Repair of Ballistically Impacted Carbon Fibre Reinforced (CFR) Laminates

by J.B. YOUNG and N. MATTHEWS

College of Aeronautics
Cranfield Institute of Technology
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5 SEP. 1983
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ISBN 0 902937 84 7
£7.50

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ABSTRACT

This report shows that thin monolithic Cabron Fibre Reinforced Panels impacted with small arms projectiles can be adhesively repaired restoring the damaged panel tensile strength to between 85% and 90% of the original. The compressive buckling strength of the thin CFC panels was found not to be significantly affected by ballistic impacting. The repair is primarily concerned with single-sided repairs utilizing a wet laminating repair patch technique and a precured patch technique. The repair research showed that for the particular impact damage sustained by the CFC panels there existed optimum repair patch physical parameters (i.e. overlap length and ply composition) for strength restorations above 85%. The research highlighted a number of repair techniques which adversely affected strength restorations. These techniques involved the removal of the damage area, use of precured square patches and the use of narrow repair patches.

In addition to the damage panel repair analysis the report also details the construction and the testing of the laminates used in the research and attempts to present theoretical repair analysis which complements the actual repaired panel characteristics.
NOTATIONS AND UNITS

The notations which are frequently used in the text of the report are defined below; others are defined locally.

a  Panel width (m), Notch length (m),
b  Panel length (m)
C  Boundary condition constant, Stiffness parameter
d  Crack length (m)
D  Bending stiffness component (Nm)
E  Elastic modulus (GN/m²)
G  Shear modulus (GN/m²)
k  Initial buckling constant, Moment coefficient, Stiffness reduction factor
K  Stress concentration factor (MN/m³/²)
M  Moment (Nm)
N  Load/unit width (KN/m)
P  Direct load (KN)
Q  Composite stiffness modulus (GN/m²)
Γp  Damage zone length (m)
t  Thickness (m)
V  Volume fraction
W  Weight (kg)

Greek Alphabet

ρ  Density (g/cm³), Radius of curvature (m)
ν  Poisson's Ratio
Δc  Critical stress intensity factor (NM/m³/²)
Σ  Laminate stiffness parameter
τ  Adhesive thickness (t)
σ  Stress (MN/m²)
INTRODUCTION

With the continual rapid increase in aircraft technology, the introduction and rapid development of new materials has occurred. In particular, the use of composite materials has become increasingly popular because they have shown to be ideal where high strength to weight ratios are required. In spite of the advantages of composites it is feared by many experts that the technology and use of composites has advanced too quickly. Unlike metals, which have evolved during many decades of application, composite materials are relatively new and there appears to be few limitations on their use. However their reaction to the many and varied in-service problems have yet to be fully evaluated. These in-service problems include impact resistance, environmental resistance, repairability etc. Of particular interest to the authors was the repairability of the high strength/stiffness composites used in military aircraft.
Because of the limited time available to conduct meaningful research (approx. 4 months) into the relatively large and complex field associated with composites repairs, research was limited to repair of thin monolithic Carbon Fibre Composites (CFC) panels impacted with small arms projectiles. The repairs if possible were to restore the impacted panel strength to a usable degree (e.g. 85% to 90%) and as such could be deemed to be permanent.

COMPOSITION AND CONSTRUCTION OF TEST LAMINATES

A. Material Used

The carbon fibre material used in the construction of the laminates was a Carrfibre high strength bi-directional five-shaft satin woven cloth. This type of material is being increasingly utilized in aircraft applications because of its desirable characteristics i.e. drapability, stable articles of commerce, better production characteristics etc.

The matrix resin system used for the construction was a two-part liquid epoxy system - Ciba-Geigy XD927. This resin system in addition to the advantages of epoxy systems, namely toughness and good adherence to fibres, exhibited low viscosity which enhanced thorough impregnation and cold curing capability.

B. Production of Laminates

The laminates used for the research were constructed at RAE Farnborough using a vacuum moulding technique. Details of the apparatus and its use are contained in [1]. Laminates of eight plies were selected for construction as it was deemed to represent a typical thickness of thin laminates found in aircraft structures (i.e. skins). Fibre volume ratios of approximately 52% were obtained which are consistent with vacuum moulding technique [2].

