

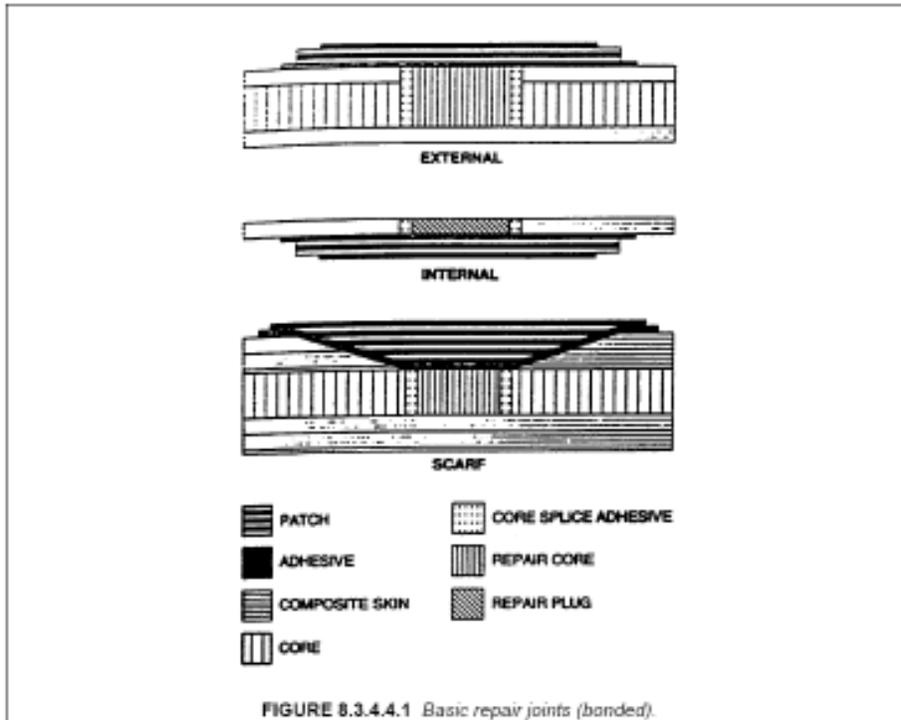
# General Aviation Composite Repair

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The repair of composite aircraft structures is similar to that of other advanced composite repair techniques. These principles will be covered briefly with emphasis and examples on applications in light General Aviation composite aircraft. This type of aircraft structure repair differs from other composite repairs in that skin thicknesses are relatively thin and usually of sandwich construction. With all aircraft repairs closer attention is usually paid to the balance between weight and safety factors, aerodynamics, quality control, and substantiation. A survey of literature available on the design of aircraft repairs found little to bridge the gap between theoretical and finite element analysis versus simpler analytical or rule of thumb solutions. In depth techniques have been developed by the US Military, Boeing, and Airbus as well as more simplified techniques used by German manufacturers of light aircraft and 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.

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. Although composite repairs may be of either a bolted or bonded nature; bolted repairs are not generally acceptable on thin laminates or sandwich structures due to the difficulty withstanding bearing loads induced by mechanical fasteners. According to Baker<sup>1</sup> 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) from *MIL-HDBK-17-3F*<sup>3</sup>. The internal patch usually is not an option due to accessibility. For simplicity the external lap is commonly used on internal component repairs such as formers, bulkheads and inner skins but for the reason of aerodynamic cleanliness as well as to minimize moment induced failure modes the flush scarf repair is preferred. Further more 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 1**

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. Generally though the specifications are known but the processes may not be duplicatable in a field situation. For example a part which was manufactured with a pre-impregnated resin system cured under high pressure in an autoclave may not be repairable in this fashion unless it can be disassembled and transported to a facility with the capabilities. Even if it could be transported to an autoclave it may not be feasible for the completed component with subassemblies to be exposed to the temperatures and pressures of the autoclave without being totally disassembled and supported within the manufactures original tooling. For this reason repairs are normally conducted with only the use of vacuum induced pressures and localized heating. Due to slightly less fiber volume ratio, the mechanical strength of the vacuum bagged lay-up will not be equivalent to the higher pressure laminate, and may require compensating the repair laminate with an additional ply. For the most part current general aviation composite manufacturing is using only vacuum bagged pre-pregs and wet lay-ups so this does not become an issue.

