Introduction: engineering requirements for aerospace composite materials

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1.1 Introduction

Composite materials have gained popularity (despite their generally high cost) in high performance products that need to be lightweight, yet strong enough to take high loads such as aerospace structures (tails, wings and fuselages), boat construction, bicycle frames and racing car bodies. Other uses include storage tanks and fishing rods. Natural composites (wood and fabrics) have found applications in aircraft from the first flight of the Wright Brothers' Flyer 1, in North Carolina on December 17, 1903, to the plethora of uses now enjoyed by man-made (engineered) composite materials on both military and civil aircraft, in addition to more exotic applications on unmanned aerial vehicles (UAVs), space launchers and satellites. Their adoption as a major contribution to aircraft structures followed on from the discovery of carbon fibre at the Royal Aircraft Establishment at Farnborough, UK, in 1964. However, not until the late 1960s did these new composites start to be applied, on a demonstration basis, to military aircraft. Examples of such demonstrators were trim tabs, spoilers, rudders and doors. With increasing application and experience of their use came improved fibres and matrix materials (thermosets and thermoplastics) resulting in CFRP composites with improved mechanical properties, allowing them to displace the more conventional materials, aluminium and titanium alloys, for primary structures. In the following sections, the properties and structure of carbon fibres are discussed together with thermoplastic and thermoset resins and the significance of the interface between the fibre and the matrix (resin).

1.1.1 Carbon fibre types and properties

High strength, high modulus carbon fibres are about 5−6 μm in diameter and consist of small crystallites of 'turbostratic' graphite, one of the allotropic forms of carbon. The graphite structure consists of hexagonal layers, in which the bonding is covalent and strong ($\sim >525$ kJ/mol) and there are weak van der Waal forces (<10 kJ/mol) between the layers [1,2]. This means that the basic crystal units are highly anisotropic; the in-plane Young's modulus parallel to the α-axis is approximately 1000 GPa and the Young's modulus parallel to the c-axis normal to the basal planes is only 30 GPa. Alignment of the basal plane parallel to the fibre axis gives stiff fibres, which, because of the

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relative low density of around 2 Mg/m^3, have extremely high values of specific stiffness ($\sim$200 GPa/(Mg/m^3)). Imperfections in alignment introduced during the manufacturing process result in complex-shaped voids elongated parallel to the fibre axis. These act as stress raisers and points of weakness leading to a reduction in strength properties. Other sources of weakness, which are often associated with the manufacturing method, include surface pits and macro-crystallites. The arrangement of the layer planes in the cross-section of the fibre is also important since it affects the transverse and shear properties of the fibre. Thus, for example, the normal polyacrylonitrile-based (PAN-based) Type I carbon fibres have a thin skin of circumferential layer planes and a core with random crystallites. In contrast, some mesophase pith-based fibres exhibit radially oriented layer structures. These different structures result in some significant differences in the properties of the fibres and of course those of the composites.

Refinements in fibre process technology over the past 20 years have led to considerable improvements in tensile strength ($\sim$4.5 GPa) and in strain to fracture (more than 2%) for PAN-based fibres. These can now be supplied in three basic forms, high modulus (HM, $\sim$380 GPa), intermediate modulus (IM, $\sim$290 GPa) and high strength (HS, with a modulus of around 230 GPa and tensile strength of 4.5 GPa). The more recent developments of the high strength fibres have led to what are known as high strain fibres, which have strain values of 2% before fracture. The tensile stress–strain response is elastic up to failure, and a large amount of energy is released when the fibres break in a brittle manner. The selection of the appropriate fibre depends very much on the application. For military aircraft, both high modulus and high strength are desirable. Satellite applications, in contrast, benefit from use of high fibre modulus improving stability and stiffness for reflector dishes, antennas and their supporting structures.

Rovings are the basic forms in which fibres are supplied, a roving being a number of strands or bundles of filaments wound into a package or creel, the length of the roving being up to several kilometres, depending on the package size. Rovings or tows can be woven into fabrics, and a range of fabric constructions are available commercially, such as plain weave, twills and various satin weave styles, woven with a choice of roving or tow size depending on the weight or areal density of fabric required. Fabrics can be woven with different kinds of fibre, for example, carbon in the weft and glass in the warp direction, and this increases the range of properties available to the designer. One advantage of fabrics for reinforcing purposes is their ability to drape or conform to curved surfaces without wrinkling. It is now possible, with certain types of knitting machine, to produce fibre performs tailored to the shape of the eventual component. Generally speaking, however, the more highly convoluted each filament becomes, as at crossover points in woven fabrics, or as loops in knitted fabrics, the lower its reinforcing ability.

