10 Repair Technology

10.1 Introduction

Carbon/epoxy (and other similar) composite airframe components are immune to the costly forms of deterioration, notably cracking and corrosion, that plague aluminum and most other alloys used in airframe structures. However, these composites are much more easily damaged in service, for example, by mechanical impact. Thus repairability of such damage[1,2] is an important consideration in the selection of composites for aircraft applications.

Table 10.1 lists the major sources of service damage. These include:

- Mishandling
- Impact, for example, by dropped tools
- Contact damage in doors, often caused by poor rigging
- Delamination damage, often caused by inadequate shimming during component assembly
- Delamination caused during fastener removal or reinstallation
- Local overheating caused by impingement of hot exhaust gases or from a lightning strike

Light weight honeycomb structures most often require repair as the thin-face skins are easily damaged by mechanical contact. Moisture penetration can occur through damaged face skins as well as through badly sealed regions, resulting in corrosion damage if the core is aluminum alloy. Damage is more of a problem with the older-generation composites as these have relatively brittle matrices. The future trend in composite aircraft structures is thus away from honeycomb to (cocured) integrally stiffened structure, although damage over stiffener regions can be difficult to repair, particularly if a post-buckled design is used; in this design approach, the skins are allowed to buckle close to limit load.

10.2 Assessment of the Need to Repair

Methods of analytical assessment of residual strength in damaged composite components are needed to ensure that only necessary repairs are undertaken. Essentially, one of the following decisions is required:

- No repair action—damage is negligible.

Table 10.1 Major Types of Service Damage Suffered by Aircraft Composite Components

Defect	Typical Cause
Manufacturing Defects	
Voids	Poor process control
Delaminations	Inclusion of release film Poor process control Faulty hole-drilling procedures Poor fit of parts during assembly
Surface damage	Poor process control Bad handling
Misdrilled holes	Incorrect drilling procedure Faulty jigging
Mechanical Damage	
Cuts/scratches/abrasions	Mishandling
Penetrations	Mishandling/battle damage
Abrasion	Rain/grit erosion
Delaminations/disbonds	Impact: tools/hailstones/runway stones Freeze/thaw expansion Aerodynamic peeling Overload, e.g., during assembly or removal Lightning strike/static discharge/laser/overheat
Disbonds	Degradation of metallic interfaces, adhesive joints
Hole elongation	Fatigue-induced bearing failure, mechanically fastened joints
Dents/crushed core	Mishandling/impact
Edge damage, doors, etc.	Poor fit/mishandling
Environmental Damage	
Surface oxidation/burns	Lightning strike/laser/overheat
Core corrosion	Moisture penetration, metallic core
Surface swelling	Incorrect use of solvents or paint strippers

- Cosmetic or sealing repair is required to correct minor damage.
- Structural repair is required (if feasible) because strength is reduced below ultimate design allowable or has the potential to be reduced in subsequent service.
- Repair is not economic and component must be replaced.

When there is penetration damage, the requirement for a structural repair is obvious.

The decision is much more difficult for less obvious damage such as cuts, scratches, and barely visible impact damage (BVID). As yet, simple analytical approaches to estimate the strength of damaged composites (similar to fracture mechanics for metals) are unavailable, so empirical methods are generally used. For BVID, quite large areas of damage (typically 25 mm diameter) can be tolerated for older-generation carbon/epoxy systems without failures occurring below the ultimate design strain allowable, generally around 3000–4000 microstrain.

Fatigue studies[3] have shown that BVID will not grow under realistic cyclic strain levels for typical carbon/epoxy laminates. This is an important point because BVID will often not be detected until a 100% non-destructive inspection is undertaken. There is also the possibility of damage growth and resultant strength degradation under hygrothermal cycling conditions, but this does not appear to be a serious concern under moderate cycling conditions. However, catastrophic flaw growth is possible under severe hygrothermal cycling. This results from expansion of entrapped moisture due to freezing, or to steam formation on heating during supersonic flight.

10.3 Classification of Types of Structure

To evaluate the need for repair, it is also necessary to establish the criticality of the structure.

Structures are generally classified as follows:

- Primary—structure critical to the safety of the aircraft
- Secondary—structure that, if it were to fail, would affect the operation of the aircraft but not lead to its loss
- Tertiary—structure in which failure would not significantly affect operation of the aircraft

Inspection, damage assessment, and repair requirements differ significantly between these classifications. However, even within a single component, the allowable damage type and size (and consequently acceptable repair actions) will vary according to the criticality of the damaged region. The component is generally zoned by the original equipment manufacturer (OEM) in the structural repair manual (SRM) to indicate these regions. Mainly, the SRM will address repairs to non-primary structure. Repairs outside the scope of the SRM, particularly to critical regions of primary structure, require engineering design and approval by the OEM (or its delegate).

10.4 Repair Requirements

Generally, the repair scheme used for structural restoration should be the simplest and least intrusive that can restore structural stiffness and strain capability to the required level and be implemented in the repair environment, without compromising other functions of the component or structure.

It is usually necessary to restore the capability of the structure to withstand a design's ultimate loads and to maintain this capability (or some high percentage of it) for the full service life. Structural requirements for the repair vary according to the component or structural element. For example, wing skins are strength-critical in tension or compression, tail skins and control surfaces are often stiffness-critical (or flutter-critical), whereas spars and (unpressurized) fuselage skins may be stability- or buckling-critical.

The functions that must be restored include: 1) aerodynamic shape, 2) balance, 3) clearance of moving parts, and 4) resistance to lightning strike. The requirement in military to restore the stealth properties of the component may also have to be considered and may influence the type of repair chosen.

Important additional requirements are that implementation of the repair should minimize down-time of the aircraft, use readily available and easily storable materials, remove as little sound material as possible, minimize degradation or damage to the surrounding region, require only simple procedures or tooling, produce minimal increases in the weight of the component.

The type of structure and its accessibility are major considerations in determining the repair approach taken. For example, honeycomb structures with thin face skins are relatively easy to repair using core inserts and simple external patches. Highly loaded thick-skin components will usually require elaborate scarf repairs.

10.4.1 Repair Levels

A major consideration in the choice of repairs is the level at which the repair can be or needs to be implemented. Repair activities are performed at one of the following levels:

- Field level is undertaken directly on the aircraft, in a situation where skilled personnel and/or adequate facilities are unavailable. Such activities will generally be limited to minor repairs to non-primary structure or non-critical repairs to primary structure. Emergency field repairs may be undertaken in a battle situation to make the aircraft operational or to ferry it back to base. Battle-damage repairs (BDR) must be implemented rapidly by relatively unskilled technicians using minimum equipment. They may subsequently be replaced with permanent repairs.
- Depot level is undertaken in a situation where skilled personnel and facilities are available (up to factory capability in some cases). However, if the component is too large or difficult to remove from the aircraft, repairs are implemented directly on the aircraft.

An alternative system of classification is organizational, intermediate, and depot. Organizational equates with field; intermediate is a special repair facility having a capability above field and below depot.

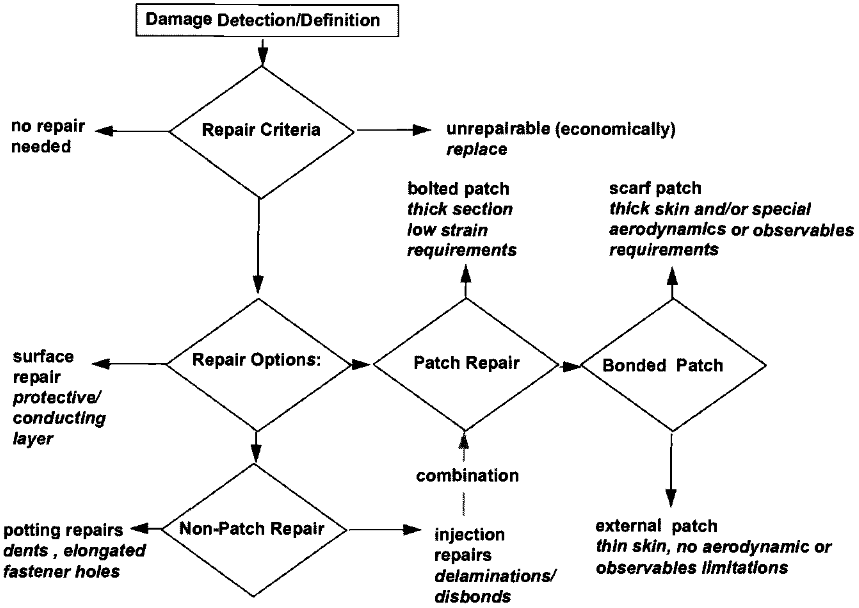

Fig. 10.1 Flow diagram indicating repair options.

Figure 10.1 shows that repairs can be broadly classified into non-patching techniques, for minor damage (detailed in Table 10.2) and patching techniques for more major damage, where restoration of the load path is required (detailed in Table 10.3). Patches can be attached by adhesive bonding or mechanically fastened by riveting or bolting.

