COMPOSITES FOR REINFORCEMENT OF DAMAGED METALLIC AIRCRAFT WINGS

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Abstract

Advanced composites, because of their inherent directional properties and ability to take complex contours are ideally suitable for repairs even in aged metallic aircraft structures. The composite materials can be suitably tailored to suit the strength and stiffness requirement at location of the damage on the parent structure. The laminate patch can be bonded to the structure. This avoids the use of large number of fasteners that are required for repair with metallic repair patch. A composite patch repair has been carried out on a military aircraft wing. The skin and the spar flange near the landing gear part were cut in order to gain access to a loosened bolt. With a request from HAL (Nasik), National Aerospace Laboratories (NAL) has taken the responsibility of designing and realizing composite repair at the damaged locations. The geometry and lay-up of the patches have been arrived at based on the strength requirements. The patches were fabricated at NAL and the repair was implemented on the aircraft. The quality of the patch bonding was checked using non-destructive testing (NDT). The aircraft has completed about 130 hrs flying with the repair. The repair was certified by the regulating authority and has been recommended to be implemented on other twenty aircraft also. This task has been completed successfully and all the aircraft are flying. This is an important achievement both for NAL and HAL (Nasik). This paper discusses the repair design, patch fabrication and implementation of the repair.

Keywords*: Composites, Autoclave Moulding, Carbon Fiber Reinforced Plastics (CFRP), Hybrid Bonding, NDT*

Introduction

Composites usage has spread from noncritical structures to strength critical structures as performance demands for new aircraft have increased and as confidence has built for composites as a structural material [1]. Some recently developed military/civil aircraft have nearly all composite airframe structures. The structural components of these aircraft are assembled using secondary bonding or co-curing techniques. This type of construction offers many advantages over other techniques, and it can produce lighter and stronger structures. Increased life expectancies and up gradation of military aircraft demands advances in repair technology. As a result, adhesively bonded patches of advanced fiber composites, such as carbon-epoxy, graphite-epoxy can provide highly efficient and cost effective repairs for metallic aircraft components, which have suffered cracking due to fatigue and stress corrosion [2,3]. Some of the advantages of crack patching with bonded composites over conventional repair procedures based on mechanically fastened metallic patches are

- Minimizes stress concentration
- Patches easily formed to complex contours
- Minimizes corrosion
- Minimizes fretting
- Minimizes undesirable structural changes by tailoring patch to suit the stress field
- Balancing problem can be minimized on control problems
- Simple Non-Destructive Inspection (NDI) is sufficient

Due to the following advantages, the use of composite patching technology is attractive.

Damage Scenario

In the present case, the military aircraft main landing beam and spar web is connected by means of bolt and nut. Due to its long service and heavy landing these bolts were loosened. During its overhaul maintenance, the bolt loosening problem was identified. Due to inaccessibility to reach the loosened bolt, the wing skin and spar flange is cut at bolt location. Fig.1 shows the repair location with wing skin and spar flange cut out. Since these damages are very close to the fuel tank zone, the conventional metallic patch repair is not advisable. Hence, Carbon Fiber Reinforced Plastics (CFRP) patches are used to reinforce the damaged wing structure. The spar flange cut out is reinforced by CFRP filler patch, using hybrid bonding technique. The filler patch size is derived based on cut out width and existing rivet locations. Few existing rivet holes are used for mechanical fastening. The wing skin damage is reinforced using CFRP external patch, which is designed to take more load at cut out zone and gradually distributed in all the directions. Fig.2 shows the external patch location with internal construction of the wing. The external patch is bonded over the wing surface using room temperature curable adhesive by vacuum bagging technique. The bonded region is qualified by Ultrasonic 'A' scan (NDE) method.

Design of CFRP Patch

The philosophy adopted in the design of the repair patch is to obtain an external patch and a filler patch of adequate stiffness and strength to replace the removed material from the parent metal structure. The composite patch of required thickness and size are designed to meet these requirements. Due to non-availability design data, from the Original Equipment Manufacturer (OEM), reverse engineering approach is used to determine the load requirements. The stress in the metal skin and the spar plate is given as 420 N/mm^2, which is ultimate stress of parent material of wing skin and spar plate. The parent aluminum structure Young's modulus is 70GPa. Using these data the equivalent strain in the metal part is calculated.

$$\varepsilon_p = \sigma_p / E_p \tag{1}$$

ε_p = 420 / 70000 = 6000 micro strain

Similarly, the load carrying capacity of the parent material is estimated. The wing skin and spar flange cut out dimensions and its thickness are given in the Fig.3.

$$P_p = \sigma_p \times A_p \tag{2}$$

Load caring capacity of wing, at cut out section is 216300 N.

