CHAPTER 8 SUPPORTABILITY

8.1 INTRODUCTION

Supportability is an integral part of the design process that ensures support requirements are incorporated in the design and logistics resources are defined to support the system during its operating or useful life. Support resource requirements include the skills, tools, equipment, facilities, spares, techniques, documentation, data, materials, and analysis required to ensure that a composite component maintains structural integrity over its intended lifetime. When the load carrying capability of an aircraft, or product is compromised, (i.e., loss of design function), the damaged structure must be restored quickly and at low cost. Customer requirements can dictate maintenance philosophy, materials availability, and repair capabilities that a design team must incorporate throughout the design process. As the contributors to this chapter were primarily from the aircraft industry, the text is slanted towards its particular needs. However, the guiding principles can be beneficial in other composite applications.

Since the operating and support cost of a vehicle continues to escalate throughout its life, it becomes imperative to select and optimize those designs that maximize supportability. Life cycle cost, being comprised of research and development, acquisition, operational and support, and disposal costs, is often a crucial customer requirement for any new weapon system or commercial transport. Often, design changes that enhance producibility, improve vehicle availability, and reduce operational and support costs, far outweigh the short-term increases in acquisition costs. Lost airline profits and reduced wartime readiness are a direct result of designs that did not incorporate supportability early in the design process. Telltale indicators of non-supportable designs include expensive spares, excessive repair times, and unneeded inspections.

Aircraft users are often constrained to perform maintenance during aircraft turnaround, after each day's usage, and during scheduled maintenance. Repair time limitations can range from several minutes to several days. In each case users of aircraft containing composite components require durable structures that, when damaged, can be repaired within the available support infrastructure including skills, materials, equipment, and technical data.

Composite designs are usually tailored to maximize performance by defining application dependent materials, ply orientation, stiffening concepts, and attachment mechanisms. High performance designs are often less supportable due to increased strain levels, fewer redundant load paths, and a mix of highly tailored materials and geometries. Product design teams should focus on a variety of features that improve supportability including compatibility of available repair materials with those used on the parent structure, available equipment and skill, improving subsystem accessibility, and extended shelf-life composite repair materials. Structural elements and materials should be selected that are impervious to inherent and induced damage especially delaminations, low velocity impacts, and hail damage. Each supportability enhancement feature results from the designer having an explicit knowledge of the aircraft's operational and maintenance environment and associated requirements and characteristics. Other design considerations also have an impact on supportability including durability, reliability, damage tolerance, and survivability. A supportable design integrates all the requirements, criteria, and features necessary to provide highly valued products in terms of performance, affordability and availability.

This section is designed to assist integrated product teams in the development of supportable products through five basic sections: 1) Introduction - which provides an overview of the Supportability chapter; 2) Design for Supportability - which provides the designer with design criteria, guidelines and checklists to ensure a supportable design; 3) Support Implementation - which defines and demonstrates those key elements of supportability that must be performed to insure mission success; 4) Composite Repairs of Metal Structure – which provides an alternative means to standard metal repair options, and (5) Logistics Requirements - which establishes the support resources needed to maintain the backbone of the support structure. Each section provides the designer and aircraft user with the supportability data and lessons learned that will reduce cost of ownership and improve aircraft availability. Other sections throughout MIL-HDBK-17 discuss the details needed to design supportable components. Sections contained in Volume 1 include material and structural testing, material types and properties, and joint types; in Volume 3 include materials and processes, quality, design, joints, reliability, and lessons learned needed to supplement those decisions that influence supportability.

8.2 DESIGN FOR SUPPORTABILITY

8.2.1 In-service experience

The first step toward designing reliable and cost-effective design details is to understand the history of composite structure. Composite materials, as we know them today, were introduced into the commercial aircraft industry during the early 1960's and used mostly glass fiber. Development of more advanced fibers such as boron, aramid, and carbon offered the possibility of increased strength, reduced weight, improved corrosion resistance, and greater fatigue resistance than aluminum. These new material systems, commonly referred to as advanced composites, were introduced to the industry very gradually and cautiously to ensure their capabilities.

The early success of the first simple components, such as wing spoilers and fairings, led to the use of advanced composites in more complex components such as ailerons, flaps, nacelles, and rudders. The increased specific stiffness and strengths of composites over aluminum, coupled with weight-driven requirements caused by fuel shortages, led to the application of thin-skin sandwich structures. Long-term durability requirements of the original aluminum parts were not fully accounted for when these composite parts were originally designed. To compound the problem further, damage phenomena such as delamination and microcracking were new and complex in comparison to traditional aluminum structure.

The original composite parts, particularly thin-gage sandwich panels, experienced durability problems that could be grouped into three categories: low resistance to impact, liquid ingression, and erosion. These parts were either control panels or secondary structure, such as fixed trailing edge panels, and given the emphasis placed on weight and performance, the face sheets of honeycomb sandwich parts were often only three plies or less with a Tedlar™ film. This approach was adequate for stiffness and strength, but never considered the service environment where parts are crawled over, tools dropped, and where service personnel are often unaware of the fragility of thin-skinned sandwich parts. Damages to these components, such as core crush, impact damages and disbonds, are quite often easily detected with a visual inspection due to their thin face sheets. However, sometimes they are overlooked, or damaged by service personnel, who do not want to delay aircraft departure or bring attention to their accidents, which might reflect poorly on their performance record. Therefore, damages are sometimes allowed to go unchecked, often resulting in growth of the damage due to liquid ingression into the core. Non-durable design details (e.g., improper core edge close-outs) also led to liquid ingression.

The repair of parts due to liquid ingression can vary depending upon the liquid, of which water and Skydrol (hydraulic fluid) are the two most common. Water tends to create additional damage in repaired parts when cured unless all moisture is removed from the part. Most repair material systems cure at temperatures above the boiling point of water, which can cause a disbond at the skin-to-core interface wherever trapped water resides. For this reason, core drying cycles are typically included prior to performing any repair. Some operators will take the extra step of placing a damaged but unrepaired part in the autoclave to dry so as to preclude any additional damage from occurring during the cure of the repair. This is done to assure they will only need to repair the part once. Skydrol presents a different problem. Once the core of a sandwich part is saturated, complete removal of Skydrol is almost impossible. The part continues to weep the liquid even in cure such that bondlines can become contaminated and full bonding does not occur. Removal of contaminated core and adhesive as part of the repair is highly recommended.

Erosion capabilities of composite materials have been known to be less than that of aluminum and, as a result, their application in leading edge surfaces has been generally avoided. However, composites have been used in areas of highly complex geometry, but generally with an erosion coating. The durability and maintainability of some erosion coatings are less than ideal. Another problem, not as obvious as the first, is that edges of doors or panels can erode if they are exposed to the air stream. This erosion can be attributed to improper design or installation/fit-up. On the other hand, metal structures in contact or in the vicinity of these composite parts may show corrosion damage due to:

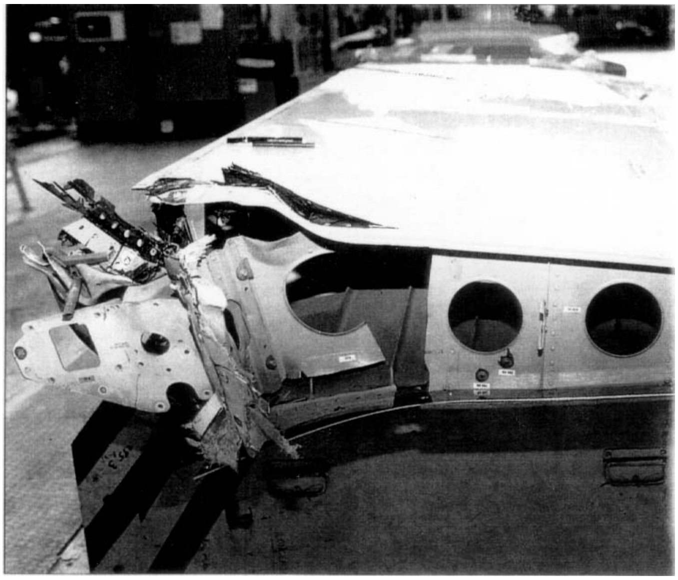
- Inappropriate choice of aluminum alloy
- Damaged corrosion sealant of metal parts during assembly or at splices
- Insufficient sealant and/or lack of glass fabric isolation plies at the interfaces of spars, ribs and fittings

Assessing operator experience with composite structure is, taken as a whole, an extremely difficult task. A survey of operators provides responses depending on the composite application ranging from horror stories for thin skinned sandwich structures, to outstanding success for thick skinned sandwich or solid laminate primary structures. Some of the facts and data that are available are the detailed reports that were received from the operators on parts involved in the NASA-sponsored Advanced Composites Energy Efficiency (ACEE) program, which supported the design and fabrication of composite parts such as the B727-200 elevators and the B737 spoilers and horizontal stabilizers. Five shipsets of B727 elevators have accumulated more than 331,000 hrs. and 189,000 cycles; 108 B737 spoilers have accumulated more than 2,888,000 hrs. and 3,781,000 cycles. Five shipsets of B737 horizontal stabilizers, which incorporated laminate torque boxes and sandwich ribs, had amassed over 133,500 flight hours and 130,000 landings as of May, 1995. The service exposure data collected for these parts have not indicated any durability or corrosion problems. One B737-200 aircraft with the ACEE stabilizers was removed from service after 19,295 flight cycles and 17,302 flight hours, and one stabilizer was acquired by Boeing for a detailed tear-down inspection. The stabilizer was found to be in excellent condition with no fatigue damage, and the only corrosion discovered was some minor pitting found in some fastener holes of the aluminum trailing edge fittings. This was determined to be due to a fastener sealing practice which has since been obsoleted. Several repairs have been satisfactorily performed on the 727 elevators and remaining 737 horizontal stabilizers which are still in service.

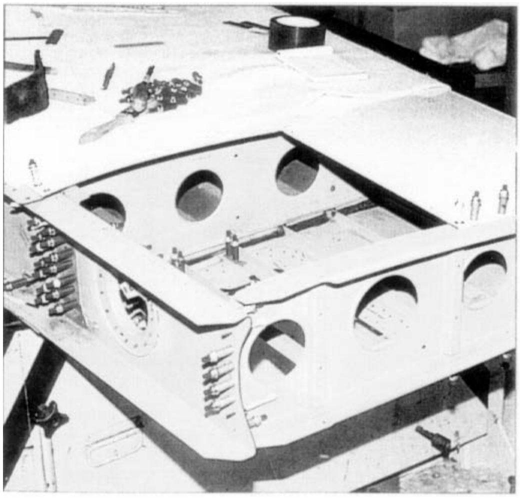
The in-service success of these ACEE components is in part due to the integrated teams which developed them. The teams for both the B727 sandwich elevators and the B737 stiffened-skin configured horizontal stabilizers considered maintainability during the developmental programs. They devised repair and inspection schemes, and for each component, Maintenance Planning Manuals were compiled and released as part of the NASA contractual obligation. The airlines, United for the ten B727 elevators, and Delta and Mark Air for the five shipsets of B737 stabilizers, were in essence part of the teams who planned these documents. As mentioned above, both of these components have been damaged and repaired using the repair schemes designed for them. In all of the instances, the repairs were satisfactorily performed in-place on the aircraft.

An in-service evaluation, launched in 1980, with twenty-two airbrakes/spoilers (14 fabricated with carbon-epoxy tape, and 8 fabricated from carbon-epoxy fabric) installed on Air France A300 aircraft, is still going on. Non-destructive inspections (visual and ultrasonic) are performed on aircraft and in the laboratory during the service life. Thirteen airbrakes are still on aircraft, and seven have been withdrawn from service for testing to assess stiffness and residual strength. As of November, 1995, these components had accumulated 405,698 flight hours and 236,588 flight cycles. The component with the most time in service had accumulated 32,069 flight hours and 16,802 flight cycles. Bolted repairs (metal patches for temporary, and composite precured patches for permanent repairs) were designed. Two components have been repaired with blind fasteners to arrest manufacturing produced disbonds between the skins and ribs. Some minor corrosion pitting was found on the aluminum (7075) spar at the central fitting splice due to the protect scheme having been damaged and not restored during assembly. A modification of the trailing edge was implemented early in the program; the rubber one being replaced by a solid carbon one.

As an example of successful thicker solid laminate structure, the ATR 72 outer wing box has accumulated 1,429,539 flight cycles and 1,163,333 flight hours since entering into service in 1989. The aircraft with the most time service has accumulated 23,343 flight cycles and 14,988 flight hours. Service experience has been very good with only one accidental damage being reported; an aircraft crashed into a hangar door at a speed of 15 miles/hr (25 km/hr). The composite outer wing box was repaired using bolted carbon-epoxy and metal patches (see Figure 8.2.1(a)), while all the metal parts of the center box were replaced due to permanent deformations. One aircraft exhibited erosion of the outer ply at the leading edge of the upper skin, and a chamfer was introduce in the design, and no other problems have been reported.



a) Damaged ATR 72 carbon outer wingbox: carbon front spar and carbon wing skins



b) Repaired carbon spar before repair of wing skins

FIGURE 8.2.1(a) *Repairs to badly damaged ATR 72 wing.*

Production carbon-epoxy sandwich parts, such as trailing edge panels, cowls, landing gear doors, and fairings have demonstrated weight reduction, delamination resistance, fatigue improvement and corrosion prevention. The poor service records of some parts can be attributed to fragility, the inclusion of non-durable design details, poor processing quality, porous face sheets (insufficient thickness), and badly installed or poorly sealed fasteners. Many of the design problems were a result of insufficient technology transfer from development programs such as NASA-ACEE. Many flight control and secondary structural panels were designed using composite materials without consideration of the applicability of composites and the service environment. As a result, many components were designed around weight-efficient sandwich configurations with face sheets of only two or three plies. Not only is the damage resistance of these components poor, but they are difficult to seal from fluids.

Fragility, so much an issue in these thin-gage sandwich structures, is much less an issue in thicker-gage primary structures (sandwich or solid laminate) such as main torque boxes of empennages and wings, and fuselages. The thicker skins of the Boeing B777 and the Airbus series of composite empennage main torque boxes, the US Air Force B-2 wings and fuselage, the ATR 72, US Air Force F-22, the US Navy/Marine F/A 18, and the RAF/Royal Navy/US Marines AV8B Harrier outboard wing boxes, to list a few examples, are much more damage resistant than the vast number of light gage sandwich flight control and secondary structural components that are currently in service, and are still being introduced to service. In addition to their highly damage resistant primary structural components, the latest Airbus aircraft and the B777 have incorporated improved light gage composite structural designs. However, they will still be more vulnerable to damage in service than the primary structural components, due to their minimum gage, mainly sandwich, construction.

8.2.2 Inspectability

During the design of composite structural components consideration should be given to the inspection methods available to both the manufacturer and the customer. Typical composite in-process non-destructive inspection (NDI) methods available to the manufacturer are: visual, through-transmission ultrasonics (TTU), pulse-echo ultrasonics, x-ray, and other advanced NDI methods such as enhanced optical schemes and thermography. Most airlines and military operators use visual inspections supplemented with both mechanical (i.e., some form of tap test) and electronic (i.e., pulse echo and low-frequency bond testing) to locate damage. Because of the predominance of visual inspections, provisions should be made during the design phase for complete external and internal access for visual inspection of all components, regardless of whether they are critical primary structural components or secondary structures such as fairings. If a visual inspection indicates potential damage, then the more sophisticated inspection techniques can be used to provide more accurate damage assessments. Additional suggestions can be found in Section 8.3.1.

8.2.2.1 General design guidelines

Whether one chooses a laminate stiffened skin or a sandwich configuration for a specific component, there are inspectability issues within each configuration category. For example, the use of closed hat stiffeners to stiffen laminate skins, while extremely efficient from a structural point of view, create three areas in the skin and stiffener that are difficult to inspect by any method (Figure 8.2.2.1(a), section (a)). A blade stiffener, on the other hand, has only the one difficult inspection area (Figure 8.2.2.1(a), section (b)). The adhesive fillets of the closed-hat stiffener, and the rolled noodle of the blade stiffener, are contributors to these inspection difficulties. These areas are difficult to inspect during the manufacturing process, and are even more of a problem for the service operator with limited access to the internal surfaces.

With a sandwich configuration there are inspection difficulties associated with potted areas, detection of fluids that have leeched into the sandwich honeycomb core, disbonds of face sheets, foam core, and damages within the core. Also difficult for operators are inspections of bondlines of stiffeners or frames that are bonded to the internal face sheets of sandwich components (Figure 8.2.2.1(b)). When airplane operators are forced to use inspection methods that are subjective, i.e., the tap test, they are handicapped by lack of knowledge of damage sizes and criticality. This is a significant problem for operators, and while sandwich structural configurations can be very efficient from a performance point of view, they tend to be fragile, easily damaged, and difficult to inspect. Interestingly some airline operators prefer sandwich over laminate stiffened skins from a repair point of view, but virtually all express frustration with the durability and inspection of sandwich structures.

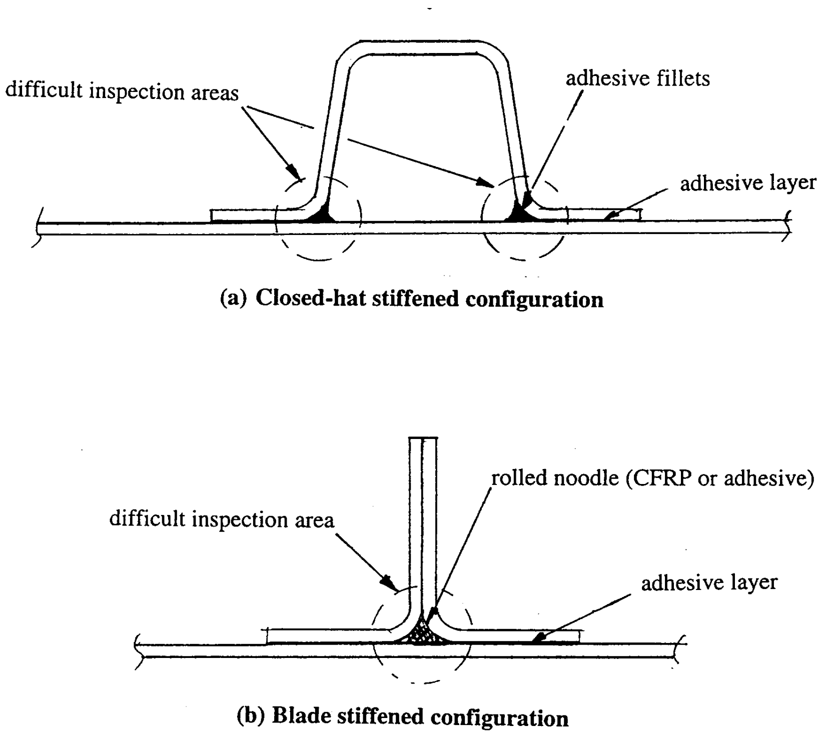


FIGURE 8.2.2.1(a) *Difficult to inspect areas on laminate skin stiffened designs.*

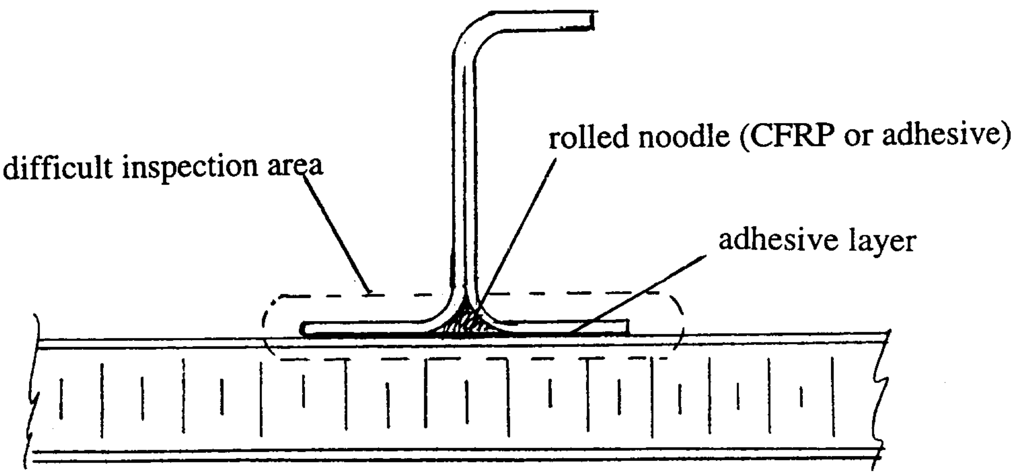


FIGURE 8.2.2.1(b) *Difficult inspection area of sandwich structural configurations.*

Most composite structural components will include metal fittings or interfaces with metal parts. It is desirable to ensure that these metal parts can be visually inspected for corrosion and/or fatigue cracking. In addition, if the mating metal parts are aluminum, then it is important to be able to inspect them for potential galvanic corrosion that may be caused by contact with the carbon fibers. This may require removal of fasteners at mating surfaces, so blind fasteners should not be used in these applications. The use of blind titanium fasteners should be kept to a minimum because, when installed, they are literally impossible to inspect to verify correct installation. They are also very difficult to remove when repairing or replacing a component.

8.2.2.2 Accessibility for inspection

Composite structural components should not be designed such that they must be removed in order for inspections to be made. Some disassembly may be unavoidable, but should be kept to a minimum. This will not only reduce the maintenance burden on the operators, but also reduce airplane out-of-service time.

All composite components should be designed to ensure visual accessibility of the external surfaces without detaching any parts, including access panels, from the airplane. In some instances, fairing panels may have to be removed, such as the horizontal stabilizer-to-fuselage fairing for access to the stabilizer skin joints-to-side-of-body rib, or spar-to-center-section attachments.

An internal inspection implies that there is visual accessibility that is achieved by removal of detachable parts, such as access plates or panels. For internal inspection of torque boxes with ribs, spars and stringers, there must be complete visual accessibility through access holes in spars and ribs. These access holes must be designed such that maintenance technicians can, through the use of flashlights and mirrors, visually inspect all of the internal structure. There must also be accessibility to critical joints or attachment fittings where pins can be removed so that they and the holes can be inspected.

8.2.3 Material selection

8.2.3.1 Introduction

Chapter 2 in Volume 3 offers an in-depth review of advanced composite materials. Each one of the composite materials described in Chapter 2 can offer benefits over metallic materials to the designer in terms of performance and costs. However, these benefits will be erased if, when designing a component, the design is focused only on the mechanical and thermal performance of the component and does not take into consideration where the part will be used and how it will be repaired if it is damaged. The goal of the designer must be to design a part that will be both damage tolerant and damage resistant as well as easy to maintain and repair. This section is offered as a guideline for the designer when selecting a material system.

8.2.3.2 Resins and fibers

When selecting a resin, it is important to look at where the resin system will be used, how the resin system has to be processed, what is its shelf life and storage requirements, and is it compatible with surrounding materials. Table 8.2.3.2 describes the common resin types, their process conditions and their advantages and disadvantages in terms of repairability. An in-depth review of these materials can be found in Section 2.2.

Refer to Section 2.3 for available fibers for composite structures.

In terms of supportability, the minimum number of resin systems and material specifications should be chosen. This will reduce the logistic problems of storage, shelf life limitations and inventory control.