MECHANICAL PROPERTIES OF THE LAMINATE AND SELECTED TEST PANELS

The testing and inspection program employed during the research to obtain the various mechanical properties, basically involved:

1) Initial testing on samples of the various laminates to determine respective laminate properties, and

2) Testing and inspection of the selected test panels to determine properties in addition to those obtained for the laminate (i.e. panel buckling and to assess the effect of ballistic impact and the effectiveness of repair).
A. Laminate Properties

The basic mechanical properties of the laminate (i.e. tensile modulus, tensile strength, shear strength, shear modulus, compressive strength etc.) were determined by conducting tests in accordance with recognized procedures. Details of these properties are contained in Table 1.

<table>
<thead>
<tr>
<th>Panel No.</th>
<th>Tensile Strength (MN/m²)</th>
<th>Tensile Modulus (Gln/m²)</th>
<th>Compressive Strength (MN/m²)</th>
<th>Shear Strength (MN/m²)</th>
<th>Shear Modulus (GN/m²)</th>
<th>Poisson's Ratio</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 Warp</td>
<td>660</td>
<td>64</td>
<td>352</td>
<td>97</td>
<td>4.0</td>
<td>.047</td>
</tr>
<tr>
<td>Weft</td>
<td>553</td>
<td>60</td>
<td>352</td>
<td>97</td>
<td>4.0</td>
<td>.047</td>
</tr>
<tr>
<td>2 Warp</td>
<td>682</td>
<td>60</td>
<td>328</td>
<td>91</td>
<td>3.9</td>
<td>.047</td>
</tr>
<tr>
<td>3 Warp</td>
<td>660</td>
<td>64</td>
<td>348</td>
<td>105</td>
<td>4.0</td>
<td>.047</td>
</tr>
</tbody>
</table>

TABLE 1 - LAMINATE MECHANICAL PROPERTIES

B. Test Panel Laminate Properties

The size of the test panels originally chosen for the research was 250 mm long by 125 mm wide. These dimensions were selected to provide an adequate area to effect repairs after impact. Because of the limited width of available mechanical/hydraulic jaws (i.e. 80 mm max.) various methods of alternative testing were employed (Figure 1 refers). These methods proved unsuccessful due to high induced peel stresses at the tab ends and the panel width had to be reduced to 80 mm. This reduction in width was initially cause for concern as it was considered that panel edge effects may introduce prohibitive stresses into the repair area. Accordingly, to determine the extent of the edge effects, a panel was strain gauged with the results showing that the edge effects extended into the panel by a distance equivalent to the panel thickness. This result is supported by [3].

The originally selected width presented no problems in compression testing of the panels. The panel was tested to destruction with compressive buckling failure occurring well below ultimate laminate compressive strength (i.e. approx. 40%).
BALLISTIC IMPACTING

All the ballistic damage was achieved with .3 calibre projectiles (i.e. standard NATO 7.62 mm small arm projectiles). Although this type of projectile damage would relate to only a small percentage of probable ballistic damage encountered by modern military aircraft, the .3 calibre projectile was selected because:

1) The limited panel width meant that ballistic damage had to be small compared with the width if effective repairs were to be incorporated.

2) The subject cartridges and range facilities were more readily available.

3) Most of the research available on ballistic impacting of fibre panels related to .3 calibre impacting which would enable comparisons to be made between this research and other work.

4) It was anticipated that the results obtained for the .3 calibre impact damage and subsequent repair could be directly applied to larger more realistic panels (i.e. those comparable with aircraft panels) with larger ballistic damage (i.e. damaged caused by 23 mm projectiles).
The ballistic impacting was accomplished by employing two methods of projectile impact. The first method of impacting which was used for the majority of the research was where the projectile was fired directly at the panel with the projectile impacting normally at the centre. The second method involved impacting of the panel with the projectile tumbling.

For both the normal impact and tumbled impact, the test panel was clamped at each end and the firing range was limited to 15 m to ensure that the unstable flight regime of the projectile was not reached (Figure 2). Tumbling of the projectile was employed to assess the significance of the projectile impacting the surface at various angles of incidence.

