12
Aircraft Applications and Design Issues

12.1 Overview

This chapter deals with the application of the technologies and materials described in preceding chapters. Its purpose is to highlight the interpretation of the strengths and limitations of polymer composites and to provide some examples of generally accepted design rules and guidelines. Although a vast amount of research has been undertaken on composite materials and structures, much of this has been done by the major aircraft manufacturers and is proprietary. Consequently, design rules vary somewhat from organization to organization, reflecting the different experiences within each. The rules of thumb given here are therefore rudimentary and should be checked with the relevant design authorities before being applied to any particular project.

The chapter also includes some examples of the applications of mainly carbon/epoxy composite structures, and it is hoped that this will show the evolution of their use, which is an inference of the experiences gained by manufacturers. More details on applications can be found in Refs. 1 and 2.

Initially, mention is made of applications with glass-fiber-reinforced polymer laminates, which were the first composite materials used in aircraft structures.

12.2 Applications of Glass-Fiber Composites

Glass-fiber composites were first used during World War II, which was about 20 years before carbon- and boron-fiber composites were used in aircraft structures. The earliest composites were made of E-type glass fabric and polyester resin, and these were used in a few niche components not subject to high loads, such as fuselage-lifting surface attachments or wing and empennage tips. At the time, the aircraft industry was reluctant to use glass-fiber composites more widely because of the low stiffness of glass-fibers and the poor strength and toughness of polyester resins, particularly at elevated temperature. The development of stronger, tougher, and more durable resins, such as epoxies, led to the increased use of E-glass laminates in some aircraft. For example, virtually the entire airframe, wings, and fuselage of modern gliders are built of glass/epoxy.

In the 1960s the development of S2-type glass, which has greater stiffness and strength than E-glass, allowed a greater variety of aircraft structures and components to be made. S-glass composites are often used as the face skins to ultra-light sandwich honeycomb panels, and typical applications in commercial aircraft are wing-fuselage fairings, rudder and elevator surfaces, and the leading and trailing edges of wing panels. Glass/epoxy honeycomb sandwich panels are also used in a variety of components on modern military aircraft, such as the fixed trailing edge on the B-2 bomber. Another common use of composites with E-glass or quartz fiber reinforcement is in radomes on commercial and fighter aircraft, in bay- and wing-mounted radomes on supersonic aircraft and missiles, and in the large radar domes on Airborne Early Warning and Control (AEW&C) military aircraft. This is because of the excellent transparency of glass to radar signals. Glass/epoxy is widely used in helicopter components, such as in the spars to the main and tail rotor blades, fuselage body panels, and flooring. Glass fibers are also used in combination with carbon and Kevlar fibers in hybrid composites for a wide variety of aircraft components, such as wing-body fairings, engine pylon fairings, and engine cowlings. Polyesters have been used in composites for cabin interiors; however, in this application, phenolics are now preferred due to their excellent flame resistance.

12.3 Current Applications

12.3.1 Fixed Wing Civil Applications

As mentioned in Chapter 1, the adoption of composite materials for aircraft structures has been slower than originally foreseen, despite the weight-saving and corrosion and fatigue immunity offered by these materials. The reasons for the restrained use include the high cost of certification and higher materials and production costs for composite components. Composite structures must not be significantly more costly to acquire[3] than those made of aluminum alloy and, to maintain the advantage of weight saving, maintenance costs also, must not be greater.

Sensitivity to impact damage and low through-thickness strength are also inhibiting factors. Other issues are the poor reliability in estimating development costs and difficulty in accurately predicting structural failure.[4]

Although a few inroads have been made in terms of reducing certification costs, recently there has been the development of more cost-efficient manufacturing methods, such as resin-transfer molding and pultrusion, and improved resin and fiber systems that provide increased toughness are making composites very strong candidates for new designs. Another important benefit is the reduction of airframe assembly costs, as composites lend themselves to the manufacture of large unitized structures.

After some years of stagnation, the use of composite materials in large aircraft structures has increased over the past half-decade as manufacturers take advantage of the unique properties of these materials and find solutions to lower the cost of production of composite structures.

As an example, Airbus Industrie has continued to increase applications of composite materials into its new aircraft programs, and in the A380 structure, composite applications amount to approximately 16% of the total airframe weight. Theoretically, this is equivalent to the replacement of about 20% of conventional aluminum structure by composites. Large commercial transport aircraft designs had, in the past, tended to limit the use of composite materials to secondary structures—ailerons, flaps, elevators and rudders—although Boeing has used the material on the tailplane and floor beams of the B777 and Airbus on the empennages of most of its fleet. More recently, several commercial airliner manufacturers have been considering and choosing composite materials for other primary structures.

The Airbus A380 will employ carbon-fiber-reinforced plastic composite materials in the massive ($7 \times 8 \times 2.4$ m) wing carry-through structure; inside the cabin the upper floor beams are pultruded 7-m long, 0.3-m deep sections. Resin infusion is used to form the rear pressure bulkhead and several of the wing panels. Leading edges will be thermoplastic to obtain improved impact resistance.

The upper fuselage skin panels (over 400 m^2) will be manufactured from a hybrid metal and fiberglass laminate, Glare; this material is discussed briefly in Chapter 1. Figure 12.1 illustrates the material used on the A380 as projected in the advanced development stage of the project. The carry-through structure represents probably the largest, most complex, and critical aerospace composite structure yet attempted in civil aircraft applications.

As technologies in both composite structures and aluminum structures advance, and with service experience, preferred options will change over time.

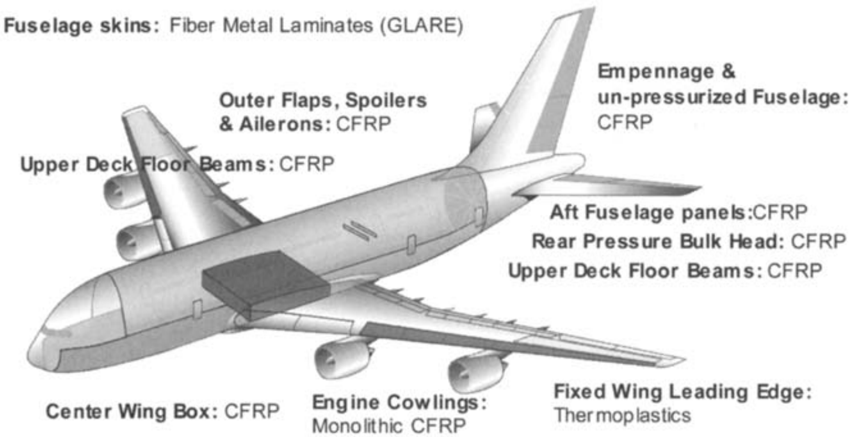

Fig. 12.1 Advanced composite materials selected for the A380. Courtesy of Airbus Industries.

Figure 12.2 shows the fluctuations of structural design selection for a number of Airbus products.

Of the smaller transport and general category aircraft, the Beechcraft Starship was the first all-composite aircraft certificated to FAR 25. Later, Raytheon products, the Premier 1 and the Horizon corporate jets, have reverted to metal wings for cost reasons; however, the fuselages remain as composite structures. In addition, new, automated methods of production are employable on surfaces of revolution. Figure 12.3 shows the Premier 1 fuselage being produced using a tow-placement process.

An attraction for the smaller fabricators is the ability to produce aerodynamically smooth surfaces with relatively low tooling costs, and many high-performance homebuilt aircraft use composite materials almost exclusively. With the drive toward lower-cost carbon fiber, promoted, in part, by the interest in the automotive industry, the use of these materials is sure to expand further.

12.3.2 Fixed-Wing Military Applications

Up to 70% of the airframe weights of some modern military airframes are manufactured from composite materials. This is due in part to the pursuit of ultimate performance, with less emphasis on cost, but also to the low radar signature obtainable through use of these materials. Perhaps the most ambitious example of the use of composites is the USAF B-2 bomber,[4] which is an almost all-composite structure. The wing, which is almost as large as that of a B-747, is

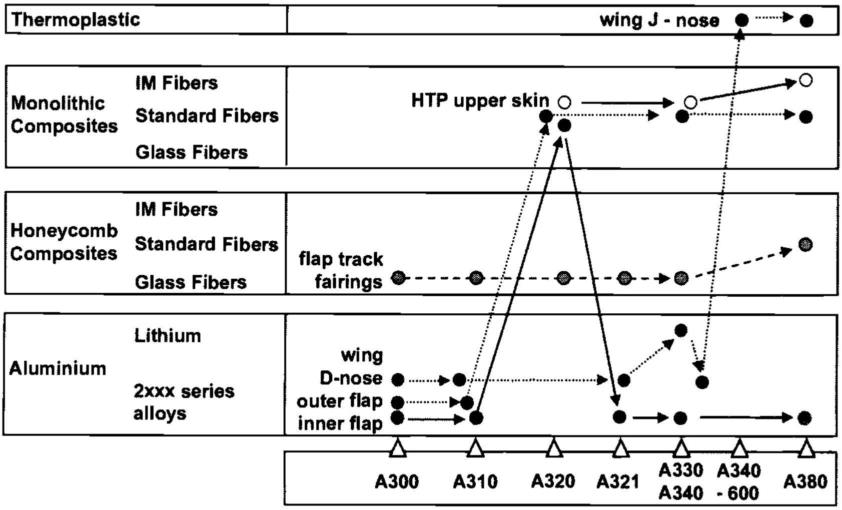

Fig. 12.2 Evolution of Airbus materials selections. Courtesy of Airbus Industries.

Fig. 12.3 Premier 1 composite fuselage. Courtesy of the Raytheon Corporation.

mostly made of carbon/epoxy, with honeycomb skins and internal structure. The fuselage makes extensive use of composites. However, this form of construction is very costly and more recently, affordability is considered to be as important as performance and is now a major design parameter.

The need for high stiffness to minimize the depth of wings and tail in high-performance military aircraft both for aeroelastic and stealth reasons ensures that all future aircraft will have composite wing and empennage skins. The requirement for stealth as well as minimum weight also ensures that most of the fuselage skin will be composite. For radar absorption, leading edges will be made of honeycomb structure with outer composite skins based on non-conducting fibers such as quartz rather than carbon in the rest of the structure. This skin material allows the radar waves to penetrate into the honeycomb core coated with radar-absorbing material, rather than being reflected back to the receiver.

Despite the structural advantages of honeycomb construction, there is a trend to replace this form of construction with stiffened cocured composite panels, because these are much less prone to damage and to water entrapment. Honeycomb is still used in some regions for stealthy structure as discussed previously and where structurally advantageous, for example, in control surfaces.

Some military aircraft such as the Harrier, have much of the internal structure of the wing made of carbon fiber reinforced plastic composite, in addition to wing skins, some in the form of sine wave spars (see Figure 12.4). However, in more recent fighter aircraft there is a trend back to metals for much of the wing substructure. This is because of the relatively high cost of composite substructure, compared to high-speed machined aluminum and the limited tolerance of composites to ballistic impact. Often titanium alloy is used for the main load-bearing spar because of its superior resistance to ballistic impact and its excellent fatigue properties.

The airframe of the F-22, as an example,[4] is made of 39% Ti 64 titanium, 16% aluminum alloy, 6% steel, 24% thermoset composite – carbon/epoxy and carbon/BMI and 1% thermoplastic. The structure is given as follows.

