**THE MECHANICAL PERFORMANCE OF REPAIRED STIFFENED PANELS:
PART I - EXPERIMENTAL CHARACTERISATION**

by

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ABSTRACT

In this paper the performance under compressive static load of I-section stiffened panels is reported. A pristine panel is compared with panels containing simulated damage and repaired panels. Modes of failure, stiffness and strains are studied and it is shown that the significant decreases in strength caused by the damage are almost fully recovered by the repair scheme used.

1. INTRODUCTION

The potential to reduce the weight of structures, and increase fatigue and corrosion resistance, make carbon fibre-reinforced plastic (CFRP) composites increasingly favoured as aircraft structural materials. A large amount of work has been done in connection with the design, construction and testing of these materials. Hence, as a result of an increased understanding of their behaviour, the relative weight of composites in aircraft has increased steadily in recent years. In the early 1970s, the Airbus A300 first used a polymer composite in its fin leading edge and other secondary structure (i.e. a structure that would not lead to the loss of the aircraft if it were to fail). Later, in 1985, Airbus was the first manufacturer to use composite materials in commercial primary structure (i.e. a structure that would lead to the partial or complete loss of the aircraft if it were to fail) with an all CFRP tail fin [1]. Today a significant amount of primary structure and nearly all moveable controls on the wings, the empennage, the cabin floor and most of the service panels of the Airbus and other large civil aircraft are made of composites. There are two main types of structure which are employed; monolithic stiffened or sandwich, chosen according to the detailed design requirements.

However, the use of CFRP materials is restricted by their poor delamination resistance which can cause severe reductions in strength and stiffness, and may lead to catastrophic failure of the structure. This characteristic can limit their use in some particular engineering applications where damage is likely to occur. Thus, repair methods for in-service or fabrication damage need to be developed to ensure that composite primary

structures are more economically viable than their metallic counterparts throughout the service life of the aircraft.

The present paper reports on one aspect of a large research programme [2] in which the repair of sandwich beams and stiffened panels was studied. Here, the experimental aspects of repairing stiffened panels are addressed, and in a companion paper [3] the related performance prediction issues are presented. Case studies from Airbus and NASA have shown that stiffened panels may be repaired in similar ways to that employed for monolithic panels [2]. Furthermore, it is generally agreed that overlap and scarf patch repairs are good candidates for the design of repairs of stiffened panels, with the scarf design usually giving the better performance [2]. In the present studies, overlap patches were selected since it was felt that a scarf patch would be too complicated to manufacture for *in-situ* repairs.

2. THE STIFFENED PANELS

2.1 Introduction

The panels were manufactured by DERA Farnborough to be 400mm long by 360mm wide. The 32-ply 4.87mm thick skin was supported by three I-section stiffeners, as shown in Figure 1. The stiffeners were 120mm apart.

Figure 1: The I-stiffened panels

The material used was T300/914 unidirectional prepreg (Toray T300 fibres and Hexcel 914 resin), from Hexcel. The quasi-isotropic $(+45^{\circ}/-45^{\circ}/0^{\circ}/90^{\circ})_{4s}$ skins were laid-up first according to the axes shown in Figure 2. The I-section stiffeners were separately

assembled from four uncured laminates, comprising a tapered foot, two C-sections back-to-back and a spar cap, all of the same $(+45^\circ/-45^\circ/0^\circ)_{2s}$ lay-up, as shown in Figure 3. The triangular spaces between the webs and the spars were filled with strips of unidirectional prepreg (known as “noodles”, see Figure 2). The foot and cap of each stiffener have unsymmetric lay-ups but, as a consequence of forming two laminates into a C-shape, the web is symmetric.