Table 10.2 Non-patch Repair Procedures for Minor Damage

Procedure	Application
Resin injection	Connected small voids Small delaminations Small disbonds
Potting or filling	Minor depressions Skin damage in honeycomb panels Core replacement in honeycomb panels Fastener hole elongation
Fusion	Delaminations in thermoplastic-matrix composites
Surface coating	Seal honeycomb panels

Table 10.3 Patch Repair Procedures for Major Damage

Patch Approach	Comments
Bonded external patch – precured carbon/epoxy – precured carbon/epoxy bonded layers – cocured carbon/epoxy – titanium alloy foil – carbon/epoxy cloth patch (wet lay-up)	Suitable for repairs to laminates up to 16 plies thick Excellent restoration of mechanical properties Easy to implement; well suited for field applications Unobtrusive, minimum further damage to structure
Bonded scarf patch – cocured carbon/epoxy patch – precured carbon/epoxy patch	Suitable for repairs to thick laminates Excellent restoration of mechanical properties Obtrusive; requires significant material removal Difficult to implement; suited only to depot-level repairs
Bolted external patch – titanium alloy (usual) – aluminum alloy – carbon/epoxy cloth patch (wet lay-up)	Suitable for repairs to thick laminates Limited restoration of mechanical properties, but may be sufficient Obtrusive; requires large number of extra fastener holes Easy to implement; well suited for field applications

10.5 Non-patch Repairs

10.5.1 Filler or Potting Repairs

Potting repairs, illustrated in Figure 10.2, are made to fill the defective region with a resin. They may be used to fill minor indentations in a laminate, provided that non-destructive inspection (NDI) has shown that no extensive internal matrix cracking or delaminations are present. In the case of lightly loaded honeycomb panels where the composite skin has been penetrated, potting repairs may be made to stabilize the skin and to seal the damaged region. In this case, the damaged skin around the penetration is removed, together with the damaged part of the core—which is usually undercut to entrap the potting compound. The potting compound (which usually consists of a compatible resin, such as an epoxy, with either a chopped-glass-fiber or ultra-fine-glass-sphere filler) is then applied and cured. An alternative approach is to plug the cavity with glass-cloth/epoxy pre-preg or wet lay-up, but this imposes a high weight penalty that may not be acceptable for a control surface, for example.

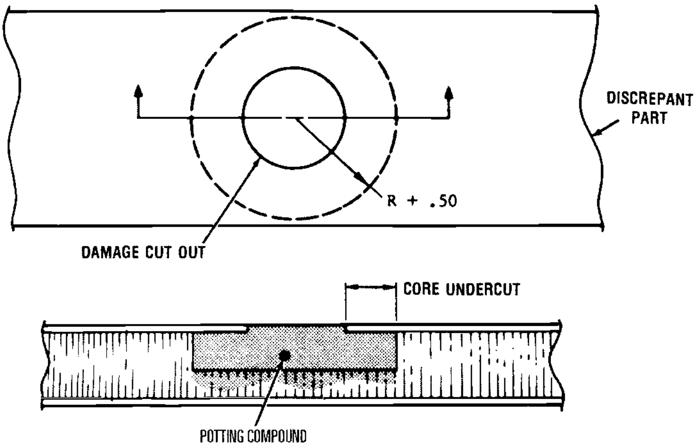

Fig. 10.2 Potted repair to a damaged honeycomb region.

Provided bearing loads are not too great, damage to attachment holes, such as minor hole elongation or wear, can be repaired by filling the hole with a machinable potting compound. A superior approach, particularly if bearing loads are high, is to adhesively bond a metal rod into the hole and redrill.

10.5.2 Injection Repairs for Delaminations

Resin-injection repairs, illustrated in Figure 10.3, are used for minor disbonds and delaminations in laminates or joints. The effectiveness depends on whether

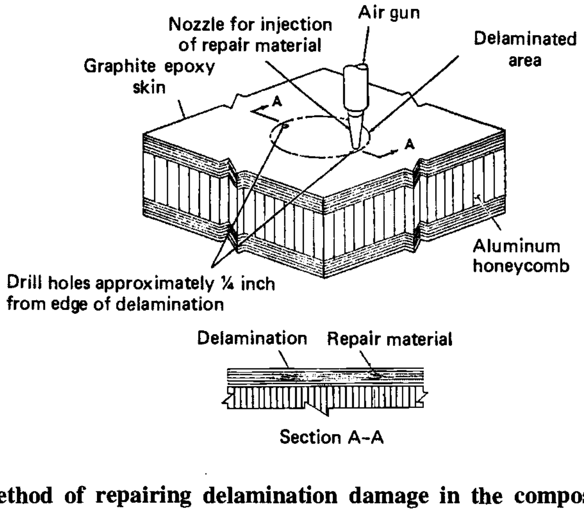

Fig. 10.3 Method of repairing delamination damage in the composite skin of a honeycomb panel by resin injection.

the defect arose during manufacture or was caused by mechanical damage during service. Manufacturing delaminations caused by a local lack of bonding pressure or contamination of the bonding surface have surfaces that are very difficult to bond without pretreatment, making injection repairs generally unsuccessful. In contrast, internal flaws caused by mechanical damage have surfaces that can be bonded reasonably well, provided contamination has not occurred by service fluids such as fuel or hydraulic oil; water can be removed by drying.

The repair procedure involves the injection of a compatible thermosetting resin under pressure directly into the delaminated or disbonded region. This can be done directly if the delaminations are exposed, as in the case of edges or fastener holes, or, if not, through injection holes (with bleeder holes to allow displacement of air, etc.—vacuum may be applied to aid this action). It is difficult, however, to ensure that these holes break into all the delaminations as many will not be internally connected.

The resin is generally injected after the part is preheated to about 70°C to decrease resin viscosity and improve wetting capability. Pressure can be applied to the component during cure; this is achieved under temperatures of about 150°C, to improve mating of the delaminated surfaces and to maintain contour. A fastener can be installed through the damaged and injected region in an attempt to further improve transverse strength.

Resin injection, although an attractive unobtrusive method for repairing delamination damage, is currently limited to non-critical applications. This is in part because resin systems recommended in the SRMs have relatively poor flow and wetting capabilities. This results in incomplete penetration of delaminations and consequently low strength recovery. Some improvement is being obtained with improved lower-viscosity adhesives.

A special resin has been developed that is optimized for injection repairs.[4,5] This resin provides excellent penetration of voids and delaminations, using either vacuum assistance or simple positive pressure, and produces a void-free well-bonded repair with a high T_g. Full recovery of compression strength after impact damage was demonstrated even when the internal delaminations were initially contaminated with environmental fluids including jet fuel, hydraulic fluid, detergent, or seawater.

10.5.2.1 Fusion Repairs for Thermoplastics. For the reasons mentioned in Chapter 9, thermoplastics are difficult to bond with thermosetting adhesives. Thus, injection repairs to delaminations are unlikely to be successful. Because thermoplastic materials are fusible, it should be possible to repair them by applying pressure and heat to the delaminated region. Unfortunately, melting temperatures are so high (typically over 380°C) that with general heating the component must be pressurized on a suitable tool to avoid distortion and delamination. Control of cooling rate is also important to avoid degradation of the mechanical properties of the matrix. This makes in situ repairs unfeasible in most cases. Another approach is to use thermoplastic adhesives with a lower melting temperature; however, because of their very high viscosity, thermoplastic adhesives are not suitable for injection repairs, though they may be suited as film adhesives for patch repairs. With mechanical abrasion, or better, with more elaborate surface treatment procedures, patching-type repairs using thermosetting adhesives may be feasible.

Several heating methods, based on either magnetic-induction, electrical-resistance, infrared, microwave, or ultrasonic welding, have been considered or evaluated in attempts to provide in situ heating. Induction[6] is probably the most promising for fusion of delamination damage.

10.6 Patch Repairs: General Considerations

Patch (reinforcement) repairs are intended to restore the load path removed by the damage, ideally, without significantly changing the original load distribution. Table 10.4 provides a summary of the performance requirements for bonded patch repairs. For a preliminary analysis, these repairs can be modelled as one of the simple joints discussed in Chapter 9.

The level of recovery of operating-strain possible, by a repair, is dependent on the stiffness of the laminate. The actual load to be transmitted by the joint is:

$$P = e_a E_x t \tag{10.1}$$

where e_a, E_X, and t are the allowable or ultimate design strain (often 4000 microstrain), the modulus in the primary loading direction, and the laminate thickness, respectively.

There are four main patching approaches:

(1) external bonded patches,
(2) flush or scarf-bonded patches,
(3) bolted patches, and
(4) bolted and bonded patches.

Adhesively bonded patches provide the most effective load transfer and, because fastener holes are not required, external patches minimize further damage to the structure. Bonded repairs are capable of restoring the original strength of the composite; however, these advantages are offset by the greater degree of complexity and skills required and the longer time required to complete the repair. Other problems with bonded repairs include the need to dry the composite before bonding to avoid porosity associated with the formation of water vapor during cure and the limited storage life of adhesives and other repair materials.

Bonded external-patch repairs are generally restricted to thin-skin applications (for example, up to 16 plies, around 2-mm-thick carbon/epoxy), whereas flush or bolted external patches are applicable to repairs for thick sections.

Scarf repairs are used when the thickness of composite exceeds the repair capability of simple external patches, around 2–3 mm of carbon/epoxy. Scarf

Table 10.4 Some Requirements for Patch Repairs to High-Performance Military Aircraft (Requirements Marked with * Refer to Bonded Repairs)

Requirement	Thick Primary Structure	Thin-skin Structure	Composite Substructure
Service temperature	−54–104°C	−54–104°C	−54–82°C
Ultimate strain/strength recovery	±4000 microstrain	±4000 up to ±6000 microstrain	Web shear and strength recovery; web cap strength and cap pull off recovery
Stiffness change	No decrease; moderate local increase acceptable		
Weight change	Minimum	Minimum; critical on control surfaces	Minimum
Aerodynamic smoothness	Maximum 2 mm change; less than 1 mm if stealth requirement	Less than 1 mm if control surface or for stealth requirements	No particular limitation
Skin configuration	Typically curved with varying thickness including ply drop-offs		
Moisture absorbed	1% prior to repair		
Spectrum loading	Durable for two lifetimes of severe loading		
Environment	Exposure to high humidity, fuel, and hydraulic fluids		
Maximum patch cure temperature*	180° C, T_g must be similar to that of parent material		
Maximum cure pressure*	Usually non-autoclave, generally atmospheric or below		
Accessibility to damaged region	Usually from outside only		
Facilities/skills	Simple as possible, commensurate with efficient cost-effective repair		
Quality	Low porosity; low acoustic attenuation		

repairs are also used where it is required to maintain 1) aerodynamic smoothness, 2) radar cross-section, or 3) clearance—for example, where a component, such as a flap, must fit into a restricted space.