- Forward Fuselage:
  - skins and chine – composite laminates
  - bulkhead/frames resin transfer moulded composite and aluminium
  - fuel tank frames and walls- RTM composite
  - side array doors and avionics – formed thermoplastic
- Mid Fuselage:
  - skins – composite and titanium
  - bulkhead and frames – titanium aluminum and composite
  - weapon-bay doors -skin thermoplastic, hat stiffeners, RTM composite
- Aft Fuselage:
  - forward boom- welded titanium
  - bulkhead and frames- titanium
  - keel web-composite
- Wings:
  - skins composite
  - spars - front titanium, intermediate and rear- RTM composite and titanium
  - side of body fitting HIP cast titanium
- Empennage:
  - skin composite
  - core – aluminium
  - spars and ribs-RTM composite
- Duct Skins:
  - Composite
- Landing Gear
  - steel

Some general details of the construction of some current fighter are provided in Figure 1.2 for the F/18EF and in Figure 12.5 for the Joint Strike Fighter. In the JSF extensive use was planned (at the time of writing) of the lightweight aluminum lithium alloy for the wing and other substructure.

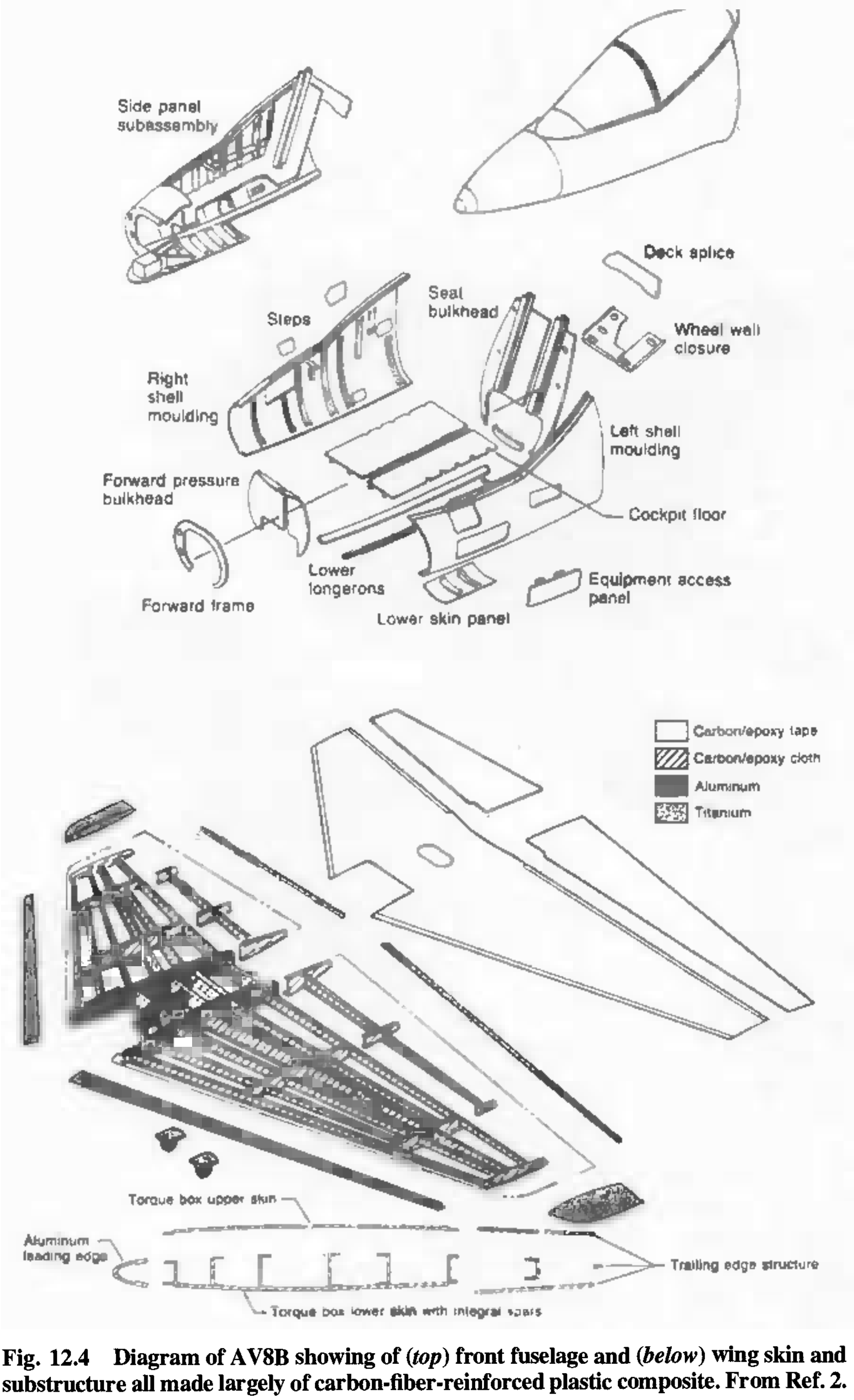

Fig. 12.4 Diagram of AV8B showing of (*top*) front fuselage and (*below*) wing skin and substructure all made largely of carbon-fiber-reinforced plastic composite. From Ref. 2.

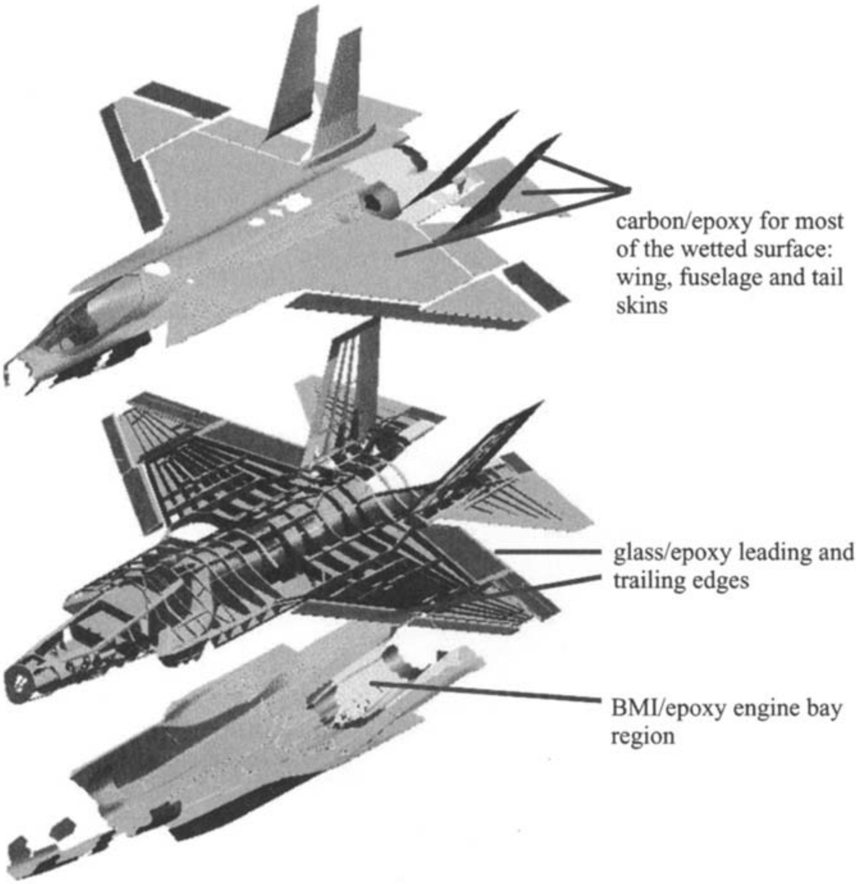

Fig. 12.5 Joint strike fighter showing extensive planned use of composite in the skins of the aircraft but use of aluminum alloy for much of the substructure.

Many future fighter and attack aircraft will be unmanned. Here the emphasis is on very high g manoeuvres to evade missiles in high-threat regions, two or three times the 7–9 g allowed in manned aircraft. Only all-composite construction could be considered for the structure in such situations and design will be based on very high strain allowables.

Both manned and unmanned aircraft will in future aircraft be continuously monitored using embedded sensors (see Smart Materials Chapter 15). These sensors will provide information on the stress, strain, temperature and any damage experienced by the structure and may also provide an indication of absorbed moisture.

12.3.3 Rotorcraft Applications

The early applications for composite materials in helicopters were in rotor blades and drive shafts. The attraction for rotor blades is the ability to produce complex aerofoil shapes and high-quality surface contours using simple construction methods. Fiberglass has often been used because the stiffness of the blades is not usually a design problem, the predominant load being tension caused by centrifugal forces. Use of composites in drive shafts is attractive but the opposite reason applies and here, torsional stiffness is an imperative and carbon fiber reinforced plastic composites offer a significant weight saving. Filament winding is an attractive manufacturing process for these components particularly for drive shafts where there is a need for ply orientations at $+/-45^\circ$ for maximum torsional efficiency.

Composite materials are now used for flex-beams in the design of 'bearingless rotor hubs' that are now becoming universally adopted. Composites allow flexural stiffnesses to be tailored into the otherwise rigid beam allowing the necessary blade flapping action arising from forward flight. Pitch cases, that transmit the pitch angle to the blade, have similar requirements to drive shafts and are also being constructed from carbon fiber reinforced plastic composite.

Over recent years, the use of carbon fiber shell structures for fuselages and tailbooms has also been spreading. The MD Explorer employs carbon fiber reinforced plastic composite for almost 100% of the non-transparent external structure (see Figure 12.6). The US ACAPS helicopter crashworthiness assessment program run in the 1980s showed the advantages of using composites in the tub structures for energy absorption under crash-landing conditions. Composite structures when designed properly have a significantly better specific energy absorbing capacity than aluminum alloy structures under crushing conditions.

The V-22 tiltrotor is an excellent example of the beneficial use of carbon/epoxy composite construction.[4] Use of composites is credited with saving 13% structural weight and reducing costs by 22%. However, to save cost, even in this highly weight-sensitive application, some of the internal fuselage structure, originally planned to be made of composite, is now made of aluminum alloy.

12.3.4 Common Configurations

Table 5.1 lists the various types of composite construction used in aircraft structure. Early composite designs tended to be of sandwich construction, featuring honeycomb cores. This construction is highly efficient structurally and, provided the core is relatively shallow, also quite cheap to manufacture. Unfortunately there have been many examples of disbonding in service. This problem in honeycomb structure is common to both metallic- and composite-skinned construction and mostly results from the ingress of moisture into the core

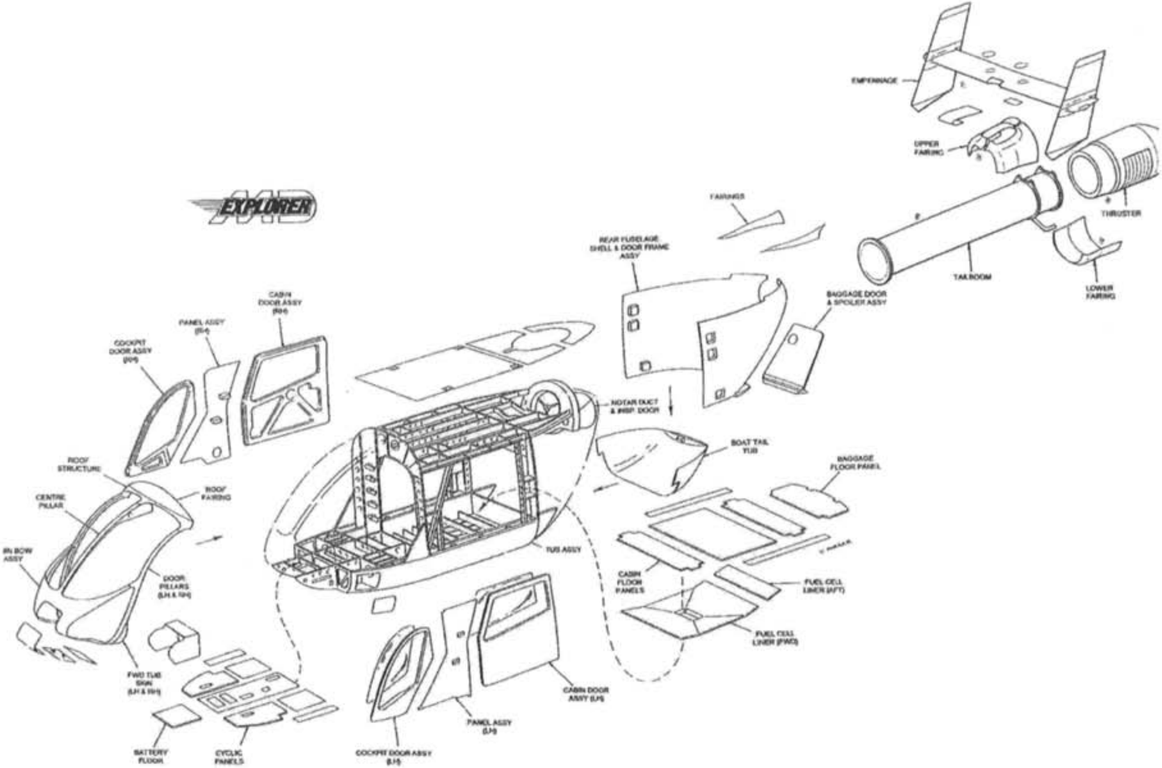

Fig. 12.6 Composite components on the MD Explorer fuselage. Courtesy of Hawker de Havilland Ltd.