A. Results of Impact Damage

The impacted panels were examined both visually and ultrasonically. The damage consisted of an area limited to the projectile cross-sectional area on the entry surface with a slightly larger square/rectangle damage area on the exit surface (Figure 3 refers). Ultrasonic inspection revealed that substrate damage was the same as revealed by visual inspection but slightly larger (i.e. for normal impact: for exit surface - 12 mm visual - 16 mm ultrasonically). Also the observation of little fibre feathering or pull out on the exit surface indicated that a reasonable fibre to matrix bond was present.

All normally impacted panels when tested in tension exhibited an overload failure across the projectile hole with a reduction of panel tensile strength of approximately 50%. The buckling and post-buckling behaviour was almost identical to that of an undamaged panel and accordingly for the remaining research only the tensile behaviour of the panels was addressed. Residual tensile and compressive buckling strengths for the panels impacted with tumbled projectiles was dependent on the amount of width related damage.

REPAIR OF BALLISTICALLY IMPACTED PANELS

As detailed in the introduction, the aim of the research was to restore the strength of the damaged panels to approximately 90% of the original strength.
(Additional Tumbling Apparatus)

Figure (2) - Ballistic Test Set-Up
Figure (3) - Projectile Damage
The repairs were accomplished by basically employing field repair techniques. In conducting the research the relevance to aircraft repairs was always prominent. Accordingly, the majority of repairs were restricted to one side. Also, because of the relatively small damage area external repairs were employed. For initial repair work the repairs were accomplished by using the same material as the base adherent, a cold curing epoxy resin and a wet lay-up technique. Once an optimum repair technique has been developed both analytically and experimentally, precured patches were employed.

A. Repair Techniques

Because patches were to be employed on only one side, the eccentricity of the patch to base adherent meant that bending would be induced in the patch and the base laminate resulting in high peel and shear stresses at both the damage edge and patch edge. It was because of the high peel and shear stresses that the first phase of the repair involved the selection of an adhesive which exhibited the best resistance to peel. Specimens were produced to ensure peel failures would occur when the joint was loaded in tension (Figure 4 refers). The choice of available adhesives which would enhance both impregnation and would cold cure was narrowed initial down to two epoxy systems Ciba-Geigy XD927 and Epikote 828. (Note the epoxy systems were selected in lieu of polyester systems because the epoxy systems enhanced better adhesion to the parent laminate). Subsequent peel tests revealed that the Epikote 828 joints were 60% stronger than the XD927 joints.

Damage area preparation was recognized as one of the most critical operations of the repair procedure. The area was prepared using a fine carborundum cloth followed by a rinse with Acetone. Wet laminated patches were layed-up with a gradual reduction in ply dimensions. This was done to produce a tapered patch effect to enhance peel stress dissipation. Precured patches were later employed in the research work primarily because they were easier to work with and they offered a lighter repair. However the initial use of precured patch was a square patch which significantly reduced repair effectiveness by approximately 25%. It was felt that this had occurred because of the combination of bending stresses both transverse and longitudinal at the patch corners. Removal of the corners (i.e. 15 mm along each edge) restored panel strength to that obtained by wet laminating patches (i.e. approx. 90%).
Figure (4) - Peel Test Lay-Up

Figure (5) - Repair Strength Vs Patch Parameters
B. Results of Repair Panels

Two trends were noted from the experimental results with the first relating to repair panel strength versus patch ply thickness. It was noted that after a nominal patch ply thickness was achieved additional plies did not assist in improving the repair strength (Figure 5 refers). The second trend related to repair panel strength versus patch overlap length. Once again when the nominal overlap length was reached further overlap resulted in little strength increase.

Other areas which became evident during the research were:-

1) Utilization of double side patches resulted in repaired panel strengths of 100% of undamaged strength. This was because bending stresses were alleviated.

2) Removal of damage area reduced panel strength by up to 30%.

THEORETICAL ANALYSIS

In conjunction with the experimental research analytical assessments were conducted. The assessments related to two main areas:-

1) Residual strength prediction of impacted panels.

2) Patch parameter prediction (i.e. ply thickness and overlap) for the repairs of impacted panels.