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) or manufacturers component drawings. A technique often 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.

With the resin system chosen the scarf joint can be designed. A design analysis of a scarf joint resembles that of a single lap bonded joint. Detailed analysis of which can be found in the references *Bonded Repair of Aircraft Structures*<sup>1</sup> or *MIL-HDBK 17-3F*<sup>3</sup>. The analysis of a bonded joint is made complex by the modulus difference of the adhesive compared to the adherends and the relative thicknesses of both which causes a nonlinear distribution of the shear forces in a lap joint with peak stresses at the ends. This is beyond the scope of this paper. Baker<sup>2</sup> offers a much simplified analysis of a scarf joint based on simple equilibrium stress transformation to relate the maximum allowable shear stress  $\tau$  to the scarf angle  $\theta$  by

$$\tau = \frac{P \sin 2\theta}{2t} \quad \text{or} \quad \theta = \frac{1}{2} \sin^{-1} \left( \frac{2t\tau}{P} \right)$$

where  $P$  is the load per unit width and  $t$  the laminate thickness. This formula could be modified to relate to more readily available material properties to yield

$$\theta = \frac{1}{2} \sin^{-1} \left( \frac{2F_{ms}}{F_{lt}SF} \right)$$

where  $F_{ms}$  is the shear strength or shear allowable of the adhesive or resin system,  $F_{lt}$  is the longitudinal strength of the lamina or  $F_{xl}$  for a laminate as the case may be and  $SF$  the safety factor.

Applying this formula to a unidirectional E-glass/ Epoxy repair with  $F_{ms}$  of 7.5 ksi and  $F_{lt}$  of 165 ksi with a safety factor of 2 would yield

$$\theta = \frac{1}{2} \sin^{-1} \left( \frac{(2)(7.5\text{ksi})}{(165\text{ksi})(2)} \right) = 1.3^\circ \approx 44/1 \text{ scarf ratio}$$

Applying the formula to a commonly used bidirectional woven fiberglass fabric 7781 with a warp tensile strength of only 53 ksi yields

$$\theta = \frac{1}{2} \sin^{-1} \left( \frac{(2)(7.5\text{ksi})}{(53\text{ksi})(2)} \right) = 4.07^\circ \approx 14/1 \text{ scarf ratio}$$

Both these scarf ratios would be slightly inadequate in comparison with industry practices. The problem with the above formula or any other simplified analysis is determination of the allowable shear stress for the overlap. As previously mentioned the shear distribution in the joint is not linear and the shear stress distribution will vary with the adherend laminate and processing. The *Handbook of Composites*<sup>2</sup> 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		.5 in.	1 in.	2 in.	3 in.	4 in.
Joint Type	Single Overlap	1500	1000	900	650	550
	Double Overlap	2000	1600	600	450	na

As can be seen from Table (1), shear strength allowables are also dependent on joint design and overlap length. 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 to the thickness of an acceptable bonded repair. Baker<sup>1</sup> 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<sup>4</sup> offer repair design recommendations based on industry practices and suggest adhesively bonded lap joints not be loaded to more than 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<sup>5</sup> 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. Table (2) is compiled from a Schemp-Hirth Repair Manual<sup>5</sup> and shows the scarf requirements for commonly used European repair fabrics with epoxy resins.

**Table 2**

Fabric Designation	Manufacturer	Fiber/Weave	Weight (g/m <sup>2</sup> ) (without resin)	Thickness (mm) (with resin)	Length (mm) scarf joint	Scarf Angle
92110	Interglas AG	E-glass 2x2 Twill	163	0.18	5	2.06° 28/1
92125	Interglas AG	E-glass 2x2 Twill	276	0.30	10	1.72° 33/1
92140	Interglas AG	E-glass 2x2 Twill	390	0.43	15	1.64° 35/1
92145	Interglas AG	E-glass Warp (Uni)	220	0.24	15	0.916° 62.5/1
98140 CF 200	Interglas AG	Carbon Plain	200	0.30	15	1.15° 50/1
98160 CF 285	Interglas AG	Carbon Plain	285	0.43	25	0.985° 58/1
98340	Interglas AG	Carbon Warp(Uni)	170	0.25	15	0.955° 60/1
98355	Interglas AG	Aramid Carbon	200	0.35	15	1.34° 43/1