TABLE 8.2.3.2 *Supportability concerns with resin types.*

Resin Type	Cure Temp. Ranges	Pressure Ranges	Processing Options	Supportability Advantages	Ease of Repair	Damage Resistance	Supportability Disadvantages
Epoxy Non-Toughened	RT to 350°F (180°C)	Vacuum to 100 psi (690 kPa)	Autoclave, press, vacuum bag, resin transfer molding	Low level of volatiles, low temp processing, vacuum bageable	Good	Poor	Time limited storage
Epoxy -Toughened	RT to 350°F (180°C)	Vacuum to 100 psi (690 kPa)	Autoclave, press, vacuum bag and resin transfer molding	Low level of volatiles, low temp processing, vacuum bageable	Good	Good	Time limited storage
Polyester	RT to 350°F (180°C)	Vacuum Bag to 100 psi (690 kPa)	Same as epoxies	Ease of processing, quick cure with elevated temp., low cost	Very good	Good	Poor elevated temp performance, health (Styrene)
Phenolic	250 to 350°F (120 to 180°C) with post cure	Vacuum Bag to 100 psi (690 kPa); lower pressure gives high void content	Autoclave, press molding		Poor	Poor	Water off gassing, high temp cure/post cure, high void content
Bismaleimides (BMI)	350F (180°C) with 400 to 500°F (200 to 260°C) post cure required	45 to 100 psi (310 to 690 kPa)	Autoclave, press molding, RTM	Lower pressure processing than polyimides	Poor	Poor	High temperature processing
Polyimides	350 to 700°F (180 to 370°C) post cure required	85 to 200+ psi (590 to 1400+ kPa)	Autoclave and press molding		Poor	Poor	Cost, availability of adhesives, high pressure
Structural Thermoplastic	500°F+ (260°C+)	Vacuum bag to 200 psi (1400 kPa)	Autoclave and press molding	Reformable	Poor	Very good	High temperature processing

8.2.3.3 Product forms

A detailed description of available composite product forms can be found in Section 2.5.

The goal when repairing a composite part is to return it to its original performance capability while incurring the least cost and weight gain. Therefore, the ease of repairing different product forms should be taken into consideration when selecting the material system. Figure 8.2.3.3 shows the relative ease of repairing various product forms.

8.2.3.4 Adhesives

Table 8.2.3.4 provides descriptions of issues for use of adhesives in repairs.

8.2.3.5 Supportability issues

Table 8.2.3.5 offers a list of Material Support issues for your consideration.

8.2.3.6 Environmental concerns

Health and safety: There are recognized hazards that go with advanced composite materials. Knowing about these hazards, one can protect oneself and others from exposure to them. It is important to read and understand the Material Safety Data Sheets (MSDS) and handle all chemicals, resins and fibers correctly. Refer to SACMA publication "Safe Handling of Advanced Composite Materials" for additional information (Reference 8.2.3.6).

Disposal of scrap and waste: When selecting materials, consideration must be given to the disposal of scrap and waste. Disposal of scrap and waste should be specified under federal, state and local laws. See Section 8.2.5.6 on how to dispose of uncured materials.

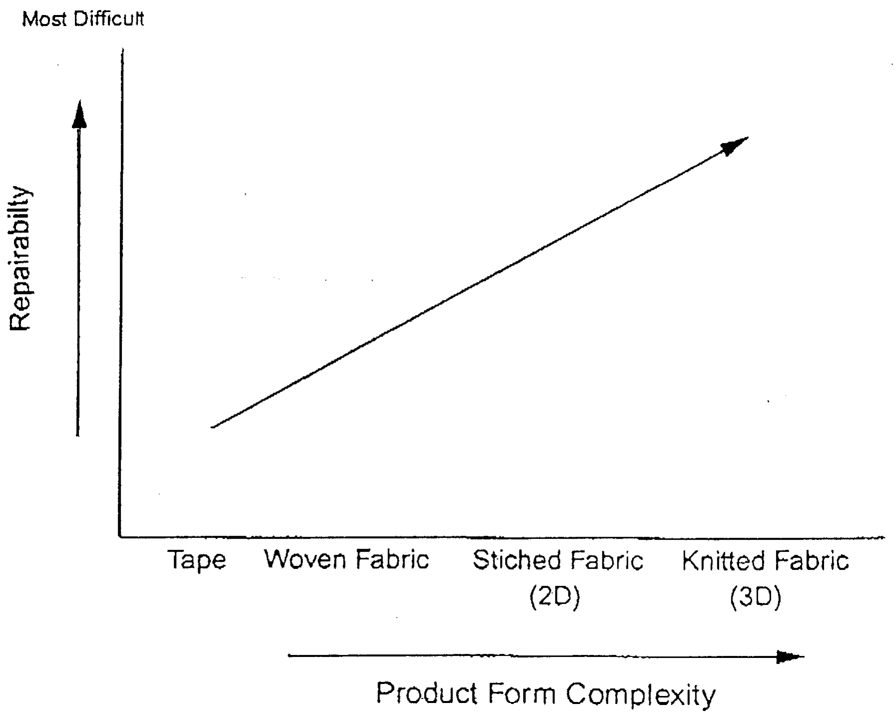


FIGURE 8.2.3.3 *Difficulty of repairing product forms.*

TABLE 8.2.3.4 *Repair adhesive considerations.*

Consideration	Response
Performance properties	The adhesive system must be capable of transferring structural, thermal, acoustic loads through a patch material and back into the parent structure. The adhesive system must also be capable of transferring those loads while operating within the vehicles environmental envelope (i.e., presence of hydraulic fluid, fuel, and dirt, and vibro-acoustic conditions).
Service temperature	The maximum surface temperature a structure will operate over the vehicle life. Exhaust sections and leading edges typically will operate at 50 - 500% higher temperatures than surrounding areas. The surface preparation method, adhesive primer, cure profile, heat sinks, and coatings and treatments can all influence the maximum temperature of the structure and associated repair.
Compatibility with surface preparation technique	Surface preparation can be anything from nothing to an electrochemically etched surface containing a commingled primer system. In addition the surface could be dirty, contain oxidation, hydrocarbons or moisture, or not lend itself well to chemically bonding with the adhesive.
Wetability	The ability of an adhesive to flow within all areas of the repair. Improvements in wetability reduce resin-starved areas and associated porosity, maintain bondline tolerances, and in general produce more reliable bonds.
Porosity of bondline	Curing without external pressure (i.e., vacuum bags) increases the potential of trapping volatiles created during the cure process. Application of heat and vacuum/pressure in the correct sequence will minimize porosity and, therefore, provide better bonds.
Tolerance of temperature deltas across repair area	All repair areas have varying thermal densities (substructure, patch ply drop-offs) which create a wide range of temperature deltas during adhesive cure. Adhesives that can cure well over a broad temperature range are more suited for repair applications. In addition, during repair only a small area of the structure is heated while the remaining structure is at ambient temperature which could be as low as -10°F or as high as 180°F.
Outtime at ambient temperature	Repairs can take a long time to assemble before the cure starts. Adhesives that are stable and fully thawed for several hours at ambient temperature will produce better and more reliable repairs.
Tolerance of bondline thickness	Uniform bondlines produce the best load transfer medium. Maintaining a uniform bondline thickness is difficult on structures that are wavy and have ply discontinuities. Adhesives that perform well with bondlines from 3-15 mils will produce the best repair performance.
Cure time	Ideally, cure time should be as short as possible to reduce vehicle downtime. Adhesives that can be heated at 5-7°F/min and dwelled at the cure temperature for less than 2 hrs. are optimum.
Cure pressure	In repair applications the only patch compaction force available is from atmospheric or mechanical pressure. Since autoclaves and associated tooling are not readily available and components are difficult to remove, vacuum bags or mechanical clamps will be the pressure devices of choice.
Cure temperature	A rule of thumb for repair applications is to use an adhesive with the lowest cure temperature that meets all the performance constraints. As temperatures increase, the tolerance of acceptable cure decreases. In addition, most hot bond control units manage the upper temperature limit, therefore, the cure temperature variance should be +0 and -40°F.
Storability at ambient temperature	Since many materials must be cold stored to minimize the effects of crosslinking, an adhesive that is tolerant of sustained outtime at ambient temperature is more suited for the repair environment. In addition, some repair facilities lack the cold storage equipment necessary and must rely on temporary cold storage methods such as iced coolers or just in time delivery of repair materials from distribution centers.

TABLE 8.2.3.5 *Material support issues.*

Issue	Support Impact
Autoclave only cure	1. Equipment not available in the field and at small repair facilities 2. Part has to be removed and disassembled for repair
Press curing	1. Equipment not available in the field and at small repair facilities. Part has to be removed and disassembled for repair
High temperature cure	1. Damage to surrounding structure in repair on aircraft 2. Protective equipment needed to handle high temperatures
Transit	1. Dry ice packing requirements may be problematic
Freezer storage required	1. Equipment not available not available in the field and at small repair facilities

8.2.4 Damage resistance, damage tolerance, and durability

In normal operating condition, components can be expected to be subject to potential damage from sources such as maintenance personnel, tools, runway debris, service equipment, hail, lightning, etc. During initial manufacturing and assembly, these components may be subject to the same or similar conditions. To alleviate the effects of the expected damage, most composite components are designed to specific damage resistance, damage tolerance, and durability criteria. How these design criteria affect supportability are discussed in this section. (Ideally, a supportable airframe structure must be able to sustain a reasonable level of damaging incidents without costly rework or downtime. Sustainability is being defined as showing no damage after such incidents and having the required residual strength and stiffness capability.)

8.2.4.1 Damage resistance

Damage resistance is a measure of the relationship between the force or energy associated with a damage event and the resulting damage size and type of damage. A material or structure with high damage resistance will incur less physical damage from a given event. For composite airframe structures repair actions are based on visibility, hence, if the damage is not visible, a repair activity is not needed. Therefore, to reduce repair activity, damage resistance levels should be such that at low impact energy levels (4 ft-lb) the damage is not visible and is negligible in high susceptibility areas. This can be accomplished by zoning the structure based on regions that have high or low susceptibility to damage and its residual strength and stiffness requirements. In defining the requirements, the type of structure (primary or secondary structures), construction method (sandwich or solid laminate), and whether its a removable or non-removable structure are pertinent. In practice, damage resistance is a critical design parameter for supportability, particularly for thin-skinned components. A more detailed discussion of damage resistance can be found in Section 7.5.

Damage resistance maybe improved by increasing laminate thicknesses and for sandwich applications by using denser core. However, the decrease in visibility may lead to the increase in nonvisible damage that must be considered for aspects of damage tolerance. The selection of reinforcement fibers that have high strain capability can also have a positive effect. Additionally, the selection of toughened matrix material can greatly enhance damage resistance. The selection of integrally stiffened panels over a honeycomb sandwich construction usually results in a more damage resistant configuration as the skin thicknesses are usually thicker and the impact energy is absorbed by the bending action for the integrally stiffened panel as compared to sandwiches. Possible water ingress into the sandwich panel after impact damage is another supportability drawback of sandwich construction.

Other items to improve damage resistance include the use of a layer of fabric as the exterior ply over tape to resist scratches, abrasion, softening of impact, and reduction of fiber breakout during drilling of fastener holes. Laminate edges should not be exposed directly into the air stream that could possibly subject it to delaminations. Avoidance of delaminations is achieved by using non-erosive edge protection, replaceable sacrificial materials or locating the forward edge below the level of the aft edge of the next forward panel.

Areas prone to high energy lightning strike should utilize replaceable conductive materials, provide protection at tips and trailing edge surfaces, and make all conductive path attachments easily accessible.

8.2.4.2 Damage tolerance

Damage tolerance for structural parts is a measure of the ability of such a part to maintain functionality, sufficient residual strength and stiffness, with damage for required loadings. In aircraft design, damage tolerance is a safety issue but does affect supportability. A very damage tolerant structure will require large area repair capability, although it may be of low frequency. On the other hand, a structure that can tolerate only small damage sizes will require frequent repair actions.

Damage tolerance is achieved by reducing allowable strain levels in damage and strength/stiffness critical areas and/or providing multiple load paths. For civil-aircraft composite parts, it is a requirement that the structure can sustain ultimate load with any damage less or equal to the barely visible size. Therefore, the designer of a structure highly resistant to surface damage has to make sure that the structure is also damage tolerant to hidden damage. Larger damages have lesser load residual strength requirements. See Section 5.12.1 for a more detailed discussion.

Except in areas where the margin of safety is near zero, a composite structure can tolerate larger than visible damage while still able to sustain ultimate load. A reduction in the number of repair actions would be possible if the part manufacturer provides a map of permissible damage sizes. Such a map would have to include not only the effect of static loads but of durability and functional requirements.

8.2.4.3 Durability

Durability of the structure is its ability to maintain strength and stiffness throughout the service life of the structure. In general, structural durability is inversely related to maintenance cost. A durable structure is the one that does not incur excessive maintenance cost during its service life. A composite structure that was designed for damage resistance will have excellent durability as carbon composites have excellent corrosion resistance characteristics (assuming no galvanic corrosion) and fatigue characteristics when compared to metals.

In composites, fatigue damage due to repeated mechanical loads usually initiates as cracks in the matrix material at laminate edges, notches, and stress discontinuities and then may progress as interlaminar delaminations. For currently designed structures with low allowable strain levels, in part due to damage tolerance and repair requirements, the fatigue loads are generally below the levels that would cause extensive matrix cracking. One exception is in the vicinity of fastener holes, where, if the bearing stresses are high, hole elongation may cause bolt fatigue failures and other anomalies due to internal load redistribution. Thus, good supportability design should feature low bearing stresses (see Section 5.3.2.3). A general discussion on durability can be found in Section 5.12.2.

8.2.5 Environmental compliance

Many aspects of the design, repair and maintenance of polymer matrix composites are impacted by environmental rules and regulations. Many people associate environmental compliance with the correct disposal of hazardous wastes. This is certainly an important factor, but is by no means the only factor to consider. In fact, by the time we are concerned with the disposal of hazardous waste, we have missed a tremendous number of opportunities to reduce the generation of waste in the first place. The concept of reduction of hazardous waste before it is generated, known as pollution prevention, can begin as early as the initial design phase. It can greatly reduce labor, cost, and paperwork associated with the disposal of hazardous wastes generated by repair and maintenance of the component throughout its life cycle. This section will identify factors to consider during the design and repair design phase to facilitate true life cycle pollution prevention.

8.2.5.1 Elimination/reduction of heavy metals

The requirement for heavy-metal containing coatings and treatments not only presents environmental compliance difficulties during manufacture, but presents additional challenges every time the coating needs to be removed, repaired or replaced. Traditional requirements for chromic acid anodizing or alodine processes impact mostly metal components, however, we encounter similar issues with polymer matrix composites as well. Typical culprits include cadmium plated fasteners, and chromated sealants and primers. When designers consider environmental compliance along with cost and quality when specifying design materials, we may be able to eliminate the specification of these materials in the first place.

Non-chromated sealants and primers are currently available and research and development initiatives are underway to evaluate their suitability for long term use on military aircraft. The specification of a non-chromated primer or sealant in the design of a component will create benefits throughout its life cycle by reducing hazardous waste and personnel exposure to hazardous materials.

8.2.5.2 Consideration of paint removal requirements

During the design of polymer matrix composite components, consideration must be given to removal of coatings. Chemical paint removers are not acceptable for most polymer matrix composites because the active ingredients that attack the organic coating also attack the matrix. Abrasive paint removal techniques, such as plastic media blasting, have proven successful on polymer matrix composites but their use can be limited by substrate thickness and specialized surface treatments or coatings. The consideration of paint removal techniques during manufacture may highlight minor changes in design that can affect major savings in maintenance over the life cycle of the component.

8.2.5.3 Shelf life and storage stability of repair materials

A significant portion of a waste stream is made up of materials that cannot be used within their useful life. In its worst case, this involves materials that are purchased, sit on the shelf and are then disposed of as hazardous waste without ever making it to the work center. At best, it represents containers that have been opened but not finished before the shelf life expires. The following are some of the ways this waste stream can be minimized by design decisions.

a) *Specify common materials*. Maintainers have difficulty “using up” materials if they are specific to a single aircraft or component. In many cases material manufacturers establish “minimum buys” of their product dictating the minimum purchase of several gallons when only a pint is required. The excess material often simply sits on the shelf until it is no longer useable and is then sent to disposal. This issue can be alleviated through the specification of materials that are already in the inventory, or that are used on a wide variety of components.

b) *Specify long shelf life, and/or room temperature storable materials.* Obviously, the longer the shelf life of a product, and the less restrictive the storage, handling and transportation requirements, the better the chances that the material can be consumed before its shelf life expires. Designers should be aware that even though these materials may be slightly more expensive, or may not be the material of choice during the manufacture of the product - they may well be suitable for repair and/or maintenance.

8.2.5.4 Cleaning requirements

Cleaning is one of the primary maintenance processes that create hazardous waste. The construction of the component often dictates the cleaning options available for that part. Many of the cleaning processes that previously utilized ozone depleting solvents and other hazardous chemicals are being replaced with aqueous cleaning processes. If a component is constructed such that water intrusion is a concern, then aqueous cleaning of the part may also be a problem. The requirement for solvent cleaning places a heavy burden on the maintainer - which will continue to worsen as environmental restrictions tighten. Designing components so they can tolerate aqueous cleaning will facilitate maintenance requirements throughout the life cycle.

8.2.5.5 Non-destructive inspection requirements

The requirement to perform non-destructive inspection on a component often requires cleaning and paint removal (resulting in hazardous waste generation) that would not otherwise be necessary. Often, non-destructive inspection requirements set during the design phase are maintained throughout the life cycle regardless of whether defects are ever found during the inspection. Periodic reviews of inspection requirements will present the opportunity to eliminate non-value added requirements thereby saving money, time and hazardous waste generation.

8.2.5.6 End of life disposal considerations

Unlike the situation for metals, there is not a widespread market waiting to buy composite materials from scrap aircraft. There are several initiatives underway to find uses for these materials. Designers need to stay abreast of these initiatives so that if a market is identified for certain polymer composites, this can be given consideration when selecting design materials.

Machining of carbon fiber laminates during cutting and trimming operations produces particulates that are nominally considered a nuisance dust by bio-environmental engineers. TLV (Threshold Limit Values) limits were updated in 1997 by the American Conference of Government and Industrial Hygienists (ACGIH) to define loose composite fiber/dust exposure limits for composite workers. Excessive exposure may require the use of NIOSH-certified respirators with HEPA filters. Resins used in composite materials and adhesives may cause dermal sensitization in some workers, thus silicon-free/lint-free gloves should be mandated for use. This will also ensure a contaminant-free laminate.

Uncured prepregs and resins are treated as hazardous materials in waste stream analyses. Scrap materials should be cured prior to disposal to inert the resins and reduce the HazMat disposal costs. It is important to ensure that scrap materials containing carbon fibers are sent to non-burning landfills; pyrolized carbon fibers freed by resin burn-off can represent a respiratory and electrical hazard.

8.2.6 Reliability and maintainability

The maintainability of a structure is achieved by developing schemes for methods of inspection and maintenance during the design phase. The designer with the overall knowledge of the performance and operational characteristics of the structure should access, based on the construction method, configuration, material selection, etc., whether the structure is maintainable. Such factors in assessment would include development of cradle-to-grave inspection methodology, techniques, protection schemes and defined inspection intervals for maintenance.

8.2.7 Interchangeability and replaceability

A composite structure can be maintained using a variety of methods, each dependent on the support plan and maintenance concept selected. One of the first design considerations that must be determined is the ease with which a damaged piece of structure can be repaired. Large integral structural elements, such as a wing skin panel, cannot be readily removed from the aircraft and, therefore, must be repaired in-place. Many panels, however, can be removed and the damaged panel replaced with a new panel.

Ease of maintenance can have a direct impact on the design and surrounding structure. By developing the maintenance concept early in the design process, tradeoffs can be made before the design is finalized that will provide the aircraft operator with more maintenance options and provide aircraft that are potentially more available to perform their design function.

The design of removable panels can have significant impact on the ease of maintenance and the associated maintenance and downtime cost. There are two commonly used types of panels used for structural maintenance - interchangeable and replaceable.

The **interchangeable panel** is one that can be installed onto the aircraft without any trimming, drilling, or other customizing. Interchangeable parts are designed through a selection of materials, tolerances, and fastening techniques to fit a production run of aircraft within the same model series.

Replaceable panels may or may not fit between different aircraft and usually require trimming and drilling on installation.

Figure 8.2.7 shows the differences between interchangeable and replaceable panels.

Interchangeability and replaceability (I&R) requirements for non-repairable, high-unit cost, frequently damaged, or highly loaded components need to be assessed early in the design process to ensure cost and operational effectiveness. Typically, I&R components are more costly to manufacture due to the close tolerances, materials and design attributes. Designers must be assured that form, fit, and function are fully realized with removable parts and realize that thermal and material mismatches, part number changes, and different manufacturing techniques, may alter a component's ability to be replaceable or interchangeable. In some cases, I&R are design requirements and can easily be met using loose tolerances and numerically controlled master tooling during the manufacturing process.

Components that must be removed frequently (<1000 flight hours) to facilitate other maintenance actions are typically good candidates for interchangeable panels. Components that are large and contain a variety of inner mold line geometries and fastener configurations (fuselage skins and components that have attachment fittings, i.e., landing gear doors) are good candidates for replaceable panels. Mil-I-8500, The Use of Interchangeable Components, provides requirements and guidance.

8.2.8 Accessibility

Accessibility is an important factor when designing structures for repair. Sufficient access should always be provided to properly inspect, prepare the damage structure, fit and install the repair parts and use repair tools and bonding equipment. Limited access may dictate the repair approach, i.e., use of precured patches, use of mechanical fasteners in lieu of cocuring, etc. If feasible, two-sided access is preferred.

8.2.9 Repairability

Designing for repairability is an essential element in the effective use of composite materials in aircraft structures. Selecting a repair approach during the design phase will influence the choice of lay-up patterns and design strain levels. It is important that the repair philosophy be set during the conceptual design stage and that the repair designs be developed along with the component design. Candidate repair designs should be tested as part of the development test program. Repair concepts and materials should be standardized to the maximum extent possible, and repair considerations are appropriate for concept development of any aircraft structural component. This section lists recommendations for design approaches that will improve the repairability of composite aircraft structural components.

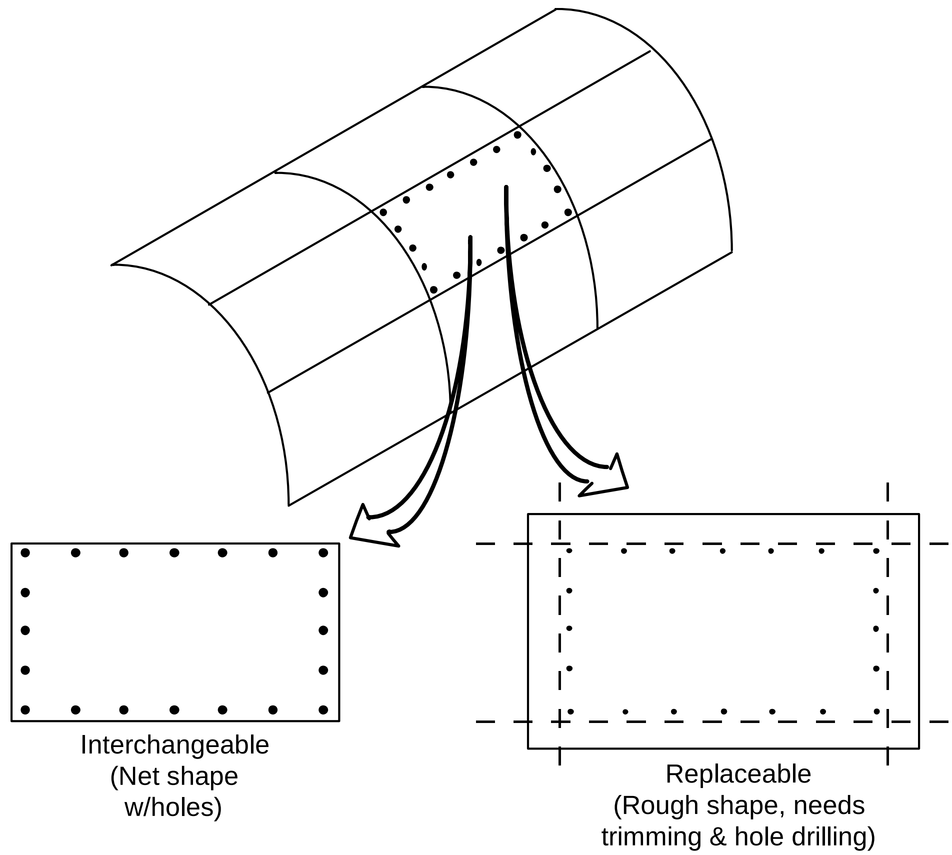


FIGURE 8.2.7 *Differences between interchangeable and replaceable panels.*

8.2.9.1 General design approach

The approach for the composite structures design team needs to be based upon input and knowledge gained from a working relationship established between the design team and airline maintenance personnel. This can be accomplished through repair workshops, or inquiries, involving airline and OEM customer support personnel, engineering personnel and involvement with the Commercial Aircraft Composite Repair Committee (CACRC). Reference 8.2.9.1 is a product of this committee and provides general guidance. The time spent within these efforts will provide a broader understanding of the overall environment in which operators operate. OEM involvement in the CACRC has contributed to addressing the problems that operators voice. The CACRC is pioneering standards and recommendations for the design and maintenance of future composite structure based on current and past experience.