through poorly sealed ends during ground-to-air pressure changes. It can also be a problem with thin composite skins, which can allow moisture to penetrate through microcracks. However, transport of moisture through the composite skins by diffusion does not seem to be a problem. Moisture penetration is particularly serious when the core is made of aluminum alloy because corrosion and bond separation result. Ingress of moisture can also cause de-bonding of the skins caused by expansion of entrapped moisture on freezing when operating at altitude.

A good design practice with honeycomb panels is to envelop the sandwich in a thermoplastic film such as Tedlar, which acts as a moisture barrier, that can be cobonded with the laminate. Cuts and darts in this film should be avoided, otherwise moisture can penetrate to the composite surfaces where it can then be absorbed into the substrate. Skin thicknesses should also not be less than 0.6 mm for the same reason. The use of an appropriate sealant must be applied to all cut edges.

Honeycomb-sandwich structures are also more prone to impact damage, and for these reasons, although accepted for secondary structures, some aircraft companies will not sanction the use of sandwich construction in primary structures. Closed-cell rigid foam cores are possible substitutes; however, the low melting temperature of PVC foams restrict its use to lower-temperature-curing (and hence lower-performing) systems. Higher-temperature-curing foams such as PEI may overcome this problem; however, some observations of the material cracking under cyclic strains have been reported, and care must be taken to ensure that the foam is completely dried before processing.

The alternative is a stiffened monolithic construction, and here the main issue is the means of attachment of the stiffeners. Some alternatives for attaching stiffeners are shown in Figure 12.7.

Although honeycomb construction is generally lighter than stiffened structure, this situation is reversed if the structure is allowed to buckle at limit-load. Compared with unbuckled stiffened structure, honeycomb saves approximately 20% weight, but post-buckled structure[5] can save approximately 30%.

From a structural point of view, the integral cocured design is the most effective solution, particularly if the stiffeners must endure buckling of the skins without disbonding; however, lay-up costs are higher. To some extent, this cost may be offset by the reduction in parts count. With bonded discrete stiffeners (although cheaper to manufacture), care needs to be taken in matching the stiffness of the panel with the attaching flange, and avoiding excessive through-thickness stresses to avoid the possibility of peel failures. Thorough surface preparation is also essential to ensure a good bond.

Conventional mechanical fastening can be used with bonding as a conservative solution to improve through-thickness strength. Alternatively, z-pinning (Chapter 14) is a novel method in which small-diameter composite pins are inserted through the thickness of the laminate for the same purpose.

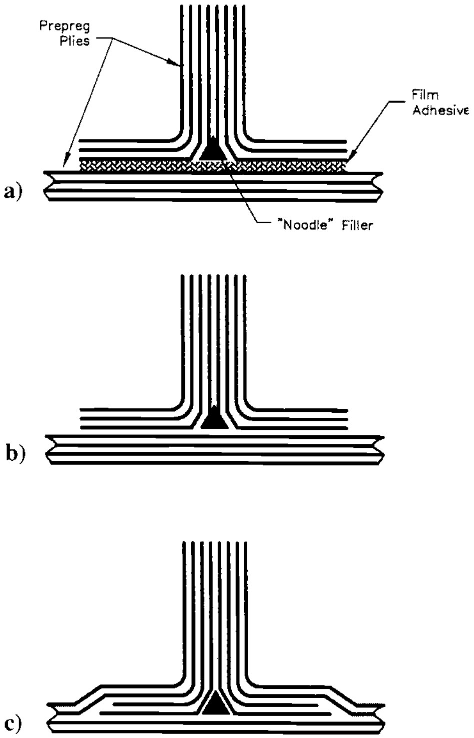

Fig. 12.7 ***a*) Secondary bonded blade stiffener; *b*) cobonded blade stiffener; *c*) integrally cured blade stiffener.**

A major NASA initiative in recent years has been to develop a cost-effective stitched stiffener wing plank using dry preforms and a resin film infusion process[6] (see also Chapter 14).

The design of composite details has to be made with a clear view of the proposed assembly procedure. A major advantage of using composite materials is the possibility of reducing parts count by making very large components (and hence subsequent joining assembly costs and problems). "Single-shot" structures, such as has been achieved on the Boeing/NASA wing plank, in which components that would previously have been individually manufactured and assembled are molded in a single operation, are becoming the goal for many designers. Although this approach brings assembly savings, the additional complexities in NDI also needs to be considered.

Another key attraction of designing with composite materials is the opportunity to tailor the design through orientation of the fiber in the direction of the load. Although it is possible to optimize structural performance through fiber alignment and by providing ply buildups at load concentration points, the value of these measures must always be considered against the increased manufacturing and certification costs. Many designs have reverted to quasi isotropic lay-ups with the aim of reducing costs. The advantages of near quasi isotropic lay-ups for optimizing strength in mechanically fastened components are discussed in Chapter 9.

12.4 Design Considerations

12.4.1 Choice of Materials

There are wide ranges of choice for both the reinforcement and the resin materials of the composite. This subject is covered more fully in Chapter 8. A summary is provided here. The most common combination for aerospace applications is an epoxy resin with carbon-fiber tape or fabric reinforcement, although BMI resins are used for high-temperature applications. In addition to these two thermosetting resins, there have been some successful applications of thermoplastic matrices such as PEEK, PEI, and PPS. Parts manufactured from thermoplastics are usually used in the smaller details due to the high forming pressures required that limit the size of part that can be formed in a conventional press. These can then be welded together to form larger components, a process not possible with thermosetting details. Thermoplastic resins were seen as attractive in the past due to their higher toughness and consequent improved resistance to impact damage. However, this advantage has been eroded somewhat by later-generation toughened epoxies. In addition, some thermoplastic resin matrices lose some toughness under some in-service conditions. PEI, in particular, has been shown to embrittle when exposed to prolonged high temperatures and furthermore is susceptible to attack from various chemicals occurring in standard aircraft fluids.

The costs of high-temperature thermoplastic materials are also considerably greater than those of competing thermosetting materials, as are the processing costs. As a result, thermoplastic systems have not been widely adopted in aerospace structures at this point.

Composite materials are usually supplied with the reinforcement pre-impregnated with resin (pre-pregs) or, less frequently, separately. In the latter case, the resin is introduced after the dry reinforcement has been placed into a mold using some form of liquid-molding process. Details of these processes are covered in Chapter 5.

Because of the high cost of material qualification, aerospace companies are typically conservative when choosing materials and tend to select early-generation materials rather than those with improved properties to avoid additional material qualification costs. Unfortunately, there are no common materials data shared between users and, in many cases, a single material is qualified to similar requirements for several different customers. An attempt is currently being made through MIL-HDBK 17 to deliver sets of properties for standard materials; however, this is as yet not comprehensive.

Carbon fiber is by far the most commonly used reinforcement material for aerospace composites. Boron fiber continues to be used for some older applications, particularly in the United States (e.g., the F-15); however, its high cost and the difficulty of processing into convenient reinforcement forms (e.g., woven and braided fabrics) and the difficulty of drilling or machining has very severely limited its application. Kevlar aramid fiber from DuPont had found some early applications, however, the limited compression strength of Kevlar composites and its tendency to absorb high proportions of moisture have led to a declining interest. It is now only mainly used for applications in which high-energy impact containment is required. The properties of composites based on these fibers are discussed in Chapter 8.

Reinforcements can be provided in a variety of woven or braid styles as well as in unidirectional plies. The latter provide the highest in-plane mechanical performance (stiffness and strength) due to the straightness and uniformity of the tows. Most weaves and all braids have "crimped" tows that reduce in-plane properties. This is particularly the case for compression strength that is very sensitive to fiber straightness. Nevertheless, braids provide a more convenient form for parts to be laid-up by hand.

Non-crimp weaves in which layers are stitched together into a carpet give properties somewhere between unidirectional and woven reinforcement, because the fibers are not held as straight as unidirectional tows. Non-crimps are highly drapeable and provide considerable advantage by reducing the number of individual plies to be laid. Currently they are not available as a pre-preg material and must be processed using liquid-molding techniques. Chapter 14 describes these forms in more detail.

12.4.2 General Guidelines

Composite structural design should not be attempted without a good working knowledge of the manufacturing limitations applying to composite materials. Generally, concurrent engineering is practiced whereby designers and manufacturing engineers work toward solutions that satisfy both design intent and production needs.

When specifying lay-ups (laminate ply stacks) and design details, some basic guidelines should be followed:

- Use balanced laminates to avoid warping
- Use manufacturing techniques that produce a minimum fiber content of 55% by volume;
- Use a minimum of 10% of plies in each of the principal directions (0°, 90°, $\pm$ 45°) to provide a minimum acceptable strength in all directions
- Use a maximum of four adjacent plies in any one direction to avoid splitting on contraction from cure temperature or under load

- Place $\pm 45^\circ$ plies on the outside surfaces of shear panels to increase resistance to buckling
- Avoid highly directional laminates in regions around holes or notches because stress concentration factors are significantly higher in this ply lay-up
- Add ply of woven fiberglass barrier between carbon and aluminum alloy for galvanic protection
- Drop plies where required progressively in steps with at least 6 mm (0.25 in) landing to improve load redistribution
- Where possible, cover ply drops with a continuous ply to prevent end-of-ply delamination
- Maintain three-dimensional edge distance and four-dimensional pitch for mechanical fasteners to maximize bearing strength
- Where feasible, avoid honeycomb in favor of stiffened construction, because honeycomb is prone to moisture intrusion and is easily damaged
- Avoid manufacturing techniques that result in poor fiber alignment, because wavy fibers results in reduced stiffness and compression strength
- Minimize the number of joints by designing large components or sections because joints reduce strength and increase weight and cost
- Allow for impact type damage (see later discussion); this may vary with risk (e.g., upper horizontal surfaces are at greatest risk).
- Exploit the non-isotropic properties of the material, where feasible.
- Ensure that the design reflects the limitations of the manufacturing processes to be used.
- Predict the failure loads and modes for comparison with test data
- Minimize or exclude the features that expose the notch-sensitivity of the material.
- Allow for degradation due to the environment.
- Provide for ready inspection of production defects.
- Allow for repair in the design.
- Predict and minimize, by design, out-of-plane loading.
- Include consideration of residual stresses in the cured laminate when calculating strength.