Compared with bolted or bonded external patches, scarf repairs to thick skins are difficult to apply and usually require depot-level facilities and highly skilled personnel. Scarf repairs to thick laminates have the serious disadvantage of requiring removal of a large amount of sound material because of the shallow scarf angles required, generally about 3°. Finally, (as discussed later) creep or stress relaxation may be a concern under prolonged high stress levels, particularly at elevated temperatures.

Bolted repairs are highly suited for the repair of thick composite skins. These repairs are simple to apply by technicians familiar with standard repairs to metallic airframe structures, require no drying, and involve minimal removal of the parent structure. However, bolted repairs are limited to relatively low strain levels, below 4000 microstrain, and usually involve the drilling of many extra fastener holes.

Bolted repairs are, however, unsuitable for the repair of honeycomb structure because (unless very carefully sealed) the bolt holes provide a path for environmental ingress and generally do not provide a sufficient strength recovery. They are also usually unsuitable for the repair of post-buckled structure because they lack the required flexibility.

10.7 Bonded Patch Repairs

10.7.1 Design Approaches

Repairs are often designed, at least to a first approximation, by modelling the repair region as a representative joint, as shown in Figure 10.4. Thus an external patch repair is modelled as an overlap joint, and a scarf repair is modelled as either a scarf or a step-lap joint. This one-dimensional approach essentially assumes the repair to be a single-load path joint. In a real repair situation, load shed by an inadequately stiff repair can be carried by the parent structure, provided its degraded strength allowables are not exceeded.

An important design input is the ultimate design strain in the repair region. This information is often difficult to obtain, particularly at short notice. If such information is unavailable, strain levels can be based on an assumed ultimate allowable design strain for the parent structure. A reasonable estimate for current (military) designs is an ultimate design strain level of between $\pm$ 3000 and 4000 microstrain. For the purpose of this chapter, the higher level is assumed for conservatism in the design of repairs.

For more realistic designs, several other factors need to be considered, including:

- The allowable strain at the edges of the (reinforced) cut-out, typically over 10,000 microstrain
- The geometry of the repair joint
- Through-thickness or peel stresses
- Extra load attracted to the region due to local stiffening by the repair

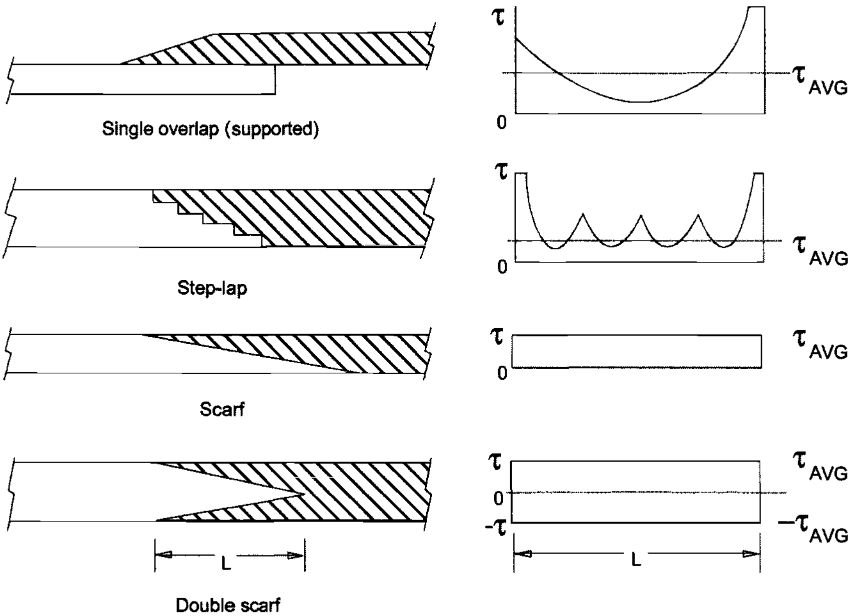

Fig. 10.4 Main types of joint configuration used for the bonded patch and resulting shear-stress distribution in the adhesive; *t* is shear stress and τ_{AVG} is the average shear stress.

- Creep/stress relaxation in the adhesive (function of temperature and absorbed moisture)
- Secondary bending arising from the neutral-axis offset caused by external reinforcements
- Development of residual stresses where the coefficients of thermal expansion of patch and parent material differ
- Proximity of other repairs—multiple repair interaction
- Scale effects

To deal with most of these complications, particularly where significant through-thickness (peel) stresses are anticipated, structural analysis using a three-dimensional finite-element approach is probably the only option and in even more sophisticated designs may need to include non-linear to include geometrical effects and adhesive plasticity.

10.7.2 External Patch Repairs

Typical external patch repairs to thin-skinned honeycomb panels are illustrated in Figure 10.5. Patches generally cover a circular cut-out made to

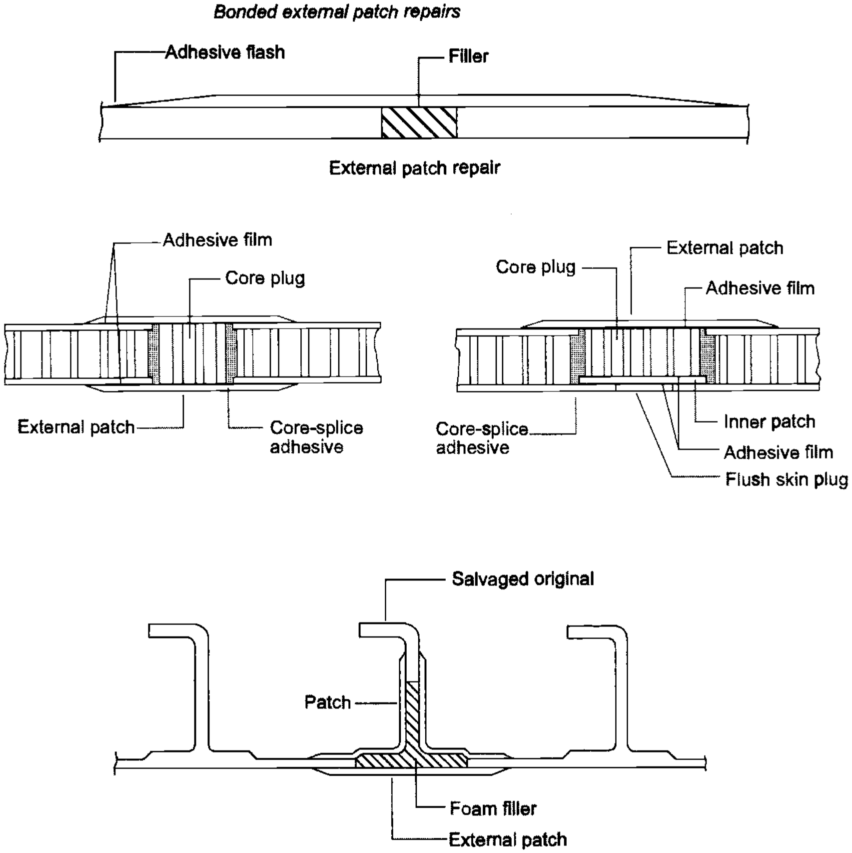

Fig. 10.5 Typical external patch repairs to honeycomb and to stiffened panels.

remove the damaged material. This shape is normally chosen for machining convenience and to minimize stress concentrations; however, a circular cut-out does not necessarily provide the lowest edge strain.

As a first approximation, an external patch repair can be modelled as one half of a double-overlap joint, assuming sufficient support is provided by the honeycomb core or substructure to react-out secondary bending. It is assumed for this analysis that:

- The patch is appropriately stepped at its ends to minimize peak shear and peel stresses (very important for patches thicker than about eight plies, or about 1 mm)—one ply drop per 3–4 mm about 3° effective taper.
- The edges of the cut-out are not tapered—this is a conservative assumption tapering improves the load-carrying capacity of the joint.

- The strength of the joint is limited by the load-carrying capacity of the adhesive in shear; this is most likely to be the case under hot/wet conditions. Under ambient or cold conditions, strength is more likely to be limited by the peel strength of surface resin or surface ply in the composite. This is a more complex scenario and therefore will not be considered here; however, in repair design it has to be ensured that peel is not the weakest failure mode. If it is, repair design will be based on a fracture-mechanics-type failure criterion related to ply or interface peel failure. The reality is that failure criteria for these modes have not been validated—this is an area of current research.

Two simple one-dimensional design approaches are considered here:

In the first, it is assumed that a patch of stiffness equal to that of the parent material is used and the strength of the joint is checked to see whether it will carry the ultimate design load. This assumes that the strain at the edges of the cut-out do not exceed the allowable ultimate design conditions.

In the second approach, the patch is designed so that the effective stiffness of the gap (cut-out) matches that of the parent material and then the strength is checked. This implies that there will be no strain concentration at the edges of the cut-out.

10.7.2.1 Simple Strength Analysis Based on Adhesive Strength. In Chapter 9 the maximum load-carrying capacity per unit width of a balanced joint is given as:

$$P_{\max b} = 2[t_A E t \tau_P(\gamma_e/2 + \gamma_P)]^{1/2} \tag{10.2}$$

where τ_p is the effective yield stress of the adhesive; γ_e and γ_p are, respectively, the elastic strain to yield and the plastic strain to failure; t_A is the adhesive thickness; t is the thickness of the patch (and the parent material); and E is its modulus.

