Review of Aviation Composite Repair in General and Design of Emergency Aircraft Repairs Using Composite Patches

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Abstract

This review paper is the outcome of study course "Engineering Mechanics of Composite Structure" completed under the guidance and teaching of Prof Dr Zaffar M. Khan, PhD (Aerospace).The ever-changing development of repair techniques of composite aircraft structures and advances in repair technology for cracked metallic structures has now significantly encompassed fatigue life extension of both military and civilian aircraft. Such developments were primarily triggered as the conventional structural repairs may significantly degrade the aircraft fatigue life and lower its aerodynamic performance. Adhesively bonded composite reinforcement is a new technology of great importance due to the remarkable advantages obtained, such as mechanical efficiency and repair time and cost reduction. In this article, these principles will be covered briefly with emphasis and examples on applications in light General Aviation composite aircraft. Likewise the bonded composite patch repairs can be designed for quick application to aircraft under emergency conditions, such as aircraft battle damage repair (ABDR). A formulated method has been developed, to be applied when damage has to be restored quickly, without restrictions to safety. However the utilization of Finite element analysis (FEA), taking into account of specific parameters of the structure under repair is most essential. Based on the FEA results, a quick design procedure using composite patch repairs for the most frequent damage cases is thereby proposed finally.

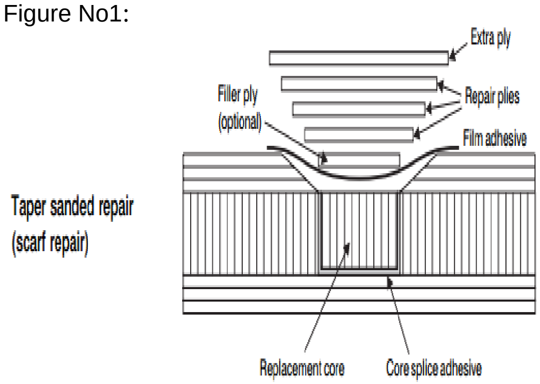
I. Introduction

A survey of literature available on the design and evaluation of aircraft repairs found little to find appreciable gap between theoretical and finite element analysis versus simpler analytical or rule of thumb solutions. In depth techniques have already been developed by the US Military, Lockheed, Boeing, and Airbus as well as more simplified techniques used by German manufacturers of light aircraft and however yet much of the industry practice appears to be based on general rules of thumb that have been proven acceptable over 40 years of industry testing and use. Likewise the dwindling world economic conditions are pressing the operation of both military and civilian aircraft well beyond their original design life, thereby calling for innovative repair techniques. The recent development of high-strength fibers and adhesives has even led to the invention of a new methodology for the repair of metallic structures, by the adhesive bonding of patches manufactured of composite materials. Primarily the bonded repairs are mechanically efficient, cost effective, and can be applied rapidly to produce an inspectable damage-tolerant repair [1- 3]. The actual objective of the repair of a cracked or corroded metallic structure by an adhesively bonded composite patch is, practically, the transfer of loads from one side of the material to the other, via the patch.

Aircraft repairs are often classified under the headings of: nonstructural, secondary structural and primary structural. The nondestructive inspection techniques that are used to examine a structure vary widely. For the purpose of this short paper all considerations will focus on obvious primary structural repairs and not hidden damage inspection.

The intent of any aircraft airframe repair is to return the structure to its original strength and stiffness as well as to keep within prescribed mass balance limitations and aerodynamic requirements. According to Baker[1] bolted repairs should not be used on laminates less than *8mm thick*. In addition the modern general aviation composite aircraft takes advantage of composites to fabricate laminar flow airfoils and smooth structures on which the use of bolted repairs would be unacceptable. This makes the bonded repair the preferred method for this discussion. The bonded repair can take the form of either an external patch, internal patch or a flush scarf repair; see Figure (1). The internal patch usually is not an option due to accessibility. Furthermore on composite control surfaces which have critical mass balance limitations, the lighter weight flush scarf repair is often the only acceptable means of repair. For these reasons the flush scarfed repair is the generally accepted method used on general aviation composite aircraft and will be the focus of this paper.

Figure No1:



II. Repair Design Concept

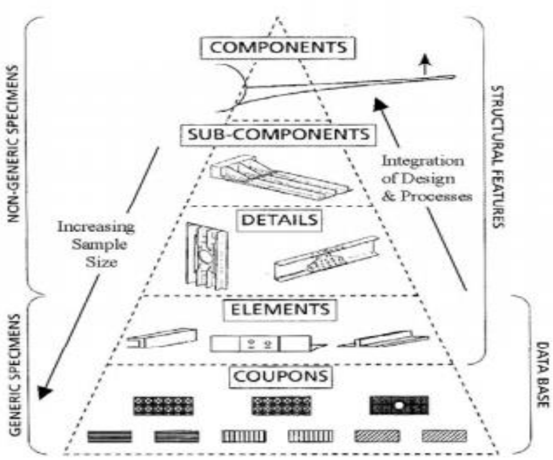
Almost all recommended procedures inform the first step in the repair of any aircraft composite damage is to identify the extent of the damage and the materials and processes used in the original part fabrication. These specifications are usually available from the original manufacturer .If original specifications are not available a more in depth engineering analysis must be done in order to quantify the design ultimate loads, fatigue and environmental exposure requirements per Federal Aviation Regulations, (FAR's) 23.305, 23.307 and 23.573.