12.5 Design of Carbon-Fiber–Based Components

12.5.1 Static Strength

Carbon/epoxy in conventional ply configurations generally has significantly higher static strength than aluminum alloys. However, because of the brittle nature of the fibers, the composites are essentially elastic materials with very limited ability to redistribute loads at structural features such as fastener holes.[7] The result is that they are quite notch-sensitive under static loading. As may be expected, the higher the fiber modulus, the higher the notch-sensitivity, because the stiffer fibers have a reduced ability to accommodate high local strains. By contrast, aluminum alloys (and other structural metals) can redistribute stresses at mild stress concentrators by local yielding, so strength loss is often simply due to the reduction in net section.

The performance of laminates in the vicinity of holes and joints is highly affected by the lay-up.[8]

Figure 12.8 shows the variation in stress concentration at the edge of a circular hole with ply lay-up. This shows that the estimated stress concentration factor increases with the proportion of fibers oriented in the load direction (e.g., if there are no $\pm 45°$ fibers and 100% 0° fibers, $K_t = 8$). Composites also have relatively low bearing strengths and quasi isotropic laminates are preferred in the area of bolted joints to ensure that there is at least some 0° fibers support the bearing loads regardless of load direction.

For these reasons, bonded joints are a better structural solution for composites; however, there are issues of maintenance and assurance of adequate bonded joint quality that must be taken into consideration. Also, bonded joints in thick section composites are complex and costly to manufacture. Joints are an extremely important design consideration, and Chapter 9 is devoted to this topic.

It is important to note that prior cyclic loading markedly reduces notch sensitivity of the composites by the formation of microcracks in the matrix and micro-delaminations between plies in regions of high initial stress concentration.

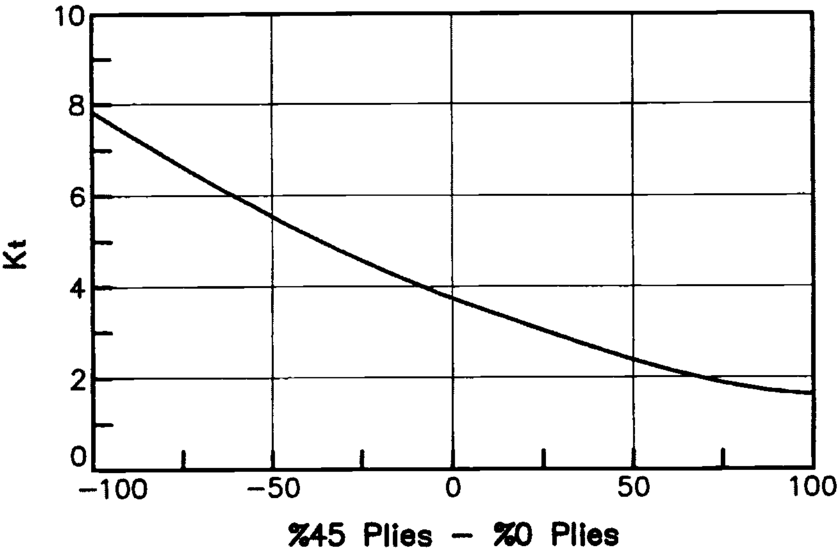

Fig. 12.8 Stress concentration factors in laminates with varying proportions of on- and off-axis plies. Based on Ref. 8.

However, this reduction may not be allowed for in assessment of static strength for certification purposes.

12.5.2 Through-Thickness Strength

The foregoing comments refer to in-plane strength properties for typical two-dimensional reinforcement. Through-thickness (or z-direction) strength is about an order of magnitude lower than that of metals (Fig. 12.9), limiting application of laminated composites to two-dimensional loading situations. It should be realized that even two-dimensional loading can result in through-thickness or peel stresses at ply drop-offs, stiffener run-outs, or edges.

Particular care is required when designing curved sections as interlaminar tension stresses that arise will often result in unexpected failure. Some examples of these situations[9] are shown in Figure 12.10.

The following simple equation may be used for approximating the through-thickness stresses in curved sections under bending.

Maximum radial (interlaminar tension) stress:

$$\sigma_r(\max) = 3M/2t(R_iR_o)^{1/2} \tag{12.1}$$

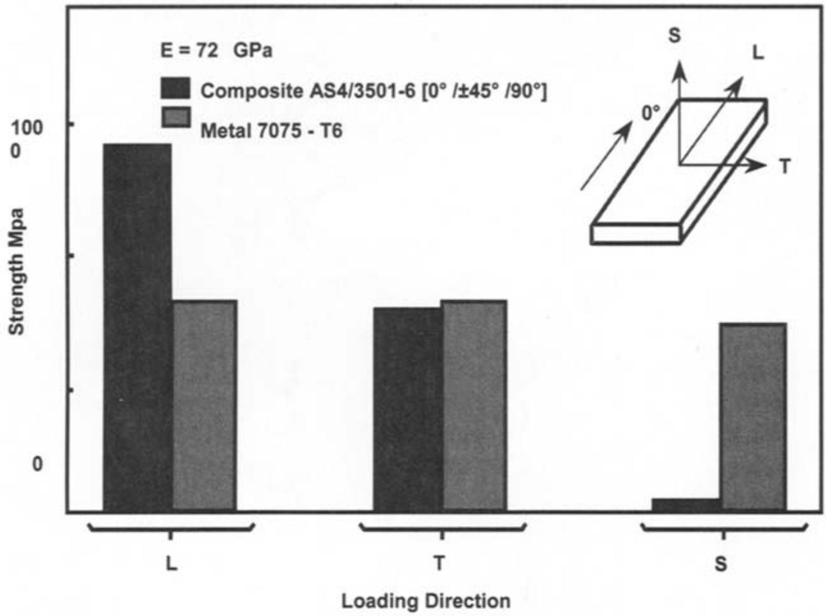

Fig. 12.9 Comparisons of strength of aluminum alloy and carbon/epoxy laminates in various loading directions. Note the very low through-thickness strength of the composite. Adapted from Ref. 7.

In addition, a temperature change in a cured curved laminate such as the drop from cure temperature to room temperature will result in the following distortion and residual radial stress:

$$\gamma = (\alpha_\theta - \alpha_r)\Delta T\pi/2 \tag{12.2}$$

and

$$\sigma_r(\max) \sim (t/R_m)^2(\alpha_\theta - \alpha_r)\Delta T\ E_\theta/R_m \tag{12.3}$$

where:

M = applied moment
R_i = inner radius
R_o = outer radius
R_m = mean radius
t = thickness
γ = springback in degrees
α_θ = circumferential coefficient of thermal expansion
α_r = radial coefficient of thermal expansion
ΔT = temperature change
E_θ = circumferential modulus of elasticity

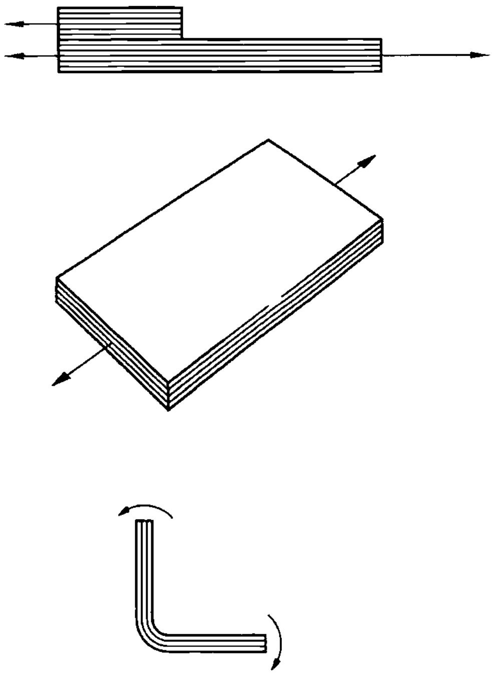

Fig. 12.10 Sources of delamination. Based on Ref. 9.

More accurate results can be obtained from finite element analyses; however, whichever method is used to calculate induced stresses, the actual failure stresses need to be established through a calibrated test.

Joints, tapers, and ply drop-offs also give rise to significant through-thickness or peel stresses that can result in the formation of delaminations. The development of unexpected or higher than expected through-thickness stresses are major reasons for the formation of delaminations in large components. In many cases, these problems only arise when full-scale or large components are tested, or even in service, because they are often not detected at the coupon or structural element scale.

Detailed two-dimensional or three-dimensional finite-element analysis is used to determine the state of stress in complex full-scale components. However, modelling at the ply level can be prohibitively time-consuming and in any case may not correctly represent the "as fabricated" component.

12.5.3 Manufacturing Defects

The mechanical properties of composite structures are influenced by the presence of defects in the material arising from inconsistencies in manufacturing processes and controls. Typical defects include resin-rich or resin-dry areas, fiber misalignment, porosity, delaminations, and the inclusion of foreign materials, such as peel ply.

Most aircraft parts are inspected using automated equipment, set to scan the work at a discrete interval. A defect smaller than the interval may not be detected. On large parts, the interval is often set at approximately 6 mm, consequently defects smaller than 6 mm diameter may be missed on successive passes.

Other forms of defect can be inadvertently introduced at the assembly stage. Exit-side fiber damage and delamination can occur on drilled holes, for example, particularly if insufficient support is provided. The extensive use of composite materials in recent years and the development of drill bit technology has minimized these effects; however, it is important to ensure that test specimens used to obtain design allowables are representative of the accepted production practice. Handling damage and damage due to excessive force fit are also possible during the assembly stage.

Typical manufacturing defects must be allowed for in design, but allowance for impact damage as described in the next section will usually cover this requirement.

12.5.4 Impact Damage

Impact damage in composite airframe components is usually the main preoccupation of designers and airworthiness regulators. This is in part due to the extreme sensitivity of these materials to quite modest levels of impact, even when the damage is almost visually undetectable. Chapter 8 describes the mechanisms involved in impact damage and also provides more background on the influence of mechanical damage on residual strength.

Horizontal, upwardly facing surfaces are obviously the most prone to hail damage and should be designed to be at least resistant to impacts of around 1.7 J. The value represents the energy level generally accepted to represent extreme value in (1% probability of being exceeded) hail conditions.[10]

Surfaces exposed to maintenance work are generally designed to be tolerant to impacts resulting from tool drops.[11] Figure 12.11 provides impact energy levels for a variety of different tool-drops, and Figure 12.12 indicates that monolithic laminates are more damage resistant than honeycomb structures. This is due to their increased compliance. However, if the impact occurs over a hard point such as above a stiffener or frame, the damage may be more severe, and if the joint is bonded, the formation of a disbond is possible.