Using, for a typical carbon/epoxy laminate and structural film adhesive (FM300), the following properties:

$\tau_p = 20$ MPa (hot/wet)	$t = t_p = 1.5$ mm (12 plies)
$E = 72$ GPa, typical for carbon/epoxy laminate	$t_A = 0.125$ mm
$G_A = \tau_p/\gamma_e = 0.4$ GPa	$\gamma_e = 0.05$, $\gamma_p = 0.5$

gives $P_{max\,b} = 0.75$ kN mm^{-1}.

It is necessary to show that the load capacity of the joint $P_{max\,b}$ exceeds the allowable load (per unit width) in the patch or parent material P_u given by:

$$P_u = E e_u t \tag{10.3}$$

where e_u is taken here as 4000 microstrain.

This gives $Pu = 0.43$ kN mm^{-1} showing that the joint has adequate strength. However, a safe margin is not obtained for external patch repairs of carbon/epoxy laminates above about 16 plies thick.

If it is considered that the strength of the repair is adequate, the next step is to determine the overlap length L. The total patch length is twice this plus the diameter of the hole. The minimum design overlap length (excluding the length of the taper) is given by:

$$L_{\min} = \frac{Ee_u t}{\tau_p} + \frac{4}{\beta} \tag{10.4}$$

where $\beta = \sqrt{2G_A/t_A Et}$ and $G_A = \tau_p/\gamma_e$

This gives L_{min} of about 40 mm on each side of repair cut-out.

Use of a generous overlap length is essential in repair designs. A long overlap ensures that the elastic trough (See Figure 9.10) is fully developed, thereby stabilizing the joint against creep in the adhesive and providing allowance for manufacturing defects, such as voids and disbonds, and service damage.

10.7.2.2 Stiffness Analysis. For a one-dimensional model of this repair joint the displacement Δ of the gap (cut-out), length h, is given by:

$$\Delta = 2\gamma t_A + he_R \tag{10.5}$$

where e_R is the strain in the patch. The first component of Δ is the displacement due to shear of the adhesive, while the second is due to strain of the patch.

To obtain equivalence in strain between the parent material and gap, we have:

$$\frac{\Delta}{h} = \frac{2\gamma t_A}{h} + e_R = e_P \tag{10.6}$$

where, based on parent material stiffness E_p and thickness t_p, the strain e_p in the parent material is given by:

$$e_P = \frac{P}{E_P t_P} \tag{10.7}$$

Assuming elastic behavior in the adhesive the shear strain γ is given by τ/G_A which, in terms of the applied load P, is given by:

$$\gamma = P\beta/G_A(1 + E_p t_p/E_R t_R) \tag{10.8}$$

Elastic/plastic behavior should be assumed for the adhesive because this is more representative of the conditions in the adhesive at ultimate design loads; however, for simplicity, only elastic conditions are considered here.

Substituting equations (*10.8*) and (*10.7*) into (*10.5*) and letting $S = E_R t_R/E_p t_p$ yields the following equation, which can be solved to obtain $E_R t_R$ as a function of h and adhesive properties.

$$(S-1)^2(S+1)\frac{h^2}{4}\frac{G_A}{t_A}\frac{1}{E_p t_p} = S \tag{10.9}$$

This shows that S is a function of the gap size h, the adhesive thickness, the shear modulus, and the absolute value of stiffness $E_P t_P$ of the parent material.

For the same joint properties as in the previous example, assuming h values of (a) 5 mm (b) 10 mm (c) 20 mm, and (d) 40 mm gives, for the stiffness ratio S, (a) 2.7 (b) 1.8 (c) 1.4, and (d) 1.2 respectively.

Thus only when h is greater than 40 mm (for this relatively thin skin) is a balanced stiffness approach reasonable. Because the joint strength is reduced by having an unbalanced joint [See equation (*9.20*)] ideally h should be chosen so that $S = 1$, provided that the cut-out is not excessive.

10.7.2.3 Modified Load Path: Two-Dimensional Effects. A more realistic model of an external patch repair needs to account for the increase in stiffness of the patched region and the load diverted around the patched hole. Generally, a finite-element approach would be needed to undertake an analysis of this complexity; however, simple analytical approaches can be used to obtain a useful first estimate.

The influence of the stiffness of the patch on the strain in the repair region (assuming the cut-out region is small in size compared with the patch) can be estimated using an inclusion model developed by Rose[7] for composite patching of metallic components. The model, which allows for the shape and elastic properties of the patch material, shows that due to extra load attracted into the region, the strain in the parent material is somewhat higher than would be estimated on the basis of patch/parent stiffness ratio. For example, for a circular patch of equal stiffness to the parent, the strain in the repair region may be reduced by about 0.60 rather than the 0.5 expected. This implies an effective increase in P, the load on the joint per unit width, of around 20%.

The influence of the load carried around the hole can be estimated using the approach of Hart-Smith.[19] This simple analysis considers a hole reinforced by a central strip. Because the unreinforced hole can carry load, an estimate can be made of the load that the hole can carry and therefore the remaining load that the strip must carry on the basis of the relative stiffness. Because a hole experiences a deformation three times that of the surrounding sound material, it has an effective stiffness of $1/3E_p t_p$. Thus, if the patch has stiffness $E_R t_R$, the effective applied load per unit width is P and the hole diameter d is such that load transfer effects can be neglected (See preceding section), the actual load P' that the patch must carry is given by:

$$P' = P/(1 + E_p t_p/3E_R) \tag{10.10}$$

Thus, assuming $E_R t_R = E_p t_p$ the load P is reduced by 20%. Therefore, for this example, the two effects approximately cancel.

10.7.3 Scarf Repairs

Scarf patch repairs are illustrated in Figure 10.6. Note that the configuration is actually scarfed in the parent material but a small step-lap in the repair, as the

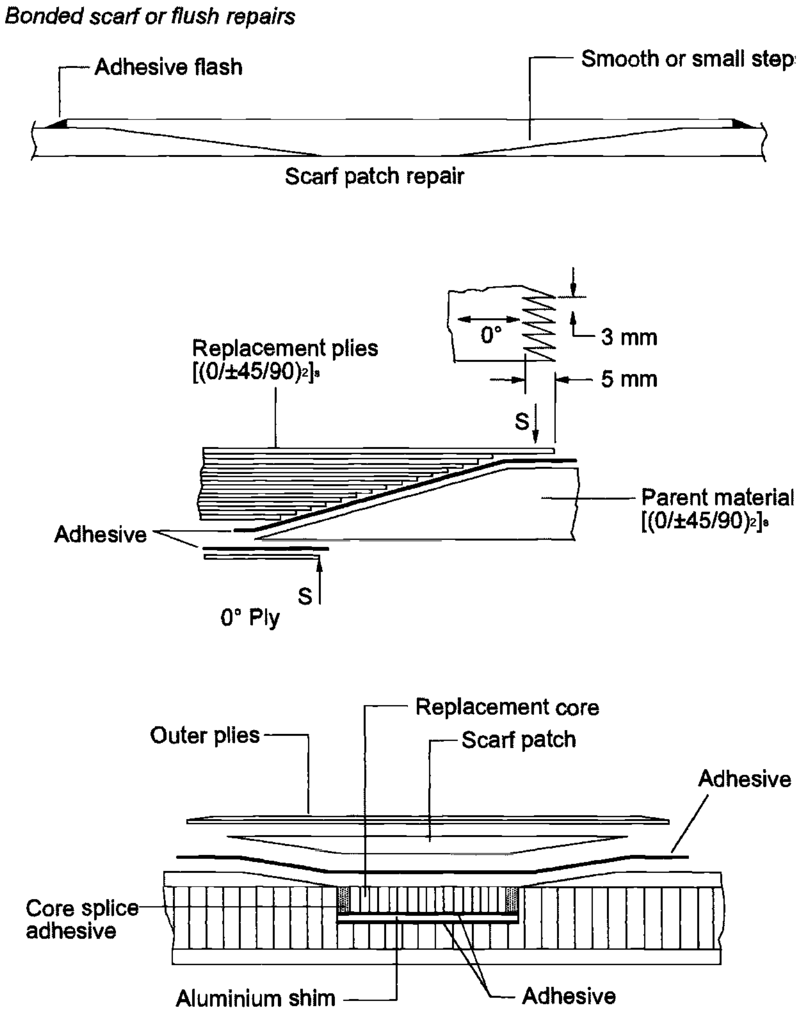

Fig. 10.6 General illustration of a simple scarf repair and, below, exploded views of a scarf repair to a thick-skinned honeycomb panel. S indicates serrated 0° plies sometimes used to reduce peel stresses in the outer plies.

repair patch is built up from individual plies. The condition of uniform shear stress in the adhesive layer holds only if the scarf is taken to a feather-edge. If the edge breaks away during repair implementation, or is damaged during service loading, significant stress concentrations arise that could lead to failure of the joint. To avoid this problem a thin external patch is bonded over the scarf region; the 0° plies may be serrated to reduce the peel stresses even further.

Scarf repairs can be modelled to a first approximation as a simple scarf joint. If the patch matches the parent material in stiffness and expansion coefficient, simple theory gives:

$$\tau = P \sin\theta \cos\theta / t \tag{10.11}$$

At small θ the normal stress σ is negligible. The required minimum value of scarf angle θ for an applied load P can be obtained from the following, taking τ_p as the peak shear stress:

$$P = Ee_u t = \tau_p t / \sin\theta \cos\theta \tag{10.12}$$

For small scarf angles, the condition for reaching the allowable strain e_u in the adherends is $\theta < \tau_p / Ee_u$ rad.