Figure 2: Dimension of the panels (mm)

Figure 3: Panel's lay-up sequence

2.2 Materials

The unidirectional T300/914C system cures in an autoclave at a pressure of 7bar and a temperature of 175°C for 1hour, plus a 4hour post cure at 190°C. It has a cured fibre volume fraction of nominally 56% and ply thickness of 0.152mm. The mechanical properties of this widely used aerospace material can be found in the literature, e.g. reference [4]. Hexcel Redux 319 was used to secondarily bond the stringers to the skin, and its mechanical properties were determined by Charalambides et al. [5].

2.3 Repair Procedures

2.3.1 Introduction

As stiffened panels are good candidates for primary aircraft structures it is of concern to establish a technique for the repair of panel elements. An impact over a stiffener can lead to a significant reduction in the compressive strength of panels designed for high buckling strains [6]. (It should be noted that aerospace primary composite structures are typically designed very conservatively, to an ultimate strain of 4000 μ strain (fully factored), which implies limit working strains at two-thirds of this level in real structures. Work to improve the design critical performance, which is driven by damage tolerance and notch sensitivity, seeks to increase this strain to 6000 μ strain (fully factored), which equates to a maximum strain of 4000 μ strains in the working structure. The theoretical buckling strain of the undamaged and repaired panels tested in this study was 6000 μ strain, with failure occurring at higher strains, see Table 2 later.)

The panels were not impacted in the current work, but ‘damage’ was simulated by making a circular cut-out with a diamond-coated tool. The damage represented an impact to an area immediately underneath the central stiffener at the mid-length of the panel. In one case the ‘damaged’ region was taken to be a 55mm circular area to the central stiffener plus skin. The second ‘damage’ type was similar but was confined to the skin only and the stiffener was left intact. It is generally agreed that overlap and scarf patch repairs are good candidates for the repair of stiffened constructions, with the scarf configuration usually giving better performance [2]. Here, overlap patches were chosen as it was felt that a scarf patch would be too complicated to manufacture for *in-situ* repairs.

2.3.2 Repair materials

Adhesively bonded repair systems are usually the strongest when the repair materials are identical to the parent materials [2]. A repair system identical to the parent materials was therefore selected: the unidirectional T300/914C prepreg supplied by Hexcel. An unsupported Redux 319 epoxy adhesive was used to bond the repair patch to the parent. The patches were cured *in-situ*, according to the manufacturer’s recommendations (i.e. 1 hour at 175°C, plus a 4 hour post cure at 190°C), but under vacuum only to be representative of an in-the-field repair.

2.3.3 Repair designs

Two overlap repair schemes were evaluated. One involved the repair of the back and front face of the panel, including the stiffener, while the second involved the repair of the back face (skin) only.

The patches were square in plan form, and their lay-up was identical to that of the parent, although with only half the number of plies in order to halve their membrane stiffness. Thick overlap repairs with straight edges have been reported to fail prematurely [7]. Tapering of the plies’ ends was however shown to limit the development of peel stresses, and hence increase repair efficiency. Here, the plies were graded in size, with an overlap length 30 times the ply thickness (e.g., $30 \times 0.15 = 4.5\text{mm}$). Pre-cured ‘plugs’ (i.e. a piece of skin and/or a stiffener cut to the appropriate shape and size), as shown in Figures 4 and 5, were used to fill the previously machined holes. The extent of the repair to the back and stiffener face is given in Figure 6(a), and of the repair to the damage to the back face only is shown in Figure 6(b).

Figure 4: Repair patch design

Figure 5: Repair plugs

Figure 6: Repaired panels

2.3.4 Manufacture of the repair patches

The repair area was first grit-blasted and subsequently thoroughly cleaned with acetone. The patch to the back face was applied directly over the prepared surface. The plugs were then placed in the cut-out regions, and enveloped by adhesive to ensure that no gap existed between the plug and the structure wall (an initial clearance of 0.5mm exists between the plugs and the wall). The patch was then applied to the stiffener, with the help of wooden formers to make sure that the patch followed the shape of the stiffener. The formers were subsequently removed and the patches were vacuum bagged and autoclave cured (but without pressure) as before.