12.5.4.1 BVID, VID, and Energy Cut-off Levels. The authorities have generally divided impact damage into two categories. The categories are

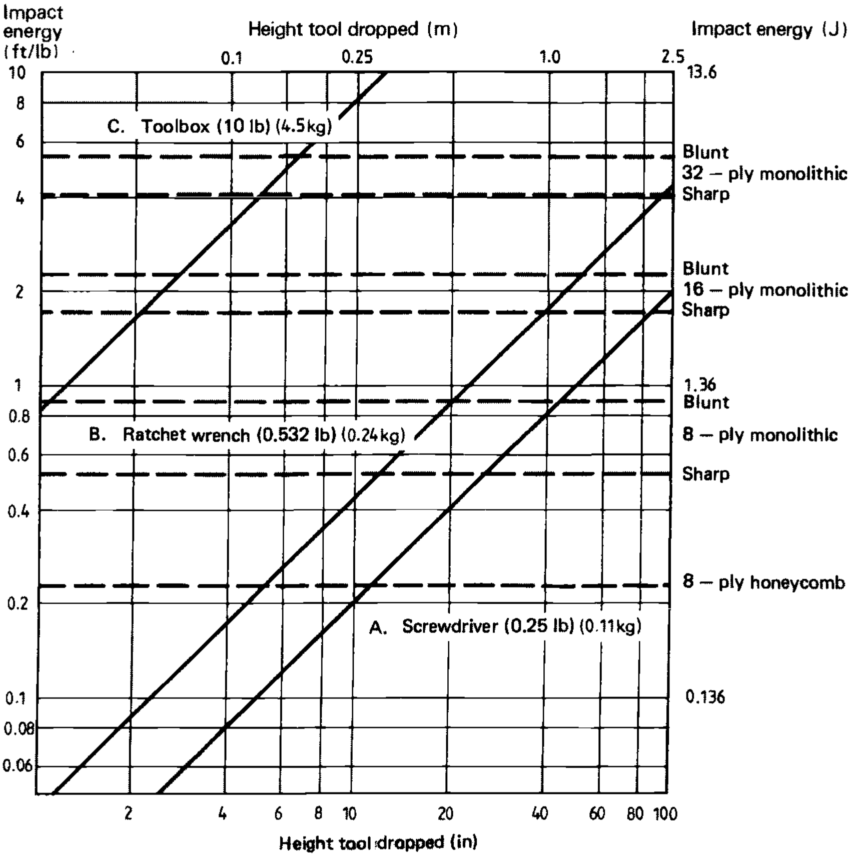

Fig. 12.11 Impact energy of dropped tools. Based on Ref. 11.

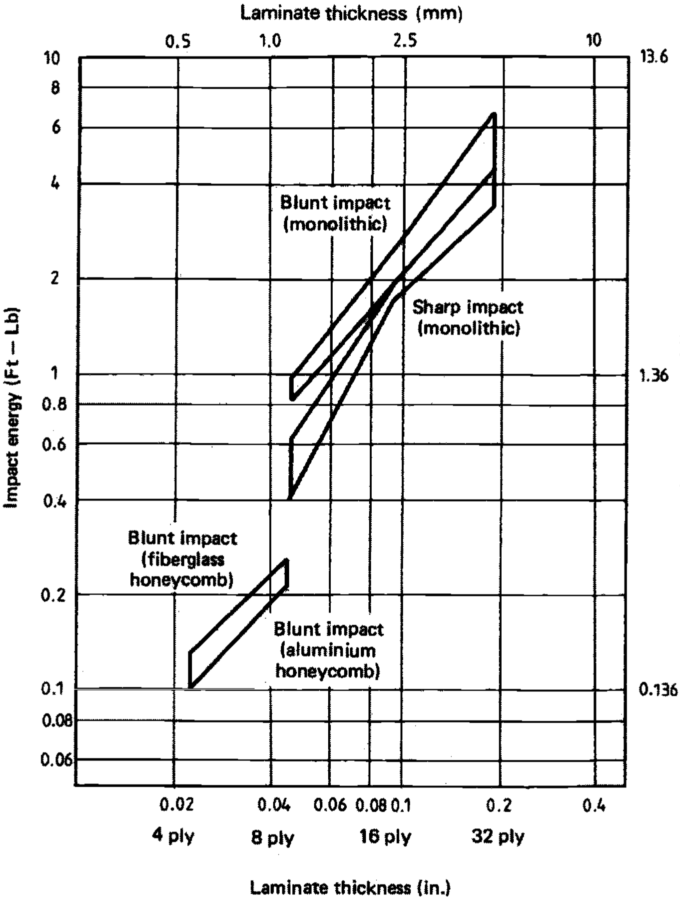

Fig. 12.12 Impact energy for incipient damage to carbon/epoxy laminates.

delineated by the ease of visibility (by the naked eye) of the damage rather than the energy of the impact: barely visible impact damage (BVID) and visible impact damage (VID). The definition of visibility is difficult to quantify because it depends on access, light conditions, and differences in human capability. Damage to an external surface could be expected to be more readily detectable; however, because it can be masked by paint. Quite often, backside damage fiber-break is more apparent than the corresponding impression on the impacted face. For airworthiness certification, the structure is expected to demonstrate an acceptable strength margin with BVID because this may not be detected for some time. It is not usually the surface condition that promotes a subsequent static failure, but more the associated underlying delaminations.

There is no current universally accepted definition of the term *barely visible*. Some authorities accept surface indentations of 1 mm; others give more qualitative requirements, for example, that an indication be observable from a given distance (say, 1 m). It is invariably agreed that structures must be able to sustain ultimate load with this level of impact damage present in the structure and that it be able to withstand limit-load with damage that is clearly visible.

The USAF has accepted an upper threshold of impact energy of 100 ft lbs (around 135 J) as equal to a dropped tool box—a once-in-a-lifetime expected event. Consequently, if the structure is capable of withstanding this without reduction in strength below an acceptable margin, the above criteria are not imposed. Figure 12.13 illustrates the situation. Other authorities such as the Joint Airworthiness Authority (JAA) have nominated 50 J as the energy cut-off.

12.5.5 Residual Strength

As noted, the compression strength of a composite laminate is substantially reduced subsequent to an impact event causing visible or even non-visible damage. For example, with laminates less than 3 mm thick, typical of control surface structures, compression strength can be reduced by more than 50% with BVID (Fig. 12.14). These reduction factors are often established at the coupon level through a standard compression-after-impact test, as discussed in Chapter 7. These tests generally involve impacting the test coupon with a specified energy level rather than specifying a degree of damage and were initially devised to provide a means by which different materials could be compared. They have, however, been widely adopted to establish allowable values for design.

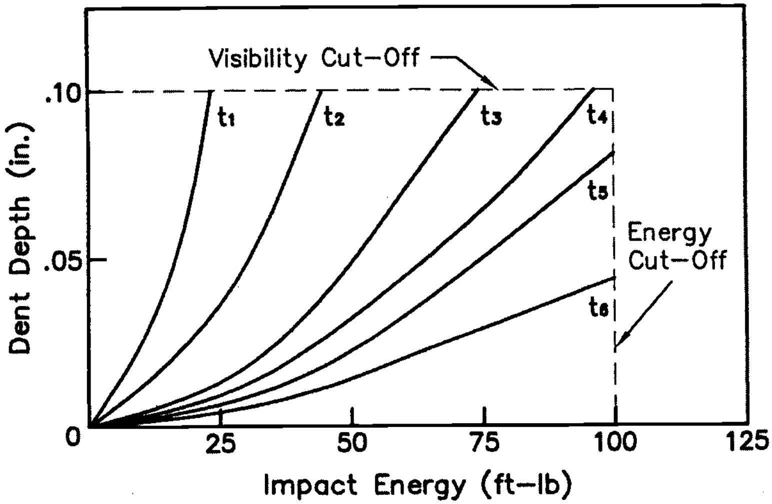

Fig. 12.13 Impact damage assumptions. The symbols t_1, t_2, etc. indicate increasing laminate thickness. Adapted from Ref. 7.

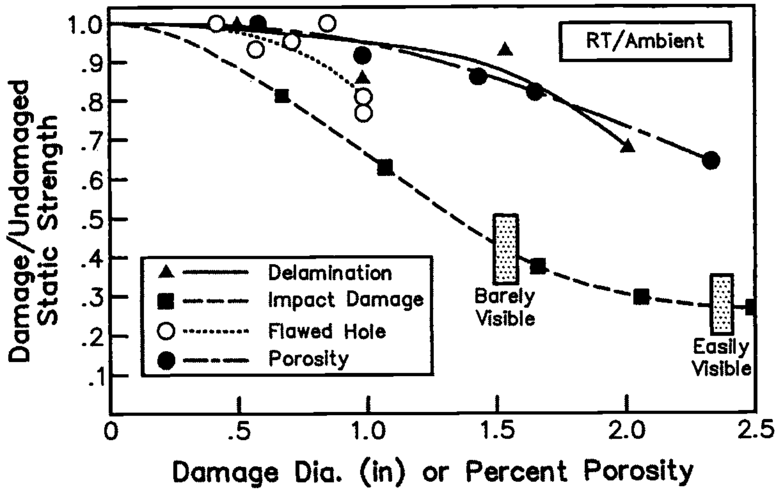

Fig. 12.14 Strength loss associated with impact damage.

In contrast, the residual strength after impact damage under tension is not usually considered as significant as other geometric characteristics, for example, fastener holes and notches, which are more critical. The case of the pressurized fuselage is an exception in which fail safety must be demonstrated in the presence of significant damage.[11] In such cases, the nature and size of damage is prescribed often following similar patterns to those known to occur in metal structures. Residual strength is usually then demonstrated by tests on full-scale subcomponents rather than by predictions from coupon data.

Horton et al.[13] provide more information on the subject of damage tolerance of composite laminates.

As discussed in Chapter 8, modelling tools for post-impact strength[14] are not sufficiently mature to be relied on, and certification is usually based on demonstrating (by test) that strain levels are sufficiently low and that failure will not occur even if damage is present. Thus residual strength tests, after impact (and other representative damage) are often performed at the various scale levels,[15] including full scale after conclusion of the fatigue test program. Residual strength testing may follow some further representative cyclic loading to check for damage growth.

When quantifying residual strength after impact, it is preferable to work in terms of strain, because the stiffness of the laminate does not then need to be considered. The allowable ultimate compressive strain with BVID is not much less than the ultimate strength of an undamaged laminate in the region of a 6-mm hole, and this latter allowable is sometimes used to cover both circumstances.

12.5.6 Damage Growth Prediction

As noted in Chapter 8, prediction of damage growth in composite laminates under cyclic loading is not straightforward. Consequently, design is generally based on a safe life with BVID damage assumed; in other words, there is no damage growth allowed under cyclic loading. Inspection intervals are set based on a demonstrated safe or no-growth life, suitably factored to allow for statistical variability.

12.5.7 Bird Strikes

Bird strikes are special cases, for example, in composite fan blades and leading edges, where it must be demonstrated that in the event of such an impact, safe continued flight and landing will not be impaired. As with metal structures (that must meet the same requirements) the issue is as much one of protection of systems behind the impact zone as of structural damage.

12.5.8 Damage Tolerance Improvements

Various methods can be considered to enhance the damage tolerance of composite materials. Some of these methods are discussed in the following paragraphs.

The ability of the composite structures to tolerate impact damage is largely dependent on the fiber and matrix properties. The increase in matrix material fracture toughness greatly enhances the damage tolerance of the composites. Published data[13] indicates that the residual compressive strength of composites after impact is directly proportional to the mode 1 strain energy release rate, G_{IC}. In tests on the same reinforcement with different resins, a matrix (resin) with twice the value of G_{IC} showed a 50% improvement in residual strength after impact when compared with the base system.

The use of a tougher resin system or thermoplastic significantly improves damage tolerance. For example, G_{IC} of a typical thermoplastic material is approximately 1050 J m^{-2} compared with 180 J m^{-2} for an un-toughened epoxy material.

There are two distinctly different issues in relation to the influence of matrix toughness on impact damage: resistance to damage and residual strength in the presence of damage. Generally, composites with tough matrices are resistant to delamination damage, as measured by delamination size for given impact conditions. However, for a given area of impact damage, both brittle and tough composites suffer about the same degradation in residual strength.

Fiber properties significantly influence damage tolerance: the stiffer the fiber, the less damage tolerant it will be. Composites with hybrid fiber construction [that is, where some percentage of the carbon fibers are replaced by fibers with higher elongation-to-failure ratios, such as E-glass or aramid (Kevlar)] have been shown to have improved compression and tension strengths after impact. However, their basic undamaged properties, that is, strength and stiffness, are usually reduced.