Taking e_u as 4000 microstrain, τ_p for hot/wet conditions as 13 MPa, and E as 72 GPa gives $\theta \leq 3°$

For a typical horizontal stabilator skin, for example, 4 mm, on a honeycomb core (tail skin) the minimum length of the scarf is about 80 mm, which with a hole size of, say, 80 mm, gives a total repair length of 240 mm. Because a bonded repair must be used in this case, the scarf repair would probably be accepted.

For a typical wing skin thickness around 13 mm, the minimum length of the scarf is about 250 mm which, for a hole size of 100 mm, gives a total patch length of 600 mm—far too long to be feasible in most cases. The problem is much less serious if the damage extends only part way through (a reasonable expectation in such a thick laminate) so that the repair length would be considerably a bit shorter. Full-thickness repairs on such a thick laminate would be based on bolted repairs, as described later. The strength recovery in this case should be acceptable if (as is usually the case for at least one of the skins) the wing skins are attached by bolts to the substructure.

10.7.4 Studies on Scarf Joints Representing Repairs

The focus of this program[8] is the scarf repair of the relatively thick-skinned region of the F/A-18 horizontal stabilator—AS4/3501–6 carbon/epoxy, over 35 plies thick in this area. This repair is particularly challenging because the design ultimate strain is 5200 microstrain, unlike most of the other composite components on the aircraft, where design ultimate strains are usually around 3500 microstrain. The aim was to evaluate the strength of scarf joints under compression or tension loading over the design temperature range for the aircraft of +104°C to −40°C. The adhesive used in these repairs is FM 300, having the following shear yield stress (τ_p) properties: approximately 40 MPa (RT dry) 13 MPa (hot/wet).

A honeycomb-sandwich beam specimen, developed for this program, corresponds to a section through a typical repaired region of the stabilator. The beams are loaded in four-point bending with the repaired region in either tension or compression. Figure 10.7 shows the configuration of the parent and repair laminate. The scarf joint consists essentially of a ply configuration similar to that of the parent material with doublers having extra 45° plies at the bottom and 0° and 45° plies at the top.

From the results provided in Figure 10.8, it is clear that hot/wet exposure (about 0.7% moisture in the skins) markedly reduces the failure strain of the joint. In the cold/wet and ambient temperature tests, failure almost invariably occurred in the honeycomb core so that these results represent a lower bound to the strain capacity of these specimens. In contrast, in the hot/wet tests failure was mainly cohesive in the adhesive layer.

A simple analysis of the shear stress in the scarf joint can be made, assuming a uniform (average) modulus through the thickness of the carbon/epoxy laminates.

Taking strain design ultimate for the parent material to be $e_u = 5200$ $\mu\varepsilon$, E_p as 80 GPa, t as 2.7 mm, and θ as 3° (allowing for the load taken by the outer plies of the patch),s the average shear stress τ_{av} in the joint is estimated from equation *(10.12)* to be $\tau_{av} = 19$ MPa, just above the hot/wet yield.

However, because the in-plane stiffness of the carbon/epoxy skin varies in the through-thickness direction, and the configuration is actually a step-lap on in the repair plies, the shear stress along the "scarf" may not be constant. A simple first approximation of the variation in shear stress along the scarf is obtained by assuming that the shear stress in the adhesive adjacent to each ply step on the

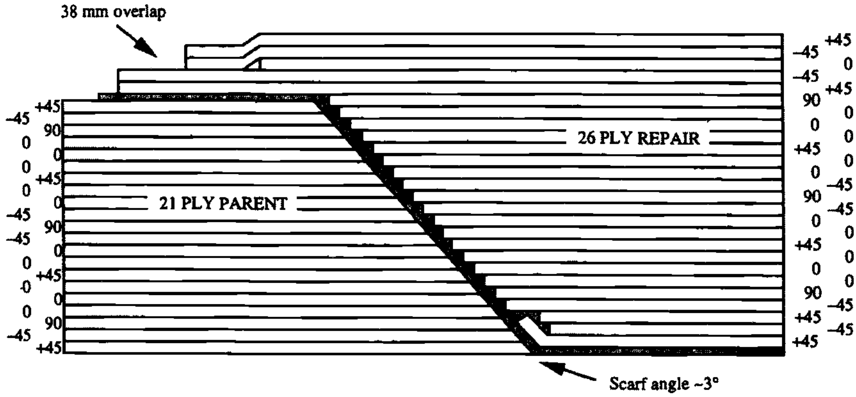

Fig. 10.7 Ply configuration of parent laminate (21 plies) and repair scarf laminate (26 plies). The adhesive (shown shaded) is either FM300 or FM300K structural film adhesive, and the scarf angle is nominally 3°. Taken from Ref. 8.

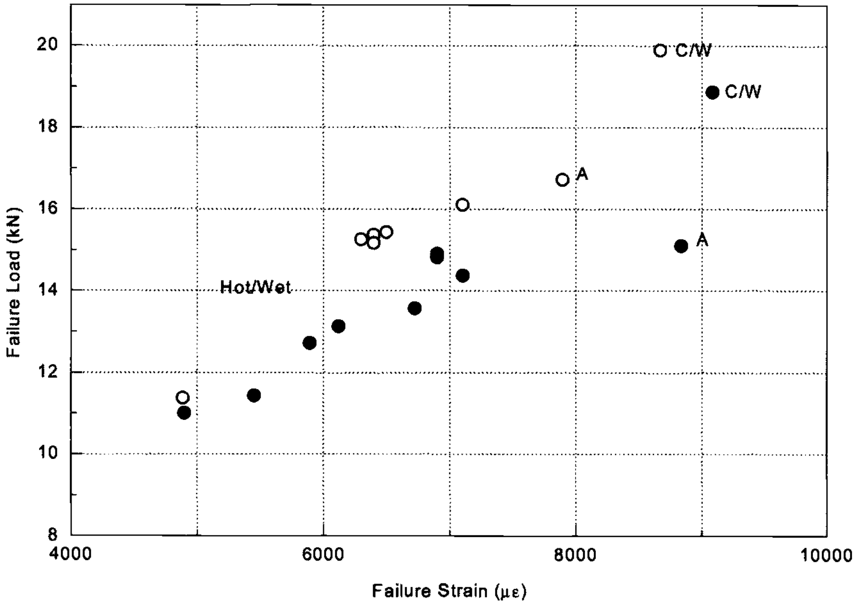

Fig. 10.8 Plot of failure loads and strains for the carbon/epoxy scarf specimens, *A* and *C/W*, respectively, refer to exposure at ambient and cold/wet conditions. Open holes or filled holes respectively refer to tension or compression loading. Taken from Ref. 8.

repair is proportional to the load carried by that ply, which is in turn assumed to be proportional to the relative stiffness of the ply. An analysis on this basis should give an upper bound (i.e., an overestimate) for the variation of adhesive shear stress along the scarf.

The average shear stress τ_{av} is given by:

$$\tau_{av} = \frac{[n_{0^\circ}\tau_{0^\circ} + n_{45^\circ}\tau_{45^\circ} + n_{90^\circ}\tau_{90^\circ}]}{n_{total}} \tag{10.13}$$

where τ_{0°, τ_{45°, and τ_{90° represent the shear stress in the adhesive adjacent to 0°, 45°, and 90° plies, respectively, and n is the number of plies.

The ratio of the laminate stiffnesses is 1 (0°), 0.23 (45°), 0.07 (90°), and therefore the equation can be rewritten as

$$\tau_{av} = \frac{[n_{0^\circ}\tau_{0^\circ} + 0.23n_{45^\circ}\tau_{0^\circ} + 0.07n_{90^\circ}\tau_{0^\circ}]}{n_{total}} \tag{10.14}$$

Inserting $\tau_{av} = 19$ MPa from above and solving for τ_{0° gives $1.74 \times \tau_{av} = 33.5$ MPa, with $\tau_{45^\circ} = 0.23 \times \tau_{0^\circ} = 7.7$ MPa and $\tau_{90^\circ} = 0.07 \times \tau_{0^\circ} = 2.3$ MPa.

The peak shear stress value of 33.5 MPa for τ_{0° is well above the hot/wet yield allowable for FM300 of 13 MPa, and, from this very simple analysis, the scarf may be expected to yield locally before a strain of 5200 $\mu\varepsilon$ is reached in the skin.

Although local yielding in the adhesive does not necessarily lead to immediate failure, it could lead to failure through creep. Thus it was conjectured[9] that final failure in the joint specimen under hot/wet conditions was initiated by strain build-up in the outer doublers as the scarf region experienced creep deformation in the adhesive layer. The outer doublers would be anchored against creep by the elastic trough in the adhesive layer.

The scarf specimens differ significantly from repairs in that they represent a single load path situation. Thus creep deformation in a repair will be much more limited and should result simply in load redistribution—not failure of the joint. Failure of the component would result only if the strain at the edges of the repaired hole exceeded the critical value due to the increased compliance. Thus the scarf results can be considered a conservative estimate of repair performance.

The shear stress distribution in the bondline for the ply-by-ply model, estimated using a finite-element model (elastic adhesive) is shown plotted in Figure 10.9. Results are included for both room temperature and hot/wet adhesive properties and for reference a similar aluminum-skinned beam. The shear stress in the composite-skinned beam is observed to vary with position along the bondline, with the maximum stress occurring adjacent to the ends of 0° plies, as expected from the simple model previously outlined. The magnitude of the variation in shear stress along the bondline is, however, much less than that predicted by the simple analysis.