3. Experimental Programme

3.1 Introduction

Five stiffened panels were supplied by DERA and were tested in direct compression. The panels tested were as follows:

- One undamaged panel (P1)
- One panel damaged with a hole through the skin and the central stiffener (P2)
- One panel damaged with a hole through the skin, underneath the central stiffener (P3)

- Two repaired panels; one of P2 and one of P3 (P4 and P5, respectively)

The panels were loaded to failure, and the load, cross-head displacement and back-to-back axial surface strains recorded during the tests.

3.2 Compression test

Compression tests were carried out in a 2500kN testing machine, which has been specially designed to have the high stiffness necessary for buckling and post-buckling experiments [8].

The ends of the panels were first machined parallel and flat, to ensure an even loading, giving a length after machining of 398mm. As it is necessary to protect the loaded ends from local damage and crushing, the ends were ‘potted’ in 50mm deep aluminium channel sections, using a filled epoxy resin (CY219 from Ciba (UK)). The free length of the panel was 308mm between the potted ends.

The testing machine comprised two platens that are 1m long and 0.5m wide. The panels were placed at the centre of the platens. To further ensure that the specimens were loaded uniformly across their cross-sectional area, a thick adhesive paste (Ciba Araldite 2011 epoxy paste adhesive bulked up with slate powder) was applied to the top end of the panels, after they were placed in the machine. A thin aluminium plate was then placed on top of the adhesive. The panels were loaded to 10kN and left under load overnight. The excess adhesive was removed and the panels were then subsequently unloaded. Once this operation was done, the panels were not moved from the machine until testing had been completed. The tests were performed at a cross-head speed of approximately 0.005mm/s.

3.2.1 Data acquisition

A data acquisition system was linked to the testing machine. The data were recorded onto disc at a rate of one data point every five seconds, until failure of the specimen. The individual data points were then treated in an Excel spreadsheet and were corrected to ensure zero displacement at zero load.

3.2.2 Application of strain gauges

Strain gauges were used for the determination of surface strains. An instant cure cyanoacrylate adhesive was used to bond the gauges to the specimens having first hand-abraded the surfaces of the specimens with silicon carbide grit paper, and then carefully cleaned them using a solvent. The gauges were positioned carefully onto the surface and pressure was applied for a few seconds. TML FLA-6-11-1L strain gauges were used. They had a gauge factor of 2.12 and a 120 Ω resistance, for a 6mm gauge length. The gauges were connected to the data acquisition system, which comprised eighteen Wheatstone 1/4-bridge input channels with 1V applied voltage.

Although twenty-two locations were defined, only sixteen strain gauges per panel were used on average. The locations of gauges on the undamaged panel are shown in Figure 7.

Fourteen back-to-back gauges were used and they served a double purpose. Eight back-to-back gauges were placed half way along the panel length, whereas six of them were placed at one quarter of the length. The uniformity of load within the panel was checked by comparing the strain readings from the aforementioned groups of gauges. Furthermore, the extent of any out-of-plane deformation can be determined from differences between the back-to-back strain measurements.

The locations of strain gauges on the damaged panels are shown in Figure 8. Some back-to-back strains were recorded as previously, but other gauges were placed in the vicinity of the 'damage' in order to determine the extent of any strain concentrations. The locations of strain gauges on the repaired panels were the same as on the damaged panels, and are shown in Figure 9.

Figure 7: Location of strain gauges on the undamaged panel

Figure 8: Location of strain gauges on the damaged panels

Figure 9: Location of strain gauges on the repaired panels

3.2.3 Shadow Moiré interferometry

The shadow Moiré fringe technique was used to assess the extent of any out-of-plane deformation during testing. The back face of the panels was painted white and placed in front of a thick Plexiglas grating (0.4mm spacing between lines). The grating was placed as close as possible to the panel and was illuminated by a projector at 60° to the normal to the surface. Any fringes were photographed using a CCD camera.