Impacted laminates with higher percentages of plies oriented in the loading direction typically fail at lower strains than laminates with more off-axis plies. This is demonstrated in the case of open-hole strengths (Fig. 12.15). This shows typically how laminate strain-to-failure varies with lay-up and load orientation. Open-hole compression (OHC) and filled-hole tension (FHT) values are plotted against the percentages of bias plies in laminates. Similar data would be obtained from residual strength testing. This presentation is popular among several U.S. aerospace company design groups and is referred to as the angle-minus-loaded (AML) ply curve. It allows the establishment of relationships between lay-up and strength and enables projections and interpolations to be made, thus minimizing the testing that would otherwise be necessary. The horizontal axis is the percentage of bias ($\pm 45°$ plies) minus the percentage of on-axis ($0°$ plies).

The designer needs to perform trade-off studies to optimize the lay-up; however, increasing percentages of softer plies in the load direction may improve the failure strain but reduce the load-carrying capability of the laminate. Even if failure occurs at a lower strain in a stiffer laminate, the higher modulus may result in higher stress-to-failure and thus higher load.

As noted earlier, laminated composites suffer from relatively poor through-thickness strength and stiffness. One of the more novel attempts to improve this is by through-thickness stitching of the fabric. Stitching is performed on a dry preform that is subsequently impregnated with resin using a liquid molding or RTM process (see Chapter 5). Stitching has been found to improve the

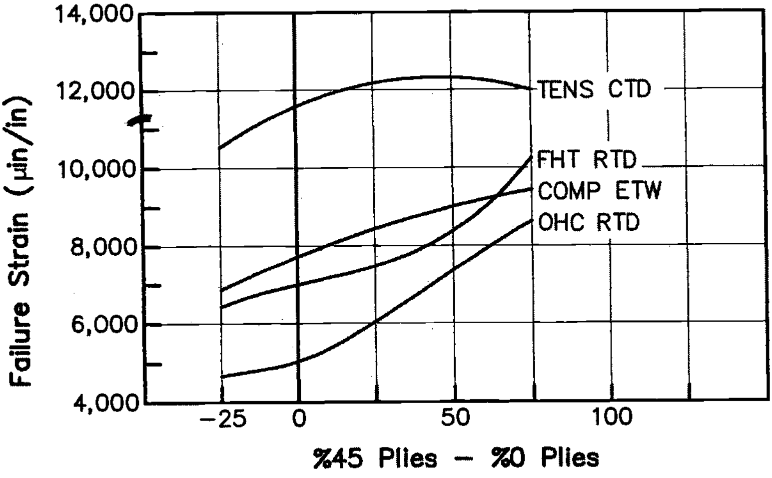

Fig. 12.15 Effect of lay-up on failure strain.

delamination fracture toughness[16] and in some cases, also improves the impact resistance and tolerance. Some studies[17] have shown little improvement in damage resistance (measurement of damage after impact) of composites (laminates 1–3.5 mm thick) made from stitched carbon woven fabrics compared with non-stitched fabric laminates. Stitching was shown to improve impact damage tolerance; however, this was offset by the reduction of undamaged compressive strength of the stitched laminate. The investigation of failure modes has revealed that stitching may offer benefits where unstitched damaged material fails by sub-laminate buckling. Where the failure mode is predominantly transverse, stitching does not provide any benefit.

Other textile preforming techniques such as knitting, braiding, and three-dimensional weaving also improve residual strength, however, again, their in-plane properties degrade appreciably. Composites with three dimensional reinforcement are discussed in Chapter 14.

12.5.9 Elevated Temperature and Moisture Exposure

Probably the most critical environmental exposure condition for composite materials is the effect of elevated temperature. This is particularly the case for composites with thermoset matrices because these polymers absorb moisture when exposed to hot-humid conditions, further reducing elevated temperature properties. Chapter 8 covers more fully the mechanics of property degradation under elevated temperatures and moisture absorption. Thermoplastic matrices, by contrast, absorb little moisture; however, they soften at elevated temperatures and are often vulnerable to chemical attack (see again Chapter 8).

Exposure extremes vary depending on the intended operation conditions, but typically chosen values for subsonic aircraft are 70 °C and 85% relative humidity. Under these conditions, thermoset composites will absorb up to 1% by weight of water over time with a corresponding reduction of glass-transition temperature, T_g, of around 25 °C. The moisture plasticises the matrix-reducing stiffness at elevated temperatures.

The effect of the matrix softening on the composite is a reduction in matrix-dominated properties, such as shear or compression strength. Figure 12.16 shows a comparison of the marked effect of temperature on compression strength for a typical thermoset composite and for comparison an aluminum alloy, where the loss in strength is seen to be minimal.

Because of the dramatic reduction in properties above T_g, the certification authorities specify a separation K between T_g and a maximum operating temperature of 25 °C (JAA) or 50° F MIL-HDBK 17 Figure 12.17.

It is normally required that property knockdowns for design are established after the material has become moisture saturated under the extreme operating conditions. Because of the slow absorption rate, particularly noticeable in thick specimens, conditioning can take many months. Recent efforts are investigating the possibility of testing a dry specimen under a higher temperature to

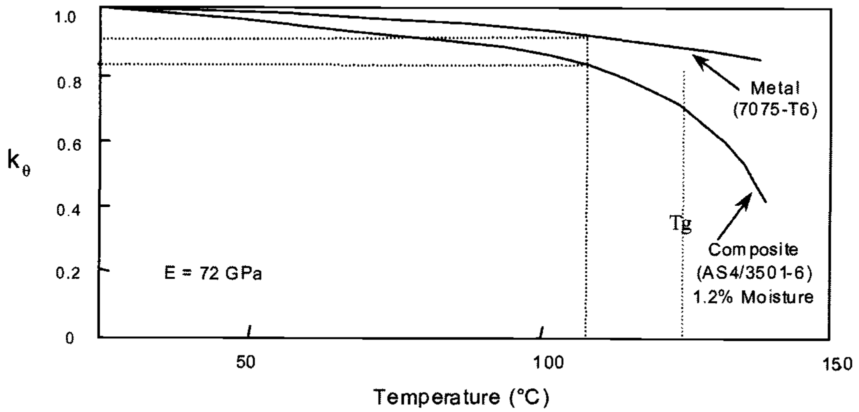

Fig. 12.16 Influence of temperature on compression strength of carbon/epoxy laminate. Adapted from Ref. 7.

compensate for the lack of moisture, however the validity of such an approach has not yet been proven.

Fiber-dominated properties, for example, tension strength, are not adversely affected by resin plasticization. In fact, the tensile properties of woven (crimped) materials are increased. However, fiber-dominated properties are adversely affected by embrittlement arising from exposure to very low temperatures. A typical tensile strength reduction for a carbon-fiber-reinforced plastic material exposed to temperatures existing at very high attitude is around 20–25%.

12.5.10 Lightning Effects

Carbon-fiber reinforced plastic composites are conducting materials, but because they have a significantly lower conductivity than aluminum alloys, the effect of direct lightning strikes is an issue of concern to airworthiness authorities. The severity of the electrical charge profiles[18,19] depend on whether the structure is in a zone of direct initial attachment, a "swept" zone of repeated attachments or in an area through which the current is being conducted. Survivability of structures in the direct attachment or swept zones will require some form of protection. The most effective methods involve the incorporation of a metal, bronze, copper, or aluminum, mesh, or foil co-bonded on the outer skin of the laminate. This mesh must make direct contact with the carbon-fiber material to be effective. Particular attention must be paid to the electrical bonding (connectivity) of the panel to the adjacent structure. Current will gravitate to points of high conductivity such as mechanical fasteners, and good electrical contact between the fastener, protective mesh, and subsequent electrical

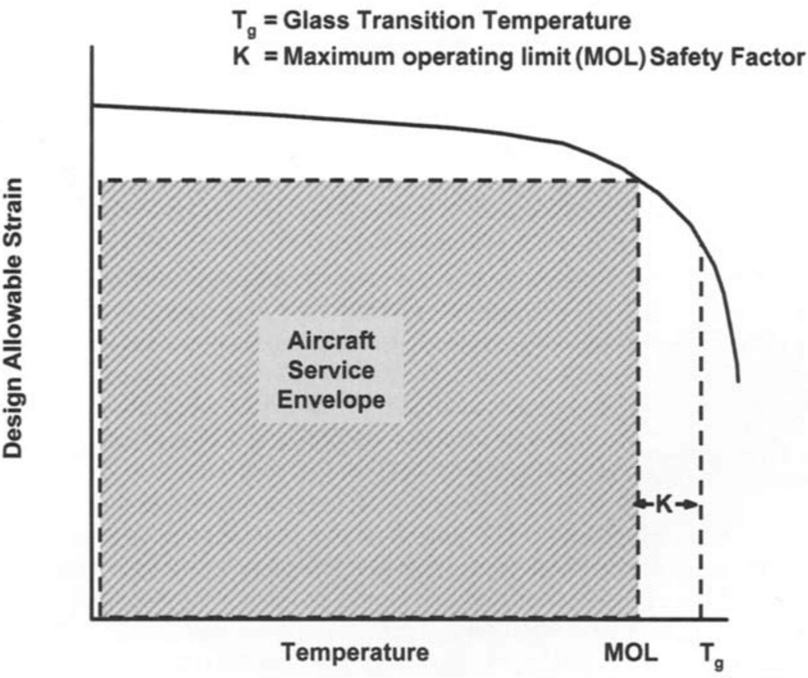

Fig. 12.17 Allowable design range for a carbon/epoxy composite as a function of strain and temperature. Adapted from Ref. 7.

path must be ensured. Severe burning around the fasteners will otherwise occur.

Composite panels with a suitable protective conducting coating in many cases out-perform thin-gauge aluminum alloy panels in terms of resistance to puncture by lightning.

12.6 Design Methodologies

The term *design* in relation to the design of composite structures refers to the process of establishing an appropriate laminate configuration (e.g., ply lay-up, built-up regions, etc.) to perform the given function. Functional requirements are usually given in terms of strength or stiffness. In the latter case, elastic properties can be reduced from coupon tests and laminate theory or the approximations thereto presented in Chapters 6 and 7. These properties can then be used to calculate strains, deflections and/or frequencies of vibration by standard techniques. Where it is necessary to base the design on a prescribed minimum strength, there are a number of analysis methods that can be used, each involving a different set of assumptions; these are discussed in Chapter 6. The choice of method will dictate the details of the laminate and qualification testing that will subsequently be required to validate or show that the various assumptions that have been made are adequately conservative.

A common assumption when analyzing rods and beams is that plane sections remain plane. In this case, for the condition of "no bending," strain is constant through the thickness, whereas stress varies from ply to ply depending on the modulus and orientation of each ply. For convenience, this leads to the use of strain analyses rather than stress analyses. Similarly, it is assumed that strains vary linearly through the thickness of plates in bending, an assumption that is reasonable, particularly for thin-shell aircraft structures. It enables the laminate to be treated as a homogeneous material and for the strains in the 1-1, 2-2, and 1-2 directions to be calculated (see Chapter 6). The simplest method of the subsequent strength prediction then introduces the assumption that the strength of any laminate is limited to a value pertaining when the strain in any one ply in the laminate exceeds a prescribed value. This is known as the first-ply failure method. Some variations of this method set the limiting value as a principal strain or maximum directional strain, whereas others base the ply failure on more complex relationships of bi-axial strain—see for example the Tsai-Hill criterion (also see Chapter 6). There has, long been debate over which of these criteria provide the best estimates of strength, however, this is likely to depend on the particular materials under consideration, the relative strains-to-failure of the reinforcement and matrix systems, and the loading. For many laminates, the maximum directional strain criterion is often used. Failure strain values are established from coupon tests on standard laminates in which the plies are orientated in the direction of the (uniaxial) load.