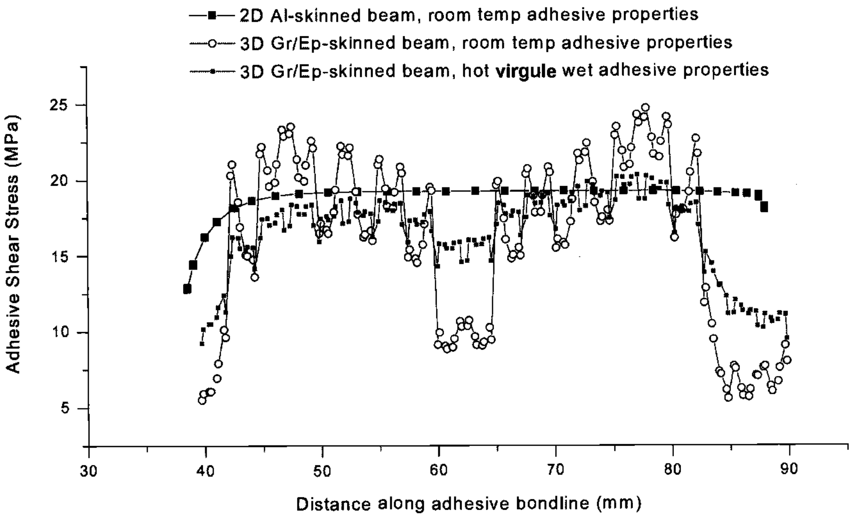

Fig. 10.9 Estimated shear stress distribution along the scarf for *FE* models of aluminum and carbon/epoxy scarf repair beam specimen. Taken from Ref. 8.

10.8 Materials Engineering Aspects

10.8.1 Bonded Patch System Options

The options for external patches for carbon/epoxy components are 1) carbon/epoxy, based on cocured or precured prepreg materials; 2) carbon/epoxy, based on wet lay-up with carbon cloth and resin; and 3) titanium alloy, based on laminated sheet. For scarf repairs, the main materials options are carbon/epoxy pre-preg, co-cured or pre-cured or, for lower stress requirements, wet lay-up.

Titanium foil can be used to produce a patch with good mechanical properties. In this case, layers of foil interleaved with adhesive film are bonded to the parent laminate. This approach appears to work well for repairs to fairly flat thin-skinned regions. To obtain strong durable repairs with titanium alloy patches, it is essential that the foil be correctly surface-treated and primed and that it not be contaminated during handling.

Where pre-preg materials or titanium foil cannot be used because of storage or forming problems, the wet-lay-up approach, using carbon fiber cloth and liquid-epoxy resin, is the best option. The manufacturing flexibility of wet lay-up procedures makes it well suited to the repair of complex stiffened internal structure, provided access is available. With suitable resins, wet lay-up is also highly suited for rapid field repairs, such as the repair of battle damage.

The advantages of the wet lay-up approach include:

- Long-time ambient-temperature shelf life for the resin because hardener and resin are stored separately
- High patch formability—patch can be formed over complex contours
- No separate adhesive required—the resin forms the matrix and the adhesive layer
- Processable even without vacuum bagging, although properties are markedly improved if bagging is used.

10.8.2 Patch Materials and Adhesives

The storage and processing properties of the repair adhesives and matrices (for cocured composite patches) strongly influence composite supportability.[10]

As well as having mechanical properties matching the parent structure, an ideal adhesive or matrix resin will be storable for long periods at ambient temperature, curable in short times at modest temperatures, and processable under simple vacuum bag conditions. As may be anticipated, no available materials meet all these requirements, and various compromises must be made.

Structural film adhesives provide excellent mechanical properties and durability in repair joints, provided they are processable under repair conditions. Film adhesives are also very convenient and reliable to use because, unlike with paste adhesives, there is no requirement for weighing and mixing of resin and hardener.

Film adhesives may be used for repair work in a partially precured condition; this is called B-staged or (when even more advanced) C-staged. The aims of staging the adhesive are to:

- Reduce the amount of flow during the cure, avoiding the danger of excessively thin bond lines
- Minimize voiding—high resin viscosity results in high hydrostatic pressure during the cure, discouraging void formation
- Improve storage life—a C-staged film adhesive may be storable at ambient temperature for several months with minimal property change

To provide an easy exit route for entrapped air and volatiles during cure, the film adhesive may be embossed by using a honeycomb panel as a caul plate for the staging.

Unfortunately, structural film adhesives suitable for repairs to carbon/epoxy (such as FM300, usable up to 110°C) have the major disadvantage of limited storage life, even under deep-freeze conditions, nominally 6–12 months at −20°C. A related problem with film adhesives is high cost and the large minimum order quantity. Furthermore, lead time for reordering some adhesives can be over 3 months. To maximize life, refrigeration (usually in dry ice) is essential, even during transportation. More recently, film adhesives specially developed for repair applications with long ambient-temperature storage capability have become available (FM275 is an example), which could greatly aid use of these materials for repairs.

In contrast, two-part paste adhesives have considerable advantages for long-term storage because the reactive components can be kept separate until required. The pressure requirements to cure paste adhesives is much lower than for film adhesives so that vacuum bag pressure is sufficient. Generally, low (not full) vacuum is used because this minimizes void formation in low-pressure regions. Special systems have been developed suitable for repair applications (such as Hysol EA 9391) with properties comparable with structural film adhesives (e.g., FM300) and good processing qualities.

A major disadvantage with paste adhesives (or resins) is the need to weigh the components accurately and to mix them thoroughly. The problem can be substantially reduced with the use of suitable packaging. Each pack contains the two components, preweighed for a single application and separated by a breakable seal; the seal is broken and the components mixed while contained in the outer envelope. The mixed adhesive is then ejected after cutting the envelope. Although most components of two-part systems have a long storage life under ambient conditions, refrigeration significantly increases life.

The repair approach that provides the best combination of formability and mechanical properties is to cocure the patch in pre-preg form with an appropriate structural film adhesive. However, the highest mechanical properties are obtained by precuring the patch in an autoclave under optimized processing conditions and then bonding in a separate operation. This is sometimes called the hard patch approach, and can also be used with scarf repairs.[11]

Pre-preg materials similar to those used in the manufacture of the composite component provide optimum properties, provided that they can be processed under repair conditions. Generally, however, the pressure requirements to process standard pre-preg are higher than can be achieved by vacuum bag processing. The result is a relatively high void content and low fiber/volume fraction. Often the patch is so porous that attenuation is too great for the use of ultrasonic NDI. However, using a combined pre-consolidation and de-gassing approach known as double-vacuum processing,[12] it is possible to reduce voids to less than 3% and markedly increase fiber/volume fraction. This procedure, which reduces the pressure required during patch application and minimizes porosity, involves placing the patch lay-up with the vacuum bag inside a sealed box to which a vacuum can be applied. First, the vacuum is applied under the bag and to the box, which allows the lay-up to degas under no external pressure, maximizing the removal of trapped air and volatiles. The vacuum in the box is then vented and normal vacuum allowed to consolidate the patch lay-up.

The double-bag technique can also be used to pre-process wet lay-up patches resulting in a marked reduction of voids and improved fiber/volume fraction.[13]

10.8.3 Repair Joint Preparation

The damaged region is first outlined as a geometrical shape that allows the accurate preparation and installation of a patch. Usually, the shape will be elliptical and will encompass the area of damage, as determined by NDI and visual inspection, but will include as little as possible of the sound material.

Carbon/epoxy is best cut with tungsten-carbide-tipped tools; conventional high-speed tools can be used, but their life is short. Most forming operations for repair purposes can be performed with an end-mill cutter or router mounted on an air motor on a portable base. Air-driven tools are preferred because carbon fibers are known to short-circuit electrical motors. A template may be used to control the outline of the shape of the cut and shims used to control its depth. Taper cuts may be made, using shims to allow cuts of one ply at a time. Alternatively, a sanding drum (alumina or silicon carbide grit) may be used to cut a smooth taper. In this case, the tool may be hand-guided (controlled by the operator's observation of the ply exposure) or, preferably, template-guided (robotic machines for cutting tapers are under development). The taper may extend fully through the thickness of the laminate in the repair of penetration damage, or only part-way through in the repair of delamination damage.

For scarf repairs, a taper angle of approximatly 3° provides acceptable levels of shear stress in the adhesive. If ground with a reasonable surface finish, the ply sequence in the repair region is clearly visible.

Instead of a cutting or grinding operation, it is possible to form the taper using a peeling procedure. This involves using a sharp knife to cut through the composite, one ply at a time, and then using grips to peel each ply back to the cut. By this approach, a step-lap joint is produced.

10.8.4 Pre-Bonding Surface Preparation

It is vitally important for the success of bonded repairs that all bonded surfaces be correctly prepared. Surface treatment is required for precured composite patches, metallic patches, and the parent material.

The first step in preparing a component for repair is to remove paint and major surface contaminants. This is generally done by a solvent degrease with MEK and then removal of the original surface material with abrasive pads. To improve environmental health and safety, the aim is now to avoid volatile organic solvents (VOCs) by using water-based solvents.

For thermosetting-matrix composites, the most effective surface-treatment for strong, durable bonding is to grit-blast with alumina or silicon carbide particles. When done correctly, this process provides a clean, uniform, high-energy surface without removing too much of the original surface resin.

Thorough abrasion of the surfaces with silicon carbide paper is a reasonable alternative, but is less satisfactory because minor depressions in the surface are left untreated unless a considerable amount of surface material is removed. This results in a weaker joint because little surface resin remains and surface fibers are exposed and damaged.