Figure 8.2.9.1(a) shows the maintenance development philosophy established during Phase B of the Boeing/NASA Advanced Technology Composite Aircraft Structures (ATCAS) composite fuselage program. Maintenance procedures such as inspection and repair, which are applicable to a service environment, must be considered during design selection. Considerations should be made for bolted repairs, for example, there should be sufficient edge distances on stiffener and frame flanges, and sandwich edge bands to allow for repair bolts. Skin thicknesses should be sufficient to prevent knife-edges when using countersunk repair fasteners. Fabric outer plies should be considered to help reduce breakout when drilling holes for repair bolts in laminates or face sheets. Any lightning strike protection systems that are needed on specific components should be designed to be repairable. It should be assumed that bolted and bonded repairs would follow general practice guidelines.

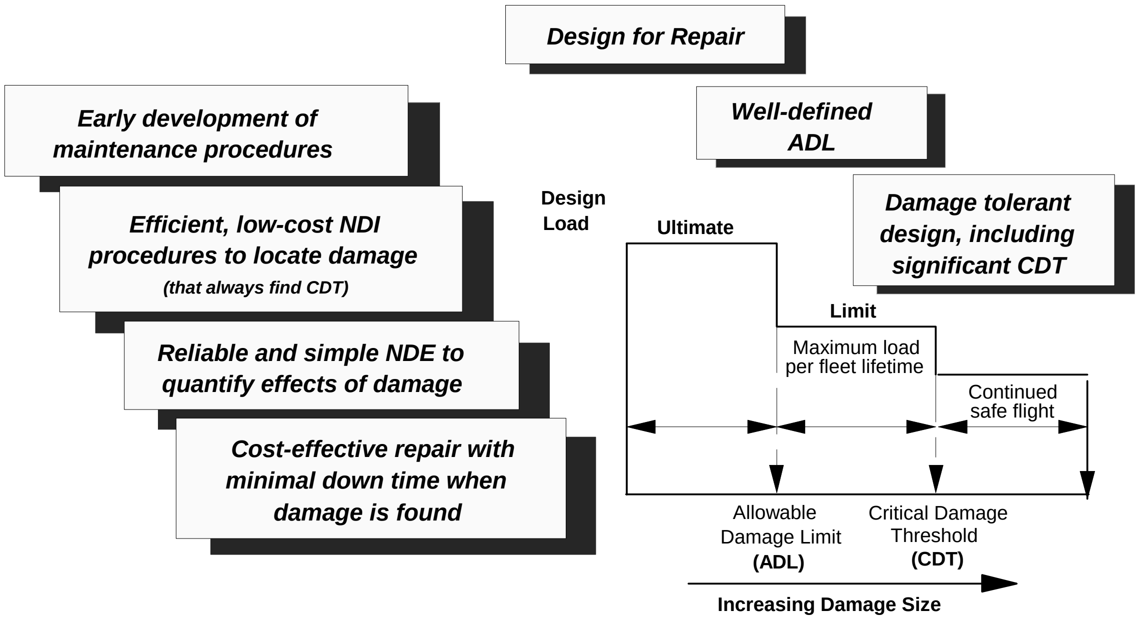


FIGURE 8.2.9.1(a) *Rules for maintainable composite structures.*

Some composite structural details, while weight and cost efficient, are difficult to repair. Closed hat-section stringers, for instance, are compatible with inexpensive manufacturing techniques and minimum weight, but pose difficulties relative to inspection and attachment in repair applications. The use of blind fasteners should also be kept to a minimum because of difficulties in removing them in order to make a repair or a replacement. Where fasteners are necessary, the removable types are preferable. Quite often, when removing fasteners performing a repair, the drilling out of blind fasteners the surrounding structure is damaged, thus incurring more cost and down-time. Material choices may also be affected. The designer should avoid the use of different material systems with different curing temperatures on one part. For instance, skins and stiffeners are sometimes precured at 350°F (177°C) and then, for manufacturing ease, secondarily bonded with 250°F (121°C) adhesive. This can present problems when the skins or stiffeners are repaired at 350°F (177°C); the integrity of the 250°F (121°C) adhesive at the bond interface may be compromised with no indication of degradation.

Design concept developments should include parallel efforts to establish maintenance procedures. Maintenance procedures established *after* design features for manufacturing scale-up are set will typically result in unnecessarily complex repair designs and processes.

Another important aspect of concept development critical to maintenance is damage tolerant design practices. The allowable damage limits (ADL) and critical damage thresholds (CDT) defined in Figure 8.2.9.1(a) must be established to support the structural repair manual and inspection procedures.

The former allows rapid determination of the need for repair during scheduled inspection, while the latter should be sufficiently large to allow safe aircraft operation between inspection intervals. Knowledge of residual strength and inspection capabilities should allow determination of both ADL and CDT as a function of structural location. Damages smaller than the ADL limit may never be discovered, whereas the CDT damage level must always be found by visual inspection.

The design of some areas of the structure can be controlled by manufacturing and durability considerations. Specific examples of these considerations are minimum gage (to provide a minimum of impact damage resistance and avoid knife-edge at countersink fasteners), stiffener, rib and frame flange width, bolt spacing and edge distance requirements, and avoiding rapid ply drops and buildups. Areas of the structure designed to these considerations will, therefore, have higher margins for damage tolerance. Figure 8.2.8.1(b) shows the minimum margins of safety for a composite fuselage side panel, illustrating the "over-designed" regions. These regions have ADLs and CDTs larger than the rest of the fuselage section. Zoned ADL and CDT information should prove useful to operators desiring minimum maintenance costs. Structural Repair Manuals quite often point to critical zones on components for special directed inspections, so zoned ADL and CDT information could be included in these manuals.

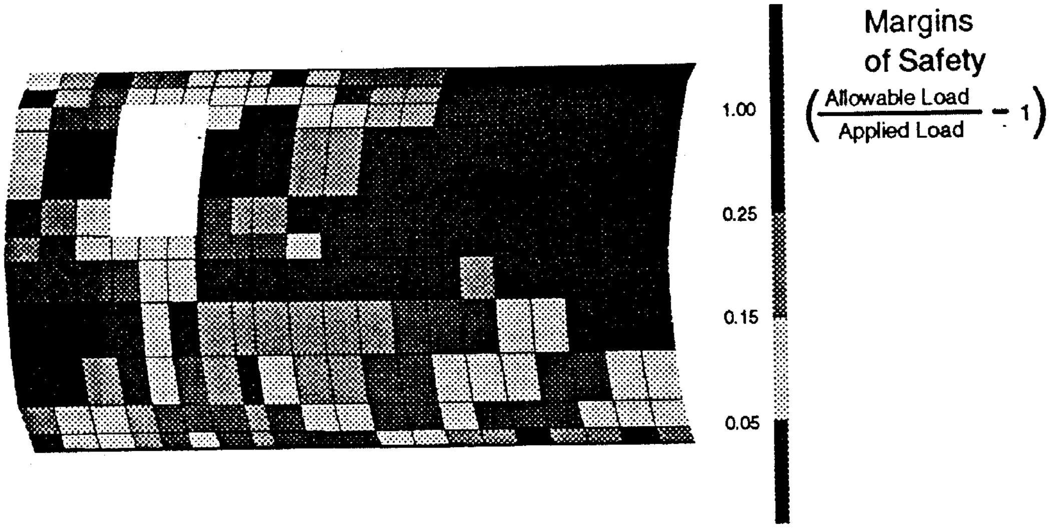


FIGURE 8.2.9.1(b) *Strength margin of safety distribution for a fuselage side panel subjected to ultimate loads.*

Returning to Figure 8.2.9.1(a), another requirement for maintainable composite structure is the establishment of non-destructive inspection (NDI) and evaluation (NDE) procedures for practical damage location and quantitative assessment, respectively, during scheduled maintenance. The latter, which may require ultrasonic methods, should only be required to assess the effects of damage found by more easily performed procedures (e.g., visual).

Damage Levels. When damage is found, efficient repair procedures are needed that the operators can accomplish with available resources (tooling, equipment, etc.) and with a minimum amount of airplane down time. In order to develop repair concepts for a broad range of damage scenarios, the repair design philosophy is focusing on more generic repairs that are not damage-specific. This approach will be beneficial because generic designs and corresponding repair procedures can be developed for various levels of damage which are, within certain limits, independent of specific damages. This is intended to greatly reduce the need to develop repairs for each damage event as it occurs, providing a higher level of maintainability. Initially, three damage levels have been defined and are shown in Table 8.2.9.1(a) as they apply to a skin/stringer configuration.

TABLE 8.2.9.1(a) *Example of skin/stringer damage level definitions.*

Designation	Damage Description	Repair
Level 0	Edge skin delamination or disbond from stiffening elements	Fastener restraint or injection resin repair
Level 1	Critical damage to a single structural element (skin or stiffener)	Mechanically fastened or bonded patch and/or splice
Level 2 (and higher)	Multiple occurrences of Level 1 damage	Same as Level 1

Designs should address repair in such a way that each bay is looked upon as a unit, or building block. Restoration of that unit (rib, frame, stringer, and/or skin) should be designed so that larger multiple-bay damages can be handled with less effort. Structural units are less easily defined for sandwich structure; however, the same general philosophy applies. The strategy behind this approach is to address the repair scenarios for a large range of damage at the beginning of the design process to ease the maintenance burden.

Multiple Options. Another aspect of the approach is to provide operators with multiple options for a given repair situation. Options might include, as examples, temporary vs. permanent repair, bonded composite patches versus bolted composite or metal patches, or wet lay-up or prepreg patches versus precured bonded patches. An operator's choice might depend on the severity of the damage, the time available to perform the repair, the operator's facilities and capabilities, inspection/overhaul schedules, and/or current field environmental conditions.

Durability versus weight trades. The understanding derived from residual strength analyses and tests will also ultimately lead to cost and weight trades that affect all of the total direct operating costs (DOC). Small increases in manufacturing cost and structural weight may be traded against increased damage tolerance and durability to reduce maintenance costs. Decisions may be required to balance the ADL and CDT. For example, test results for laminate tensile notch sensitivity may show an inverse relationship between small and large notch strength. Under such circumstances it may be desirable to have some ADL capability to avoid having to repair small damages but not at the expense of CDTs that allow sufficiently long inspection intervals and satisfactory fail-safe behavior.

8.2.9.2 Repair design issues

Skin/stringer structure repair issues. Solid laminate skin/stringer designs are quite often repaired using mechanically fastened external skin patches and nested substructure splice angles. Mechanically fastened repairs require care and accuracy in the drilling of holes and the alignment of parts during assembly. Fastener hole breakout is a characteristic problem, commonly solved by using a layer of fabric as the outermost ply for all laminates. Typically, even though there may be other methods to avoid fastener hole breakout, there are numerous situations in the real world that challenge a good mechanic's ability to consistently drill high quality holes. Provisions to locate the position of the drilled holes in the structure include alignment marks and templates. Each skin/stringer component design should have laminates lay-ups that have sufficient thickness and numbers of plies in each of the 0°, 90°, and 45° directions so that they are repairable with mechanically fastened patches.

Sandwich structure repair issues. Sandwich structure is generally repaired with insitu processed bonded scarf or stepped patches. The typical scarf/step taper ratios employed when repairing thin face sheets of control panels and fixed secondary structure are quite shallow (e.g., 20:1). When repairing sandwich structures with thicker face sheets in more highly loaded areas, however, scarf repairs with these traditional shallow taper ratios result in the removal of a large amount of undamaged material, and hence, very large patch sizes. In these situations, repairs may be combinations of scarfed and external patches, so that the repair sizes can be minimized. Flush repairs may be required for some components for aerodynamic reasons or to prevent chafing. Also, thick face sheets require thick patches, which may require special processing to achieve proper consolidation. Patch and bondline porosity are of particular concern with normal field processing, which is accomplished with vacuum pressure and heat blankets. Lower temperature cures are generally preferred due to concerns over causing additional damage via vaporization of water that has infiltrated the core. Also, the surrounding structure may act as a heat sink, making it difficult to achieve and control the higher temperatures with heat blankets, and may contribute to thermal gradients that can result in warpage or degradation of the surrounding structure. For thick sandwich, heat blankets on both sides of the structure may be required to control the through thickness temperature. Still, the shorter processing times generally associated with higher temperature cures are very attractive in terms of minimizing the out-of-service time for a damaged airplane.

Sandwich moisture ingression issues. Consideration must be given to moisture ingression when designing maintainable, repairable sandwich structures. Sandwich designs must address the effects of moisture in the core, both by minimizing the degree of moisture ingression, and by determining what its presence does to the performance of the structure. Moisture ingression can occur through face sheet damages, and part edge and end seals, so special care must be taken to design durable sandwich parts. Unfortunately, to make durable face skins, additional thickness is needed, and this may not be desirable from a performance point of view. Durable edge and end seals can be designed, see Ref. 8.2.1.1. When repairing damaged sandwich structures, a drying cycle is typically performed prior to the accomplishment of any bonded repair. This is performed so that any retained moisture does not interfere with the curing cycle. There have been numerous cases of face skins blowing off sandwich components during the vacuum bag heating cure cycle.

8.3 SUPPORT IMPLEMENTATION

A repair has the objective of restoring a damaged structure to an acceptable capability in terms of strength, durability, stiffness, functional performance, safety, cosmetic appearance or service life. Ideally, the repair will return the structure to original capability and appearance.

The design assessment of a repair for a given loading condition involves the selection of a repair concept, the choice of the appropriate repair materials and processes, then specifying the detailed configuration and size of the repair. Most repairs are basically designed as a joint to transfer load into and out of a patch. To ensure that the repair configuration will have adequate strength and stiffness, the repair joint must be analyzed to predict its strength.

The selection of the type of load-transfer joint to be used for a patch/strap is a tradeoff between simplicity, strength and stiffness. The easier configurations are generally not as strong as the more difficult ones. It is critical that the materials and process information is available prior to the system being put into place.

8.3.1 Part Inspection

Presence of damage in aircraft composite parts is usually found in the course of a routine on-line inspection, depot inspection, or, for large damages, noticed by the pilot. The predominant mode of inspection is visual with more sophisticated modes of inspection performed at the depot. Once damage is identified visually in-service, the damage should be characterized quantitatively by measuring dent depth, extent of surface damage, and length of scratches before proceeding to more complex NDI. This will generally consist of tap testing to define the boundary between damaged and undamaged portions of the structure and followed, for major repairs, with instrumented NDI techniques (ultrasonics, radiography, etc.) to locate the through the thickness characteristics of the damage. At a depot other NDI methods, such as shearography or thermography, may be available. A good general reference on inspection methods is SAE ARP 5089 “Composite Repair NDI and NDT Handbook” (Reference 8.3.1). A summary of common nondestructive test methods and their utilization is shown in Table 8.3.1.

TABLE 8.3.1 *Common non-destructive test methods.*

METHOD	STRUCTURE	DAMAGE DETECTED	RELIABILITY
Visual	All	Surface damage	Good
Tap test	Thin laminate	Delaminations near surface	Good
	Thin face sheet	Lack of bond	Good
		Disbond near surface	Good
		Voids	Poor
		Blown core (core damage)	Poor
		Lack of tie-in at closure	Good
		Lack of tie-in at core splice	Poor
Ultrasonics	All	Delaminations	Good
		Lack of bond	Good
	Sandwich	Crushed core	Poor
		Blown core (core damage)	Poor
		Water in core	Poor
Radiography	All	Disbonds/delaminations	Poor
		Delaminations in corners	Good
	Sandwich	Node separation	Good
		Crushed core	Good
		Blown core (core damage)	Good
		Water in core	Good
Shearography	All	Disbonds/delaminations	Good
Thermography	All	Disbonds/delaminations	Good
	Sandwich	Water in core	Good

8.3.1.1 Visual

Nondestructive inspection by visual means is by far the oldest and most economical NDI method. Consequently, visual inspection is performed routinely as a means of quality control and damage assessment for both the manufacturer and the repair technician. Fortunately, most types of damage either scorch, stain, dent, penetrate, abrade, or chip the composite surface making the damage visually verifiable. Once detected, the affected area becomes a candidate for closer inspection. Flashlights, magnifying glasses, mirrors, and borescopes are employed as aids in the visual inspection of composites. They are used to magnify defects which otherwise might not be seen easily and to allow visual inspection of areas that are not readily accessible. Resin starvation, resin richness, wrinkles, ply bridging, discoloration (due to overheating, lightning strike, etc.), impact damage by any cause, foreign matter, blisters, and disbonding are some of the discrepancies readily discernable by a visual inspection. Visual inspection can not find internal flaws in the composite, such as delaminations, disbonds, and matrix crazing. More sophisticated NDI is needed to detect these, although an experienced (with the part and composites in general) technician can often surmise if there is any internal damage. Additionally, tight surface cracks and edge delaminations may not be detected visually.

Therefore, visual inspection techniques need to be supplemented by other methods of nondestructive testing. Because many of the defects associated with composites are hidden within the composite component's structure (i.e., within the ply lay-up or common to the honeycomb core), special techniques dealing with the analysis of sound attenuation are utilized to assure structural continuity within the composite.

8.3.1.2 Tap testing

Sometimes referred to as audio, sonic, or coin tap, this technique makes use of frequencies in the audible range (10Hz. to 20Hz.). A surprisingly accurate method in the hands of experienced personnel, tap testing is perhaps the most common technique used for the detection of delamination and/or disbond. The method is accomplished by tapping the inspection area with a solid round disk or lightweight hammer-like device, as shown in Figure 8.3.1.2 and listening to the response of the structure to the hammer. A clear, sharp, ringing sound is indicative of a well-bonded solid structure while a dull or thud like sound indicates a discrepant area. The tapping rate needs to be rapid enough to produce enough sound such that any difference in sound tone is discernable to the ear. Tap testing is effective on thin skin to stiffener bondlines, honeycomb sandwich with thin face sheets or even near the surface of thick laminates such as rotorcraft blade supports. Again, inherent in the method is the possibility that changes within the internal elements of the structure might produce pitch changes that might be interpreted as defects, when in fact they are present by design. This inspection should be accomplished in as quiet an area as possible and by experienced personnel familiar with the part's internal configuration.

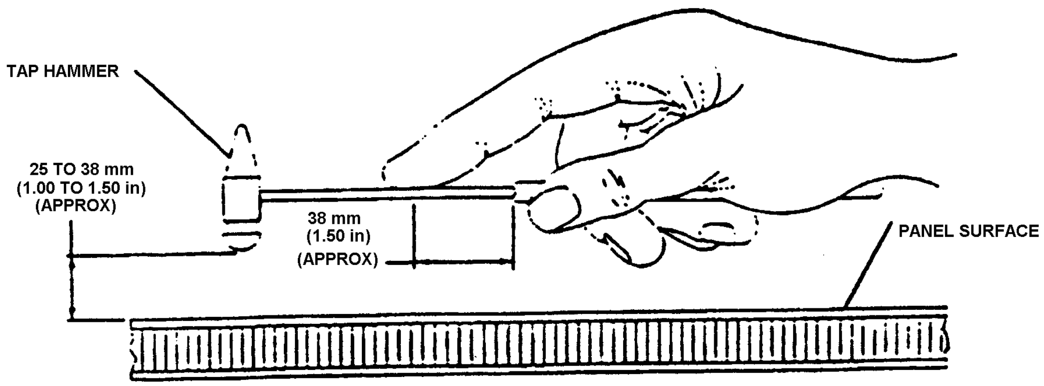


FIGURE 8.3.1.2 *Tap testing.*

8.3.1.3 Ultrasonics

Ultrasonic inspection has proven to be a very useful tool for the detection of internal delaminations, voids, or inconsistencies in composite components not otherwise discernable using visual or tap methodology. There are many ultrasonic techniques, however, each technique uses sound wave energy with a frequency above the audible range. A high frequency (usually several MHz) sound wave is introduced into the part and may be directed to travel normal to the part surface, or along the surface of the part, or at some predefined angle to the part surface. Different directions are used as the flow may not be visible only from one direction. The introduced sound is then monitored as it travels its assigned route through the part for any significant change. Ultrasonic sound waves have properties similar to light waves. When an ultrasonic wave strikes an interrupting object, the wave or energy is either absorbed or reflected back to the surface. The disrupted or diminished sonic energy is then picked up by a receiving transducer and converted into a display on an oscilloscope or a chart recorder. The display allows the operator to comparatively evaluate the discrepant indications against those areas known to be good. To facilitate the comparison, reference standards are established and utilized to calibrate the ultrasonic equipment.

The repair technician must realize that the concepts outlined here work fine in the repetitious manufacturing environment, but are likely to be more difficult to implement in a repair environment given the vast number of different composite components installed on the aircraft and the relative complexity of their construction. The reference standards would also have to take into account the transmutations that take place when a composite component is exposed to an in-service environment over a prolonged period, or has been the subject of repair activity or similar restorative action. The two most common ultrasonic techniques applicable to damage definition are discussed next.

Through Transmission. This technique may be utilized when both sides of the part to be inspected are accessible. The basic principle of through transmission ultrasonics is shown in Figure 8.3.1.3. Pulsed high voltage is applied to a piezoelectric crystal contained within the transducer. This crystal transforms the electrical energy to mechanical energy in the form of ultrasonic sound waves. The ultrasonic waves are propagated through the part to the receiving transducer where the mechanical energy is transformed back into electrical energy. A couplant other than air is needed for the method to work. In production environment the part is immersed in water or a squirted water system is utilized. Caution must be exercised when using couplant material other than water so as not to contaminate the laminate. Water soluble couplants work well. New techniques are being developed which do not need a couplant. The output may be plotted on a recording system or displayed by a meter or an oscilloscope. Defects within the test article will disrupt or absorb a portion of the energy and thereby change the amount of energy detected by the receiving transducer. The defects resultant diminished energy then becomes discernable on the display.

