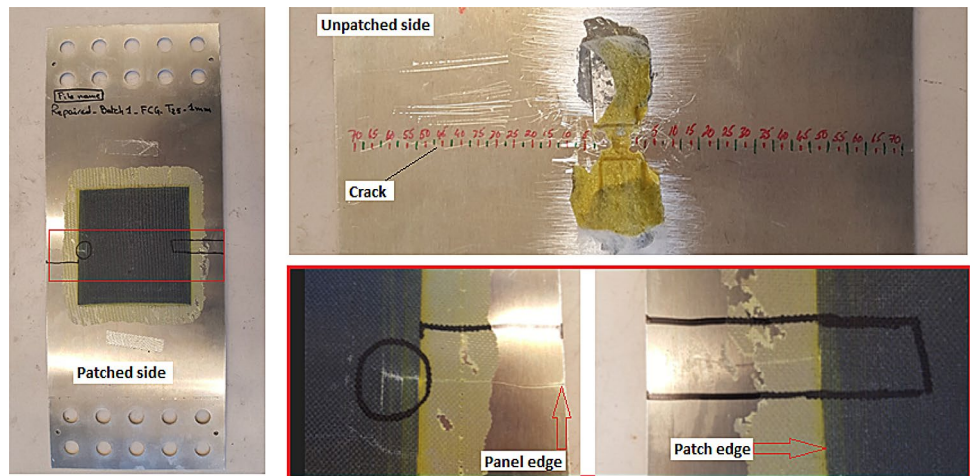
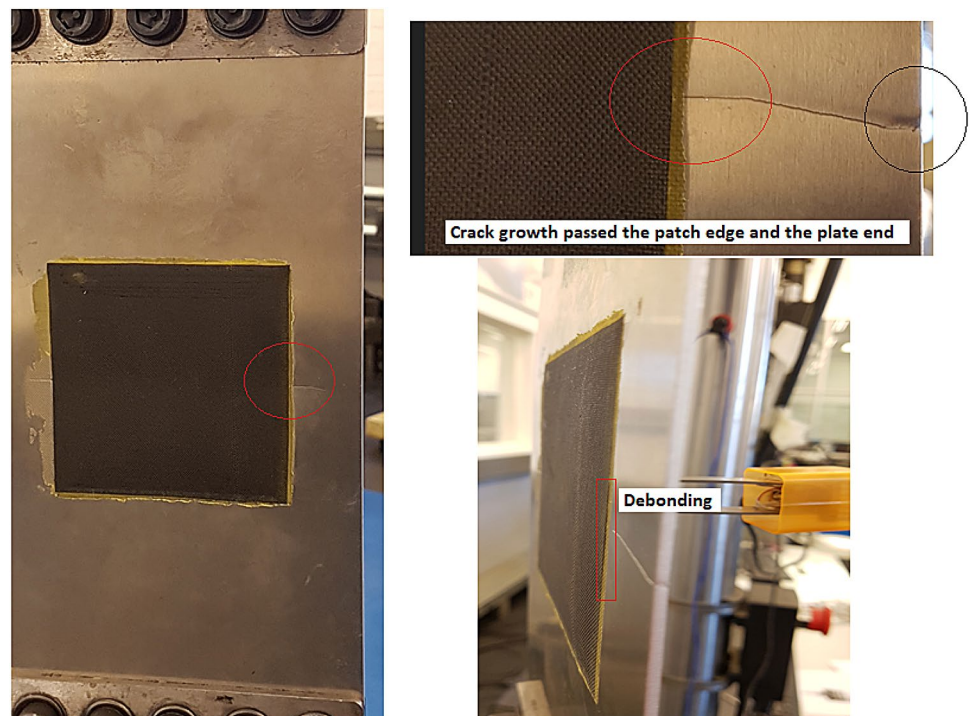


Fig. 12 Effect of patch strength after plate failure



presence, which confirms the results in the improvement of the fatigue life with the bonded composite repair.

The observed debonding in Fig. 12 creates unbound interfaces. Such a defect can give rise to preferential sites at the initiation of rupture or detachment. Similarly, when the stress supports are not on the same plane a bending moment appears, thus creating in the joint area and the supports stress normal to the bonding, which is superimposed on the shear stresses. If the rigidity of the supports is low, the latter will flex under the action of bending moment, which increases the normal stresses at the ends of the bond line and risks endangering the bonded area by initiating the separation. Stresses in the adhesive bond layer are increased due to the mismatch between the patch and plate that can lead to possible peel and cleavage stresses. Tension on the repair system creates shear stress in the bond line. The composite fibre breakage in a sliding failure along a plane that is parallel to the loading direction can be seen in Fig. 13.