The scarf joint itself is normally prepared in the process of grinding out the damaged area as required. Either taper sanding or step sanding the individual plies is acceptable. In final preparation for the lay-up a solvent wipe is performed followed by water break test. In the water break test, deionized water is lightly sprayed on the surface and should flow out smoothly. If the water beads up, the repair area is contaminated and further preparations are required. The area is then dried in final preparation for the lay-up. The water break test must be carefully controlled in areas near sandwich cores to avoid moisture egress into the core. Surface preparation before the repair lay-up is of primary importance, the bond areas must be abraded, clean and dry.

If the repair is to a sandwich construction the inner skin and core are repaired in a first step. A nonstructural backing may have to be improvised to support the inner lay-up during cure. Often the inner skin is repaired with a simple external lap and replacement core co-cured in place with potting compound around the perimeter. Core replacements should be repaired with similar materials and may be of butt joint for high density foam cores. If the core is honeycomb the replacement should be of the same material with matching cell size and orientation. Large areas of core replacement with potting compound should be avoided to keep from inducing a failure due to increased local stiffness. The inner skin must be cured at elevated temperatures before repairing the outer skins.

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 hygrothermal 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.

Most repairs specifications will require curing under vacuum induced pressure. A typical vacuum bag schematic is shown in Figure(2) from *MIL-HDBK 17-3F*<sup>3</sup>. The best vacuum bag schedule will vary from one repair lay-up to another. Different repair lay-ups will require different bleeder ply schedules. If the SRM does not call out a precise bagging schematic, it may be necessary to lay-up test panels to determine that the proper amount of resin bleed is achieved. Too much bleed out will cause a weak dry lay-up with high void content, inadequate bleeding or breathing of the bag will cause trapped resin rich areas with uneven fiber volume.