10.8.5 Moisture Problems

Moisture can cause serious problems if it is not removed by an initial heat-treatment, especially with composites based on high-temperature matrices, such as bismaleimides, because repair cure temperatures will exceed 200°C. During patch application, the moisture may vaporize, split the laminate in the voided regions, and form voids in the adhesive and in the matrix of the repair laminate (if being cocured). Damage in the matrix can be severe if heat-treatment is performed above its T_g, when its strength is quite low. During cure of the adhesive or patch, moisture that has diffused in from the parent laminate will produce voids[14] if the partial pressure of the moisture exceeds the applied (hydraulic) pressure during cure. In all cases, the result may be a severe degradation in mechanical properties. The problem of moisture removal is much more difficult in a thick laminate (50 plies or more) because many days of heating may be required. However, it is not necessary to remove all the moisture; only the surface moisture causes problems in curing the patch and adhesive.

Thin laminates (16 plies or less) can be dried out fairly rapidly. However, if the laminate forms the face of a honeycomb panel, care must be taken because high internal pressure can develop[15] that can easily exceed the strength of the skin-to-core bond. The internal pressure developed in the core during cure is made up of the partial pressure of the air and the partial pressure of moisture desorbed from the skin. Typically, at a cure temperature of 175°C, pressures well over 0.5 MPa can develop, which, depending on the adhesive, can exceed the hot/wet adhesive strength, resulting in skin/core separation.

With precured patches, precured bonded plies, or titanium foil patches, the patches do not suffer moisture problems. However, the problem of adhesive porosity is even more severe because the patch is unable to absorb any of the evolved moisture.

Contrary to expectation, application of the vacuum during drying of the skin does not significantly increase the drying rate. This is because drying is controlled by the rate of moisture diffusion through the laminate. However, a vacuum is used to aid in the removal of entrapped moisture from honeycomb panels.

There are several methods for quantitatively assessing moisture behavior during drying of the laminate and subsequent cure of the patch system. However, a simple but useful method for estimating drying requirements is based on the concept[16] of drying depth. As discussed in Chapter 9, the drying time t for which moisture concentration in the composite at depth x is reduced to half its original value is given by $t = x^2/D$, where D is the diffusion coefficient.

As an example, consider a moist carbon fiber/epoxy laminate on which an external patch will be bonded at 120°C for 1 h. Taking $D = 2 \times 10^{-6}\ \mathrm{mm^2\,s^{-1}}$ (typical value) gives a value of x of 0.085 mm. To be safe, it is suggested that the composite should be initially dried to give a value of x of twice this value. For drying to be undertaken at 105°C (where $D = 3 \times 10^{-7}\ \mathrm{mm^2\,s^{-1}}$), the time for x to reach 2×0.085 mm is found to be 26 h.

If moisture is present in the core of a honeycomb panel, and particularly if cure is at high temperature, an elaborate drying cycle is required. The procedure used is to apply an external supporting pressure using a vacuum bag and then to place the component in an oven at about 90°C for about 30 h, allowing the moisture to escape by the route it entered the component. The temperature is then raised to just above 100°C and held for several hours. The gases drawn out of the vacuum bag are checked for moisture and the process is continued until no trace is found.

10.9 Application Technology: In Situ Repairs

Heat and pressure are required to cure the adhesive and obtain a uniform non-porous adhesive layer. Cure pressure requirements are most simply satisfied using a vacuum bag, [1] that provides a pressure of around 1 atm. This pressure is adequate if the patch mates well with the parent material (by pre-forming the patch on a mold made on the parent structure) or is cocured with the adhesive on the surface of the parent material.

Heat may be applied internally by encasing a heater blanket under the vacuum bag (usually an electrical resistance wire embedded in silicone rubber). Alternatively, a reusable combined vacuum bag and heater blanket may be used, consisting of silicone rubber with built-in heater wires. Silicone rubber heater blankets can be unreliable because they are prone to burn-out during a repair, particularly if used for large heat inputs or for high-temperature cures.

Figure 10.10 illustrates the vacuum bag assembly for an external patch, and Figure 10.11 illustrates the assembly for a flush or scarf patch. The illustration for the scarf patch shows its use to repair part-through damage, such as a region that has been ground.

The simple vacuum bag procedure has several major drawbacks, most of which are associated with the low pressure that may be created in some regions inside the bag. These include:

- Entrapped air and volatile materials in the resin matrix and adhesive expand, leaving large voids in the cured resin
- Moisture absorbed in the carbon fiber/epoxy parent laminate enters the adhesive, producing voids (and possibly interfering with the cure mechanism)
- Air that, is drawn into the bond region due to porosity in the parent material produces voids in the patch system
- Reduced pressure produced inside a honeycomb panel can subsequently cause the panel to collapse.

Thus, although the vacuum bag procedure generally works well, its use has dangers. A safer alternative is to use pneumatic or mechanical pressure. The problem here is to work out how to react out the resulting loads. If they cannot be reacted out by the surrounding structure, vacuum pads or adhesively bonded anchor points can be used.

10.10 Bolted Repairs

Bolted repairs of composites are based on well-established procedures for the repair of metallic aircraft components. They are suitable for field repair of thick

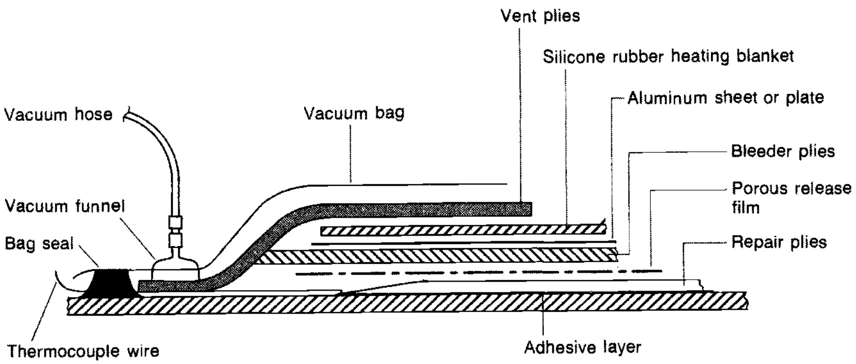

Fig. 10.10 Vacuum bag and patch arrangement to bond an external patch repair.

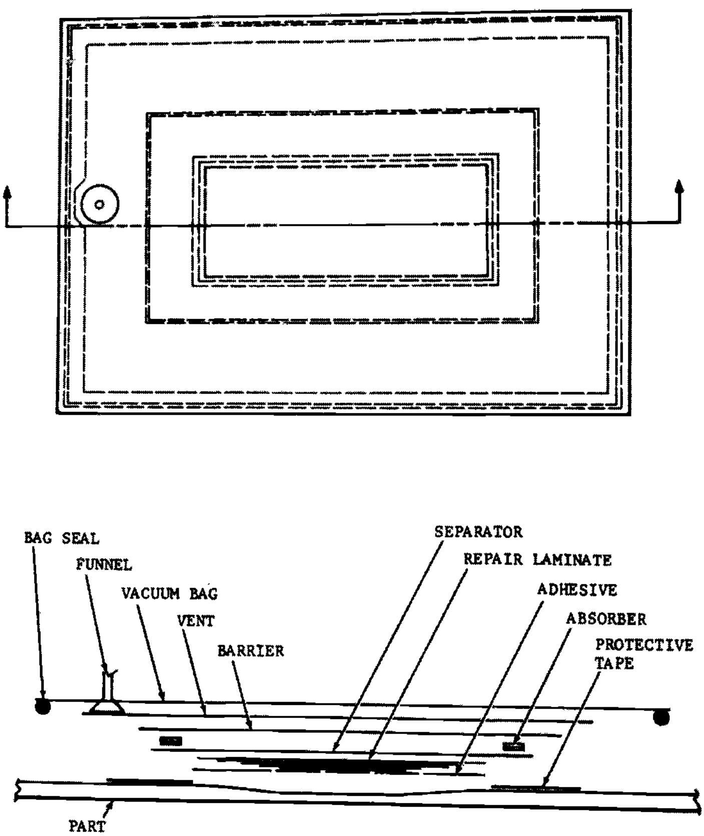

Fig. 10.11 Plan and section views of a vacuum bag and patch arrangement for application of a scarf repair.

laminates (over 3 mm) where the stressing requirement for external patch repairs exceeds the capability of adhesives. They are also suitable for depot-level repairs where the complexity of repair implementation and requirement for extensive material removal preclude the use of scarf repairs. Bolted repairs have the important advantage that no drying of the damaged component is required.

Bolted patches are generally external (sometimes called scab or boiler-plate patches) but they can also be flush, with the patch on the inside surface and the damage hole filled with some resin filler. External bolted repairs are particularly well-suited to the rapid repair of battle damage because they are easy to apply and require minimal facilities.

The stress concentrations associated with bearing loads in a bolted repair are not a serious disadvantage in a composite structure already mechanically fastened—for example, a composite wing skin bolted to a metal or composite substructure. A major advantage of bolted repairs is that the bolts provide a transverse reinforcement (with clamping pressure), which can be effective in preventing the spread of pre-existing delaminations.

An alternative option is the bonded/bolted approach. As discussed in Chapter 9, although it is not generally possible to obtain satisfactory load-sharing between the mechanical fasteners and the adhesive, there are several advantages in this approach for repairs:

- Bolts can provide an alternate (fail-safe) load path in the event of bond failure. Thus, bonding and bolting can be used in some situations where the bond quality may be questionable due, for example, to moisture problems or in thick structure where failure of the bond is the likely failure mode in an overload situation.
- Bolts can stabilize delaminations (as previously discussed), thereby minimizing or even avoiding the need to remove the damaged region before a bonded repair.
- Bolts can be used as jigs to locate and pressurize a bonded repair during the cure cycle.