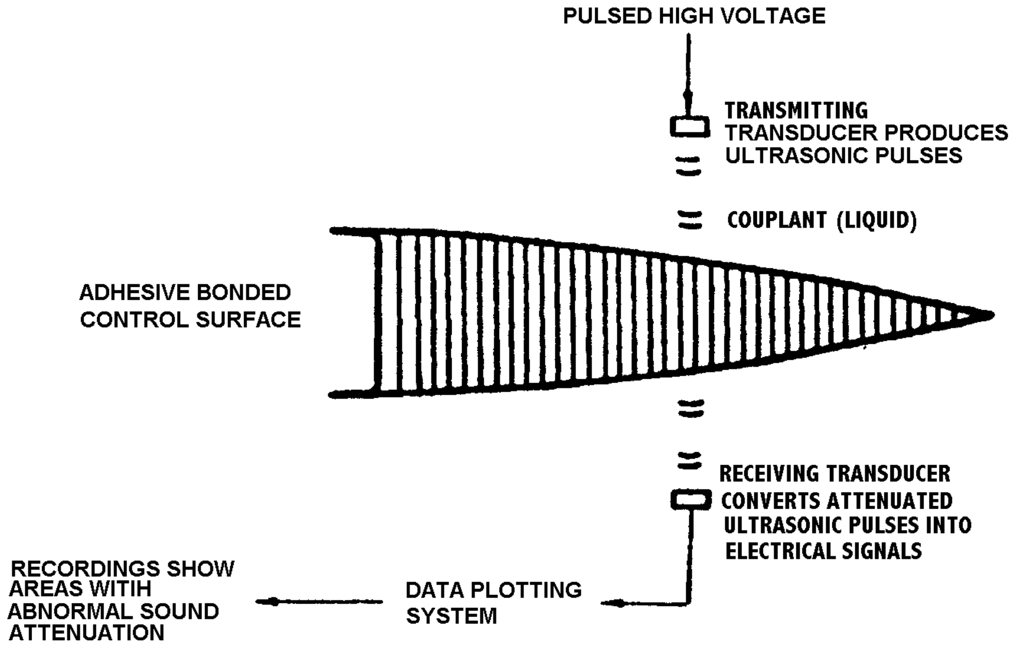


FIGURE 8.3.1.3 *Through transmission.*

Pulse Echo. Single side ultrasonic inspection may be accomplished using pulse echo techniques. In this method a single search unit is working as a transmitting and a receiving transducer which is excited by high voltage pulses. Each electrical pulse activates the transducer element. This element converts the electrical energy into mechanical energy in the form of an ultrasonic sound wave. The sonic energy travels through a Teflon or a Methacrylate contact tip into the test part. A wave-form is generated in the test part and is picked up by the transducer element. Any change in amplitude of the received signal, or time required for the echo to return to the transducer, indicates the presence of a defect. In pulse echo the couplant is directly applied to the part.

8.3.1.4 Radiography

Radiography, or X-ray, as it is often referred to, is a very useful NDI method in that it essentially allows a view into the interior of the part. This inspection method is accomplished by passing X-rays through the part or assembly being tested while recording the absorption of the rays onto a film sensitive to X-rays. A typical radiographic exposure setup is shown in Figure 8.3.1.4. The exposed film, when developed, allows the inspector to analyze variations in the opacity of the exposure recorded onto the film, in effect creating a visualization of the relationship of the component's internal details. Since the method records changes in total density through its thickness, it is not a preferred method for detecting defects such as delaminations that are in a plane that is normal to the ray direction. It is a most effective method, however, for detecting flaws parallel to the X-ray beam's centerline. Internal anomalies such as delaminations in the corners, crushed core, blown core, water in core cells, voids in foam adhesive joints, and relative position of internal details can readily be seen via radiography. Most composites are nearly transparent to X-rays so low energy rays must be used. Opaque penetrant can be used to enhance the visibility of surface breaking defects, however, it is generally not available for in-service inspections.

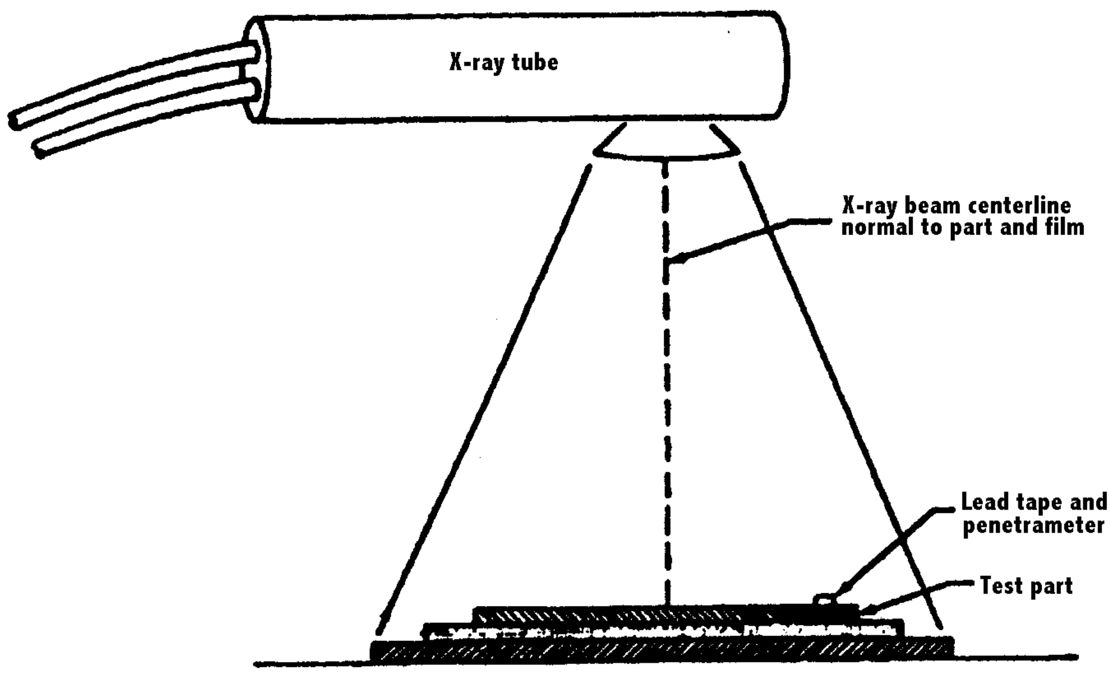


FIGURE 8.3.1.4 *X-ray setup.*

Because of the picture-like quality of the results, radiographic inspection lends itself to easy interpretation, although honeycomb X-ray radiographs are best analyzed by experienced technicians. However, because of safety concerns it is impractical to use around aircraft. Operators should always be protected by sufficient lead shields, as the possibility of exposure exists either from the X-ray tube or from scattered radiation. Maintaining a minimum safe distance from the X-ray source is always essential.

8.3.1.5 Shearography

Shearography is an optical NDI technique that detects defects by measuring the variations in reflected light (speckle pattern) from the surface of the object. Using a laser light source, an original image of the illuminated surface is recorded via a video image. The part is subsequently stressed by heating, changes in pressure, or acoustic vibrations during which a second video image is made. Changes in the surface contour caused by disbonding or delaminating become visible on the video display.

Shearography is being used in production environments for rapid inspection of bonded composite structure assemblies including carbon/epoxy skin and Nomex core sandwiches. This is accomplished by inducing stresses by partial vacuum. Partial vacuum stressing causes air content defects to expand, leading to slight surface deformations that are detected before and during stressing comparisons. Display of the computer processed video image comparisons reveals defects as bright and dark concentric circles of constructive and destructive reflected light wave interference. A schematic of an inspection system currently in use is shown in Figure 8.3.1.5.

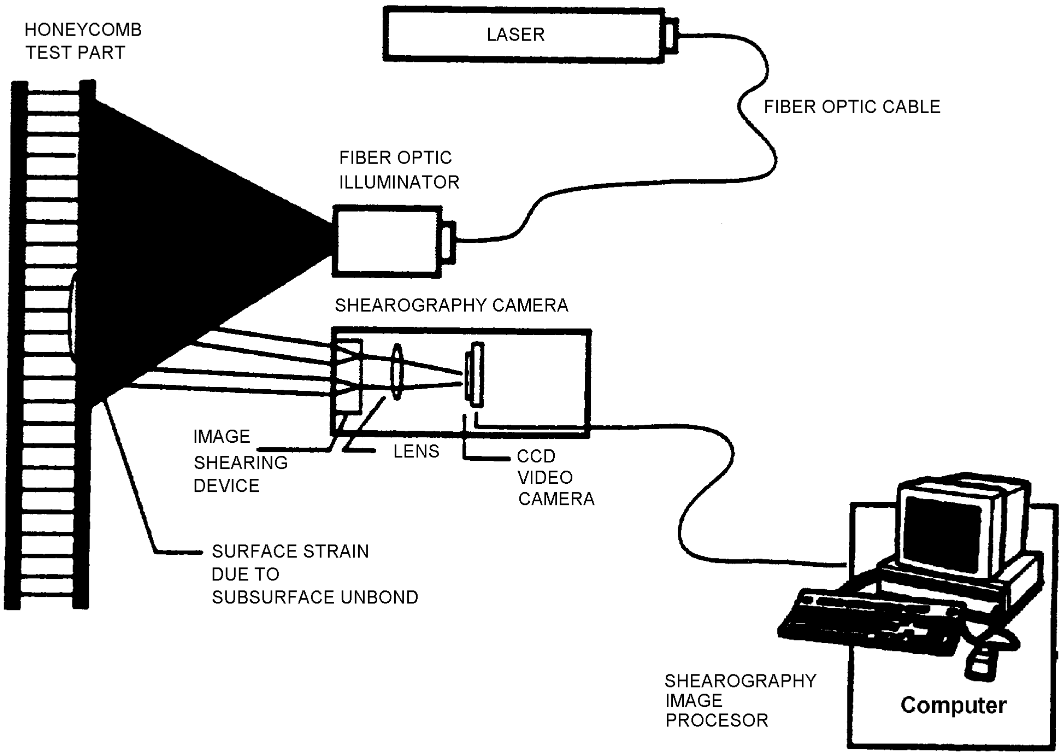


FIGURE 8.3.1.5 *Components of shearography inspection system.*

8.3.1.6 Thermography

Thermal inspection comprises all methods in which heat-sensing devices are used to measure temperature variations for parts under inspection. The basic principle of thermal inspection consists of measuring or mapping of surface temperatures when heat flows from, to, or through a test object. All thermographic techniques rely on differentials in thermal conductivity between normal, defect free areas and those having a defect. Normally, a heat source is used to elevate the temperature of the article being examined while observing the surface heating effects. Because defect free areas conduct heat more efficiently than areas with defects, the amount of heat that is either absorbed and reflected indicates the quality of the bond. The type of defects that affect the thermal properties include debonds, cracks, impact damage, panel thinning, and water ingress into composite materials and honeycomb core. Thermal methods are most effective for thin laminates or for defects near the surface.

The most widely used thermographic inspection technique uses an infrared (IR) sensing system to measure temperature distribution. This type of inspection can provide rapid, one-sided non-contact scanning of surfaces, components, or assemblies. Figure 8.3.1.6 illustrates the components of such a system that would measure near-static heat patterns. The heat source can be as simple as a heat lamp so long as the appropriate heat energy is applied to the inspection surface. The induced temperature rise is a few degrees and dissipates quickly after the heat input is removed. The IR camera records the infrared patterns. The resulting temperature data are processed to provide more quantitative information. An operator analyzes the screen and determines whether a defect was found. Because infrared thermography is a radiometric measurement, it can be done without physical contact. Depending on the spatial resolution of the IR camera and the size of the expected damage, each image can be of relatively large area. Furthermore, as composite materials do not radiate heat nearly as much as aluminum and have higher emissivity, thermography can provide better definition of damage with smaller heat inputs. Understanding of structural arrangement is imperative to ensure that substructure is not mistaken for defects or damage.

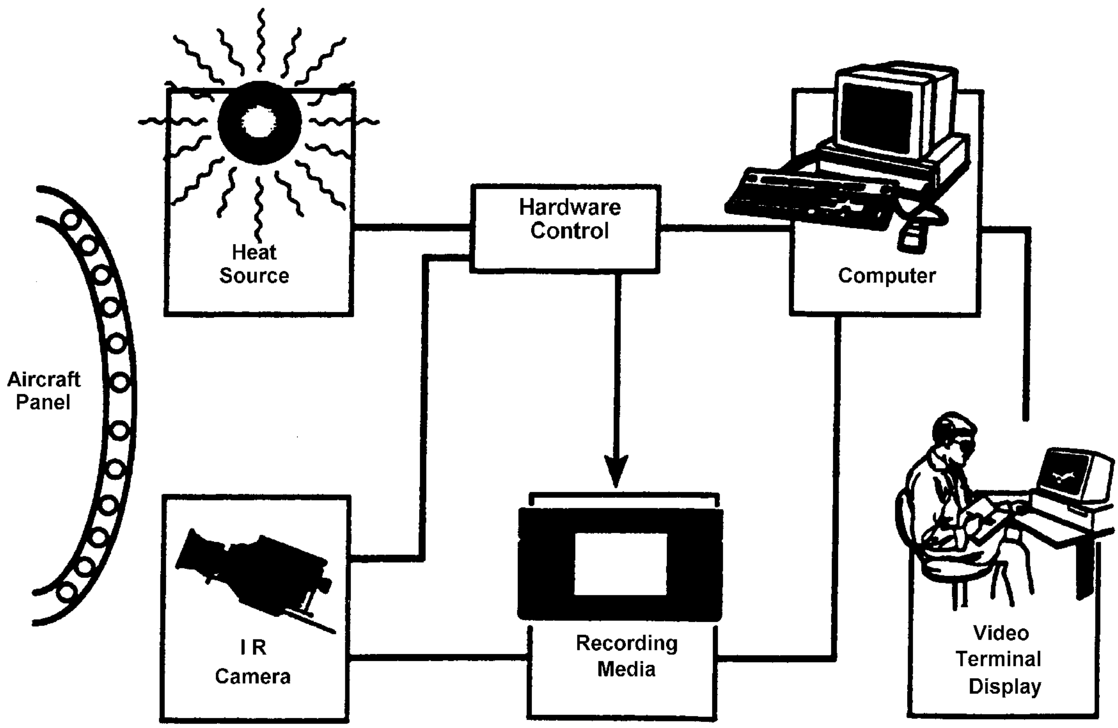


FIGURE 8.3.1.6 *Typical components needed for an advanced infrared inspection system.*

8.3.2 Damage assessment for composite repairs

8.3.2.1 General

The damage assessment is the intermediate stage between inspection and repair and includes taking a decision about if and how to repair a damaged structure, the nature of the repair (permanent or temporary), and the needed inspection after the repair and during the residual life of the repaired structure.

This decision depends upon where the damage is detected, the accuracy of damage characterization, the means available in determining the severity of the damage, and designing and performing an adequate repair.

8.3.2.2 Mandate of the assessor

The mandate of the assessor is the authority to interpret the inspection results and to decide about the needed repair and residual life of the structure. This is strongly dependent on the available information and expertise of the assessor. In the field, the mandate of the assessor is limited to following the manufacturer's instructions. In a repair station, and at the manufacturer's facilities, it can be extended, provided that engineering approvals and for civil aircraft authority approvals, are obtained. For larger damages, experimental substantiation may be required.

8.3.2.3 Qualification of the assessor

The assessment process is one of integration. The assessor should have the technical background to understand the inspection results, to understand the available design information, be familiar with the repair capabilities and have the necessary skills and experience. The demands on its technical expertise vary according to the repair location, for example, FAR Part 65, Certification: Airman other than crew members, prescribes the requirements for issuing Mechanics and Repairman certificates. The assessment of in field damage for civil aircraft are done by a certified mechanic, appropriately rated, with knowledge in composites and under the restrictions stated by the aircraft manufacturer in the Maintenance Manual and Structural Repair Manual. In other fields of activity, for example, in the military, similar criteria exist to define skill and experience requirements. In a Repair Station and at the Manufacturers Site, the assessment should involve a team, including engineering design and analysis.

8.3.2.4 Information for damage assessment

The following information is needed in the assessment process:

<u>Damage Characterization</u>

- Damage geometry, includes damage kind (delamination, cut, hole, etc.), dimension, form.
- Damage location, includes position on the part (in composite laminates in-plane location as well as depth should be considered), vicinity to other structural elements or systems, vicinity to other damages and repairs.

This information is a function of the inspection capability.

<u>Degradation of the structure due to the damage</u>

The assessor must consider the design requirements of the structure and the criteria to which it was designed before embarking on any repair actions.

<u>Repair Capabilities</u>

The repair methods recommended for composite structures are detailed in the next section. However, the repair capability and the availability of means to perform the repair have to be evaluated at this stage as part of the decision if and which repair to perform. For example, if bonding facilities are not available or if not enough time for the curing process is available, an equivalent bolted repair may be performed, or a temporary repair may be chosen, in order to arrive at a site where capabilities are available.

8.3.2.5 Dependence on repair location

The information for damage assessment depends on the location of damage detection and repair. Table 8.3.2.5 summarizes the information available at different locations.

TABLE 8.3.2.5 *Damage assessment per location.*

Location	Damage Information	Design Information	Repair Capability	Qualification of Assessor	Mandate of Assessor
In the field	Limited	Limited, manufacturer instructions	Limited, means and time, facility conditions	Mechanic or repairman*	Limited to manufacturer repair manual
Repair Station	Partial, varies according to station	Partial, manufacturer instructions and some functional and design information	Partial, varies according to station	Mechanic or repairman* and engineering support	Partial, requires manufacturer and civil authority approval or military depot engineering deposition
Manufacturer	Complete, equipment and know how available	Complete, design information, analytical capabilities, knowledge and mandate for certification	All facilities, from complicated repairs to rework	Engineering and manufacturing team	Ample, needs civil authority approval for changes to certified products

* Appropriately rated, with knowledge in composites (FAR part 65)

<u>In the field</u>: airport, airline, air force base, navy carrier, harbor (boat), road (car)

- Limited inspection equipment and, therefore, limited knowledge on the real extent of the damage (for example, due to impact damage, the upper skin and paint have visible damage, by tapping there is an indication that there is a delamination, but the exact dimensions of the delamination, their depth, number, etc., are not known).

- No knowledge how the damage affects the structural integrity of the part.

- Limited repair capabilities.

- Time limits.

In this case, it is the responsibility of the designer of the part to define maximal damage that can be repaired in field conditions and the repair method based on his knowledge of the structure and knowledge of the inspection and repair capabilities of the user. For civil airplanes, this is required by FAA regulations, Part 43, which state that repair methods and personnel should be FAA approved. Reference 8.3.2.5(a) contains a chapter relating to Repair of Laminated Structures, limited to fiberglass. A standard repair manual should include the detection method, the maximum damage which can be left unrepaired, the maximum damage which can be repaired and the repair methods.

Repair Station

Repair stations may vary according to capability and rating.

A repair station may be part of an airline, manufacturer, military repair station, or facility specializing in repair. For civil aircraft, the repair station has to be FAA approved. AC No. 145-6 (Reference 8.3.2.5(b)) provides information and guidance concerning means of demonstrating compliance with the requirements of 14 CFP parts 21, 43, 121, 125, 127, 135, and 147 regarding procedures and facilities for repair and alterations of structure consisting of metal bonded and fiber reinforced materials.

A repair station may repair damage exceeding the limits posed by the manufacturer in the repair manual under an appropriate engineering disposition. For civil aircraft, such a repair has to be FAA (or other civil authorities) approved. In the U.S. military, components requiring a repair that exceeds the limits specified in the repair manual are referred to the weapon system (aircraft) program manager for engineering disposition. Depending on the damage to the part and field capabilities, the engineering disposition may specify a field repair, the assignment of a depot repair team or transfer of the part to the depot for repair.

The following information is available at the repair station:

- Inspection capabilities of a repair station are more ample than the field capabilities, but may differ according to the type or class of station.

- Design information: engineering support (drawings, special problematic areas, lightning protection, electromagnetic transmission, etc.) should be supplied by the manufacturer. Analytical means have to be available to evaluate the performance of the designed repair.

- Repair facilities must be rated for the type of repair performed (for example, to include clean room, appropriate storage and curing facilities for bonded repairs).

Manufacturer

When a part is being returned for repair to the manufacturer, all resources concerning inspection, structural definition, and repair methods are available, and major repairs or reworks can be performed. The manufacturer may send a repair team to perform the repair in field; this will also be a manufacturer repair. Military repair depots perform remanufacture in-house to original manufacturer's specifications.

Again, the repairs have to be substantiated, and for civil aircraft, approved by the civil authorities.

8.3.3 Repair design criteria

Repair design criteria fulfill a function of assuring that the structural integrity and functionality of the repaired part are the same as that of the undamaged part. The repair design criteria should be established by the original manufacturer or cognizant engineering authority and used to develop repairs in the Structures Repair Manual (SRM). They are implicitly followed by the operator or the repair station when the repairs are made within the scope of the SRM. When a repair is designed which exceeds the limits of the SRM, the repairs must be substantiated and approved based on the specified repair criteria.

SRM's for specific aircraft frequently "zone" the structure to show the amount of strength restoration needed or the kinds of standard repairs that are acceptable. Zoning permits the use of simpler repairs in areas where large strength margins exist. Zoning also restricts operator repairs in areas where repairs are too complex and should be only repaired with original equipment manufacturer's (OEM's) involvement.

Repair design criteria for permanent repairs are fundamentally those that designed the part that is to be repaired. These are: restore stiffness of the original structure, withstand static strength at the expected environments up to ultimate load including stability (except for postbuckled structure), assure durability for the remaining life of the component, satisfy original part damage tolerance requirements, and restore functionality of aircraft systems. Additionally there are other criteria applicable in repair situations. These are: minimize aerodynamic contour changes, minimize weight penalty, minimize load path changes, and be compatible with aircraft operations schedule.

8.3.3.1 Part stiffness

First consideration in any repair is to replace structural material that is damaged. This means that especially for large repairs the stiffness and placement of repair material should match the parent material as closely as possible. This avoids any recalculations of the overall dynamic behavior of the component, such as flutter or structural load redistribution. Furthermore, many lightweight flight vehicle structures are designed to meet stiffness requirements that are more critical than their strength requirements. A repair made to a structure of this type must, therefore, maintain the required stiffness so that deflections or stability requirements are met.

Fixed aerodynamic surfaces, such as wings and tails, are frequently designed to have bending and torsion stiffness that are adequate to prevent excessive deflections under aerodynamic loading. This is to prevent divergence and control surface (such as aileron) reversal. Moveable surfaces are frequently sensitive to aerodynamic flutter and their stiffness may have been carefully tailored to obtain natural frequencies for which flutter will not occur. Effects of added weight are discussed in Section 8.3.3.7.

Increasing the stiffness of a control surface, especially the bending stiffness, can reduce the flutter speed to unacceptable levels; a decrease in stiffness can be equally damaging. Any significant change in stiffness must be evaluated for its effect on the dynamic behavior of structure. Stiffness can also affect the deflections of actuated doors, such as landing gear doors. Reduction in stiffness can result in excessive deflections under aerodynamic loading. These reduction may increase drag or in extreme cases cause loss of the structure.

8.3.3.2 Static strength and stability

Any permanent repair must be designed to support applied loads at the ultimate design load level at the extremes of temperature excursions, moisture levels, and barely visible damage levels. If the loads are not available, specific SRAM repair recommendations must be strictly adhered to. In the SRAM repairs, there is an implicit assumption that the specific repairs meet all static strength and stability requirements.

Load path changes are a special concern when designing repairs. When strength restoration is necessary, attention must be given to the effect of the stiffness of the repair on the load distribution in the structure. If a patch has less stiffness than the original structure, the patch may not carry its share of the load, and this causes an overload in the surrounding material. This condition can be caused by a patch made from a less stiff material, or from fasteners that fail to transfer full load because of loose fits or fastener deformation. Conversely, an overly stiff patch may attract more than its share of load, causing adjacent areas to which it is attached to be overloaded. Stiffness mismatch between parent material and the patch may cause peel stresses that can initiate debonding of the patch.