4. Results

4.1 Behaviour of the undamaged panel (P1)

The failure load of the undamaged panel (P1) was 1200kN at an applied displacement of 3.70mm. The panel collapse resulted from a sudden, explosive, failure involving in-plane compression fracture of the skin and the stiffeners, and delamination of the stiffeners away from the skin, as sketched in Figure 10 (left-hand picture is view from skin-side, right-hand picture is view from stiffener side). It is noted that extensive delamination in the stiffeners' spar caps and at the unsupported panel edges also occurred. From inspection of the panel after testing, it may be argued that the panel collapse was initiated by in-plane skin compressive failure (originating either at the unsupported edges or at the centre of the panel), which then resulted in the separation of the stiffeners away from the skin. The stiffeners were then left to support an increasing load and they immediately collapsed. However, it is difficult to comment on the details of fracture, which was so violent that much evidence of the failure processes was lost.

The load-displacement curve of the undamaged panel was linear up to a displacement of 2.25mm, after which it deviated from linearity. Strains from the back-to-back gauges at various locations on the panels were found to be uniform except near the edges where bending took place. The traces from gauges SG1 and SG2 (see Figure 7 for locations) diverged from a displacement of about 2mm onwards.

Figure 10: Representation of the locus of failure of the P1 panel (jagged line represents skin fracture, horizontal bars represent stiffener delamination, shaded area represents complete separation)

4.2 Behaviour of the damaged panels (P2 and P3)

Panel P2

The failure load of the panel damaged by a hole through the central stiffener and the skin underneath it (P2) was recorded as 515kN, at an applied displacement of 1.88mm. The locus of failure for this panel was simpler than that of the undamaged panel, since the panel's collapse resulted from in-plane compressive failure of the skin that originated at the edge of the cut-out, as sketched in Figure 11 (left-hand: skin-side, right-hand; stiffener side). The stiffeners were also seen to have delaminated from the skin.

The load-displacement curve was linear to failure. The strains in the vicinity of the damage (SG15, SG16, SG17 and SG18; Figure 8), were compared with far-field strains (SG1 and SG2; Figure 8) and it was seen that relatively more bending occurred around the damage, whereas it was more limited at the edges of the panel. The maximum strain recorded at 15mm from the damage (on the stiffener side) was about 2.4 times greater than the far field strains. It seems clear that the presence of the cut-out resulted in an increase in strains in its vicinity, which suggests that failure resulted from the high strains occurring at the edge of the hole.

Figure 11: Representation of the locus of failure of the P2 panel (jagged line represents skin fracture, horizontal bars represent stiffener delamination, shaded area represents complete separation)

Panel P3

The panel damaged by a hole only through the skin underneath the central stiffener (P3) failed at a load of 849kN at an applied displacement of 2.72mm. The collapse of the panel seems to have resulted from in-plane compressive failure that initiated at the edge of the damage, possibly due to a high strain concentration there. The stiffeners may then have separated from the skin, and failed by in-plane compressive failure, as sketched in Figure 12 (left-hand; skin side, right-hand; stiffener side). The panel's load-displacement curve was again linear to failure. The test was stopped at a load of 700kN (about 82% of the failure load) and an ultrasonic scan (And-scan) was used *in situ* to assess the extent of

delaminations around the damaged area. No damage was observed and the test was subsequently continued. However, the process resulted in the loss of the strain readings because the gauges had to be removed.

Figure 12: Representation of the locus of failure of the P3 panel (jagged line represents skin fracture, horizontal bars represent stiffener delamination, shaded area represents complete separation)

4.3 Behaviour of the repaired panels (P4 and P5)

Panel P4

The repaired panel that was damaged by a hole through the central stiffener and the underneath skin (P4) failed at a load of 1000kN at an applied displacement of 2.91mm. The sequence of failure was difficult to obtain since the repair obscures the observations. However, it seems that in-plane compressive failure, of the skin and the repair patches, initiated at the top of the repair plug, and propagated towards one unsupported edge of the panel only, as shown in Figure 13 (left-hand; skin side, right-hand; stiffener side). The stiffeners may then have separated away from the skin, which resulted in the fracture of the repair patch on the stiffener side on one hand, and its separation from the panel on the other.