10.10.1 Bolted Patch Repairs: Design Approaches

A bolted patch repair is usually modelled as a single-lap bolted joint. If backing structure is provided such that the loads in the damaged material can be shared equally between the patch and the backing structure, double-shear conditions can be assumed. In the case of a single-lap joint, the patch is assumed to be supported against secondary bending resulting from loading eccentricity.

The design procedure described in Ref. 17 is based on a compliance approach that assumes compatible deformation of a metallic patch plate and composite skin and allows for fastener flexibility and hole fit. A recent and more detailed analysis is provided in Ref. 18, based on similar principles.

The software developed in Ref. 17 is called BREPAIR and is used to estimate 1) the strain at the edge of the hole and 2) the load distribution in the bolt holes.

Three failure modes are considered: 1) net-tension at the repaired hole, 2) laminate bearing and tension interaction, and 3) fastener shear failure. The allowables for each of these modes were obtained from coupon tests on the composite system under consideration (AS/3501–6).

The net-tension allowable was simply obtained from tests on coupons with open holes of various sizes and measuring the peak strain at the edge of each hole. Failure strains fell below the required 4000 microstrain only for holes larger than 25 mm, suggesting that repairs are not required for smaller holes.

The tension/bearing interaction allowables were based on test results on repaired specimens and the fastener shear allowables obtained from manufacturers' data. The analysis proceeds as follows:

The maximum load carried by the repair joint is estimated from:

$$P = eE_st_sD \tag{10.15}$$

were D is the hole diameter, E_s and t_s are, respectively, the modulus and thickness of the skin, and e the allowable strain, 4000 microstrain in this case.

A first approximation to the number of fasteners n required to carry the load is the greater of:

$$n = \frac{P}{V_{al}} \text{ based on fastener shear allowable } V_{al} \tag{10.16}$$

and

$$n = \frac{P}{(\sigma_{bu} d t_s)} \text{ based on bearing allowable } \sigma_{bu} \tag{10.17}$$

For the given P and n, the program is used to estimate actual faster loads, laminate bearing stress, and laminate strain at the ends of the hole. These are compared with the allowables and the analysis continued until a sufficient margin of safety is achieved, for example, by increasing the number of fasteners or changing the hole pattern. Figure 10.12 shows a typical bolted repair designed using the foregoing procedure and indicates the loads in the critical fastener holes and at the edge of the circular cut-out. An edge distance of $4d$ and fastener separation of $2d$ are used.

10.11 Materials Engineering Aspects

10.11.1 Patch Materials

Titanium alloy is generally used for the metal patch because (unlike aluminum alloy) it does not suffer from galvanic corrosion when in contact with carbon/epoxy. However, aluminum alloy patches can be used if precautions are taken to avoid corrosion by insulating the repair from the carbon/epoxy structure. Aluminum alloy is significantly easier to drill than titanium and therefore better suited to rapid repair of damage where facilities are limited.

Carbon/epoxy is an alternative to titanium alloy for the patch and can be formed by wet lay-up from woven carbon cloth. However, such patches are limited to low-load applications because they have low bearing strength. Composite patches formed from pre-preg provide significantly higher bearing strengths.

Unless the patch is attached to the inner surface to form a flush repair, the holes in the patch must be countersunk and the patches chamfered on their edges to maintain aerodynamic smoothness; generally, the patch must not protrude more than about 4 mm. Countersinking limits the bearing strength because most of the bearing loads are transmitted by the non-countersunk region. To avoid bearing failure caused by insufficient shank in countersunk holes, it is important that the patch be sufficiently thick. The requirement for countersinking is a particular problem for composite patches, therefore these are best used for internal patches where countersinking is not required.

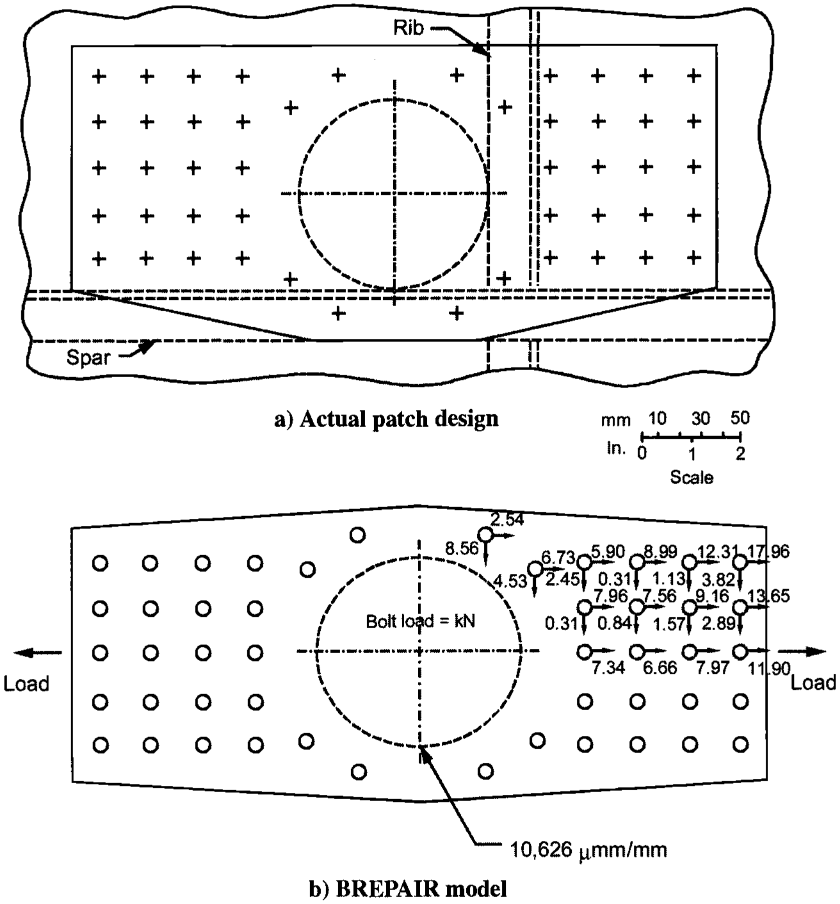

Fig. 10.12 Schematic illustration of *a*) bolted repair near a spar and rib to a thick laminate, and *b*) predictions of loading at the edge of the repair cut-out and in some of the bolt holes made using the BREPAIR program. Taken from Ref. 17.

10.11.2 Patch Installation

Titanium alloy fasteners are generally used for permanent repairs, although corrosion-resistant steel is a suitable alternative. Blind fasteners (applied from only one side) are the most critical because these are subjected to single-shear loading.

The patch is first drilled with the appropriate hole pattern and then used as a template to drill the composite skin. Aluminum alloy patches are initially drilled undersized and then drilled out to the full size when drilling the composite. Because it is much harder than aluminum, titanium is usually initially drilled to final size. Where possible, existing fastener holes are included as part of the hole pattern. To improve the bearing strength of these holes, the countersink can be filled with a metal washer.

Patches are generally wet-installed (sealant at all interfaces), particularly if the repaired structure is a fuel tank. This is a particularly important requirement if an aluminum alloy patch is used to electrically isolate the patch from the composite.

Because access is generally from only one side, blind fasteners are used wherever possible; special blind fasteners have been developed for composite applications. These provide good filling of the bolthole without developing excessive pressure that could split the composite. They are particularly effective if used with a soft-metal sleeve.

Load-transfer efficiency with single-shear fasteners is limited because the fasteners tend to rotate under load, causing edge-bearing failure in the composite skin. Thus, for improved load-transfer efficiency in highly loaded joints, it is usually necessary to use a backing plate to load the fasteners in double shear.

If the bolted patch is attached to the inner surface for a flush repair, the patch plates must be inserted through the hole. This can be a difficult operation and involves using special tools to hold the patch in position on the underside of the component while the blind bolts are inserted.

Hole-drilling in composites can be a problem, even when access is possible from both sides and a backup plate can be used. When a backup plate cannot be used, it is very easy to produce an oversized hole or, when using excessive pressure (generally with a blunt drill), to cause severe delaminations or back-face splintering. Special drill presses have been developed for hand-use that control drill pressure and speed to avoid these problems.

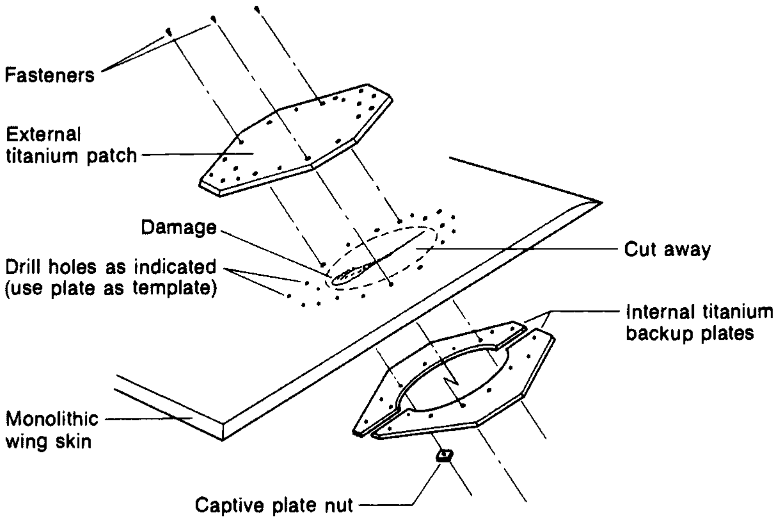

Fig. 10.13 Bolted titanium external patch repair for thick carbon/epoxy laminates, as used for the skins of an aircraft wing.

A typical bolted patch configuration is shown in Figure 10.13. The patch has a chamfered edge to minimize any disturbance of the airflow. The backup plate consists of two sections to allow blind insertion.

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