NAL decided to use unconventional approach in the design of composite patch to reinforce cut out on the metallic wing. NAL opted to fabricate the repair patch using carbon fiber prepreg. The following elastic properties are used in the patch design:

Young's modulus along fiber direction, E_L = 130GPa,
Young's modulus along resin direction, E_T = 10GPa,
In plane shear modulus, G_{LT} = 5GPa,
In plane Poisson's ratio, ν_{LT} = 0.35.

Two different CFRP patches are designed namely filler patch and external patch. Both repair patches are designed to carry the total load of 216300 N and the 0, +45, -45 and 90° layers are arranged to obtain an equivalent elastic modulus of parent material of the wing. The layup sequence is arranged in such a way the equivalent modulus of 76 GPa along the main load direction. The size of the filler patch is 210 x 50 x 3.8 mm. Classical lamination theory is used to compute the CFRP patch stiffness. For strain continuity both CFRP patches are maintained same Young's modulus and strain values are computed as ε_c = 420 / 76000 = 5526 micro strains. Since, the obtained strain is very close to the parent structure, the strain continuity is maintained. The load carrying capacity of the filler patch is 73720 N. The external patch designed to take the balance load 142580 N. For external patch design, the spar plate thickness 6.5 mm is considered. Since external patch is bonded over wing surface, sudden projection will affect the aerodynamic performance. Hence the central zone of external patch dimension is derived as 210 x 55 x 4.2mm. Thus, thickness of the repair patch at the cut out area is 8.0mm. Since the total thickness of the metal structure at cut out is 10.3mm, the load transferred through the remaining thickness 2.3 is compensated through the extra width provided in the external patch. The overall dimension of the external patch is 550 x 340mm and thickness varied from 4.2 mm to 0.3mm. The section of the patch that is in width direction normal to the spar direction is considered for the repair design. The cross section of the CFC repair patch is shown in Fig.4. The load carrying capacity of the external patch is computed as 354438 N. Total load carrying capacity of CFRP patch is estimated as 428158 N. Margin of safety is computed as 1.9. Since, the margin of safety is more than 1, design is safe from strength point of view and load transfer is ensured through the strain continuity.

The repair patch is proposed to be installed using adhesive bonding and mechanical fastening. Since the repair zone is very close to the fuel tank, adhesive bonding process is preferred to integrate patch with metallic wing. However existing rivets are replaced with higher grip length in the spar plate area and few on the skin area. Since the load transfer is through adhesive bonding, the bond area is critical. So the bond length is checked and sized the external patch. The external patch through adhesive bonding has to resist a load of 142580 N. The lap shear strength obtained in coupon level testing with room temperature curable adhesive is 11MPa (Carbon- epoxy laminate and aluminum plate as adherents). In plane shear strength design allowable 10MPa is considered for the design. Hence, the lap area required is 14258 mm^2. The external patch dimension is 550 x 340mm. Lap length available = (550-210)/2=170 mm. Bonding width available = (290-50)/2=120 mm. Available bonding lap area is 20400mm^2. Since, available bonding area is 20400 > and this is greater than 14258. It is enough to resist the remaining load. Further, as a conservative approach mechanical fastening is done all around the patch, to prevent the peeling of the patch. To effective load transfer stitch riveting concept is employed here.

One of the ways to check the repair design is to check the stiffness ratio (S) of repaired structure and the attached patch, which can be computed by using the equation given below [4].

$$S = \frac{E_c}{E_p}\frac{t_c}{t_p} \tag{3}$$

Where

E_c = Young's modulus of composite patch =76 GPa
t_c = Thickness of composite patch = 8 mm (= 4.2+3.8)
E_p = Young's modulus of parent material = 70 GPa
t_p = Average thickness of parent material at repair zone = 7.1mm (= 3.8 + (6.5/2))

The stiffness ratio (S) is computed as 1.2, which is satisfying the condition of $1 \leq S \leq 1.5$; the proposed repair design is safe. Due to the adhesive bonded repair, the smooth load transfer is ensured and few bolts and rivets were used to integrate composite patch with skin. This mechanical connection is ensured the structural integrity of the wing.