Furthermore a practical, simple and quick way is to find the appropriate thickness and stacking sequence of the composite patch that will be applied on a damaged area, when knowledge of the detailed load spectrum normally carried is limited. The aircraft's skin thickness, the damage dimensions, and the material properties are considered as input parameters for the design of the repair. In most actual repair cases the damaged material is removed by performing a circular cut-out, in order to relieve any sharp edges that might create stress concentrations.

Critical Issues for Composite Designs: Proceeding with the modus of repair following critical issues needs to be considered:

- Integration of structural design detail with repeatable manufacturing processes ie the material and process control
- Design details, manufacturing flaws and service damage, which actually cause local stress concentration such as Strength, fatigue & damage tolerance, Dependency on tests & Scaling issues
- Environmental effects, ie Temperature & Moisture content
- Maintenance inspection and repair.

Following figure shows the hierarchy and integration of composite material:



Similarly some structural design details causing local stress concentration an d redistribution

- Bolted joints
- Bolted joints
- Doors and windows
- System provisions (penetrations and attachments)
- System provisions (penetrations and attachments) Access and drain holes
- Attachment tabs
- Stringer terminations (run-outs)
- Bonded attachments
- Ply drop-offs

Woven and unidirectional fabric materials and orientations are usually able to be duplicated. If not, an analysis by classical lamination theory should be performed to confirm that any substituted lay-up is equivalent in both modulus and strength in all loaded directions. Usually this lay-up schedule is available in the *Structural Repair Manual* (SRM like 1C-130A-3) or manufacturers' component drawings. However a technique oftenly used to identify the lay-up schedule on large repairs is to cut away a small piece of the area to be repaired and remove the resin by burning; this leaves the fiber materials behind for easier identification.

With the knowledge of the laminate characteristics known, the next step is the choice of the adhesive or resin system to be used in the repair. This will usually be dictated by the SRM but if unknown it must be chosen based on the required laminate characteristics, service temperature requirements and available process and cure temperature capabilities. Obviously the cure temperature of the resin system must not exceed the maximum exposure or glass transition temperature t_g of the component. Often the repair can be completed by a vacuum bagged repair using the original pre-preg material system as the adhesives and laminates. More often in field repairs, refrigerated storage of the pre-pregs will not be available and the repair will have to be performed with an equivalent epoxy resin system and a vacuum bagged wet lay-up.

The *Handbook of Composites*[2] provides a brief table of overlap length versus tensile shear strength for fiberglass epoxy laminate joints, reproduced on the following page in Table(1).

Table 1

Effect of Overlap Length for Fiberglass Epoxy Laminate/Epoxy Joints

		Tensile Shear Strength (psi)		
	Overlap Length	1 in	2 in	3 in
Joint	Single Overlap	1500	1000	900
Type	Double Overlap	2000	1600	600