The more rigorous approach recognizes that the laminate strength is influenced by lay-up and stacking sequence. These influences are not altogether well understood. Some credit is given to differences in residual stresses remaining in the laminate after cure; however, predictions of residual stress and subsequent laminate strength do not always provide improved estimates. As it stands currently (if these effects are to be included) laminate capacities have to be established by tests on individual laminates. The difficulty with this approach is that there are often many different laminates in a single structure, and there may be several different load vectors applied. This means that each laminate may need to be tested under each loading combination, and to satisfy issues of variability, a number of coupons are required to establish each data point, leading to hundreds, and in some cases thousands, of tests. The larger companies have established such databases over long periods of time, and this explains the reluctance of many to change systems even when improved or cheaper materials become available.

The integration of testing into the overall design process is illustrated in Figure 12.18. (Note here the emphasis on trade-off studies that will establish an appropriate balance between cost and weight. These are essential if cost-effective design solutions are to result.)

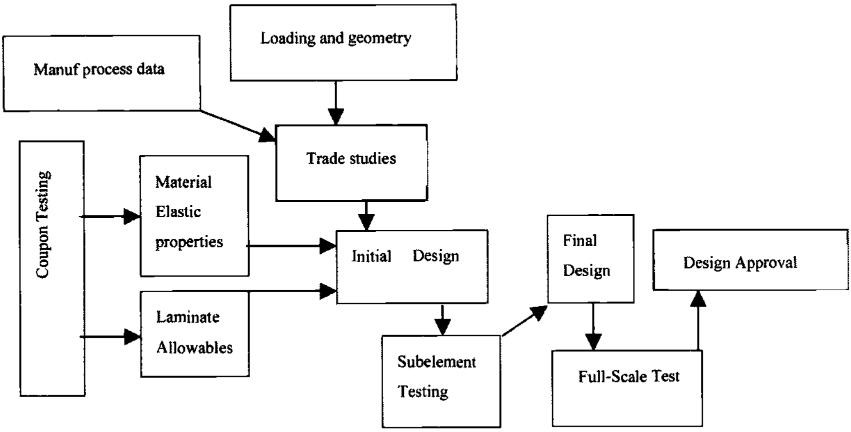

Fig. 12.18 An outline of the design process.

12.6.1 Use of Knockdown Factors

Figures 12.16, 12.19 to 12.21 illustrate property reductions of a typical composite material as compared with metal. As shown in the various figures, the knockdown factors used are:

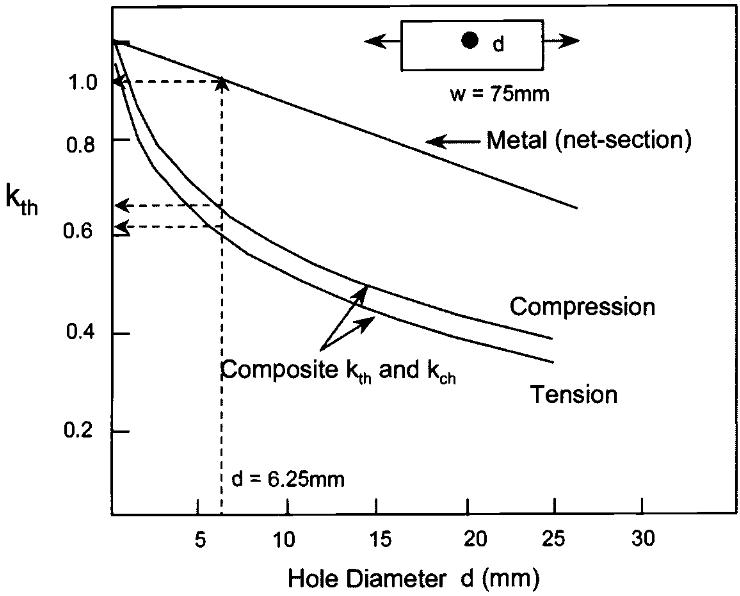

Fig. 12.19 Open-hole knockdown factors for quasi isotropic carbon/epoxy laminates compared with aluminum alloy. Adapted from Ref. 7.

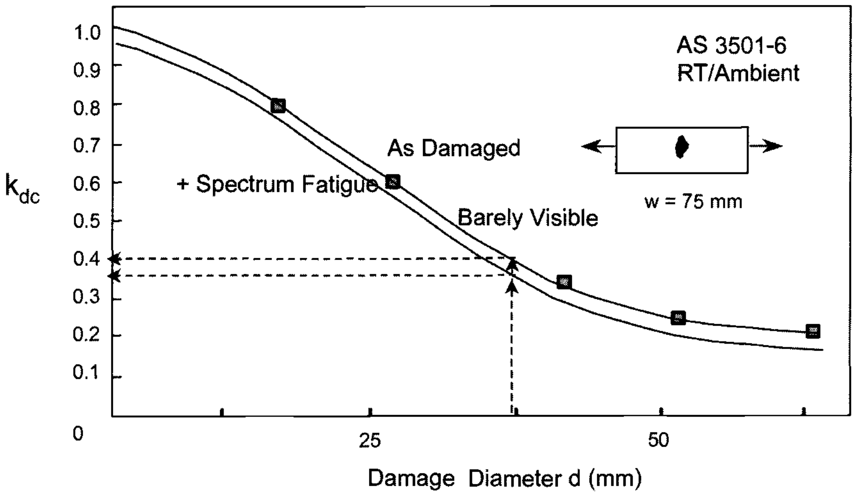

Fig. 12.20 Knockdown factor compression residual strength for impact damaged carbon/epoxy laminate after spectrum loading. Adapted from Ref. 7.

k_θ: temperature; k_{th}, k_{ch}: open-hole tension and compression, respectively; k_{dc}, k_{dt}: impact damage tension and compression, respectively.

The values to be attributed to these knockdowns will vary with material and lay-up, and the values provided in these figures are only a guide.

It is a common practice to multiply these factors to obtain combined affects. For example, in the case of a combined factor for a specimen with a 6.25 mm open hole, the compression allowable under hot/wet conditions would be:

$$k_{ch\theta} = k_{ch} \times k_\theta = 0.65 \times 0.85 = 0.55$$

Combining these factors in this manner tends to be highly conservative and for this reason is generally acceptable to airworthiness authorities.

Typical maximum strain values used in design are between 4000–5000 microstrain (strain $\times\ 10^6$) in tension and 3000–4000 microstrain compression. These values take into account combinations of environmental conditioning and impact damage or other stress concentrations.

In addition to point strain, other potential failure modes such as local and general instability (buckling), interlaminar strain, and bearing require consideration. Local instabilities by themselves may not be limits to load capacity; however, their presence will elevate point strains due to local bending and ultimately the maximum allowable ply strains may be exceeded. Non-linear finite element analyses are required for investigation of these conditions. Chapters 6 and 16 provide further information on this topic.

In the case of carbon/epoxy laminates, if the static strength has been established with due account for knockdown effects and the usual ultimate/limit load factor, it is not usually necessary to consider fatigue because the design limit

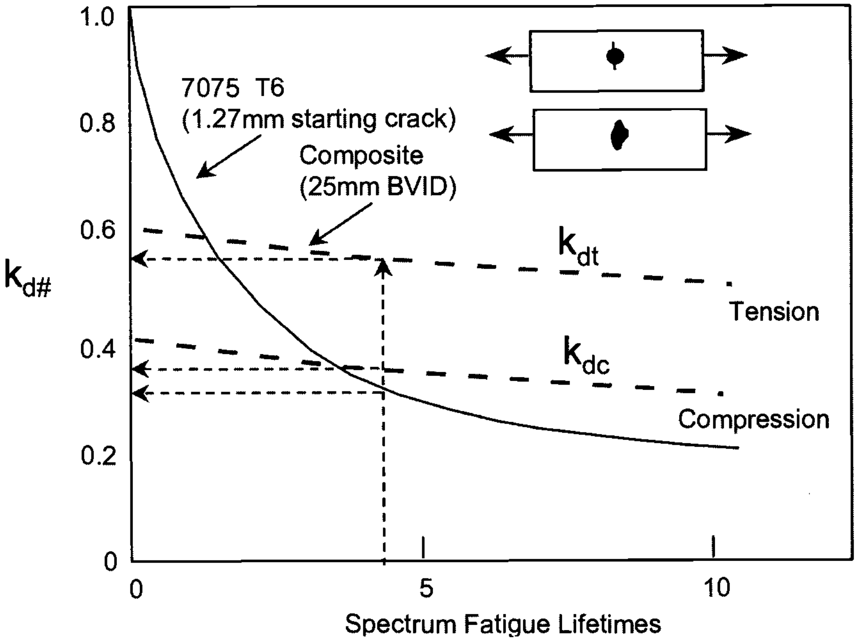

Fig. 12.21 Knockdown factor for tension- and compression-dominated fatigue spectrum loading following BVID for a carbon/epoxy laminate compared with an aluminum alloy with cracked fastener hole. Adapted from Ref. 7.

stresses will be below the fatigue limit of the laminate. This is not, however, the case for glass-fiber-reinforced composites. A discussion on fatigue properties of composite materials and structures is given in Chapter 8.

12.7 A Value Engineering Approach to the Use of Composite Materials

The demands of today's marketplace for new aircraft are significantly different from the past, where the pressure was to improve performance to deliver operational benefits. What both commercial and military aircraft operators now seek is a significant reduction in the initial purchase price of aircraft. Additionally, the ability to rapidly evolve an aircraft variant to meet an emerging niche application or opportunity is highly desirable. The latter has implications for material selection and new product development times.

The aircraft industry is going through a global rationalization of suppliers while product variety is increasing in a similar way to the automotive industry, and most large aircraft manufacturers are modelling their approach to the market and their suppliers on the automotive industry. With regard to composite materials, this means (for a given application) providing a comparable or better performance at a reduced cost in a shorter time.

12.7.1 Cost/Performance Trade-Offs

Composite structures have clearly been cost-effective in enhancing aircraft performance because their weight fraction in aircraft has steadily increased with time. But compared with standard aluminum alloys, they are relatively expensive, therefore there is continued price competition from metal com-ponents, particularly those produced by improved manufacturing methods. Further, it is believed that current design approaches do not fully utilize the potential of the unique material properties of these composites.

Table 12.1 provides cost and other property data for typical competing aircraft materials. This data has been obtained from several sources and is only accurate to the first order. The k terms are the knockdown factors described in Section 12.6.1.

C_r and C are the raw material cost and finished component cost (neglecting scrap), respectively. Note that these values have been obtained from one aircraft manufacturing company at one point in time and are reproduced here only as a guide to the approach that should be taken to material trade-studies.

Both the underlying material price and how efficiently material is processed are issues. Many metal components in modern aircraft are produced in high-speed machining centers where material removal can be achieved very rapidly. This has the effect of encouraging large amounts of scrap, for example, when a wing rib is hogged out of an aluminum alloy billet. Typically up to 90% of the material is removed, and consequently, the real cost per kilogram of the fly-away material is 10 times the raw material cost. In examples such as these, composite materials can be very cost-effective provided that the subsequent processing costs can be minimized. In other cases, the raw material costs can still rule out the composite materials option.

The following is an example of a typical approach to evaluating the effectiveness of a material choice for a given application.

A list of typical aircraft components is listed in Table 12.2 together with each component's design failure modes and the percentage of its weight contribution to the overall structure. Note that a fighter aircraft has been chosen in this example. For transport or other aircraft categories, the failure modes will be much the same, however, the weight percentages will be somewhat different.