The load-displacement curve was seen to deviate from linearity for displacements over 1.5mm. The strains in the repair patch and close to the damage (SG17 and SG18; Figure 9) were compared with far-field strains (SG1 and SG2; Figure 9). It was seen that while bending of the panel occurred, it was not as pronounced as for the undamaged panel. It may be further noted that the repair patch had a beneficial stiffening effect, since the strains in the repair (i.e. SG17 and SG18) were lower than the back-to-back far field strains (i.e. SG1 and SG2), although they were 1.4 times greater than the far field strains at failure for the damaged panel. The repair patch allowed the panel to fail at a load of 1000kN, which may be compared with the failure load of 515kN for the equivalent damaged panel. The present overlap repair scheme therefore recovered an appreciable proportion of the pristine, undamaged, panel strength.

Figure 13: Representation of the locus of failure of the P4 panel (jagged line represents skin fracture, horizontal bars represent stiffener delamination, vertical bars represent patch delamination, shaded area represents complete separation)

Panel P5

The repaired panel damaged by only a hole to the skin underneath the central stiffener (P5) failed at 1074kN, at an applied displacement of 3.31mm. Again it is difficult to comment on the sequence of failure. However, it seems that the collapse of the panel resulted from a compressive failure that initiated at the centre of the repair patch, as shown in Figure 14 (left-hand; skin side, right-hand; stiffener side).

As for the undamaged panel, the load-displacement curve deviated from linearity for displacements over 2.25mm. Again, and as for the repaired P4 panel, the repair patch resulted in a decrease in the strain in the vicinity of the damage. Also, the SG17 and SG18 (Figure 9) traces decreased suddenly at an applied displacement of about 3mm, which was the result of an in-plane compressive failure of the repair patch at that displacement. The crack was seen visually during the test and was estimated to span the central stiffener width. The crack then stabilised and the panel carried on supporting the load until total collapse.

Again, the repair scheme used has been shown to be successful and restored an appreciable fraction of the original strength of the panel. The panel's failure load was 1074kN, whereas the corresponding damaged panel failed at a load of 849kN.

Figure 14: Representation of the locus of failure of the P5 panel (jagged line represents skin and patch fracture, horizontal bars represent stiffener and patch (skin side) delamination, shaded area represents complete separation)

5. Comparison of the panels' performance

The failure loads of the undamaged, damaged and repaired panels are shown in Table 1. The reduction in failure load from the undamaged failure load is also shown. The undamaged panel failed at a load of 1200kN. The failure load of the damaged panels was greatly reduced; the panels P2 and P3 failed at 515kN and 849kN, respectively. As expected the damage that consisted of a hole through the central stiffener led to the biggest reduction in strength; the P2 panel lost 57% of the undamaged strength. In comparison, the damage that consisted of a hole through the skin resulted in only a 30% decrease in strength. It may be seen from Table 1 that the present repair schemes have been very successful. The two repaired panels failed at loads above 1000kN, recovering 83% and 89.5% of the strength for the P4 and P5 panels, respectively.

Table 1: Tests results for the I-stiffened panels

	Failure Load (kN)	Decrease in failure load (%)	Cross-head displacement at failure (mm)
P1 (Undamaged)	1200	0	3.70
P2 (Damaged: through-hole)	515	57	1.88
P3 (Damaged:skin-hole)	849	30	2.72
P4 (Repaired: through-hole)	1000	17	2.91
P5 (Repaired: skin-hole)	1074	10.5	3.31

The load-displacement curves of the undamaged and damaged panels are compared in Figure 15. The load-deflection curve of the pristine panel may be seen to be the steepest (i.e., the highest stiffness). For the damaged panels, the cut-out has resulted in a change in the panels' load path that ultimately resulted in a loss in structural stiffness. Figure 16 presents similar curves, but for the repaired panels. The load-displacement curves of the undamaged and the repaired panels may be seen to overlap. The overlap repairs successfully restored the stiffness of the panels.