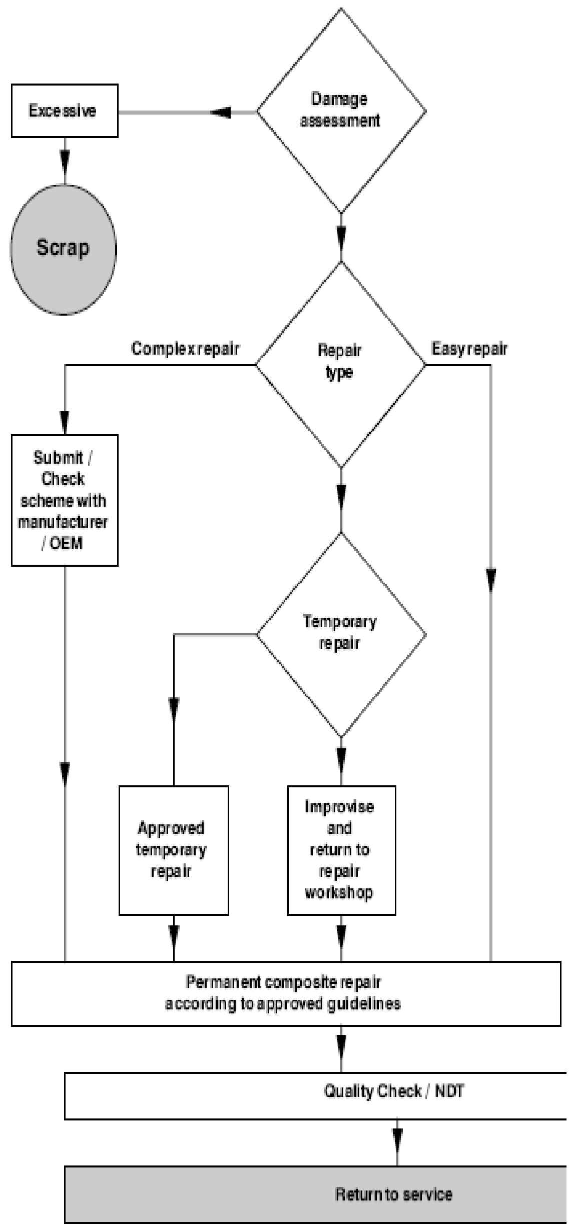
Another important point is of standard stacking sequence which is normally was defined, namely, $[0; 90; +45; -45]_s$, which gives pseudo-isotropic design properties. Patches with this stacking sequence would be able to carry both shear and tensile loads. The stacking sequence of the patch recommended to be symmetric to avoid problems of warp during manufacturing and coupling between membrane and bending effects during operation. The shape of the patch chosen to be circular, given that the actual load direction is unknown, in order to have the same strength available at all directions.

The allowable shear is not a simple function of average shear stress and falls far below the maximum shear strength F_{ms} of the resin system. Also noticeable is the average allowable shear stress goes down as the thickness of the repair and required overlap length goes up. This implies a practical limit the thickness of an acceptable bonded repair. Baker[1] suggests a limit for monolithic laminates of 10mm. This is not normally a problem in general aviation composite repair but strongly demonstrates that shear allowable values should be determined from actual structural testing. Also the above method does not allow for elevated temperature, creep loading, fatigue loading or peel stresses.

Armstrong and Barrett[4] offer repair design recommendations based on industry practices and suggest adhesively bonded lap joints not be loaded to more that 15% of shear strength F_{ms} and that scarf joints for fiberglass generally are done at taper ratios of 50/1 with additional layers overlapping the ends to account for peel stresses. Schemp-Hirth Flugzeugbau Gmbh[5] a German manufacturer of composite sailplanes recommends a scarf slope of 50/1 for unidirectional glass fibers and 100/1 for unidirectional carbon fiber or Aramid laminates.

Two different methods of scarf joint lay-ups are currently being used. In general, military specifications and Boeing use a lay-up starting with the smallest ply down first and building up to the largest ply last on the outside. Airbus and many European aircraft manufactures use a reverse method and start with the largest ply down first and the smallest ply on last. In the Airbus method the orientation of the laminate schedule must be reversed and the lay-up becomes a mirror image of the original skin. At first one would be concerned with the asymmetrical nature of the repair, but at the lower thermal stress levels of general and commercial aviation service, the asymmetry is apparently not a problem. The advantage to the Boeing method is that the peel stresses at the edge of each ply are restrained by the next layer. Regardless both methods recommend a final layer overlapping the entire repair and this adequately restrains the peel stresses and provides environmental bond sealing in the Airbus method. The major advantage to the Airbus methods is often overlooked and is in the practical nature of finish sanding a repair on a laminar flow surface. With the Boeing method the technician must not finish sand the final lay-up or the most critical larger structural plies will be damaged, compromising the repair. With the Airbus method the repair may be finish sanded in a technique known as back scarfing. This effectively fairs the repair into the surrounding surface. The final overlap layer can then be a very thin fabric which is faired into the final surface with a sandable primer returning the surface to its laminar profile.

The following flow chart displays the key stages of composite repair:



III. Recommendation

1. Different damage cases needs to be investigated such as on C-130 Aircraft Nose Radome.

2. Finite element analysis (FEA) utilization, taking into account of specific parameters of the structure under repair is most essential to finalize repair scheme.

3. Based on the FEA results, a quick design procedure using composite patch repairs for the most frequent damage cases needs to be proposed finally.

4. Concept and practical aspect of *SMART* patches needs to be studied.

IV. Conclusions

In this study lot many literature and research papers were consulted which informed that the repair of composite aircraft structures is similar to that of other advanced composite repair techniques. A design method enabling quick application of composite patch repairs can be duly developed, in order to be used in emergency conditions requiring quick aircraft repairs, such as ABDR. A table permitting quick determination of the appropriate precured patch to be used for each damage case with the knowledge of specific parameters, such as the skin's thickness and the damage's characteristic length, was composed. It should be noted that by manufacturing only three types of patches, the whole spectrum of damage cases that were examined is covered, limiting significantly the amount of required stock of prefabricated composite patches.

References

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