Figure 13 shows the fractured panels after the tension test. At first, it can be seen that debonding between the aluminium base plate and the adhesive happened in cases for cure condition A, B, and C. The same type of debonding was not evident for cure condition D. Furthermore, it can be seen from the area around the fatigue crack that the yellow adhesive is visible underneath the boron/epoxy layer, meaning that there also was debonding at the adhesive-boron/epoxy interface. The increased fatigue life of cure condition D is

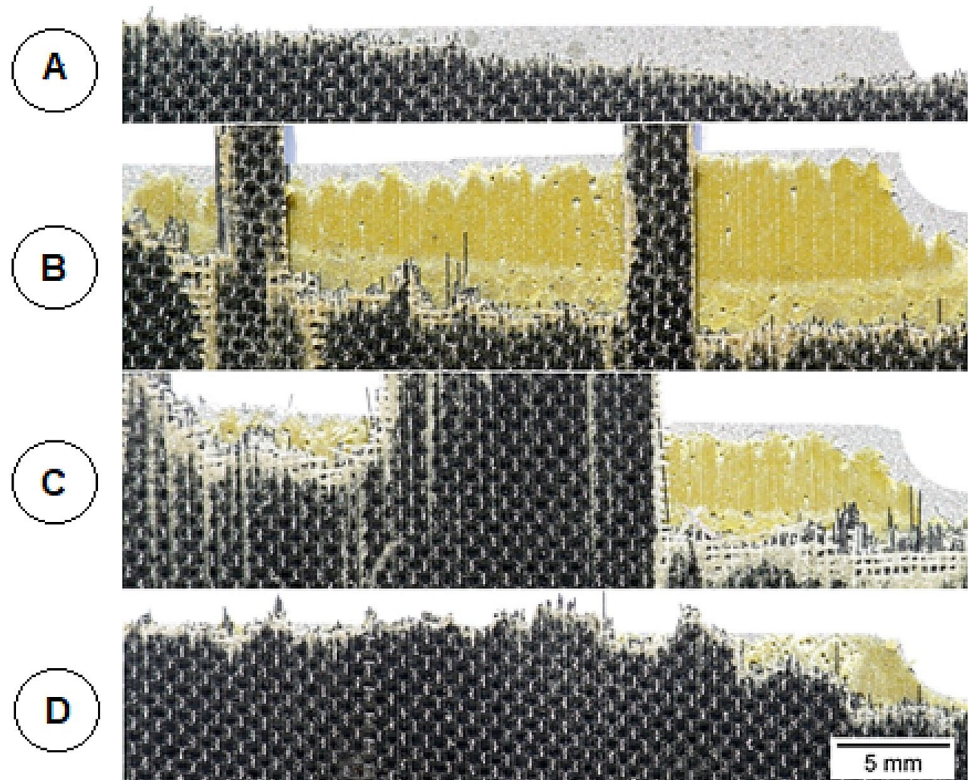
a good example that infers a proper interface bond between the adhesive layer and the boron/epoxy layer. That is because of the improved bonding enhanced by the elevated temperature. It has a high degree of distortion, but it also has the best fatigue life (Fig. 8). However, excessive distortion may not be suitable for aero-structure skins due to aerodynamics purposes for in-service conditions. The bending behaviour might also influence the fatigue life of the structure in a cyclic mode.

4 Conclusions and Recommendations

This paper presents an experimental study on the fatigue patching efficiency for notched AA7075-T6 aluminium alloy panels reinforced with single-sided boron/epoxy patches. The novelty of this paper is that the patches are adhesively bonded to the panels with four different chosen cure regimes. To enhance the repair bonding durability and reliability, curing conditions and processes of bonded composite patches repair with structural adhesive were investigated. The findings are summarised as follows:

- The repair of the cracked plate by means of bonded composite patches is effective in decreasing the crack growth rate and thereby enhancing the fatigue life of the structure.

Fig. 13 Fractured samples in shear mode after tension test



- The various cure parameters affect the repair efficiency and fatigue life of the structure. The order of fatigue life cycles for cure conditions is $A < B < C < D$.
- The fatigue life with a composite patch is greater than without a patch.
- Bonded patch repair reduces stresses near the tip of the crack by bridging between the cracked plate and composite patch to slow down or stop the crack growth.
- An average of 70% decrease in the crack growth rate from unpatched panel to the patched one was found
- The increased fatigue life of cure condition D is a good example of the enhanced adhesion between the metal surface and the composite patch due to the use of elevated temperature and duration.
- The curing condition parameters affect the structural distortion and secondary bending influences the behaviour in single side reinforcement/repair.
- The fatigue life decreases as the plate distortion increases, that is, for cure conditions A, B, and C. The exception is cure condition D, which has the largest plate distortions, but the longest fatigue life.
- The optimum combination of temperature and duration is of importance to reduce the distortion.
- The selection of the best cure parameters involves close consideration of multiple criteria.
- There are opportunities to optimise the stress distributions within repairs using finite element packages. Validation of these experimental data is required and further development to ensure that they can predict the effect of various parameters accurately. The ability to accurately calculate the magnitude of thermally generated residual stresses would be an advantage for certain repairs.
- The structure's curvature in the repair area would require detailed investigation if applied to fleet aircraft.

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5 6. Recommendations

Behaviour prediction of the fatigue crack growth in metallic structures is a highly involved and complex process. When these structures are repaired with bonded composite patches, the complexity is (summy text).

- The behaviour prediction of the fatigue crack growth in metallic structures is a highly involved and complex process. When these structures are repaired with bonded composite patches, the complexity is further compounded. Despite this complexity, there is a need to establish a generalised approach, of sufficient accuracy, to model the crack growth rates after repair
- Thermomechanical analysis (TMA) and dynamic mechanical analysis (DMA) are required to understand more in-depth the thermal and mechanical behaviour of the involved material.
- For aircraft damage repair with composites patches and aircraft composites components, it is also important to perform those analyses starting from low temperatures ranges ($< 18^\circ\text{C}$). That will conform to the in-service aircrafts temperatures and environment to predict the failures before maintenance to detect defects or damage during inspection.

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