A. Residual Strength Predictions

Experimental data has shown that the application of fracture mechanics to fibre composites is extremely difficult because the failure of composites are dependent on the numerous variable properties of individual composites. These variables include the mechanical properties of the fibres and resin, ply orientation etc. Because of the difficulties in the definition of the exact failure mechanisms, many models have been proposed for residual strength prediction of damaged composites under tensile loading. These models can be basically divided into two main groups - those which have adopted the basic concepts of Stress Concentration Factors (SCF) and those which have adopted the basic concept of Linear Elastic Fracture Mechanics (LEFM). Numerous papers have been written on the various SCF and LEFM models and these papers adequately describe approaches undertaken in producing the models ([4] and [5] refers). However, it is worthwhile to note that all the models assume some stress distribution either by a blunting mechanism or by the forming of a damage zone. Accordingly to employ any of the models at least two parameters need to be known. These are the ultimate strength of the undamaged specimen
and the size or characteristic of the damage zone. Therefore it must be assumed that the available modes are empirical. Fortunately relevant data on size and characteristics of the damage zone for carbon-epoxy laminates was available which permitted the use of available strength prediction models.

The models selected for use were Nuismer and Whitney [11] average stress criteria SCF and LEFM models. These models provided relationships as follows:

\[ \sigma_c = \sigma_0 \left( 2(1-p_2)/(2-p_2^2-p_2^a+K_T-3)(p_2^a-p_2^a) \right) \]

\[ \text{SCF} \]  \[ \text{LEFM} \]  \[ \text{(1)} \]  \[ \text{(2)} \]

where \( K_T \) = stress concentration factor

\( p_2 = R/(R+a_\infty) \)

\( R = \text{radius of damage} \)

\( a_\infty = \text{material property independent of laminate construction and stress distribution} \)

Also

\[ \sigma_c = \sigma_0 (1+2a/a_\infty)^{\frac{1}{2}} \]

\[ \text{LEFM} \]  \[ \text{(2)} \]

Application of these models yielded predicted results within 5% of experimental results. (Table 2 refers).

B. Theoretical Repair Analysis

As detailed in the preceding text experimental results indicated that for a single sided external patch repair, there existed an optimum patch ply overlap and patch ply thickness.

The patch ply thickness is primarily a function of the amount of load being transferred through the patch. However the exact amount of the load through the patch and the load path is extremely difficult to define. The lack of definition can be attributed to the effect of the patch on the laminate stiffness around the damage. The available models, in the main, adopt a simplistic 2D approach which relates patch loading to the relative linear stiffness of the patch in the damage/repair area. (Figure 6 refers):

\[ \text{i.e. load through the patch} = \frac{\text{patch stiffness}}{\text{total stiffness}} \times \text{applied load} \]

\[ \text{(3)} \]
Although the linear stiffness of the patch can be readily defined once the patch lay-up is selected, the difficulty arises in defining the laminate stiffness in the damage area. This basically means that the laminate stiffness around the hole (i.e. 3D) is being defined via a 2D model ([7] and [8] refers). However utilizing this basic concept the following model for patch load prediction was developed.

\[
\frac{P_p}{P} = \frac{((E_{tp})_p)}{(E_{tp})_p + (E_{t})_L/K} \\
\]

where:
- \((E_{tp})_p\) is the patch linear stiffness
- \((E_{t})_L/K\) is the laminate linear stiffness around the hole
- \(K\) is the laminate stiffness reduction factor and is the ratio of the ultimate panel load over the net panel load.
- \(P\) is the applied load.

Having determined the patch thickness, the optimum overlap was obtained by adopting the fundamental analysis of a single lap joint contained in [6j. This analysis, in relationship to peel failure predicts that the maximum peel stress is related to the average laminate stress outside the joint by:

\[
(k/(2f((3E'c_t)/E_n))^{1/2} \\
\]

where:
- \(k\) is the moment co-efficient of the load path = \(2M_o/P_t\)
- \(E'c\) is the adhesive peel modulus
- \(t_1\) is the adherent thickness
- \(t\) is the laminate thickness
- \(n\) is the adhesive thickness