Fabrication of Composite Patches

Composite patches are manufactured by using autoclave moulding process. Since there is no contour in the filler patch, it is made in the flat surface plate, where as the external patch has to match with external surface of the wing. So, the specially designed composite (wet lay-up processes) mould is used. This is helped to avoid coefficient of thermal expansion mismatch problems and also tool part interaction based spring in and spring out problems are avoided. As per design requirements, the filler and external patches are made out of unidirectional carbon fiber prepreg of 300mm width and orientation of fibers are (0,+45,-45 and 90) tailored to get required Young's modulus.

The most important in the manufacturing of composite patch is the process planning. All process parameters need to be maintained as per quality requirements and some of the process variables need to be inspected. The filler patch and external patch are fabricated as per the design lay-up sequence in respective mould and they are vacuum bagged and cured as per material manufacturer's recommendation. Temperature cycle is monitored and controlled using thermocouples at the time of curing. Cured parts are post cured as per specified post cure cycle. Fig.5 depicts, how the external patch looks with aerodynamic profile and also clearly visible the layer drops on the surface of the patch. Developed composite patches are qualified by Ultrasonic 'C' scan which scan plot was shown in Fig.6. The mechanical properties of the process are checked by using the traveler coupon which is cured along with patches. The patches are qualified in all aspects as per Aircraft Design Standards (ADS) and accepted for repair of metallic wing.

Repair Implementation

Installation of Filler Patch

Filler patch is matched in the cavity, where the wing skin has been cut to the shape of parallelogram with corner radius. While matching the filler patch, few pilot holes are transferred from the spar plate. Fig.7 shows the filler patch installation on the wing cavity. After matching, both the spar plate surface and filler patch bonding surface have been prepared for adhesive bonding since CFRP patch has a peel ply on the surface, this can be removed just before bonding. But the metallic surface is abraded using metallic wire brush and cleaned by acetone to remove any oil from surface. Alan Baker has discussed different surface preparation methods for adhesive bonding process and effects of those process also highlighted in his paper [5,6]. Room temperature curable, epoxy liquid shim (AV 138 and HV 998) is used as adhesive to bond the filler patch on the spar plate cavity and allowed to cure for 12 hours, by applying sufficient pressure from the patch (8 kg). Few rivets holes also used hold the patch in position over the wing.

External Patch Bonding

Before bonding the external patch, the filler patch pilot holes are to be transferred to the external patch. Transfer foil is prepared from the Mylar film, in which boundary of filler and external patch and location of pilot holes of filler patch and new proposed rivets for external patch are marked in different colours. The pylon location is marked on the external patch and trimmed. Vent holes of dia 1.6mm on the external patch with a centre and pitch distance of 50 mm are drilled to remove any entrapped air/ volatiles during vacuum bag bonding.

Wing surface is cleaned by brushing wheels, free from any paint / primer up to patch bonding surface [6,7]. The wing surface is also cleaned by acetone. Since the external patch has large area for bonding, good wettability VK-9 adhesive material is used. Gauss cloth of 150 micron is used at interface to maintain uniform glue line thickness. Protruding head rivets are used to locate external patch over the filler patch using pilot holes to avoid any dislocation of external patch. Pylon hole is also blocked using dissolvable material, to avoid vacuum leakage during vacuum bag bonding process. External patch is bonded by applying atmosphere pressure using vacuum bag. After curing, extra adhesive material spilled over the surfaces and edges of the patches are removed and smoothened.

Figure 8 shows the vacuum bagging over the wing surface and Fig.9 shows adhesively bonded external patch on the wing with pylon hole is blocked with dissolvable material. All pilot holes are opened and reamed to the actual rivet size. The various diameter rivets are installed at required position with interface sealants [8]. Fig.10 shows the completely repaired composite patch on the wing surface.

Qualification of Repair

Composite patches are certified as airworthy components, hence only installation need to be qualified. The filler patch and external patch bonding is qualified by ultrasonic ‘A’ scan. Bonding process is qualified by mechanical destructive testing (Lap shear test). Single lap joint specimen is prepared with aluminum and composite adherents using VK-9 adhesive and cured. The repair joint is validated by destructive and non destructive testing. Durability of the repair is validated by periodic inspection and NDT. Alan Baker described in his article about F111 lower wing repair and the author explained about fatigue testing of repaired wing using boron-epoxy composite patches. He concluded that the wing was survived for more than 5000 hours of spectrum loading with no further detectable crack growth [4]. The present wing repair was done to extend 500 flight hours only. Hence there is no fatigue studies were done as a part of qualification of this repair.