Structures loaded in compression or in shear, such as some wing skins, webs of spars or ribs, and fuselage structure, including both external skins and internal bulkheads, may be stability and not strength critical at ultimate design load. Two types of stability failure are possible:

1. Panel Buckling - The panel, such as a section of wing skin, buckles between its major supports, for example, spars and ribs. The repair must account for the stiffness of the panel and the amount of support provided by the attachment to the substructure. Some portions of structure,

i.e., wing skins between spars, are permitted to buckle below ultimate design load. These types of structures develop specific post-buckling behavior which redistributes the load and allows the structure to carry ultimate load. Any repair of a stability critical structure, and especially a structure that is permitted to buckle, should be considerate in not affecting its buckling and post-buckling modes. Matching of stiffness of the parent material is of utmost importance here.

2. <u>Local Crippling and Buckling</u> - This is buckling of the cross section of a member or its component, such as a spar cap, by distortion of the cross section rather than the overall buckling along its length or width. Restoration of local crippling strength must be considered when making repairs to substructure.

Composite laminates under compressive load can fail when individual fibers or bundles of fibers buckle where delaminations or penetrations result in fibers with reduced support. Because of the danger of microbuckling or local ply buckling, resin injection repair that fills a delamination without adequately bonding the delaminated plies together could be unsatisfactory.

8.3.3.3 Durability

Durability is the ability of a structure to function effectively throughout the life of the vehicle. For commercial transport aircraft, the design life can be greater than 50,000 cycles; military fighter aircraft are designed for 4,000 to 6,000 flight hours. Included among the factors affecting durability are temperature and moisture environments (covered in Sections 8.3.3.8 and 8.3.3.9).

Although the parent composite structure may not be durability critical, structural repairs may be more susceptible to damage caused by repeated loads during their service lives. This is because the repair process is not as well controlled and the repairs themselves create solitary joints and discontinuities in areas that are exposed. For bolted repairs, high bearing stresses on fastener holes should be avoided as they may elongate under repeated loading and lead to fastener fatigue. Bonded repairs should be well sealed as they can develop disbonds after being weakened by environmental effects. All found delaminations exceeding the acceptance/rejection criteria of the SRM should be repaired as unrepaired delaminations may grow under compressive or shear loading. Bolted repairs of sandwich structure must be sealed.

8.3.3.4 Damage tolerance

Composite structures are designed to be damage tolerant to accidental damage. In practice, this is accomplished by lowering design strains so that the structure with impact caused damage can withstand ultimate load. Repairs must also be capable of tolerating a predetermined level of impact damage. The level of impact damage is usually established by OEM's with concurrence of certifying agency. When using metal for large damage repair, damage tolerance requirements for metallic structure must be followed. Metallic doublers and parts will also require protection against galvanic corrosion and lightning strike.

8.3.3.5 Related aircraft systems

In addition to satisfying structural criteria, compatibility with related aircraft systems may also be required of the repair. These systems include:

1. <u>Fuel System</u> - Structure is frequently used to contain fuel, as in the "wet" wings of many aircraft. A repair must seal adequately to prevent leakage of the fuel. The repair may also be subjected to fuel pressure loading. Repair material must be compatible with fuel.
2. <u>Lightning Protection System</u>- Some composite structure has provision for conducting lightning strikes by use of flame-sprayed coatings, bonded metallic strips, wire mesh, etc. A repair to the structure must restore the electrical continuity as well as the structural strength. Bolted repairs around fuel tanks must avoid creating an electrical path.
3. <u>Mechanical System</u> - Components that are mechanically actuated, such as landing gear doors or control surfaces, must function correctly after repair. Clearances and fit-up to adjacent fixed structure may be critical. Re-rigging or re-balancing may be required after repair.

8.3.3.6 Aerodynamic smoothness

High-performance flight vehicles depend on smooth external surfaces to minimize drag. During initial fabrication, smoothness requirements are specified, usually by defining zones where different levels of aerodynamic smoothness are required. Most SRM's specify smoothness requirements for repairs consistent with initial part fabrication.

The most critical zones typically include leading edges of wings and tails, forward nacelles and inlet areas, forward fuselage, and overwing areas of the fuselage. The least critical zones typically include trailing edges and aft fuselage areas. In addition, intermediate zones may be specified. For the most critical zones, forward-facing steps are usually limited to 0.005 to 0.020 in. (0.13 to 0.51 mm)at permanent butt joints. At removable panels, mechanical doors, and major joints, forward-facing steps from 0.010 to 0.030 in. (0.25 to 0.76 mm) are typically allowed. At installed equipment, such as antennas and navigation lights, steps up to 0.020 to 0.040 in. (0.51 to 1.02 mm) are permitted. All sharp edges as the result of patch ply termination should be smoothed and feathered.

Whatever the requirements, each exterior repair should restore aerodynamic contour accurately and smoothly as structurally and economically feasible. Trade-off exist between accepting a slight reduction in performance in order to accept a repair that is more structurally sound and that is easier and quicker to accomplish.

8.3.3.7 Weight and balance

Compared to the overall weight of the vehicle, the weight added by most repairs is insignificant. Exceptions may exist for very large repairs or for space vehicles.

The weight of repair becomes a major concern when the repair changes the mass balance of components sensitive to dynamic response, such as moveable control surfaces, rotor blades, and rotating shafts. In such cases, it may be possible to remove as much damaged material as will be added by the repair so that there is little change in weight and moments of inertia. If that is not possible, the part must be re-balanced after repair.

8.3.3.8 Operating temperatures

Most flight vehicles experience extremes of temperature during use. Repairs to such flight vehicles must be acceptable for the temperature extremes for which the vehicle was designed. Low temperatures result from high-altitude flight or from extremes of ground storage in cold climates. Many aircraft are designed for a minimum service temperature of -65°F (-54°C). Elevated-temperature requirements vary with the type of vehicle. The maximum temperature for commercial transport aircraft and most rotary wing vehicles is 160°F (71°C) and generally occurs during ground soak on a hot day. However, components experiencing significant loads during takeoff and initial climb may require validation of design ultimate loads at temperatures up to 200°F (93°C). Supersonic transport, fighter, and bomber aircraft typically experience aerodynamic heating of up to 220°F (104°C) or in special cases as high as 265°F (130°C), especially on the leading edges of lifting surfaces. Components exposed to engine heat, such as nacelles and thrust reversers, may be required to withstand even higher temperatures in local areas.

Operating temperature influences the selection of repair materials: resin systems for prepreg repairs, resins for wet lay-up repairs, and adhesives for bonded repairs. Materials that develop adequate strength at the required temperature must be selected. The combination of temperature extremes with environmental exposure (especially moisture) frequently is the critical condition for which the repair must be designed.

8.3.3.9 Environment

Repairs may be exposed to many environmental effects, including those listed below:

1. Fluids - salt water or salt spray, fuel or lubricants, hydraulic fluid, paint stripper, and humidity
2. Mechanical loading - shock, acoustic or aerodynamic vibration, and operating loads
3. Thermal cycling

Moisture is particularly critical to the polymeric matrix composites. At elevated temperature absorbed moisture reduces the ability of the matrix to support the fibers, thereby reducing the strength of the laminate for compressive or shear loading. This effect is considered in the original design, and allowable loads are frequently limited by "hot-wet" conditions. The same considerations pertain to bonded repairs.

Absorbed moisture can affect bonded repairs in three ways. These must be considered in the selection of a repair procedure.

1. Parent Laminate Blistering- As a "wet" laminate is heated to cure a bonded repair, the absorbed moisture may cause local delaminations or blisters. Pre-bond drying at lower temperatures, slow heat-up rates, and reduced cure temperatures all diminish the tendency to blister.
2. Blown Skins/Core of Sandwich Structure - Moisture in the cells of honeycomb sandwich structure expands when the part is heated to cure a bonded repair and develops sufficient pressure to separate the skin from the core, especially if the strength of the adhesive has been reduced by temperature and moisture. Similarly, this process may be sufficiently severe to rupture cell walls in the low density core. Pre-drying is normally used to prevent bondline failure of this type.
3. Porosity in Bondlines - As a repair is bonded to a "wet" laminate, the moisture tends to cause porosity in the bondline. This porosity can reduce the strength of the bondline. This problem can be minimized by pre-drying, reduced temperature cure, and selection of moisture-resistant adhesives.

8.3.3.10 Surrounding structure

In the course of the repair process it is imperative that the surrounding structure does not sustain any damage. The predominant sources of damage are dropped tools, scratches caused by prying of bagging material, and the application of high temperatures during the cure of the repair. If there is a potential for the latter damage, resins should be selected that cure at sufficiently low temperature while still capable of hot, wet performance.

8.3.3.11 Temporary repair

Repair design criteria for temporary or interim repairs can be less demanding, but may approach permanent repairs if the temporary repair is to be on the airplane for a considerable time. Most users of aircraft and OEM's prefer permanent repairs, if at all possible, as the temporary repairs may damage parent structure necessitating a more extensive permanent repair or part scrapping. All temporary repairs have to be approved before the aircraft can be restored to operational status.

Temporary repair will restore functionality of the aircraft and its systems but on the temporary basis. Static strength requirements may be reduced to limit load or maximum load in the spectrum. Stiffness requirements may be reduced to a level where they do not cause overall buckling or flutter. Damage tolerance and durability goals are often severely reduced or not considered but are compensated by shorter inspection intervals.

A special subset of temporary repairs are those associated with aircraft battle damage repair (ABDR) and other emergency repairs. In this situation repair design criteria will require sufficient strength, stiffness, and functionality restoration to permit the aircraft to fly to a repair facility or sustain 100 hr of limited flight envelope, or in the ABDR scenario fly one more mission. In the military, there exist ABDR manuals which suggest the types of repairs to be implemented. These repairs are usually required to be accomplished within 24 hours.

8.3.4 Repair of composite structures

8.3.4.1 Introduction

The task of repair begins only after the extent of the damage has been established by cognizant personnel using inspection methods described in Section 8.3.1 and damage assessment as described in Section 8.3.2. The repair has the objective of restoring the damaged structure to a required capability in terms of strength, stiffness, functional performance, safety, service life, and cosmetic appearance. Ideally, the repair will return the structure to original capability and appearance. To start the repair process the structural makeup of the component must be known and the appropriate design criteria should be selected from the considerations described in Section 8.3.3. The continuity in load transfer is re-established in a damaged part by attaching new material by bolting or bonding thus bridging the gap or reinforcing the weakened portion. Thus the repair is in reality a joint where a load is transferred from the parent material into and out of the patch.

Repair design criteria, part configuration, and the logistic requirements will dictate whether the repair should be bolted or bonded. Some of the main drivers that determine the type of repair being more suitable are listed below.

Condition	Bolting	Bonding
Lightly Loaded, Thin (<0.10 in. [2.5 mm])		X
Highly Loaded, Thick (>0.10 in.[2.5 mm])	X	X
High Peeling Stresses	X	
Honeycomb Structure		X
Dry and Clean Adherend Surfaces	X	X
Wet and/or Contaminated Adherend Surfaces	X	
Sealing Required	X	X
Disassembly Required	X	
Restore Unnotched Strength		X

In any case, the Structures Repair Manual (SRM) for the particular component will provide guidance as to the type of repair to be applied.

8.3.4.2 Damage removal and site preparation

Once the repair perimeter has been established around the damage, the task of damage removal begins. The first step is the removal of finish topcoat by hand sanding or other mechanical means. The use of chemical paint stripper is prohibited as it can attack the composite resin system and can also become entrapped in the honeycomb core. Once the topcoat and primer are removed and the damaged plies clearly defined, the damaged plies are then removed either by sanding or other mechanical means, if the damage is partial through the thickness, or by trimming, if the damage is through the laminate. In either case, a well-prepared site should have a well-defined geometric shape with smoothed out corners. Damaged core must be cut out, with special care taken not to damage the inner surface of the opposite (non-damaged) composite skin.

Once the damage has been removed, the repair area should be checked for evidence of moisture and/or contaminants. Contaminants, such as hydraulic fluids or engine oils, will saturate the composite materials making it extremely difficult to obtain a clean bond surface. They may also degrade the mechanical properties of the composite materials. Undetected moisture will turn into steam during elevated temperature cure. The steam will seek an escape path from within the panel causing blown core and laminate disbonding. It has also been shown that patches bonded to parent composite material containing more than a nominal (0.3% moisture content by weight) experience lower adhesive bond strengths. For honeycomb parts cured at room temperature, presence of moisture is undesirable, particularly if the core material is aluminum. SAE ARP 4916 (Reference 8.3.4.2(a)) and ARP 4977 (Reference 8.3.4.2(b)) give guidelines how the composite part should be cleaned and dried before proceeding with the repair.

For bonded repairs, site preparation for installation of repair usually involves taper sanding or step cutting of plies. This is done so that there is gradual introduction of load into and out of the repair material. For external patches, additional consideration for step patching is to minimize intrusion into the air stream. SRM's usually specify the taper angle, overlap and step lengths.

8.3.4.3 Bolted repairs

8.3.4.3.1 Repair concepts

Bolted repairs can comprise an external or an internal patch that results in a single shear joint, or two patches, one on each side that leads to a double shear joint, see Figure 8.3.4.3.1. In both cases the load is transferred through the fasteners and the patch by shear forces, but in the case of the two-patch repair, transfer load eccentricity is minimized. The main disadvantage of bolted repairs is that the new holes created in the parent structure weaken the structure by creating stress concentrations that become damage initiation sites.

The external bolted patch is the easiest repair to fabricate. The patch overlaps the parent skin with sufficient area to install the required amount of fasteners to transfer the load. For large repairs the patch may be stepped and different size fasteners may be used in different rows to ease the load transfer. The external patch thickness may be limited by aerodynamic considerations and by the induced load eccentricity due to neutral axis offset. However, this type of repair does not need backside access as the fasteners can be blind, i.e., being able to be installed from one side only. If the external patch is unfeasible, an internal patch can be applied. When backside access is not possible, the patch is split to allow insertion through an elliptical or circular cutout in the skin. In some cases the damage must be enlarged in the direction of the primary load to effect the repair. Because of hardware, internal bolted patches may have interference problems with substructure members. The two-patch repair using external and internal patches is a desirable repair from the load transfer point-of-view, however, the repair is more complicated and is heavier.

For complex repairs, multi-row fastener patterns will be required to gradually introduce the load from the part being repaired into the repair patch. It is virtually impossible to distribute the load evenly between all the fasteners in a multiple row pattern, but careful design of patch geometry, fastener diameter and spacing can alleviate the high loads at the first fasteners. Such complex repairs are not usually identified in the approved repair manuals or procedures (SRM, TO, or TM) and normally need engineering input for design.

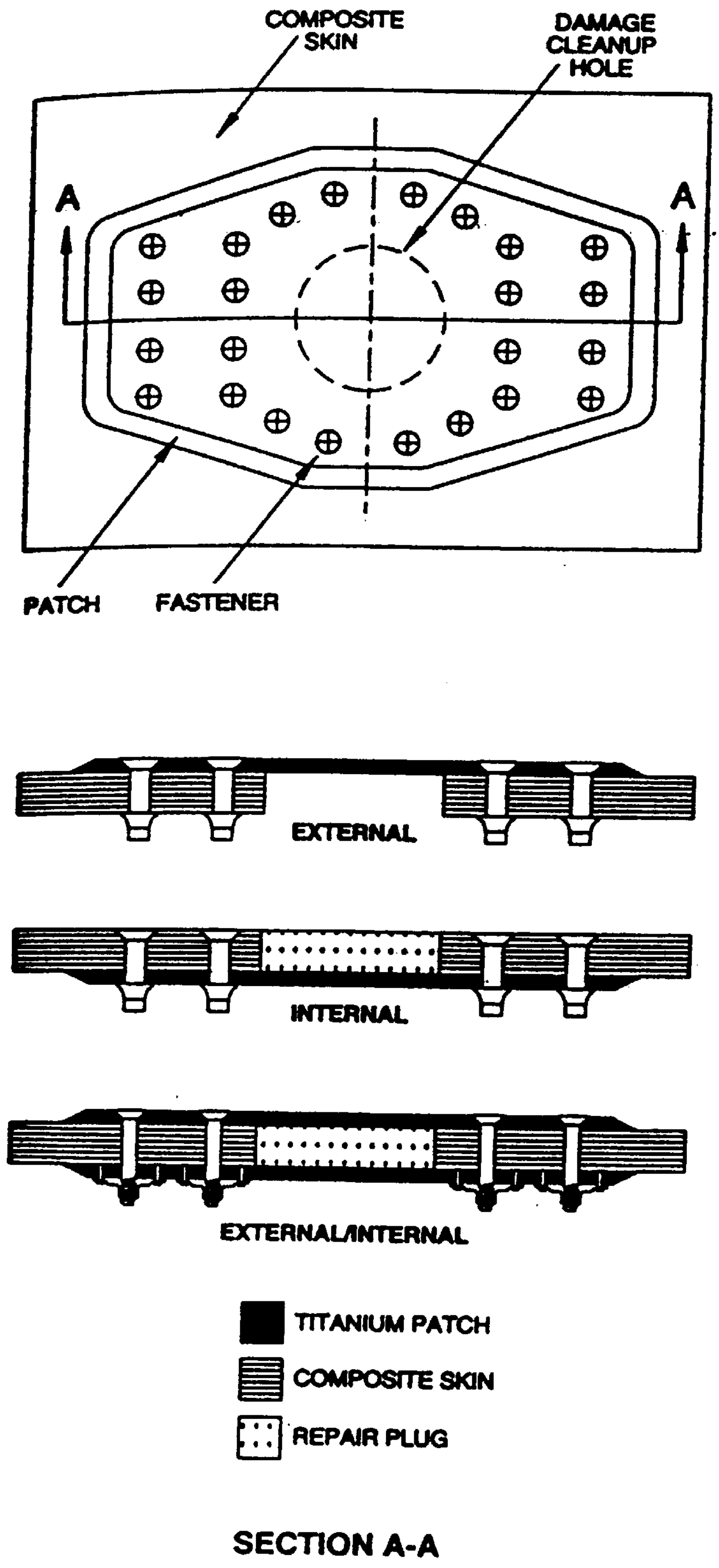


FIGURE 8.3.4.3.1 *Basic repair joints (bolted).*

8.3.4.3.2 Repair materials

For bolted repairs, there is only the need to select patch material and fasteners. Patches may be aluminum, titanium or steel, or pre-cured composite, carbon/epoxy or fiberglass epoxy. For aluminum patch repair on carbon parent material, a layer of fiberglass cloth is placed between them to prevent galvanic corrosion. For repair of highly loaded components, titanium or pre-cured carbon/epoxy patches are usually preferred. For repair of high strain structure coupled with severe fatigue load environment, carbon/epoxy patches can be more effective. Pre-cured carbon/epoxy patches will have the same strength and stiffness as the parent material as they are usually cured and inspected similarly. The major disadvantages of this type of patch are that they do not conform to curved or irregular surfaces and that warpage during pre-cure can result in poor fit requiring shimming.

For repair of composite parts the choice of fasteners is limited to titanium, Monel, or stainless steel. The choice of fastener type is strictly controlled by the SRM. A discussion on fasteners for composites can be found in Chapter 6 of this volume.

There is a general misconception that bolted repairs require very little logistics support in terms of materials. This is false, as many types of fasteners with different grip lengths need to be stored. As fasteners for composites are expensive, the inventory can be costly. If pre-cured carbon/epoxy patches are used, different patch sizes and thicknesses have to be available, as cutting to size requires specialized equipment.

8.3.4.3.3 Repair analysis

Analysis of a bolted repair follows the guidance lines of the analysis of a bolted joint, Volume 3, Chapter 6.3. In the following, the main steps will be presented, with emphasis on items specific for repairs.

a) Estimation of load transferred through the repair

As defined in the Introduction, the repair is a joint where load is transferred from the parent material into and out of the patch. The estimation of the transferred load through the repair is the first stage in the repair analysis.

The two situations where there is need for analyses of repair are during the writing of the SRM or when damage that exceeds the allowed SRM limits has to be repaired (Section 8.3.2 addresses the repair mandate and certification requirements). The SRM is written by the manufacturer, who has all the needed information from the analysis of the undamaged structure. In the second case, the load information has to be obtained from the manufacturer. In special occasions, especially for temporary repairs, loads can be approximated by reverse engineering, utilizing the known design of the parent structure. Care should be taken to use conservative approximations that are based on the maximum load that can be sustained by the geometry and lay-up of the parent structure.

b) Load sharing in the repair

After the load transferred through the repair is known, the distribution of this load between the various fasteners, and then, in the region of each fastener between the parent structure, the patch and the fasteners, has to be evaluated. The analysis is done according to Volume 3, Chapter 6.3.2.1.

c) Analysis of local failure

- *Parent structure*: The parent part of the joint may not be adequate to accommodate the mechanically fastened joint. It may not have the adequate thickness or the proper lay-up to provide the bearing resistance. As the lay-up cannot be changed, the only recourse is to bond additional plies. However, care must be taken so as not to end-up with a highly unsymmetrical lay-up. Care must also be taken to properly estimate the bearing/by pass ratio and to consider all possible laminate failure modes (Volume 3, Figure 6.3.2.3(a)), in order to avoid increasing the damage by failing the periphery of the repair. Analysis techniques follow Volume 3, Chapters 6.2.2.2 and 6.3.2.3. For the case of repairs to be incorporated into the SRM, a test program is usually performed to verify the analysis and substantiate the repair.

- *Patch structure*: In the patch design there is freedom to select composite material, lay-up, and thickness according to the analytical results. Patches can be prepared to provide the accurate strength, stiffness, edge distance and bolt spacing. In cases where composite patches are being used the analysis can be performed according to Volume 3, Chapters 6.2.2.2 and 6.3.2.3.

- *Fasteners*: Fastener stiffness should be determined by test or analysis and subsequently used in the analysis of the overall repair. Fastener tensile and shear stresses should be determined as to their adequacy for static strength and for fatigue loading. Fastener selection is addressed in Volume 3, Chapter 6.3.3.3.

8.3.4.3.4 Repair procedures

This section will describe general procedures to complete a bolted repair. Specific repair procedures are given in SRM's, NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)), and Air Force TO 1-1-690 (Reference 8.3.4.3.4(b)). An example of typical bolted repair will be described at the end of this section. Bolted repair procedure consists of six distinct steps: (1) patch preparation and pilot drilling holes, (2) laying out hole pattern on the parent skin and pilot drilling skin holes, (3) the transfer of the holes in the skin to the patch if the patch covers some existing skin holes, (4) drilling/reaming of patch and skin, (5) patch and fastener installation, and (6) sealing of the repair.

The first step is to cut, form and shape the patch before attaching the patch to the damaged structure. In some cases the repair patches are stocked pre-shaped and pre-drilled. If cutting is to be performed, standard shop procedures should be used that are suitable for the patch material. Metal patches require filing to prevent crack initiation around the cut edges. When drilling pilot holes in the composite, the holes for repair fasteners must be a minimum of four diameters from existing fasteners and have a minimum edge distance of 2 1/2 fastener diameters. This is different than for metals where the edge distance of two is standard practice. Specific pilot hole sizes and drill types to be used should follow specific SRM.

To locate the patch on the damaged area, two perpendicular centerlines are drawn on the part that define the principal load or geometric directions. The hole pattern is then laid-out and the pilot holes in the skin are drilled. The principal directions of the patch are then aligned between the patch and the parent structure. The edges of the patch are marked so that it can be returned to the same location. After the patch is removed, it is advisable to check if there is sufficient edge distance between the patch perimeter and the outer holes. The pilot holes in the patch are then enlarged.

Composite skins should be backed-up to prevent splitting. The patch is then reattached through the interior fasteners so that the corner fastener holes can be enlarged. All holes are then reamed. A tolerance of (+0.0025/-0.000 in. [+0.06/-0.00 mm]) is usually recommended for aircraft parts. For composites this means interference fasteners are not used.