12.7.1.1 Weight-Saving as a Function of Failure Mode. The starting point of this analysis is Table 12.1, based on the analysis developed in Eckvall et al.[21] for comparing two materials 1 and 2 for weight-saving. Material 1 is the benchmark and is taken here to be aluminum alloy 2024 T3. Material 2 is any of the other referenced materials.

The thickness, and therefore the weight, of each component of the airframe is determined by the primary loading it is required to support and the design failure modes. No alternative failure mode under these loads can be any weaker; if it is, then the thickness (and therefore the weight) must be based on this alternative mode.

Table 12.1 Properties Assumed for Candidate Airframe Materials

Material Type	Code	C_R \$/Kg	C \$/Kg	ρ Kgm^{-3}	E GPa	σ_e MPa	k_{ht}	k_{hc}	k_{dt}	k_{dc}	k_θ
Al Alloy	2024T3	10	229	2800	72	325	0.94	0.94	0.31	0.94	0.90
Al Alloy	7075 T76	10	229	2796	72	483	0.94	0.94	0.29	0.94	0.90
Al Alloy	A357	5	58	2800	72	276	0.94	0.94	0.30	0.94	0.90
Al/Li Alloy	8090 T3X	50	329	2530	80	329	0.94	0.94	0.39	0.94	0.90
Ti Alloy	Ti6Al4V	300	398	4436	110	902	0.94	0.94	0.20	0.94	1.00
Al Laminate	GLARE 1	100	550	2520	65	545	0.94	0.94	0.69	0.90	0.85
Carbon/epoxy	3501/6	160	788	1600	67	736	0.61	0.65	0.55	0.38	0.83
Carbon/epoxy	3501/6	160	788	1600	80	880	0.55	0.62	0.55	0.38	0.83

Table 12.2 Typical Fighter Aircraft Structural Breakdown (Based on Ref. 20)

Category	Component	% Of Structure	Design Failure Mode
1	Lower wing skin, wing-attachment lugs, longerons	18.6	Tensile strength
2	Upper wing skin	3.5	Compressive strength
3	Spar caps, rib caps	19.5	Crippling
4	Wing upper surface	9.7	Column and crippling (compression surface)
5	Horizontal tail torsion box	18.1	Buckling (compression or shear)
6	Fin torsion box, aft fuselage	11.6	Aeroelastic stiffness
7	Fin box	19.0	Durability and damage tolerance

The equations used for each failure mode are based on the equations provided in Ref. 20, but expressed in terms of the knockdown factors. The terms S_1 and S_2 represent any of the mechanical properties, and ρ_1 and ρ_2 represent their respective densities.

Using the data from Table 12.2 with the equations in Table 12.3, the weight ratio W_1/W_2 for each of the failure modes can be estimated. These data are provided in Table 12.4, where the lowest value is the most desirable. It is seen that the carbon-fiber-reinforced plastic composites are generally the optimum choice. This is particularly true for optimally designed orthotropic laminates (O), however this remains generally the case even for the less-than-optimum quasi isotropic (QI) ply configuration. The exception is for damage tolerance in compression where the carbon-fiber composite is similar to the standard aluminum alloy.

12.7.2 Cost Value Analysis of Weight-Saving

Unfortunately, it is not sufficient to choose a material based on weight-saving alone; the cost must also be considered. The value of saving a kilogram of weight will depend on the actual application, and for the relative comparison made here, the values provided in Table 1.2 in are used.

The analysis of the value of weight-saving is made as follows. For material 2, let the required thickness per unit area,

$$t_2 = t_1\left(\frac{S_1}{S_2}\right)^n$$

[Note: For tensile, compressive strength and aeroelastic stiffness $n = 1$, whereas for buckling, $n = 1/3$.]

Table 12.3 Weight Ratio Equations for Various Failure Categories (Based on Ref. 20)

Category	Failure Mode	Weight Ratio (W_2/W_1)
1	Tensile strength	$\frac{\rho_2}{\rho_1}\frac{\sigma_{e_1}}{\sigma_{e_2}}\left[\frac{k_{th1}}{k_{th2}}\frac{k_{\theta 1}}{k_{\theta 2}}\right]$
2	Compressive strength	$\frac{\rho_2}{\rho_1}\frac{\sigma_{e_1}}{\sigma_{e_2}}\left[\frac{k_{ch1}}{k_{ch2}}\frac{k_{\theta 1}}{k_{\theta 2}}\right]$
3	Crippling	$\frac{\rho_2}{\rho_1}\left[\frac{E_{s_1}\sigma_{e_1}}{E_{s_2}\sigma_{e_2}}\right]^{0.25}$
4	Compression surface column and crippling	$\frac{\rho_2}{\rho_1}\left[\frac{E_{s_1}E_{t_1}\sigma_{e_1}}{E_{s_2}E_{t_2}\sigma_{e_2}}\right]^{0.2}$
5	Buckling compression and shear	$\frac{\rho_2}{\rho_1}\left[\frac{E_1}{E_2}\right]^{\frac{1}{3}}$
6	Aeroelastic stiffness	$\frac{\rho_2}{\rho_1}\frac{E_1}{E_2}$
7	Durability and damage tolerance	$\frac{\rho_2}{\rho_1}\frac{\sigma_{e_1}}{\sigma_{e_2}}\left[\frac{k_{d1}k_{\theta 1}}{k_{d2}k_{\theta 2}}\right]$

Then, the weight change per unit area

$$\Delta W = W_1 - W_2 = t_1\rho_1 - t_2\rho_2$$

Substituting for t_2,

$$\Delta W = t_1\left(\rho_1 - \left(\frac{S_1}{S_2}\right)^n \rho_2\right)$$

The cost change per unit area

$$\Delta C = t_1\rho_1 C_1 - t_2\rho_2 C_2$$

where C_1 or C_2 is cost per unit weight and ΔC is cost change per unit area.

Substituting again for t_2,

$$\Delta C = t_1\left(\rho_1 C_1 - \left(\frac{S_1}{S_2}\right)^n \rho_2 C_2\right)$$

Thus, the cost per unit weight change is

$$\frac{\Delta C}{\Delta W} = \frac{\left(\rho_1 C_1 - \left(\frac{S_1}{S_2}\right)^n \rho_2 C_2\right)}{\left(\rho_1 - \left(\frac{S_1}{S_2}\right)^n \rho_2\right)} = C_W$$

Table 12.4 Weight Ratios for Candidate Airframe Materials for the Various Failure Categories

		Weight Ratio $(S_1/S_2)^n(\rho_2/\rho_1)$						
Material Type	Code	Cat 1	Cat 2	Cat 3	Cat 5	Cat 6	Cat 7a	Cat 7b
Aluminum alloy	2024T3	1.0	1.0	1.0	1.0	1.0	1.0	1.0
Aluminum alloy	7075 T76	0.7	0.7	0.9	0.9	1.0	0.7	0.7
Aluminum alloy	A357	1.2	1.2	1.0	1.1	1.0	1.2	1.2
Aluminum lithium	8090 T3X	0.9	0.9	0.9	0.9	0.8	0.7	0.9
Titanium alloy	Ti6Al4V	0.5	0.5	1.1	1.0	1.0	0.8	0.5
Aluminum laminate	GLARE 1	0.6	0.6	0.8	0.8	1.0	0.3	0.6
Carbon/epoxy	3501/6 QI	0.4	0.4	0.5	0.4	0.6	0.2	0.7
Carbon/epoxy	3501/6 O	0.4	0.3	0.4	0.4	0.5	0.1	0.6

Let the value of unit weight saved be C_V using the data provided in Table 1.1 in units of \$/kg.

[Note: Compared with C_1 and C_2, the values for C_V, are taken as negative. All costs are in \$/kg.]

Thus, to break even: $C_W = -C_V$

Then

$$-C_V = \frac{\left(\rho_1 C_1 - \left(\frac{S_1}{S_2}\right)^n \rho_2 C_2\right)}{\left(\rho_1 - \left(\frac{S_1}{S_2}\right)^n \rho_2\right)}$$

This leads to

$$\frac{W_2}{W_1} = \frac{\rho_2}{\rho_1}\left(\frac{S_1}{S_2}\right)^n = \left(\frac{C_1 + C_V}{C_2 + C_V}\right)$$

Thus, the difference between cost ratio and weight ratio is an index of value. The difference is zero for break even, positive for better than break even, and negative for worse than break even, with the magnitude giving an indication of the degree in each of the non-zero cases.

Finally, Table 12.5 presents the results of these calculations for the fighter aircraft applications chosen here as an example.

Although the above analysis indicates the trade-off between acquisition cost and structural performance, it should also be noted that through-life support costs will also have a bearing on the final selection. As previously indicated, well-designed composite structures can be expected to be more durable than metal structures; however, conversely, they can be more costly to repair.

Further material adoption might be approached from two perspectives:

(1) As operational experience builds confidence in non-aerospace materials, these materials, or the processes used to make them, may be adapted for aerospace use.

(2) As an aerospace material becomes more widely used, volume factors may help reduce the price.

In addition to reducing material cost, it is also necessary to minimize material usage for a given application. Composite materials have finite shelf lives and purchasing and production must be managed to ensure that all stock is consumed in a timely fashion. This becomes an issue in providing support for low-volume production or out-of-production items. The other factor related to consumption is the minimization of scrap, both in production components and in off-cuts and process-related consumable materials (e.g., bagging film). Here again, pursuing closer associations with customers and suppliers to optimize the formulation of composite materials within the specification and to deliver configurations that help optimize utilization is appropriate.

Table 12.5 Value Indices for a Typical Fighter Aircraft

		(Cost Ratio–Weight Ratio) Fighter						
Material type	Code	Cat 1	Cat 2	Cat 3	Cat 5	Cat 6	Cat 7a	Cat 7b
Aluminum alloy	2024T3	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Aluminum alloy	7075 T76	0.3	0.3	0.1	0.1	0.0	0.3	0.3
Aluminum alloy	A357	0.2	0.2	0.4	0.4	0.4	0.2	0.2
Aluminum lithium	8090 T3X	0.0	0.0	0.0	0.0	0.0	0.1	0.0
Titanium alloy	Ti6Al4V	0.3	0.3	−0.3	−0.2	−0.3	0.0	0.3
Aluminum laminate	GLARE 1	0.1	0.1	−0.2	−0.1	−0.4	0.4	0.0
Carbon/Epoxy	3501/6 Q	0.1	0.1	0.0	0.1	−0.1	0.4	−0.2
Carbon/Epoxy	3501/6 O	0.1	0.2	0.1	0.1	0.0	0.4	−0.1

There is also an issue in relation to the disposal of composite material waste. Metal waste has some residual sale value and can be recycled. This is not currently the position with most composite materials.

12.8 Conclusion

There is an increasing amount of applications for composite materials in aircraft structures; however, their very different properties from traditional metallic materials need to be thoroughly understood to produce a satisfactory design. The increasing scrutiny of costs means that careful consideration must be given to the total cost of ownership before deciding on a particular application.

In the aerospace industry, companies do not have full freedom to choose the materials to be used in a particular design. The costs of testing and analysis needed to qualify a new material are high, and this can lead to an impasse, even where lower-cost or higher-performance possibilities emerge.

As indicated, material costs typical for airframe alloys can be significantly lower than for composite materials. Further, metal products can be produced by automated processes with minimal quality assurance testing required, thus reducing the labor cost of manufacture. Many composites manufacturing processes remain labor intensive and require extensive non-destructive testing, therefore process cost issues are also critical to broader composite structure applications and usage.

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