As can be seen from the inclusion of the moment co-efficient in the equation (5) the maximum stresses within and outside the patch area are greatly influenced by the value of the bending moment, \(M_o\), induced just outside the overlap, by the eccentricity in the load path. This induced moment is a function of the applied load, \(P\), through the adherent and the adherent stiffness can be expressed by the following relationship:

\[
M_o = Pt/(2(1+\Sigma C+\Sigma C^2)/6) \\
\]

where:
- \(\Sigma^2\) is the laminate stiffness parameter and is equal to \(P/D\)
- \(C\) is half the patch overlap from the centre of the damage
- \(P\) is the load/width
Figure (6) - Patch Repair Analysis

Figure (7) - Predicted and Actual Patch Overlap

<table>
<thead>
<tr>
<th>Panel No.</th>
<th>Type of Impact</th>
<th>Predicted % of Ult.</th>
<th>Actual % of Ult.</th>
</tr>
</thead>
<tbody>
<tr>
<td>1, 2 &amp; 3</td>
<td>Normal</td>
<td>49</td>
<td>48</td>
</tr>
<tr>
<td>2E</td>
<td>Tumbled</td>
<td>44</td>
<td>42</td>
</tr>
<tr>
<td>2F</td>
<td>Tumbled</td>
<td>46</td>
<td>43</td>
</tr>
</tbody>
</table>

TABLE 2 - RESIDUAL STRENGTH COMPARISONS
Therefore, once the maximum peel stress has been established, based on the maximum peel strength of the patch adhesive (obtained experimentally), the patch overlap can be predicted.

By employing the above theoretical analysis a ply thickness of 4 layers was predicted for the repair patch. As exhibited in Figure 5, a 4 ply patch was experimentally found to be the optimum patch thickness. Comparison of predicted patch overlap to experimental results are contained in Figure 7.

**DISCUSSION**

Although the limited time available for the research did not permit investigation of areas pertinent to the repairs accomplished (i.e. fatigue, environmental effects etc) sufficient time was available to complete the main objective of the research. Before dealing with specific observations it is considered prudent to point out that the term 'strength' used throughout the report refers to tensile strength. Although the normal impacted panels did not reflect changes in compressive characteristics and therefore compressive behaviour was neglected for the repairs, this approach could not be taken where thicker panels are used or damage is greater.

Earlier in the report it was stated that the laminates constructed for this research were analogous to those on aircraft. Although this is not entirely correct as most skins contain a substantial amount of ±45% plies, ballistic impact damage for the various laminates are the same. [9] and [10] refer. Also since the direct strength of the research panels are greater than those containing angled plies, ballistically impacted thin monolithic panels can be repaired using single sided external patches.

With regard to actual repair techniques employed during the research a number of important characteristics were observed. These characteristics were the necessity to ensure proper adhesion between the patch and substrate, the advantage of retaining the damage area and the significant strength degradation when using square patches.

Finally one of the most significant repair characteristics observed was that there appeared to exist optimum patch parameters to achieve a permanent repair (i.e. 90% of undamaged tensile strength). These parameters were patch ply thickness and patch overlap length. By employing various 2D analysis the majority of which is based on joint
analysis, the experimental results were paralleled by theoretical analysis (Figure 7). The inherent problems associated with the use of 2D analysis for 3D situations has been discussed in the main part of the text.

**CONCLUSION**

The research highlighted a number of important finds which can be best summarized as follows:-

1) Although it is generally accepted that when carbon laminates are damaged their compressive strength is greatly affected, the research revealed that the compressive buckling strength of the thin monolithic panels was not affected. This revelation is particularly important when one considers that the thin monolithic panels are representative of a large majority of aircraft type panels which have or would be affected by ballistic damage. However, the tensile strength was significantly affected with panels exhibiting only a 20% reduction in cross section area due to ballistic damage suffering degradation of tensile strength in excess of 50%.

2) Single sided external repairs could be utilized to restore impacted panel strength to a usable degree (i.e. 90% of original).

3) There existed optimum path parameters (i.e. ply thickness and overlap). These became evident during experimental research.

4) These optimum patch parameters can be predicted analytically.

5) Residual tensile strength of impacted panels can also be predicted analytically.

6) Square patches, damage area removal and, poor surface preparation have a significant and detrimental affect on restored strength.
REFERENCES