Once fastener holes are drilled full size and reamed, permanent fasteners are installed. Before installation the fastener grip length must be measured for each fastener using a grip length gage. As different fasteners are required for different repairs, SRM should be consulted for permissible fastener type and installation procedure. However, all fasteners should be installed wet with sealant and with proper torque for screws and bolts.

Sealants are applied to bolted repairs for prevention of water/moisture intrusion, chemical damage, galvanic corrosion and fuel leaks. They also provide contour smoothness. The sealant has to be applied to a clean surface. Masking tape is usually placed around the periphery of the patch parallel with the patch edges leaving a small gap between the edge of the patch and the masking tape. Sealing compound is applied into this gap.

8.3.4.3.5 Example of a bolted repair

External patch bolted repair of through penetration of the composite skin taken from NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)) is used here as an illustration example. The repair, shown schematically in Figure 8.3.4.3.5, is applicable to repair holes up to 4 in.(100 mm) in diameter of a thick monolithic skin. A single metallic plate is used to span the hole fastened to the skin by 40 blind fasteners. The repair assumes there is single side access. A scrim cloth is used to prevent galvanic corrosion. The applicability of this repair for specific application depends on loading conditions and laminate thickness.

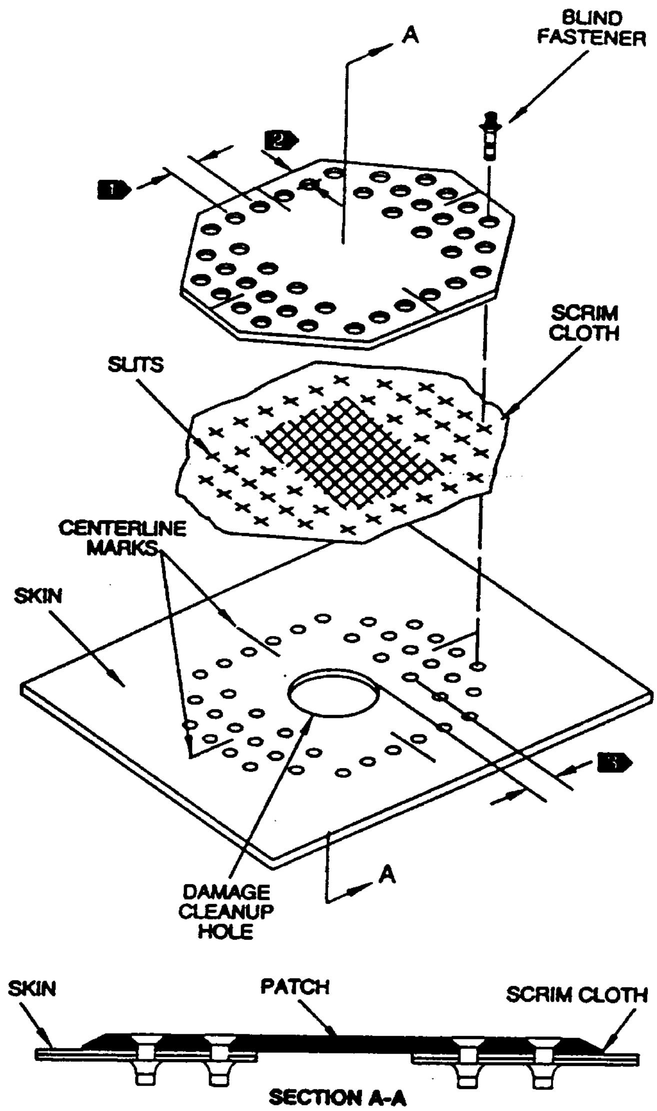
8.3.4.4 Bonded repairs

8.3.4.4.1 Repair concepts

The two most common bonded repairs use external patches or are internal patches that are made flush with the parent material, both shown in Figure 8.3.4.4.1. Combinations of both types of repairs are also common. Although the external patches are usually stepped, the internal repair can be stepped or more commonly scarfed. The scarf angles are usually small to ease the load into the joint and to prevent the adhesive from escaping. This translates into a thickness to length ratios between 1/10 to 1/40. The adhesive placed between the repair material and the parent material transfers the load from the parent material to the patch by shear. The external patch repair concept is the easier of the two to accomplish. Its drawbacks are eccentricity of the loading causing peel stresses and protrusion into the air stream. The stress concentration at the edge of the patch can be reduced by stepping or tapering the patch as shown in Figure 8.3.4.4.1. Because inspection of bonded repairs is difficult, bonded repairs, as contrasted with bolted repairs, require a higher commitment to quality control, better trained personnel, and cleanliness.

The scarf joint, Figure 8.3.4.4.1, is more efficient from the viewpoint of load transfer as it reduces load eccentricity by closely aligning the neutral axis of the parent and the patch. However, this configuration has many drawbacks in making the repair. First, to maintain a small taper angle, large quantity of sound material must be removed. Second, the replacement plies must be very accurately laid-up and placed in the repair joint. Third, curing of replacement plies can result in significantly reduced strength if not cured in the autoclave. Fourth, the adhesive can run to the bottom to the joint creating a non-uniform bond line. This can be alleviated by approximating the scarf with a series of small steps. For these reasons, unless the part is lightly loaded, this type of repair is usually performed at a repair facility, where if the part can be inserted into the autoclave, this type of repair can result in part strength as strong as the original part.

Although it may seem that there are only two common concepts, it is somewhat misleading as the two repair joints can be made by many different methods. The patch can be pre-cured and then secondarily bonded to the parent material. This procedure most closely approximates the bolted repair. The patch can be made from prepreg and then co-cured at the same time as the adhesive and lastly the patch can be made using dry cloth, paste resin, and co-cured. This latter repair is called “wet” lay-up repair. The curing cycle can also vary in length of time, cure temperature, and cure pressure, thus increasing the number of possible repair combinations.



LEGEND

1. FASTENERS MUST HAVE MINIMUM SPACING OF 4D AND MAXIMUM OF 6D
2. FASTENERS MUST HAVE MINIMUM EDGE DISTANCE OF 3D
3. FASTENER TO EDGE OF DAMAGE CLEANUP HOLE MUST BE A MINIMUM OF 3D

NOTE:
DRILL PILOT HOLES (0.128 INCH DIAMETER) IN PATCH FIRST. ENLARGE HOLES TO FINAL SIZE AFTER TRANSFERRING PILOT HOLES TO COMPOSITE SKIN.

FIGURE 8.3.4.3.5 *Repair arrangement, bolted repair, external patch.*

8.3.4.4.2 Repair materials

Bonded repairs require selection of both the repair material and adhesive. The selection cannot be independent as the curing parameters of the adhesive and the repair material must be compatible for co-cured repairs. Bonded repairs also require materials that are used in the processing of the repair but not remain with the repair. Many materials used in bonded repairs require special handling, are storage time and temperature sensitive, and may require controlled environment during the repair process.

A very good description of materials that are available and needed for bonded repair is provided in Air Force TO 1-1-690 (Reference 8.3.4.3.4(b)) and NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)). It should be noted that the mechanical properties of repair materials, other than metals, depend very strongly on the curing process used. As the repair curing process is usually different than the process used for fabricating the original part (lower curing temperature and/or pressure), material suppliers have developed unique materials that are optimized for the repair process. It should be noted that the repair materials are usually lower in strength and stiffness than the original part materials. Volume 2 contains a special section with repair material properties.

Metal patches for bonded repairs are constructed using sheet material that is bonded to each other to form a stepped patch. The same method is used for pre-cured composite patches where the sheets are made of two or more unidirectional plies of fabric or tape. Because the pre-cured patches can be cured in the autoclave, they are made from composite materials that were used in construction of the original part.

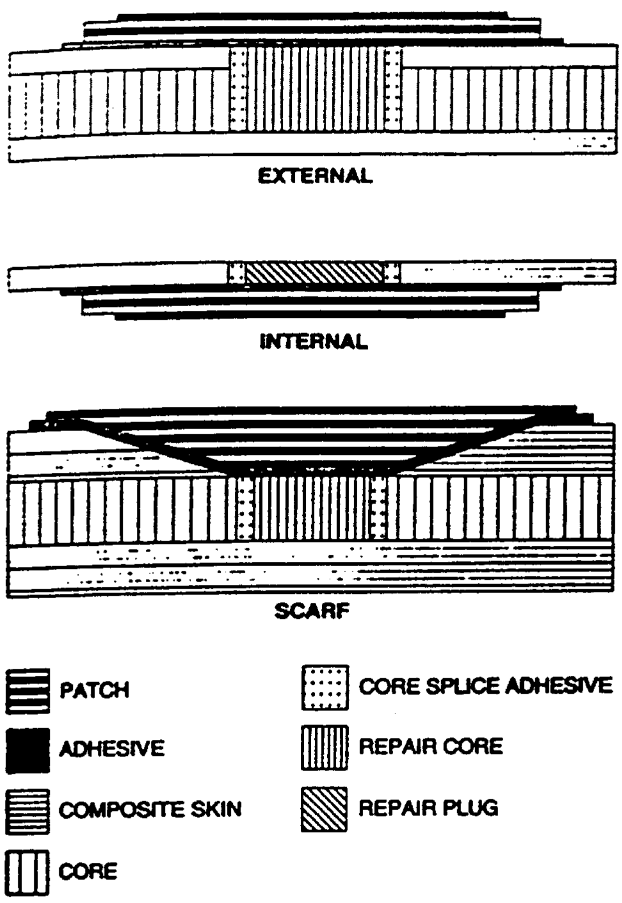


FIGURE 8.3.4.4.1 *Basic repair joints (bonded).*

Co-cured bonded repairs use parent material prepreg, if the repair can be cured in the autoclave, or repair material prepreg, or dry fabric with paste resin. The use of the latter materials defines 'wet' lay-up repair. The prepreg provides a uniform distribution of resin in the composite, but requires refrigeration for storage. The resin for the wet lay-up repair usually consists of two parts that do not require freezers. However, mixing of the two parts and spreading the mixed resin on the dry fabric requires strict adherence to written protocol and experienced personnel to effect consistent repairs. All composite repair materials require incoming material control or re-testing of key properties to assure the integrity of the material being used. AC 145-6 (Reference 8.3.2.5 (b)) has a discussion on incoming repair material requirements.

Adhesives for bonded repairs are discussed in Section 8.2.3 as to their desirable properties. Here the discussion will be limited to types of adhesive that are available and are being used. Two categories of adhesives are films and pastes. Films come with and without mesh carrier cloth with typical thickness between 0.0025 to 0.01 in. (0.064 to 0.25 mm). The carrier cloth provides improved handling, results in more uniform bondline, and helps reduce galvanic corrosion. Although films provide a more uniform bondline thickness than paste adhesives, repair part inaccessibility or a lack of refrigerated storage equipment sometimes necessitates use of paste adhesives. Wet lay-up repairs almost always use paste adhesives as they are more compatible with paste resins in terms of curing characteristics. As the paste resins, the paste adhesives consist of two separate parts that have a long shelf life. Conversely, film adhesives are more prevalent when prepreg is used to form repair patches as they usually require higher temperature and pressure for curing.

Bonding repairs require many ancillary materials. They do not become part of the repair and are removed and discarded after the repair is complete. They include items such as vacuum bag materials, scrim cloths, bleeder/breather materials, release films, tapes, wiping materials, and solvents. The specifications for these materials are usually given in the specific SRM.

8.3.4.4.3 Repair analysis

A bonded repair is from a structural point of view a bonded joint. As in a joint the load is transferred from the parent structure by the bond to a patch (single lap) bypassing the damaged portion of the parent structure. The geometry is usually two dimensional. If a sandwich structure is repaired, the core, repaired original or new replaced, forms a substrate which provides support for the out-of-plane loads. This is why bonded repairs are very efficient for sandwich structures. The repair analysis of a bonded repair follows the guidance lines of the analysis of a bonded joint, Volume 6, Chapter 6.2.3. The main steps as for bolted repairs (Section 8.3.4.3) are as followed:

a) Estimation of load transferred through the repair

As for bolted repairs, Section 8.3.4.3.3.

b) Load sharing in the repair

The load flow in a bonded repair is continuous. It depends on the elastic properties of the adherends and the adhesive and on joint geometry. In some cases, the geometry can be approximated by the use of models of lap or strap joint. A two dimensional finite element model can be used to calculate load distributions in the skin, patch, and adhesive layer. A nonlinear solution can be used to account for the nonlinear stress strain behavior of the adhesive (Volume 3, Chapter 6.2.3.6).

Several specially developed computer codes can be used for analyzing bonded repairs. In Reference 8.3.4.4.3(a) the codes PGLUE (Reference 8.3.4.4.3(b)), A4EI (Reference 8.3.4.4.3(c)) and ESDU8039 (Reference 8.3.4.4.3(d)) are discussed. The PGLUE program contains an automatic mesher which creates a three dimensional finite element model of a repaired panel containing three components - a plate with a cutout, a patch, and an adhesive connecting the patch and the plate. Plasticity of the adhesive is considered in the analysis. However, the version available through ASIAC does not consider peel stresses, which can be critical. Traditional bonded joint codes, such as A4EI and ESDU8039, model only a slice through the repair and do not consider the two dimensional effects of stiffening of the sides of the panel. Both bonded joint codes allow the patch to be stepped. A4EI considers plasticity in the adhesive shear stress but does not predict peel stress, while ESDU8039 predicts peel stress in the joint but does not consider plasticity.

c) *Analysis of local failure*

- *Parent Structure*: As in the case of bolted repairs the parent structure is a given item in the repair design. The advantage of the bonded repair is that loads are introduced into the parent structure in a continuous way without inducing any stress concentrations in the parent structure and thus there is no need for increase in thickness in the joint region. After establishing the stress distribution in the parent structure, the stress and failure analyses are performed according to Volume 3, Chapters 5.3 and 5.4.

- *Patch Structure*: as for the parent structure.

- *Adhesive*: Volume 3, Chapter 6.2.3 deals extensively with stress analysis of adhesive joints, however, failure criteria are not covered presently. The following should be taken into consideration:

- The joint should be designed in such a way that the adhesive layer is not the critical joint element.
- Peel and transverse shear stresses should be minimized by design (tapered or stepped adherends, filleting, etc.).
- Incorporation of nonlinear stress-strain behavior of the adhesive (usually approximated by elastic-plastic stress-strain curve).
- Dependence of the measured elastic mechanical properties of adhesive on its thickness.
- Change of adhesive properties as a function of the environment as well as long term degradation.

8.3.4.4.4 Repair procedures

This section will describe general procedures to complete secondarily bonded, co-cured with prepreg, and wet lay-up repairs. Specific repair procedures are given in SRM's, NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)), and Air Force TO 1-1-690 (Reference 8.3.4.3.4(b)). Bonded repairs require close control of the repair process and the repair environment. Structural integrity of the bonded joint is strongly dependent on the cleanliness of the work area and its ambient temperature and humidity. Other important factors are workmanship and geometrical fit of mating parts. An example of typical bonded repair will be described at the end of this section.

The four major activities to effect a bonded repair consist of patch and parent surface preparation, adhesive application, bagging, and curing. Each of these activities may be different for the type of bonded repair being attempted, materials used, and the part being repaired. Size of the repair may be limited by the allowable out-time of the adhesive. A drawing of the patch is used to lay-up the composite tape or fabric, sheet metal or dry fabric materials. Standard shop procedures are used to make the composite patch laminate from prepreg. Dry fabric plies for wet lay-up are cut first to size before impregnating with resin. This is done to minimize repair time. How to properly mix the resin is described in SAE document ARP 5256 - Resin Mixing (Reference 8.3.4.4.4(a)) and NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)). The impregnation of the dry fabric with the mixed resin is described in SAE document ARP 5319 - Impregnation of Dry Fabric Application of Repair Plies (Reference 8.3.4.4.4(b).

Before adhesive application, the repair patch and the parent surfaces must be wiped clean with solvent and allowed to dry. At this point the composite surface should be abraded. A light grit blast gives a more uniform abrasion than hand sanding. The surface is then wiped dry with a clean, lint free cloth. Metal sheet patches have special surface preparation requirements depending on whether the patch is aluminum or titanium. They are specified in detail in the SRM and MIL-HDBK-337 (Reference

8.3.4.4.4(c)) and should be followed closely. Film adhesives are first attached to the patch, trimmed, and then applied to the damaged area.

Bagging is an operation wherein the repair is enclosed for the curing operation. As most of the repairs are done outside the autoclave, the process described here will address only vacuum bagging. This allows the repair to be cured under atmospheric pressure. When the repair can be cured in the autoclave, additional pressure and higher, more uniform heat can be applied. Figure 8.3.4.4.4(a) shows a typical bagging arrangement in which patch plies of prepreg are co-cured with a layer of adhesive and a heating blanket is used to supply heat. Starting from the top of the patch, the repair bagging assembly contains porous separator release film to prevent bleeder plies sticking to the repair plies, bleeder plies to absorb extra resin (it is assumed that the prepreg is not net resin type), Mylar separator ply perforated to allow venting, caul or pressure plate to help provide smooth finish to the repair, breather plies to provide for the air to be initially inside the bag to be drawn off by the vacuum source, and finally a rubberized vacuum bag. The vacuum bag is sealed on the periphery using tape. For a bonded repair with a metallic or pre-cured composite patch, bagging would still be needed to apply vacuum pressure to the adhesive but would be simpler.

An integral part of the bagging process is the placement of the thermocouples to monitor part and repair temperatures during cure. Thermocouples on the part are needed to make sure that the part is not overheated. Figure 8.3.4.4.4(a) shows only one thermocouple wire. (The more common practice is to place the heat blanket within the vacuum bag.) For larger repairs, more thermocouples are needed to map the temperature distribution for the complete repair area. Distributing the heat evenly on the repair is one of the goals of proper bagging technique. In some cases a thin aluminum or copper sheet is inserted inside the bag for that purpose. Care must than be taken not to puncture the bag. NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)) has a good description where the thermocouples should be placed. SAE ARP 5143 - Bagging (Reference 8.3.4.4.4(d)) gives guidance as to proper bagging techniques.

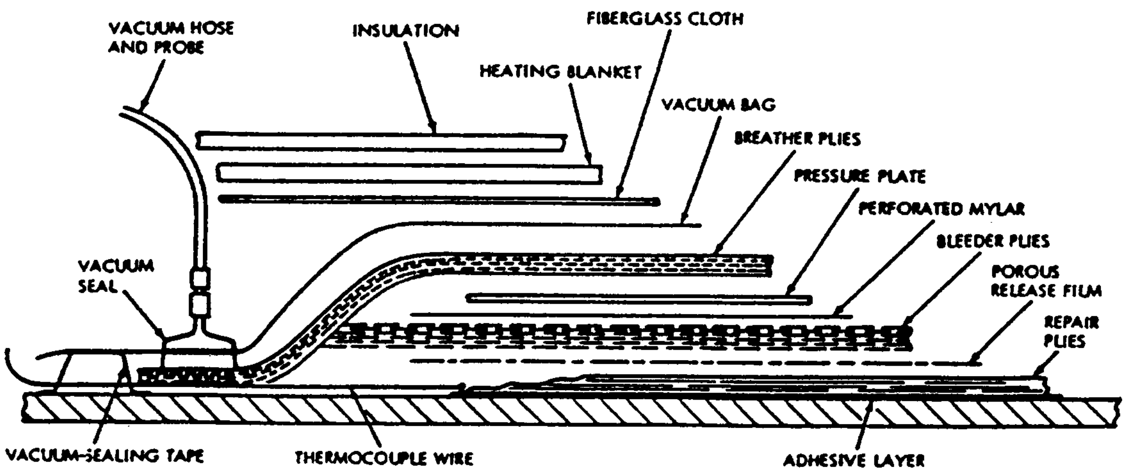


FIGURE 8.3.4.4.(a) *Schematic cross section of bagging lay-up.*

The process of curing structural adhesives and composite resins is achieved by a chemical cross-linking accelerated by heat. Therefore, cure temperatures should be sufficiently high to achieve this, but care must be taken not to reach temperatures that may damage the original structure. Keeping the cure temperature as low as possible to effect cure is the safest policy. The rate of heat-up is important as the resin and the adhesive undergo physical and chemical transformations. Therefore, the resin and the adhesive must have compatible cure cycles and follow prescribed time-temperature curve, i.e., rate of increase in temperature, dwell temperature duration, and rate of decrease in temperature. If the repair is an autoclave cure, pressure must be applied according to cure specifications. The maximum thermocouple reading is usually used as a control on the maximum allowed temperature. Cure time is adjusted by monitoring the minimum thermocouple reading. After the cure is completed the repair assembly is cooled before relieving vacuum pressure. More details on this subject are contained in SAE ARP 5144 – Heat Application for Thermosetting Resin Cure (Reference 8.3.4.4.4(e)).

Heat blankets, individual or integral with repair kits, are most commonly used to cure bonded repairs. Autoclaves, ovens, and quartz lamps are other acceptable methods.

The double vacuum bag processing concept is one of several alternative approaches (Reference 8.3.4.4.4(f)) that has been investigated since the early 1980's (Reference 8.3.4.4.4(g)) in an effort to develop processes that would improve the overall quality of composite laminate repair patches for thicker laminates. In 1983, work performed by the Naval Air Warfare Center demonstrated that the double vacuum bag approach produced lower levels of porosity, improved resin distribution and improved resin dominated mechanical properties in prepreg repair patches (Reference 8.3.4.4.4(h)). This program also investigated the double vacuum bag process for use as an intermediate step to debulk, compact and stage ambient storable prepreg repair patch laminates for later use in co-bonded field repair patch applications with the intention of performing the final cure on-aircraft using a single vacuum bag process. This work was expanded in 1992 to address the use of the double vacuum bag process on wet lay-up as well as prepreg repair patches (Reference 8.3.4.4.4(i)). More recent work demonstrated that optimization of double vacuum bag processing parameters such as debulk and cure temperatures, heat up rates, debulk time, vacuum level, the number of bleeder plies, etc., for the specific resin system in use can further improve overall repair patch laminate quality (Reference 8.3.4.4.4(j)).

To fabricate wet lay-up repair laminates using the double vacuum bag process, an additional step is required wherein the impregnated fabric is placed within the de-bulking assembly shown in Figure 8.3.4.4.4(b).

To begin the debulking process, air within the inner flexible vacuum bag is evacuated. The rigid outer box is then sealed onto the inner vacuum bag, and the volume of air between the rigid outer box and inner vacuum bag is evacuated. Since the outer box is rigid, the second evacuation prevents atmospheric pressure from pressing down on the inner vacuum bag over the patch. This subsequently prevents air bubbles from being "pinched-off" within the laminate and facilitates air removal by the inner vacuum. The laminate is then heated to a predetermined debulking temperature in order to reduce the resin viscosity and further improve the removal of air and volatiles from the laminate. The heat is applied through a heat blanket that is controlled using thermocouples placed directly on the heat blanket, in order to limit the amount of resin advancement during the debulk cycle.

Once the debulking cycle is completed, the laminate is then compacted to consolidate the plies by venting the vacuum source attached to the outer rigid box, thus allowing atmospheric pressure to reenter the box and provide positive pressure against the inner vacuum bag. Upon completion of the compaction cycle, the laminate is removed from the assembly and is prepared for cure.

In the case of prepreg repair patch laminates, the prepreg plies are cut, stacked, and placed within the double vacuum debulking assembly shown in Figure 8.3.4.4.4(b). In this process, the thermocouples are placed along the edges of the laminate, to ensure that all areas of the laminate reach the required debulk and compaction temperatures. No bleeder material is used in the prepreg staging process, in contrast to the wet lay-up staging process.