Figure 15: Load-displacement curves (Undamaged and damaged panels)

Figure 16: Load-displacement curves (Undamaged and repaired panels)

Strains at various locations in the panels were measured by the application of strain gauges. As described above, as many as 18 gauges were used on each panel. However, for brevity, only the far field strains and strains around the damage area are considered here.

The back-to-back far field strains at failure, recorded on one edge of the panels, are summarised in Table 2. The strains SG1 and SG2 (see Figure 7) were read on the skin and the stiffener side, respectively. In addition, the average strain and the percentage difference in strain are given. The difference in strain was calculated to give an indication of the out-of plane bending occurring at the edges.

Table 2: Strain gauge results at failure for the I-stiffened panels

	SG1 (μ strain)	SG2 (μ strain)	Average Strain (μ strain)	Difference (% of average)
P1 (Undamaged)	-7174	-9082	-8138	23.2
P2 (Damaged: through-hole)	-3643	-3989	-3816	9
P3 (Damaged: skin-hole)	-5472	-6000	-5736	9.2
P4 (Repaired: through-hole)	-6152	-6531	-6341	6
P5 (Repaired: skin-hole)	-6550	-7072	-6811	7.6

It may be seen in Table 2 that the percentage difference in back-to-back strains reaches 23.2% for the undamaged panel, whereas it is lower than 10% for the damaged and the repaired panels. In other words, significant out-of-plane deformations occurred at the edges of the undamaged panel, but not on the others. Furthermore, it may be seen from Table 2, that the far-field strains of the repaired panels were over -6000 μ strain. Therefore, these repair schemes would appear to be very suitable for primary composite structure.

6. Discussion

6.1 Undamaged panel (P1)

The load-displacement curve of the undamaged panel was found to be non-linear for displacements over 2.25mm, and this may be attributed to two phenomena. First, as the applied strain increases, damage, such as fibre failure, matrix cracking or delamination, evolves in the skin and the stiffeners. Secondly, the eccentricity of the skin from the stiffeners guarantees that some bending occurs as the applied displacement is increased. It may be assumed that the applied load is shared between the skin and the stiffeners, in proportion to their stiffnesses (the product of their respective axial elastic modulus and cross-sectional area). For the present panel it may be calculated that, under a constant strain (i.e. for a constant shortening of the loaded ends), the skin carries 55% of the load, with the remaining 45% being carried by the three stiffeners. The evolution of damage and geometric non-linearity will result in a loss of the structural stiffness.

The unsupported edges of the panel were found to deform out-of-plane. When the back-to-back strains at the unsupported edge of the P1 panel were compared with back-to-back strain measured at the centre of the panel, it was seen that while some bending took place at the centre of the panel, out-of-plane deformations were a maximum at the panel's edge. This behaviour arises because the unsupported edges are relatively wide and unrestrained from bending. It should be noted that it has been reported in the literature [9] that the unsupported edges are sometimes made narrower, and consequently more stable to out-of-plane deformation, so that the region around the central stiffeners that is under study is under higher stress than the panel's edges.

The shadow Moiré fringe technique was used to assess the extent of any out-of-plane deformation (buckling pattern), but no major deformation was seen on the undamaged panel, nor indeed, on any of the five panels. It is therefore not clear if buckling of the panel occurred in the experiment. However, it should be noted that the grid spacing used in this study was rather coarse (i.e. 0.4mm between each line) when compared with grid spacings reported in the literature (i.e. 0.125mm between each line in [6]), and hence the present tests may not have picked up the smallest out-of-plane deformations. It is however clear that some bending took place, as noted above.