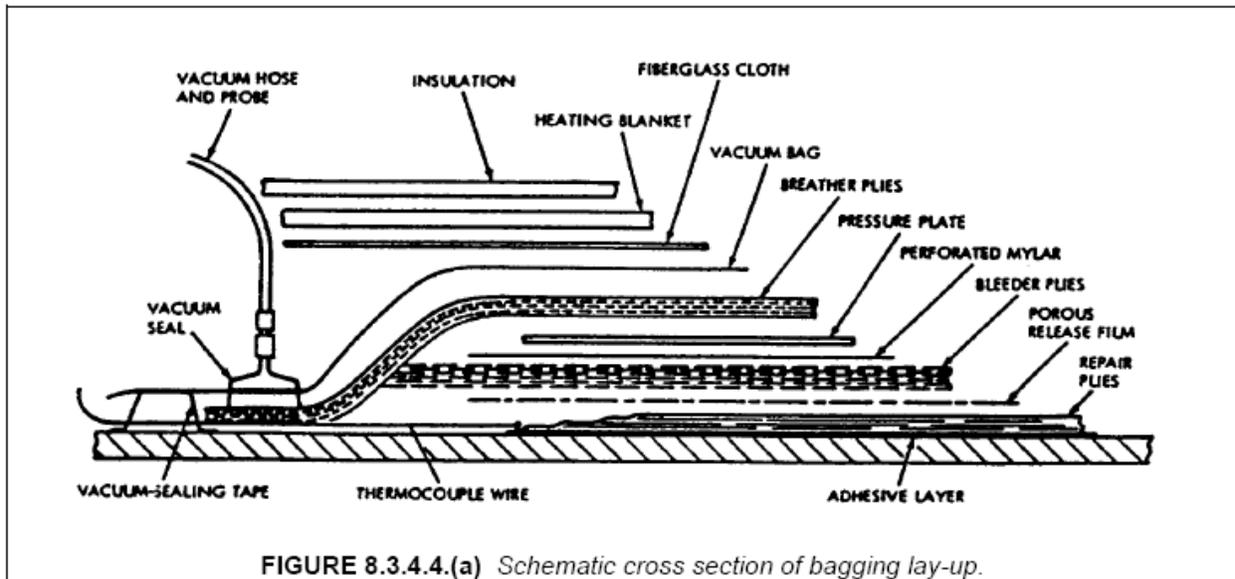


Figure 2

Depending on the resin system used the repair will require curing at elevated temperatures. These could be as low as 50-60°C (122-140F) for some room temperature wet lay-up resins to 350°F for some pre-preg systems. Each resin system will have a recommended temperature profile for the cure. Following this profile while under vacuum is important for proper out gassing of trapped volatiles and resin bleed out. Too fast of a tack cure can cause trapped volatiles which can result in voids, too slow can result in excessive resin bleed out and a dry laminate. In the case of flat laminates, heat is usually applied by means of temperature controlled electric heating blankets. Complex shaped repairs may require custom fabrication of temporary forced air ovens. In order to substantiate the repair quality a process panel should be fabricated within the repair vacuum bag and cured simultaneously. When the repair is completed the process panel can be tested for fiber volume, void content and glass transition temperature to verify the process results.

The following series of photographs are of a repair performed by the author, Larry Mansberger, on a modern general aviation aircraft. The aircraft is an all composite Columbia 350 originally manufactured by Lancair Certified Aircraft and now by the Cessna Aircraft Company. The damage was the result of a door being opened in flight becoming detached, striking the wing and the tail section on the right side and causing compression buckling on the left side of the aft fuselage skins. The repair specification was written by myself from review of original manufacturing drawings and approved by a FAA Designated Engineering Representative.

The areas of the aircraft being repaired were manufactured from vacuum bagged lay-ups of Hexcel 7781 8H satin woven fiberglass cloth pre-impregnated in a 250°F epoxy resin system. The resin system specified for the repair is MGS 418 epoxy. MGS 418 is a wet lay-up laminating resin which when post cured for 10 hours at 210°F yields a glass transition temperature in excess of 250°F with tensile and compressive strengths within tolerances of the original system.

Photo (1) shows the wing skin impact damage with core removed and an external lap repair on the inner skin. The repair laminate was [45/0/0/45/.250 honeycomb/45/45].



**Photo 1**

Photo (2) shows core replacement with process panel ready for outer skin lay-up.



**Photo 2**

Photo (3) shows the final repair under vacuum. Heat lamps were used for initial cure, followed by post cure at 200°F after removal of vacuum.



**Photo 3**

Photo (4) shows the left side of the aft fuselage with the damaged outer skin and honeycomb core removed. The inner skin had only minor damage and remained intact.



**Photo 4**



**Photo 5**

Photo (5) shows the honeycomb core replaced.

Photo (6) shows the completed scarf mapped out on clear bagging film. This will have fiber orientation marked for use as a cutting template of the individual 7781 plies. The actual repair laminate was [45/0/45], plus an additional localized reinforcement [45/45/45] from 5.5" below to 6.5" above horizontal stabilizer



**Photo 6**

Photo (7) shows one method of building a temporary forced air oven from a high temperature urethane foam box. This was used for post curing the tail repair at 200°F.



**Photo 7**

Photo (8) shows the repair after back-scarfing and initial primer surfacer application.



**Photo 8**

Photo (9) shows the completed repair after painting and reinstallation of stabilizer.



**Photo 9**

## References

- 1) Baker A.A., Jones R. *Bonded Repair of Aircraft Structures* , Martinus Nijhoff 1988
- 2) Lubin G. *Handbook of Composites* , 1<sup>st</sup> ed. Van Norstrand Reinhold 1982
- 3) MIL-HDBK-17-3F *Composite Materials Handbook* US Department of Defense 2002
- 4) Armstrong K.B., Barrett R.T. *Care and Repair of Advanced Composites* SAE 1998
- 5) *Repair Instructions for Sailplanes* Schemp-Hirth Flugzeugbau GmbH