To begin the staging process, the inner vacuum bag and outer box are evacuated. The prepreg laminate is then heated to the debulking temperature. Once the debulking cycle is completed, the laminate is then compacted at temperature to consolidate the plies. Upon completion of the compaction cycle, the staged prepreg laminate is removed from the assembly and is either prepared for storage or immediately cured.

The double vacuum bag approach has been shown to produce repair laminates with low void content and good compaction approaching that of autoclave processed laminates. However, there are several limitations in application of the process due to the required use of a rigid outer vacuum box. In the case of co-bonded applications, the patch laminate must be debulked, compacted and staged off-aircraft. As the staged patch remains formable, the patch can then be transferred to the aircraft, formed to contour and co-bonded in place using a single vacuum bag process for the final cure. This two step process is necessary since using the rigid vacuum box assembly on-aircraft creates a peeling load that may be sufficient to further damage the parent structure. Likewise, the vacuum box assembly is difficult if not impossible to set up on a contoured surface. Overall patch dimensions are also limited by the maximum practical size of the rigid outer box (in practice, relatively portable vacuum boxes have been a maximum of approximately 24” wide by 24” in length).

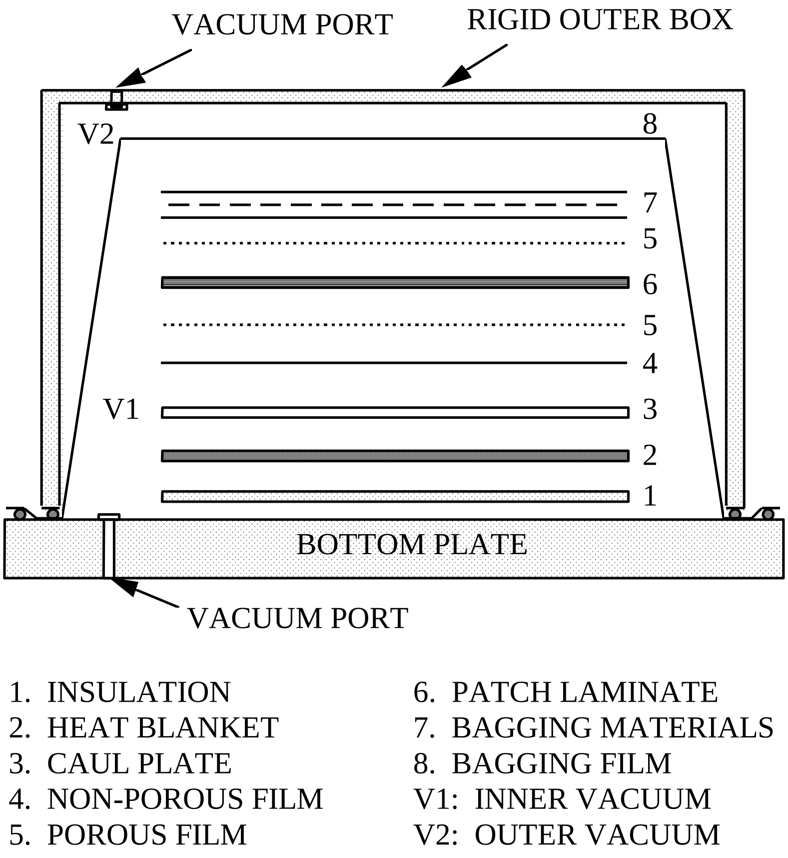


Figure 8.3.4.4.4(b) *Typical double vacuum bag debulking assembly.*

8.3.4.4.5 Bonded repair examples

The first example of the bonded repair is a simple scarf repair of a penetration damage of a 16-ply laminate. The repair, shown in Figure 8.3.4.4.5(a), is from the Air Force TO 1-1-690 (Reference 8.3.4.3.4(b)). The scarf plies replace parent plies with the same orientation and thickness repair plies. The taper of the scarf is determined from the SRM. Additional plies on the outside and inside mold lines (OML and IML) are placed on top of the repair to compensate for the lower strength and stiffness of the replacement plies because of vacuum pressure cure and to protect the repair. The external plies are identical to each other to maintain symmetry. The 0° and the 45° plies are serrated to prevent peeling of the longer plies. From the ply directions that are serrated, one can assume that the primary axial load is in the 0° direction with a shear component. The edges are cut with standard pinking shears producing 1/8-in. deep serrations.

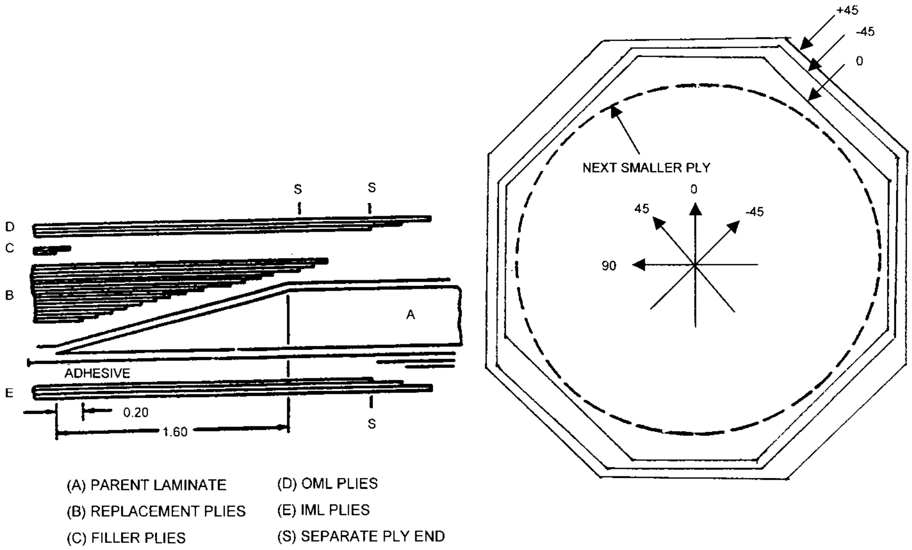


FIGURE 8.3.4.4.5(a) *Single-scarf bonded joint repair.*

An example of a more complex bonded repair is drawn in Figure 8.3.4.4.5(b). The repair is to a penetration damage to a skin and the underlying stiffener. This field repair was verified by test to restore original strength and stiffness. The skin is repaired using a circular external patch that is laid-up wet. The J-stiffener is reconstructed using Rohacell foam as the mold and the filler material over which a square composite patch is placed using wet lay-up method.

8.3.4.5 Sandwich (honeycomb) repairs

Most structural repairs that are performed due to service damage are on sandwich structure, metallic or composite. For composites, it is due to the fact that a large proportion of current components are light sandwich structures that are susceptible to damage and are also easily damaged. The repair experience gained on the repair of metallic sandwiches is applicable to composite sandwiches. Additional flexibility with composites is possible as flush scarf repairs can be accomplished.

8.3.4.5.1 Repair concepts

Because sandwich structure is a bonded construction and the face sheets are thin, damage to sandwich structure is usually repaired by bonding. Procedures to effect the repair are, therefore, similar to the bonded repairs discussed above with the additional task of restoring the damaged core. When repairing one face skin of the sandwich, one should remember that half of the in-plane load is transferred through that face sheet, and if the repair does not approximate in stiffness the undamaged face sheet extraneous bending moment could induce peel loads between the face sheets and core. Thus, external patch is usually applicable only for thin skin repairs while scarf concepts are used to repair thicker skins.

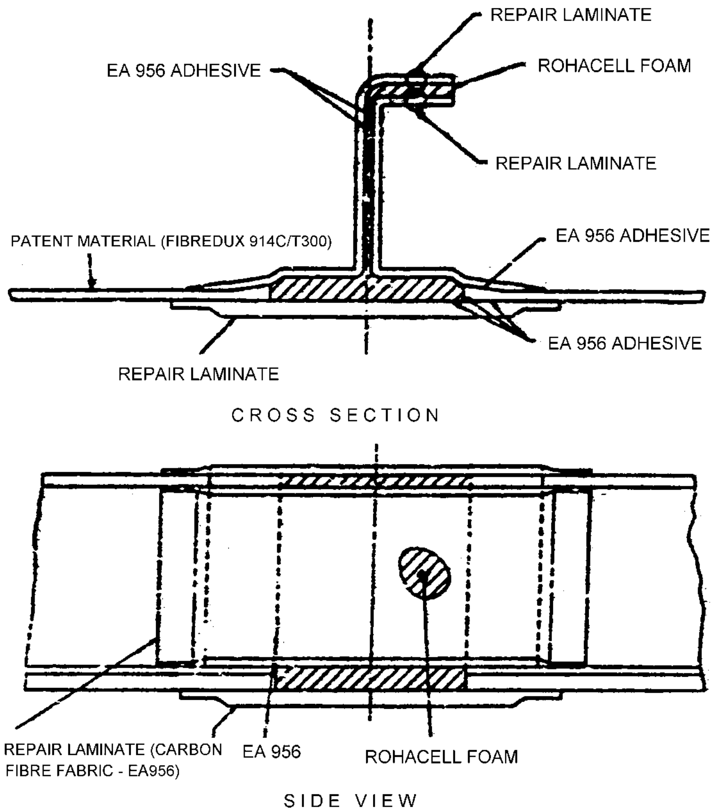


FIGURE 8.3.4.4.5(b) *Field repair of the "J"-stiffened panel.*

8.3.4.5.2 Core restoration

For full-depth core replacement there are three common methods, the core fill method, the paste adhesive method, and the film/foam method. The three methods are shown in Figure 8.3.4.5.2(a). The core fill method replaces the damaged honeycomb with glass floc filled paste adhesive and is limited to small damage sizes. The weight of the repairs must be calculated and compared with flight control weights and balance limits set out in the SRM. The other two methods can be used interchangeably depending on the available adhesives. However, the paste adhesive method results in a much heavier repair than the film/foam method, especially if the damage diameter is greater than 4 inches. The foaming adhesive required to utilize the film/foam method is a thin unsupported epoxy film containing a blowing agent which is liberated during cure causing a foaming action. The expansion process needs to be performed under positive pressure to become strong, highly structured foam. Like film adhesives, foaming adhesives require high temperature cure and refrigerator storage. Core replacement is usually accomplished with a separate curing cycle and not co-cured with the patch.

For partial-depth damage, different methods can be used to attach the replacement honeycomb to the parent honeycomb as shown in Figure 8.3.4.5.2(b). The two methods describe the prepreg/film adhesive bonding and the wet lay-up bonding. Both of these bonding methods were discussed in Section 8.3.4.4. A general description of how to perform core restorations for simple configuration is contained in SAE ARP 4991 - Core Restoration (Reference 8.3.4.5.2).

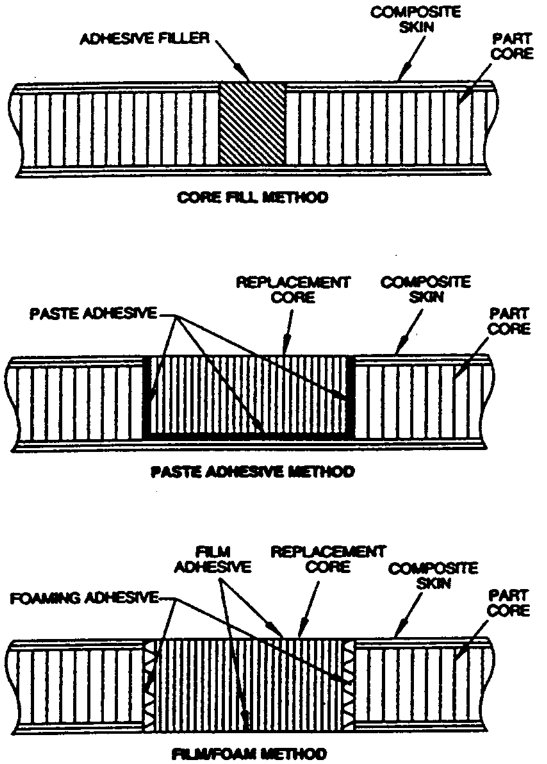


FIGURE 8.3.4.5.2(a) *Core replacement methods – full depth.*

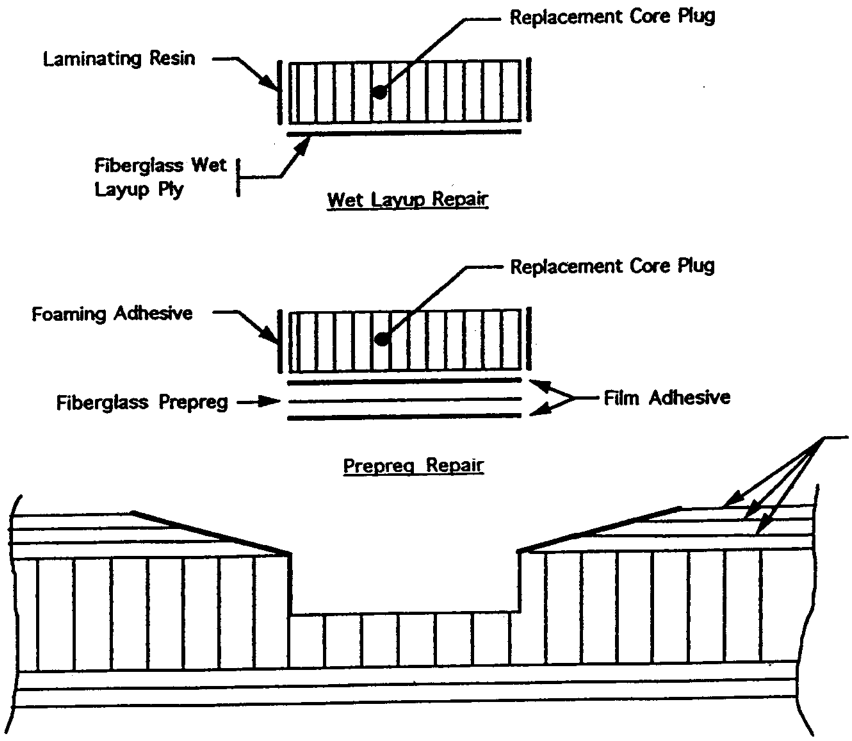


FIGURE 8.3.4.5.2(b) *Core replacement – partial depth.*

8.3.4.5.3 Repair procedures

Following the core replacement, the sandwich repair proceeds as a bolted or bonded repair of the face sheets as was described in Sections 8.3.4.3 and 8.3.4.4, respectively. One more step has to be performed before proceeding with repair of the face sheets and that is to bond a pre-cured fiberglass plug on top of the exposed core. This preserves the continuity of the bond between the core and the face sheets.

Sandwich structures are usually repaired by bonding patches. For bonded repair of the sandwich structure, special considerations that have to be adhered to are: the honeycomb must be thoroughly dried to prevent face sheet disbond during curing, and the curing pressure must be low to prevent honeycomb crushing. If it is unfeasible to dry out the honeycomb, lower temperature (200°F (93°C)) curing and be used if this has been approved in the SRM.

Occasionally, sandwich structure is repaired using bolted external patches. In this case, the honeycomb where the bolts would pass through has to be strengthened by filling the core with the same filler as for core replacement. The diameter of this area should be at least three times the diameter of the bolt. Special bolts that have limited clamping force are used for such repairs.

8.3.4.5.4 Sandwich repair example

The sandwich repair example is taken from NAVAIR 01-1A-21 (Reference 8.3.4.3.4(a)). A repair of a full depth damage, of sufficient diameter to warrant core replacement, to a control surface is demonstrated. Actual steps in repair are shown in Figure 8.3.4.5.4. These consist of removing damaged material, drying the repair area, fitting replacement core, tacking the replacement core using filled paste adhesive with sufficient glass floc to make the adhesive into consistency of a putty, machining the core to match the contour of the part, installing core with the foaming adhesive using a heat blanket, and installing external patches using another cure cycle. The details of the final cure are simplified in this example. Unless the patch is pre-cured, the bagging details would be more complicated.

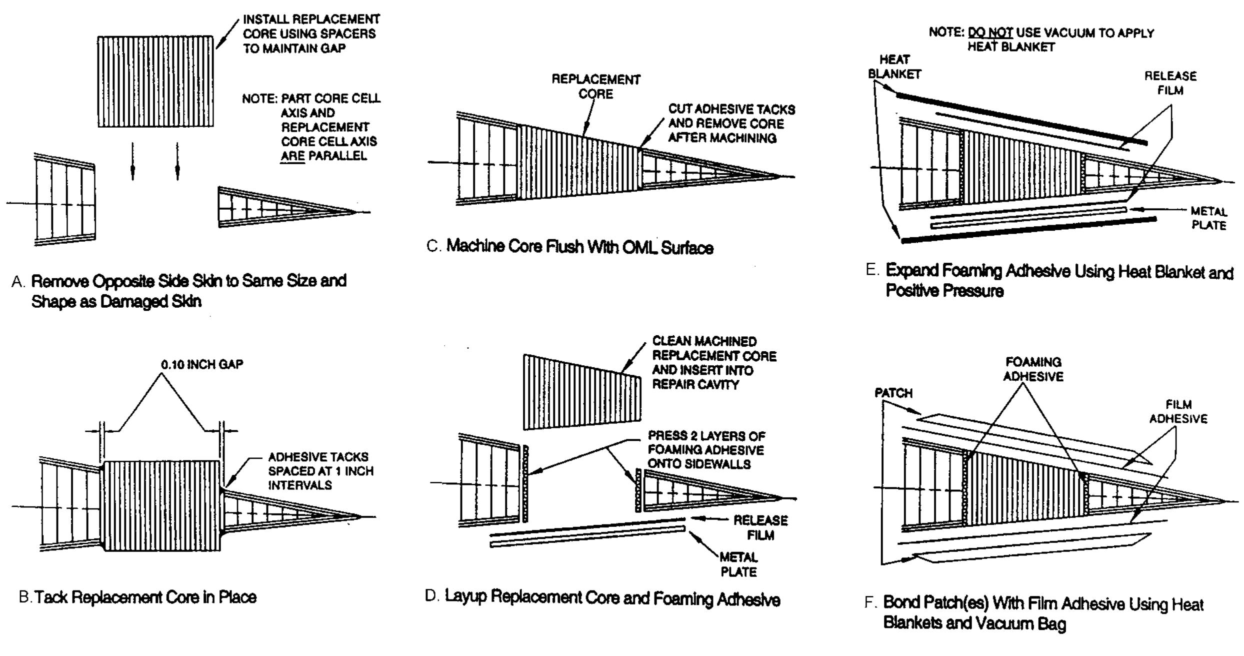


FIGURE 8.3.4.5.4 *Bonded repair of sandwich control surface.*

8.3.4.6 Repair inspection

8.3.4.6.1 In-process quality control

Bonded repairs require more in-process quality control to obtain structurally sound repairs than bolted repairs. Composite materials and adhesives require extensive record keeping to ensure they are within life, such as to storage time in the refrigerator, warm-up time, and out time on the shop floor. Lay-up operations need to be inspected as to the correct fiber orientation. Cure cycles must be monitored to assure they follow specifications. For large repairs, a small companion panel is cured with the repair. It is used for coupon testing to provide confidence in the quality of the repair, repair patch and adhesive bond.

Bolted repairs require inspection of holes for damage and size. Assembled repairs also require inspection of fastener installations.

8.3.4.6.2 Post-process inspection

Completed repairs should be inspected to determine their structural soundness. The NDI methods described previously in Section 8.3.1 are used to perform this function.

8.3.4.7 Repair validation

Successful inspection of a repair is not sufficient to guarantee that the repair will perform as designed and implemented. Repair designs need to be supported by an existing experimentally verified database and analysis. This helps to ensure the repair's capability to carry the intended loads or to replace the capability of the parent structure. The repair material allowables used in the design should be generated using approved testing and data reduction methods that reflect the amount of testing completed, material and process controls in place, and the criticality of the structure.

Both strength and stiffness must be taken into account when designing the repair as described in Sections 8.3.4.3.3 and 8.3.4.4.3. Analyses need to be done in each fiber direction with careful attention to limit the effects of hardpoints as described in Section 8.3.3.1. It must be understood that increases in stiffness do not correlate to increases in the repair factor of safety. Environments the parent material was designed to and details such as edgebands, cutouts, and fastener penetrations must also be considered.

Repair designs are based on using a specific material or a family of materials. Design properties are usually obtained by mechanical property testing specimens that mimic the particular repair process. Typically this testing is not as extensive as for the parent material and does not involve sufficient replication to obtain statistically based properties. Batch to batch variation of repair materials should be obtained which can be somewhat problematic for wet lay-up materials. Typically two practices are utilized for obtaining repair allowables. Allowables can be based on the parent material properties with knock down factors that reflect lower cure temperatures and pressures of the repair material relative to the parent material, or allowables and material properties are derived for the repair material to be used in the repair analysis. It is common to use large reductions from the mean value to the allowable design value as the process parameters such as fiber volume, pressure, and temperature of repair patches have more variability than the parent material.

For the repair material to have meaningful design allowables, rigid material purchasing and process specifications must be in place. This means that the material is purchased to a material specification with incoming material controls, that the material handling and storing is according to specification, and that the repair is performed to a process specification.

In addition to coupon testing, a variety of elements are tested to validate repair designs. These are usually performed to support repair designs included in the SRM and range from simple joint specimens, representing bolted or bonded load transfer, to tests of full-scale repairs. Simple joint specimens are used for development of repair designs. These are usually two-dimensional. Example of such elements would be single or double bolt specimens to obtain bearing, bearing/bypass and net tensile values, and lap bonded specimens to obtain joint shear strengths. The more complex elements are used to validate the repair design and repair process. These are full-scale representations of the repair.

Final validation of bolted and bonded repairs as described in Section 8.3.4 rely on strict attention to all details, including: damage removal and site preparation, repair design, appropriate use of materials, repair analysis, material and fabrication processes, inspection, and appropriate design values supported by test evidence.

8.4 COMPOSITE REPAIR OF METAL STRUCTURE (CRMS)

Composite materials can be used to structurally repair, restore, or enhance aluminum, steel, and titanium components. Bonded composite doublers have the ability to slow or stop fatigue crack growth, replace lost structural area due to corrosion grindouts, and structurally enhance areas with small and negative margins. This technology has often been referred to as a combination of metal bonding and conventional on-aircraft composite bonded repair. The U.S. Air Force and the Royal Australian Air Force have been using the technology for over 25 years on aircraft ranging from F-5s to Boeing 747s to C-130s to C-141s to B1Bs. Commercial aircraft manufacturers and airlines are starting to adopt this technology to their needs.

Boron epoxy, GLARE® and graphite epoxy materials have been used as composite patches to restore damaged metallic wing skins, fuselage sections, floor beams, and bulkheads. As a crack growth inhibitor, the stiff bonded composite materials constrain the cracked area, reduce the gross stress in the metal, and provide an alternate load path around the crack. As a structural enhancement or blendout filler, the high-modulus fiber composites offer negligible aerodynamic resistance and tailorable properties.

An understanding of fracture mechanics, durability and damage tolerance and the structural and thermal load spectra are invaluable in deciding on a CRMS application. The repair techniques and design principles used for composite structures that are described in Section 8.3 are applicable to design bonded doubler for a metallic structure. To apply CRMS technology successfully one needs an understanding of laminate theory and behavior, failure modes in both the composite patch and the metallic structure, and structural and operational loads. Design decisions based on sound material allowables, confidence in the metallic surface preparation, and a full understanding of the structural behavior after repair will result in a fully restored structure. Finite element analysis and closed form analytic solutions have been used to design and analyze bonded composite and metallic doubler repairs. The U.S. Air Force (AFRL Materials Directorate) developed a comprehensive document that captures previously published data on the technology and uses “Guidelines for Composite Repair to Metallic Structures”, AFRL –ML-TR-1998-4113, as a how-to manual for assessing, designing, analyzing and installing composite doublers. AFGROW and CalcuRep software programs in those guidelines can provide a basic understanding of a damage/repair situation and provide the initial doubler sizing.