There will be a number of locations in the panel where failure could have initiated, and in a number of possible ways. It is noted that since the collapse of the panel was explosive, the location of crack initiation is not known. However, due to the wide unsupported edges, the strains were higher at that location, and failure may have initiated there. This is consistent with the earlier observation that failure may have resulted from a compressive failure of the skin. Failure may have otherwise initiated between two stiffeners. Extensive delamination was also observed at the unsupported edges, where through-thickness effects may have helped the process of fracture initiation. Other failure modes such as skin-stiffener separation are possible since the stiffeners were completely debonded from the skin. In the present case, failure may have initiated at a crack in the skin, with the initial crack growing perpendicularly to the applied stress and leading to stiffener debonds, and ultimately collapse of the skin and the stiffeners.

6.2 Damaged panels (P2 and P3)

In Table 2 the difference in back-to-back strains at the unsupported edges of the damaged panels were seen to be about 9%, and hence much less than that of the undamaged panel. Several reasons may be identified. First, it is noted that the failure load of the P2 and P3 panel was much less than that of the P1 panel. Secondly, it may be thought that the presence of the damage resulted in the load being carried mainly by the two side stiffeners and by the skin in the area adjacent to the damage. The presence of the damage would have resulted in a change of the load path in the panels. Consequently, the area near the damage is under a higher stress than the unsupported edges.

The panel damaged by a through-hole (P2)

It was noted above that the maximum strain recorded at 15mm from the cut-out was 2.4 times greater than the far field strains. It may be assumed that failure resulted from the high strains occurring at the edge of the cut-out. It is interesting to note that the ratio of undamaged failure load to that of the P2 damaged panel is 2.33. In a very simplistic way we may argue that the P2 panel failed at a load inversely proportional to the increase in strain at the edge of the cut-out. The failure load of the P2 panel is, therefore, much as expected.

The panel damaged by a hole through the skin only (P3)

Similarly to the P2 panel, the load path in the P3 panel would be changed by the presence of the damage, but to a lesser extent since the central stiffener was left intact. The load

was carried by the three stiffeners, as for the undamaged panel. However, the area of the skin adjacent to the damage will support a higher load, resulting in an increase in stress in this area. The stress increase will, however, not be as high as for the P2 panel, since the stiffener takes some of the load. Consequently, the P3 panel fails at a lower load than the undamaged panels, but at a load higher than the panel damaged by a through-hole (P2).

6.3 Repaired panels (P4 and P5)

The behaviour of the repaired panels was the most complicated of all the panels. In Table 2 the difference in the back-to-back strains at the unsupported edges was lowest for the repaired panels. Thus, although the overall stiffness of the panels was comparable with the stiffness of the undamaged panel (Figure 16), the repairs have had a local stiffening effect.

However the stiffening effect of the patches is beneficial since the increased stiffness helped reduce the strain concentration at the edge of the damage. The patches have recovered the load-carrying capability of the central stiffener as well as the area of the skin around the damage. The excess of load, previously carried by the side stiffeners and the skin around the damage, is now being carried by the repair patch, through the adhesive layer. The strains around the damaged area were hence seen to be comparable with the far field strains. Consequently, the repair patches recovered the load path in the panels, reducing the excess of stress in the skin around the damage, and ultimately increasing the failure load. However, the local increase in material caused by the overlap patches has resulted in a local increase in stiffness which effectively prevented the panel deforming freely.

While the sequence of failure may be complicated for the P4 and P5 panels, it was observed that the panels may have failed by in-plane compressive failure of the skin that was initiated in the repair, possibly at the edge of the damage. Therefore, and as for the damaged panels, it may be assumed that failure initiated when the strain in the vicinity of the original damage reached the failure strain of the material. It may be noted that the failure of the repaired panels was not as a result of an adhesive failure. The repair procedure followed in the present work may therefore be seen to be applicable to the repair of primary structures. Although the repair was seen to reduce the strain concentration around the damage, it may be thought that the interaction of the repair plugs used in the present repair scheme with the wall of the cut-out, resulted in a local increase in strains around the cut-out.