Conclusion

Due to tailorability, advanced composites are the prime candidates for repair of the damaged metallic wing. It is found that the local reinforcement of CFRP patches on the wing has shown that they can effectively transfer load. In the absence of design data, reverse engineering employed and found that this approach is able to recover the strength and stiffness of the damaged wing. The hard patch approach used to recover the wing has shown structural integrity and uniform load transfer in the repaired wing. This work has further proved that adhesively bonded CFRP patch is a viable option for repairing metallic structures. After the success of 130 hrs of snag free flight trails, certifying agency has certified the repair and permitted to implement it on all other damaged aircraft. This task has been completed successfully and all the aircraft are flying. This is an excellent example how the carbon - epoxy composite patches and adhesive bonded technology can be used to extend the life of airframes.

Acknowledgement

The author gratefully acknowledges the encouragement given by Head and Deputy Head of Advanced Composites Division and the Director of CSIR-NAL for providing various facilities to carry out this work. We are also grateful to Mr. M. Subba Rao, former Head of ACD to giving valuable guidance and support for the project. The author wish to thank the organizations like AURDC, HAL (ND), RCMA, DGAQA and their officials, especially to Mr. B. Ganesan, Chief Manager (Design) AURDC, HAL (ND) and Mr. Atul Bakare, RCMA for their excellent support and guidance. Thanks are due to Ms. V. Rashmi and Mr. H.B. Santhosh Kumar for their help preparing this manuscript. Special thanks to HAL (ND) for sponsoring the project.

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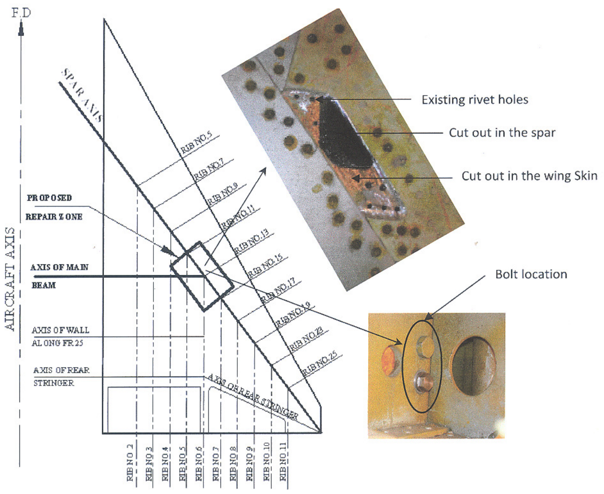