Grit blast silane and phosphoric acid anodizing using the Phosphoric Acid Anodizing Containment System (PACS) have been the only surface preparation technologies authorized by the U.S. Air Force as suitable for durable and reliable bonded doubler installation on aluminum structures. Film adhesives using a 250°F (121°C) cure are used routinely to bond the doublers to the metallic structure. Critical areas of the installation process include a good thermal cure control, having and maintaining water break-free bond surfaces, chemically and physically prepared bond surfaces, technician training and certification, and managing a quality bonding site.

Secondarily bonded precured doublers and in-situ cured doublers have been used on a variety of structural geometries ranging from fuselage frames to door cutouts to blade stiffeners. Vacuum bags are used to apply the bonding and curing pressure between the doubler and metallic surface. Autoclaves and a tooling splash from the repair area are used to prepare precured doublers.

Inspection methods have been developed to monitor damage growth under the bonded doubler and to assess the bond quality. Procedures need to be developed for each group of repaired applications and used to assess and manage the repairs while in-service. Eddy current inspection has been used to assess crack growth through bonded boron doublers. Since the boron doubler does not shield the bondline and parent structure, conventional eddy current techniques work quite well assessing crack growth. Pulse echo, through transmission ultrasonics and thermography have also been used to assess disbonds under metallic and composite doublers. These methods are described in Section 8.3.1.

In summary, to repair metallic structure with composite patches (doublers), repair procedures described in Section 8.3 are applicable except special care in metallic surface preparation must be taken. However, Section 8.3 does not cover structural analysis techniques needed to assure life of the metallic component or parent structure. Such discussions on metallic fracture mechanics and durability and damage tolerance can be found in several textbooks and handbooks on aircraft structures. Materials and design allowable data for the composite doubler and metallic structure can be found in numerous DOD, OEM, and vendor literature. CMRS also requires not only initial inspection of the adhesive joint between the repair patch and the metallic structure, but also periodic inspections for crack growth emanating from the damage site of the metallic structure.

8.5 LOGISTICS REQUIREMENTS

8.5.1 Training

Specialized knowledge and skills are required to perform sound structural composite repairs. The orthotropic and process-dependent nature of the materials requires technicians who are attentive to detail and well-trained. Unlike conventional metal repair technicians, the composite technician must be counted on to not only assemble a damaged component, but create the material properties in the process. Also needed are engineering support members who are skilled in composite analysis and composite repair.

Repair technicians must be trained in a formal program for certification since they will be expected to perform bonded repairs on diverse structures with many different types of material. The general lack of means to non-destructively inspect the strength and stiffness of a complete component once it has been layed up and cured means the repair technician cannot be someone without training. This program should include a classroom lecture to provide in-depth information into the specifics of working with composite materials in addition to hands-on instruction so that proficiency can be demonstrated in practice. SAE document AIR 4938 (Reference 8.5.1) provides a curriculum for such training. This should be a prerequisite to on-the-job training (OJT) with actual components prior to achieving certification as a composite technician. A key to maintaining a core competency in composite repair is the availability of experienced and skilled mentors who will provide continuing guidance and instruction since this skill requires continual learning and practice to become proficient. The fundamental intent of such a program is to provide the knowledge, skills and abilities so that the technician can consistently make safe and effective repairs. The Office of Personnel Management is now considering a DoD-wide composite repair technician job series to formalize and standardize the skills required for this technology. The need for personnel qualification and training for repair personnel that work in approved repair stations for civil aircraft is defined in FAA Advisory Circular AC 145-6 (Reference 8.3.2.5 (b)).

Engineers supporting the design of composite structural repair will likely require a mechanical or aeronautical engineering background with a concentration in mechanics of materials. It is necessary to have a thorough understanding of structures designed using anisotropic and orthotropic materials. The engineer should have knowledge of composite laminate theory and joint analysis so that repairs can be analytically justified. Complicated repairs, such as those accomplished on three-dimensional laminates, may require FEA modeling in addition to the more traditional analysis techniques. Many undergraduate, graduate and continuing education programs offer selected courses on these topics and can be very useful in establishing and maintaining the skills necessary for composite repair design. A materials and process engineer to support materials testing and composite processing will be needed. These engineers must have a firm understanding of thermosetting material chemistry and rheology.

8.5.2 Spares

Field repair facilities have limited space, equipment, tooling, material, trained personnel, and time. Operational requirements dictate that the aircraft be returned to service as quickly as possible, and the cost of developing a depot-level capability at each field unit is prohibitive. Thus, damaged composite aircraft components must either be easily and quickly repaired on-aircraft or in a backshop, or removed and replaced (R&R) with a spare component. In the latter situation, the damaged component, if deemed reparable by the responsible depot, will be sent to the depot facility or original equipment manufacturer for repair. In some cases, a depot team may be dispatched to the field unit to perform the repair if it is not tooling- or equipment-intensive.

During aircraft design and acquisition, estimates must be made on the likelihood of significant damage to a structure during its life cycle. This damage may come as the result of unanticipated flight loads, insufficient design or incorrect manufacture (inherent damage), or as a result of induced damage such as mechanical impacts in flight or on the ground, lightning strikes, overheating, erosion, aging, fluid ingression, chemical contamination, thermal or flight stresses. Mean Time Between Maintenance Actions (MTBMA) and Mean Time Between Failure (MTBF) estimates based on probabilistic analysis and comparison with a similar structure already in use are normally used to establish repair and replacement rates. This is then used to establish initial provisioning requirements by the aircraft procurement agency for spare/replacement components. As operational experience is gained with the parts, adjustments can be made to the on-going provisioning requirements.

Spares are normally stored at the designated aircraft depot in an enclosed storage facility. Accurate estimates when determining the number of spares to purchase are important. Too low a number may mean aircraft downtime or flight restrictions while awaiting part replacement. Too high a number of spares mean precious resources are tied up in storage and part costs.

From a logistics support perspective, components that are interchangeable are preferred. Pre-drilled panels can be taken from the supply depot or scavenged from another aircraft and easily installed on the affected aircraft. Trim-to-fit/match-drilled (replaceable) parts require additional labor for installation. Once trimmed, they may only be useable on the particular aircraft to which they were adjusted. The removed and repaired part must then either be stored for that aircraft, or the holes filled and edges rebuilt, which will probably then require depot tooling for contour matching. While interchangeable panels are usually more expensive to produce, the life-cycle cost differential between interchangeable and replaceable panels will often be minimal if these considerations are taken into account.

8.5.3 Materials

Repair materials present another logistics support issue. While most structural components on an aircraft are manufactured from prepreg composite materials, support considerations may make it desirable to use different repair materials. The materials required for repair are specified in the applicable repair manual. For the repair patch itself, the repair materials chosen may be prepreg or dry cloth with a laminating resin. Whichever material type is used, strength and/or stiffness should be matched with the parent laminate during design of the repair. Adhesives must also be available to accomplish any bonded repair. Film adhesives, while offering excellent structural properties, require cold storage. Cold storage is also required for foaming adhesives, which are used for splicing core in honeycomb assemblies, or for filling small gaps. Laminating resins and paste adhesives present a room-temperature storable alternative, but will have reduced performance at higher temperatures. If the damaged structure has a lightening-strike protection scheme, this must be restored. Thus, lightning–strike protection systems, such as copper, nickel, or aluminum mesh, must be available at the repair station. The materials and processes for the repair must be called out in the applicable structural repair manual.

The structural repair materials that are brought to the facility must be purchased and controlled by a materials specification. Some incoming material tests may be required to verify supplier material quality. AC-145-6 (Reference 8.3.2.5(b)) has a discussion on this topic.

Beyond the materials used in the actual repair, the repair station must also have additional consumable materials, which will be used during the repair process. These materials will likely consist of non-porous and porous fluorinated ethylene proplyene (FEP) release films, peel ply, breather, bleeder cloth, bagging film, and tacky tape. Tooling materials are also a requirement if bondform tools (splash molds or constructed) are required for contoured repairs, as well as mold releases.

While the direct cost of materials used in a repair is normally a small part of the total cost, significant indirect costs can result from special handling requirements such as storage, safety, process control, procurement, and waste. These will become a consideration in the overall repair scheme selected by the repair design team. Shelf-life limited adhesives and prepregs will require an incoming certification as well as recertification once the manufacturer-specified shelf-life is exceeded. This certification must be accomplished by a laboratory with trained personnel and the proper equipment to measure resin advancement and material properties. Tensile and compressive test apparatus and fixtures are required for coupon testing of material strength and stiffness. Recertification typically looks at matrix dominated properties such as compression, flexure, and transverse shear, as well as resin physical properties such as flow, gel time and glass transition temperature. Normally, this facility will be placed at a depot because of material usage rates and the cost of this capability.

In the final analysis, it is often difficult to anticipate the proper kinds and amounts of repair materials, which should be on hand at a repair facility when commencing support of a new aircraft. As a result, availability often becomes a significant factor in material selection for repair.

8.5.4 Facilities

The basic requirements for a field or depot composite repair facility consist of a lay-up area, part preparation area, a part curing area, and a material storage area. An environmentally-controlled lay-up area is required to prevent contamination of the repair surface and materials from dust, dirt, oil, and moisture when a bonded repair is implemented. Ideally, the lay-up/bonding area will be enclosed, with a slight positive pressure to prevent dust from entering. Temperature and humidity in the area should be controlled to a maximum of 75°F (24°C) and 50% relative humidity. A chart recorder should be used to track conditions in this facility. General lay-up room requirements are defined in MIL-A-83377 (Reference 8.5.4), which can be superceded by specific requirements specified in weapon-system-specific technical manuals. Civil aircraft operators must demonstrate compliance with the requirements of Title 14 of the Code of Federal Regulation parts 21, 43, 121, 125, 127, 135, and 145 regarding procedures and facilities. AC-145-6 (Reference 8.3.2.5(b)) delineates facilities that may be required.

For on-aircraft repair, humidity and temperature control is unlikely. In the best case, a hangar will be available in which to perform the repair, giving some shielding from the environment and contaminants. In the worst case, the repair will have to be performed on the tarmac. Some form of shelter should then be devised around the repair area. Repair materials should be prepared and sealed in bags in the backshop, and the bags only opened immediately prior to installation.

Because a depot repair/rework facility must perform repairs which range from beyond the field support capability to out-right component remanufacture, these facilities must essentially replicate those existing at the original equipment manufacturer. Rework facilities must have all the equipment and tooling necessary to restore aircraft components to essentially original strength, stiffness, aerodynamic, and electrical requirements. Floor space is less at a premium at the depot than it would be in a field environment; thus separate lay-up, bonding, tool manufacture, part machining, and part and tool storage areas will be available.

During remanufacture, the components are off-aircraft and mobile. Therefore, large stationary industrial cold storage, curing, machining, and inspection equipment can be used to perform repair operations at the depot. A depot should also have a three- or five-axis core cutters and numerical control machines to accurately shape fittings, core, substructure, and tooling. Ample capability exists to store and use fixtures and tooling, or manufacture them. A phosphoric acid anodizing (PAA) line for surface preparation of metal substructure in composite components must be accessible. Abrasive waterjet cutters for rapid and smooth trimming of composite panels may be a worthwhile consideration if workload warrants it.

8.5.5 Technical data

Personnel skill, facilities, and equipment make up only part of the logistics support requirement for structural repair of composites. Information, or technical data, in various forms is required to quickly and adequately support composite repair to aircraft structure. Technical data ranges from structural repair manuals/military technical manuals, to part drawings and CAD data, to loads books and finite element models for engineers. Hopefully, this data was acquired as part of the original aircraft acquisition process, and was available prior to initial operating capability (IOC), or provided soon thereafter. Airlines and repair stations may not have such data and may have to request it from the original equipment manufacturer (OEM).

Repairs for smaller damages (less than 4 inches) which are expected to commonly occur are found in Structural Repair Manuals (SRMs). A Non-Destructive Inspection Manual and NDI Standards panels should also be provided with the aircraft to allow accurate detection of damages to composite panels. The SRMs should define repair size limits for negligible (cosmetic), field-repairable, and non-repairable damage. Size and depth limits should be provided in the manual for scratches, gouges, dents, delaminations, disbonds, and partial- or full-through punctures. These limits are most useful when provided through “mapping/zoning” of the structure. A parts-breakdown graphic should be in the manual, with a listing of the materials contained within each sub component. Within the negligible and field-repairable limits, repairs should be defined which can be accomplished within the limited capabilities/conditions of the field but restore full strength and stiffness to the structure. This definition must specify repair materials, equipment, and well-written step-by-step instructions which can be understood by a structural repair technician.

For repairs not provided in the SRM or beyond field-repairable limits, cognizant engineers must be consulted. The engineers require access to design information, which define aircraft component loading conditions, and aircraft design manuals, which provide information on the design requirements and load distribution within the structure. With increasing use of finite element analyses (FEA) to design aircraft structure, the aircraft acquisition agency may find it useful to purchase the FEA models developed by the manufacturer. Otherwise, engineers will require FEA software and workstations to perform analyses for structurally-significant repairs. This means engineering will also require a fully dimensioned drawing package, to include material properties and process specifications for FEA model or table top analyses.

8.5.6 Support equipment

8.5.6.1 Curing equipment

Heat blankets, hot bonders, heat lamps, heat guns and convection ovens are examples of portable heating and curing equipment usually found in both the depot and field environments. These are usually used in conjunction with a vacuum bag, to expedite moisture removal prior to repair, and to provide some consolidation pressure to the repair. They can be used to manufacture pre-cured composite repair patches and to bond repairs to the component. Whatever the portable heat source, generous use of thermocouples must be made to closely monitor cure temperatures.

Heat blankets consist of heating elements sandwiched between temperature-tolerant and flexible materials, such as silicone. The heat blankets can be separately zoned to allow differential heating of the repair area. Blanket temperature is controlled by regulating power to the blanket, either manually through a rheostat or through a hot bonder. Heat blankets are inexpensive, and can be purchased in a variety of sizes and shapes. Blanket flexibility and element durability are limiting factors; highly contoured repair areas may make blanket use unfeasible.

Hot bonders are programmable heat and vacuum control units, which provide power to heat blankets automatically to an operator-specified cure cycle. The hot bonder monitors bondline temperatures through thermocouples placed on or near the repair area, and varies power to the heat blanket according to cure requirements. This assures acceptable cure temperatures. Hot bonders often have a vacuum pump included. The electrical draw of a hot bonder is approximately 30 amps, and a suitable power source (110V) must be available. If they are to be used on-aircraft, some form of explosion-proofing of the hot bonder system is mandated. Fuel vapors, which seep into the hot bonder case, may present an explosion hazard.

Infrared heat lamps and heat guns are also used for elevated temperature cures of composite repairs. Some are available with thermostat controls. Heat lamps can quickly heat up surfaces; monitoring and control of the cure temperature is essential to avoid overheating and damaging the surrounding structure. However, they are useful when part contours and repair size makes a heat blanket unfeasible. Heat guns can be convenient sources of heated air for smaller repairs.

Industrial ovens are a necessity at depots, and worthwhile in the field support environment, as a means of drying and curing composite parts and repairs. They are stationary pieces of capital equipment, and most depots will have several ovens of various sizes to accommodate workload. Maximum convenience is gained from ovens that are automatically controlled, allowing the programming of multi-step heating and curing cycles. The temperature capability and size of the oven will usually be defined by weapon-system specific requirements. Multiple vacuum lines and thermocouple connections within the oven, and a vacuum pump, are required for vacuum-bag repairs. A large (10' x 7' x 12' [3 m x 2.1 m x 3.7 m]) 500°F (260°C)-capable oven will require 300 Amp, 480V power supply.

Autoclaves are pressurized ovens usually required at a depot facility for part repair and remanufacture. The typical 85 psi (586 kPa) pressures required to achieve maximum consolidation of a composite prepreg laminate make large structural repairs possible. Because components are off-aircraft, the autoclave should be sized to accommodate large parts and their associated tooling. Like the ovens, autoclaves should be automatically controlled, with multiple-step cure cycles accommodated, and numerous vacuum line and thermocouple connections. The autoclave should be capable of providing 100 psi (689 kPa) of pressure and 500°F (260°C) in temperature, to accommodate both epoxy and bismaleimide cure cycles. Nitrogen gas should be used to provide pressure and inerting to prevent a fire, or more importantly, an explosion. Because of the high pressure achieved during cure, tooling and bondforms are necessary to support the aircraft component to the correct contour. Because of the infrastructure involved in them, including but not limited to tooling and bondforms, experienced operators and technicians, and fairly high operating and acquisition costs, autoclaves are restricted to depot use.

For manufacture of smaller, non-contoured precured repair patches, a heated-platen press may be a useful piece of equipment in the depot, although multiple patches can be made in an autoclave and stored. Presses can also be used in the laboratory for the creation of coupons and specimens to accomplish receiving and shelf life certification testing.

8.5.6.2 Cold storage rooms

Non-frost free cold storage equipment that can maintain material at 0°F (-18°C) or lower is required to maintain shelf-life on the vast majority of preimpregnated materials and adhesives. Frost-free freezers remove frost by temporarily heating up above 32°F (0°C), which can reduce the shelf life of the cold-storage materials. Because of moderate usage rates on preimpregnated materials, and the number of different shelf-life-limited materials necessary for weapon-system support, as well as long procurement lead-times, depots will require walk-in freezers which have sufficient capacity to meet local, and potentially field, needs for six to twelve months. Procurement and manufacturer lead times for most specification materials can mean 8-week to 4-month wait before delivery. Walk-in freezers are relatively inexpensive; thus, a field unit may also opt to purchase one if space and weapon-system requirements dictate it. In many cases, a case freezer will be adequate for field use, however. In case of power or freezer failure, an alarm system tied to a 24-hour notification site may avoid an expensive loss of shelf-life and/or material. AC 145-6 (Reference 8.3.2.5(b)) has specific details on control and operation of freezers.

8.5.6.3 Sanding/grinding booths

For paint removal and machining of cured composite materials, facilities separate from the bonding and curing area are required. The facility should have the means to safely remove dust produced during machining operations. The use of a down- or side-draft booth to remove the dust from the part and the air is preferred. Grinding and sanding areas must be configured to allow the dust unobstructed transit into the filter system. Worker tables must not shadow each other, to prevent accidentally sending the dust from one grinding operation onto adjacent work areas and workers. At a minimum, a shop vacuum system should be provided to the technician to collect the composite dust.

8.5.6.4 NDI equipment

Both field and depot facilities require damage evaluation and verification equipment for nondestructive inspection of composite components. Radiography (X-ray), thermography (infra-red), ultrasonic and laser shearography inspection equipment may be required for pre-repair damage mapping, in-progress inspection, and post-repair inspection. Field equipment will likely be small and portable to allow its movement to the aircraft, and to reduce facility and equipment costs. Depots will usually require both portable and stationary NDI equipment. The use of robotics to scan an entire aircraft for damage is practicable at a depot if workload and timeliness warrant it. Some method of archiving NDI data, whether digitally or in hard copy/film, is usually necessary to track damage growth.

The above is equally valid for assessing the electrical properties of structures and coatings, as in the case of radomes and low-observable structures. Field and depot units with a regular radome repair workload may require a static radar test range to test repaired radomes. Improved hand-held equipment to measure infrared or electrical properties of repaired low-observable structures are being developed for field use. A depot may require a separate electrical test range for low observable (LO) structure. In addition, a fly-through test range for a particular LO weapon system may be required to verify the proper behavior of multiple repairs.

REFERENCES

8.2.3.6 "Safe Handling of Advanced Composite Materials," Suppliers of Advanced Composite Materials Association, Arlington, VA.

8. 2.9.1 "Guide for the Design of Durable, Repairable, and Maintainable Aircraft Composites" AE-27, SAE, 1997.

8.3.1 "Composite Repair NDI and NDT Handbook" ARP 5089, SAE, 1996

8.3.2.5(a) "Acceptable Methods, Techniques, and Practices – Aircraft Inspection and Repair, Volume II Airframe: Non-Metallic Structure, 1994

8.3.2.5(b) "Repair Stations for Composite and Bonded Aircraft Structure", Advisory Circular AC 145-6, 1996

8.3.4.2 (a) "Masking and Cleaning of Epoxy and Polyester Matrix Thermosetting Composite Materials", ARP 4916, SAE, 1997

8.3.4.2 (b) "Drying of Thermosetting Composite Materials", ARP 4977, SAE, 1996

8.3.4.3.4 (a) "General Composite Repair", Organizational and Intermediate Maintenance, Technical Manual, NAVAIR 01-1A-21, January 1994

8.3.4.3.4 (b) "General Advanced Composite Repair Manual", Technical Manual, TO 1-1-690, U.S. Air Force, July 1984

8.3.4.4.3(a) Francis, C.F., Rosenzweig, E., Dobyns, A., Brasie, S., "Development of Repair Methodology for the MH-53E Composite Sponson," in 1997 USAF Aircraft Structural Integrity Program Conference, San Antonio, TX, 2-4 December 1997.

8.3.4.4.3(b) Dodd, S., Petter, H., Smith, H., "Optimum Repair Design for Battle Damage Repair, Volume II: Software User's Manual," WL-TR-91-3100, McDonnell Douglas Corp., February, 1992.

8.3.4.4.3(c) Hart Smith, J., "Adhesive-Bonded Scarf and Stepped-Lap Joints," NASA CR 112237, Douglas Aircraft Co., January, 1973.

8.3.4.4.3(d) "Elastic Adhesive Stresses in Multistep Lap Joints loaded in Tension," ESDU 80039, Amendment B, Engineering Sciences Data Unit International plc, London, November 1995."

8.3.4.4.4(a) "Resin Mixing", ARP 5256, SAE, 1997

8.3.4.4.4(b) "Impregnation of Dry Fabric and Ply Lay-up", ARP 5319, SAE, 1998

8.3.4.4.4(c) MIL-HDBK-337

8.3.4.4.4(d) "Vacuum Bagging of Thermosetting Composite Repairs", ARP 5143, SAE, 1997

8.3.4.4.4(e) "Heat Application for Thermosetting Resin Curing", ARP 5144, SAE, 1997

8.3.4.4.4(f) Westerman, E.A., Keller, R.L., Rutherford, P. , "Improved Processing for Field Level Repair," The Boeing Company, Seattle, WA, Air Force Materials Directorate Wright Laboratory Report No. WL-TR-97-4119, December 1997.

8.3.4.4.4(g) Burroughs, B.A. and Hunziker, R.L., "Manufacturing Technology for Non-Autoclave Fabrication of Composite Structures," Air Force Contract F33615-80-C-5080, Final Report for period October 1980-April 1984, Report No. AFWAL-TR-85-4060.

8.3.4.4.4(h) Buckley, L.J., Trabocco, R.E., and Rosenzweig, E. L., "Non-Autoclave Processing for Composite Material Repair," Naval Air Warfare Center, Warminster, PA, Report No. NADC-83084-60, 1983.

8.3.4.4.4(i) Mehrkam, P.A., Cochran, R.C., and DiBerardino, M. F. , "Composite Repair Procedures for the Repair of Advanced Aircraft Structures," Naval Air Warfare Center, Warminster, PA, Report No. NAWCADWAR-92091-60, September 1992.

8.3.4.4.4(j) Bergerson, A., Marvin, M., Whitworth, D., "Fabrication of a Void Free Laminate by Optimizing a Non-Autoclave Cure," Society for the Advancement of Material and Process Engineering.

8.3.4.5.2 "Core Restoration of Thermosetting Composite Materials", ARP 4991, SAE, 1996

8.5.1 "Composite and Bonded Structure Technician/Specialist Training Document", AIR 4938, SAE, 1996

8.5.4 MIL-A-83377