7. Conclusions

The relative static strengths of the undamaged, damaged and repaired I-sectioned stiffened panels were much as expected, based on: observations of failure modes; strain gauge readings; shadow Moire observations; ultrasonic inspection. Most importantly, the repair schemes employed recovered 83% and 90% of the undamaged strength for the through-hole and the skin-hole damage, respectively.

The load path for the undamaged panel was such that the load was carried by the skin as well as by the three stiffeners. Large out-of-plane deformations were seen to occur at the wide unsupported edges of the undamaged panel, whereas they were much smaller in the skin bay. There were several locations in the panel where failure could have initiated, and in a number of possible ways. Identification of the precise failure mechanism was, however, difficult since the collapse of the panel was explosive. However, from visual observation of the panel after testing and analysis of the experimental data, the panel collapse seems to have initiated from in-plane compressive failure of the skin that led to stiffener debonds, and ultimately to the collapse of the skin and the stiffeners.

The panel damaged by a through-hole lost 57% of its strength. By comparison, the damage that consisted of a hole through the skin only resulted in just a 30% decrease in strength. The presence of the cut-out changed the load path in the panels significantly, resulting in the load being carried mainly by the two side stiffeners and by the skin in the area adjacent to the damage. It was considered that failure may have resulted from the high strains occurring at the edge of the cut-out.

The repair patches were shown to have successfully recovered the load-carrying capability of the damaged panels. The present repair scheme recovered the load path in the panels, reducing the excess of stress in the skin around the damage and ultimately increasing the failure load of the panels. However, it was also discussed that while the stiffening effect of the patch is beneficial, since it reduces the strain around the damage, it also increased locally the bending stiffness of the panel. By altering the patch stacking sequence, whilst keeping the lay-up constant, it may be possible to produce a repair patch that maximises the membrane stiffness and minimises the bending stiffness, which may be better than one where both stiffnesses are maximised.

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REFERENCES

- [1] D. H. Middleton, *Composite Materials in Aircraft Structures*: Longman Scientific and Technical, 1990.
- [2] S. Mahdi, "The Performance of Bonded Repairs to Composite Structures": PhD Thesis, Imperial College, London, February 2001.
- [3] S. Mahdi, A. J. Kinloch, F. L. Matthews and M. A. Crisfield, "The Mechanical Performance of Repaired Stiffened Panels - Part 2: Finite Element Modelling; to be published.
- [4] S. Mespoulet, J.M. Hodgkinson, F.L. Matthews, D. Hitchings, and P. Robinson, "Design, Development, and Implementation of Test Methods for Determination of Through Thickness Properties of Laminated Composites", *Plastics, Rubber and Composites*, vol. 29, pp. 496-502, 2000.
- [5] M. N. Charalambides, A. J. Kinloch, and F. L. Matthews, "Adhesively Bonded Repairs to Fibre Composite Materials - I: Experimental", *Composites Part A*, vol. 29A, pp. 1371-1381, 1998.
- [6] E. Greenhalgh, S. Singh, and D. Roberts, "Impact Damage Growth and Failure of Carbon-Fibre Reinforced Plastics", Proc 11th International Conference on Composite Materials (ICCM 11), Gold Coast, Australia, vol. 2/6, pp. 573-582, 1997.
- [7] J. W. Deaton, "Repair of Advanced Composite Commercial Aircraft Structures" in *Engineered Materials Handbook, Vol.3: Adhesives and Sealants*: ASM, pp. 830-839, 1990.
- [8] K. A. Stevens, R. Ricci, and G. A. O. Davies, "Buckling and Postbuckling of Composite Structures", *Composites*, vol. 26, pp. 189-199, 1995.
- [9] K. A. Stevens, S. Specht, and G. A. O. Davies, "Postbuckling Failure of Carbon/Epoxy Compression Panels", Proc 11th International Conference on Composite Materials (ICCM 11), Gold Coast, Australia, vol. 6/6, pp. 695-705, 1997.