Fig.1 Wing Repair Zone with Skin and Spar Flange Cut out

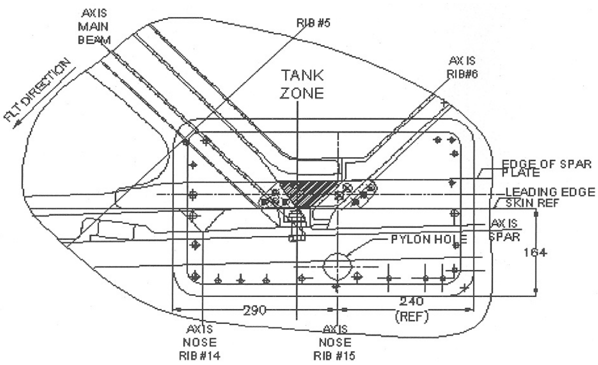

Fig.2 Interna Construction of the Wing with External Patch

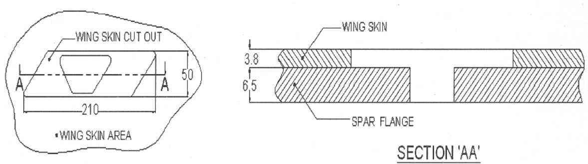

Fig.3 Cutout Wing Skin and its Sectional View

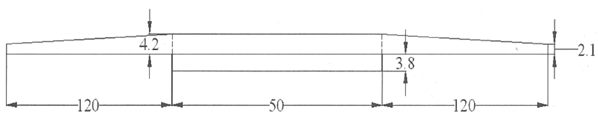

Fig.4 Section of the Repair Patch Across the Width

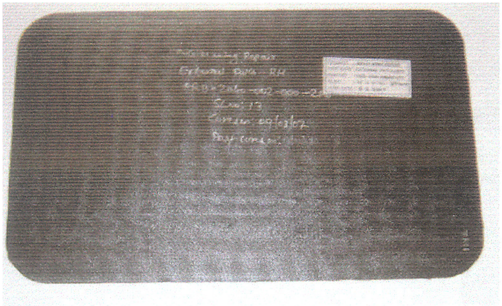

Fig.5 CFRP External Patch

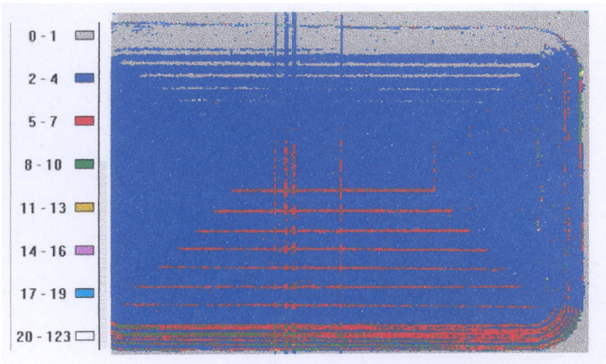

Fig.6 Ultrasonic 'c' Scan Plot of External Patch

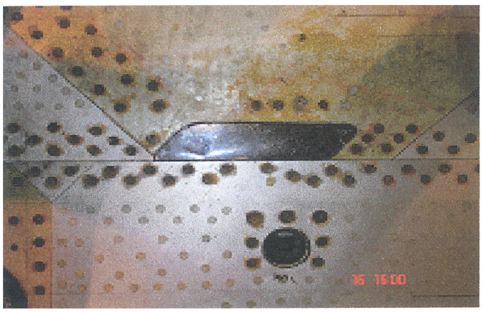

Fig.7 Filler Patch Bonded on the Wing

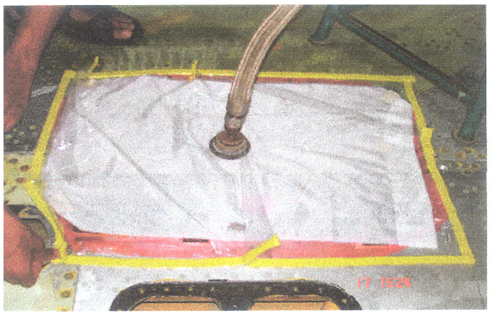

Fig.8 Vacuum Bagging on the Wing Surface

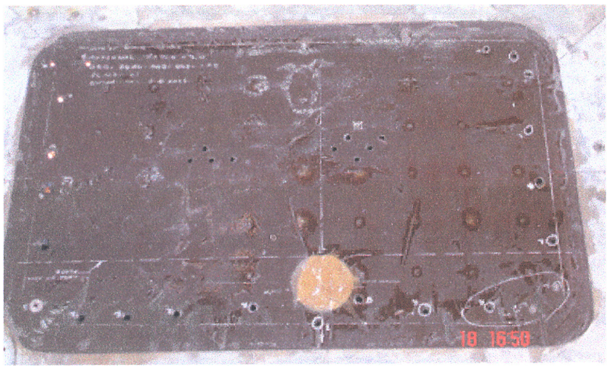

Fig.9 Adhesively Bonded CFRP Patch

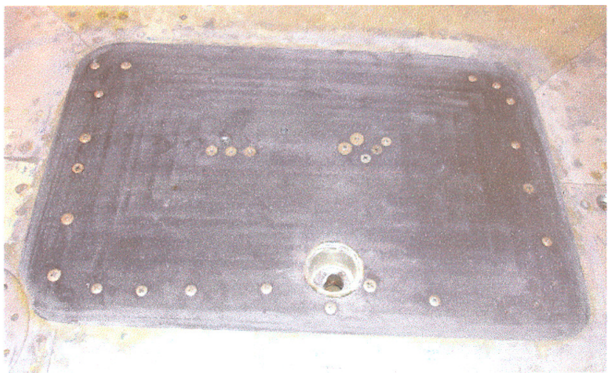

Fig.10 Repaired Wing