1.1.2 Fibre-matrix interface

The fibres are surface treated during manufacture to prepare adhesion with the polymer matrix, whether thermosetting (epoxy, polyester, phenolic and polyimide resins) or thermoplastic (polypropylene, Nylon 6.6, PMMA, PEEK). The fibre surface is roughened by chemical etching and then coated with an appropriate size to aid bonding to the specified matrix. Whereas composite tensile strength is primarily a function of fibre properties, the ability of the matrix to both support the fibres (required for good compression strength) and provide out-of-plane strength is, in many situations, equally important. The aim of the material supplier is to provide a system with a balanced set of properties. While improvements in fibre and matrix properties can lead to improved lamina or laminate properties, the all-important field of fibre-matrix interface must not be neglected.

The load acting on the matrix has to be transferred to the reinforcement via the *interface*. Thus, fibres must be strongly bonded to the matrix if their high strength and stiffness are to be imparted to the composite. The fracture behaviour is also dependent on the strength of the interface. A weak interface results in a low stiffness and strength but high resistance to fracture, whereas a strong interface produces high stiffness and strength but often a low resistance to fracture, i.e., brittle behaviour. Conflict therefore exists and the designer must select the material most nearly meeting his requirements. Other properties of a composite, such as resistance to creep, fatigue and environmental degradation, are also affected by the characteristics of the interface. In these cases the relationship between properties and interface characteristics are generally complex, and analytical/numerical models supported by extensive experimental evidence are required.

1.1.3 Resin materials

Thermoplastic materials are becoming more available, however, the more conventional matrix materials currently used are thermosetting epoxies. The matrix material is the Achilles heel of the composite system and limits the fibre from exhibiting its full potential in terms of laminate properties. The matrix performs a number of functions amongst which are stabilising the fibre in compression (providing lateral support), translating the fibre properties into the laminate, minimising damage due to impact by exhibiting plastic deformation and providing out-of-plane properties to the laminate. Matrix-dominated properties (interlaminar strength, compressive strength) are reduced when the glass transition temperature is exceeded, and whereas with a dry laminate this is close to the cure temperature, the inevitable moisture absorption reduces this temperature and hence limits the application of most high-temperature-cure thermoset epoxy composites to less than 120 °C.

Conventional epoxy aerospace resins are designed to cure at 120–135 °C or 180 °C usually in an autoclave or closed cavity tool at pressures up to 8 bar, occasionally with a post cure at higher temperature. Systems intended for high temperature applications maybe undergo curing at temperatures up to 350 °C. The resins must have a room temperature life beyond the time it takes to lay up a part and have time/temperature/viscosity suitable for handling. The resultant resin characteristics are normally a compromise between certain desirable characteristics. For example, improved damage tolerance performance usually causes a reduction in hot-wet compression properties, and if this is attained by an increased thermoplastic content, then the resin viscosity can increase significantly. Increased viscosity is especially not desired for a resin transfer moulding (RTM) resin where a viscosity of 50 cPs or less is often required, but toughness may also be imparted by the fabric structure such as a stitched non-crimped fabric (NCF).

The first generation of composites introduced to aircraft construction in the 1960s and 1970s employed brittle epoxy resin systems leading to laminated structures with a poor tolerance to low energy impact caused by runway debris thrown up by aircraft wheels or the impacts occurring during manufacture and subsequent servicing operation. Although the newer toughened epoxy systems provide improvements in this respect, they are still not as damage tolerant as thermoplastic materials. A measure of damage tolerance is the laminate compression after impact (CAI) and the laminate open hole compressive (OHC) strengths. The ideal solution is to provide a composite exhibiting equal OHC and CAI strengths, and while the thermoplastics are tougher they have not capitalised on this by yielding higher notched compression properties than the thermoset epoxy composites. Polyetheretherketone (PEEK) is a relatively costly thermoplastic with good mechanical properties. Carbon fibre-reinforced PEEK is a competitor with carbon fibre/epoxies and Al-Cu and Al-Li alloys in the aircraft industry. On impact at relatively low energies (5−10 J) carbon fibre-PEEK laminates show only an indentation on the impact site while in carbon fibre-epoxy systems ultrasonic C-scans show that delamination extends a considerable distance affecting more dramatically the residual strength and stiffness properties of the composite. Another important advantage of carbon fibre-PEEK composites is that they possess unlimited shelf life at ambient temperature; the fabricator does not have to be concerned with proportioning and mixing resins, hardeners and accelerators as with thermosets; and the reversible thermal behaviour of thermoplastics means that components can be fabricated more quickly because the lengthy cure schedules for thermosets, sometimes extending over several hours, are eliminated.

It can be seen that in the effort to improve the through-the-thickness strength properties and impact resistance the composites industry has moved away from brittle resins and progressed to thermoplastic resins, toughened epoxies, through damage tolerant methodology, Z-fibre (carbon, steel or titanium pins driven through the z-direction to improve the through-thickness properties), stitched fabrics, stitched performs and the focus is now on affordability. The current phase is being directed toward affordable processing methods such as non-autoclave processing, nonthermal electron beam curing by radiation and cost-effective fabrication [3]. NASA Langley in the United States claims a 100% improvement in damage tolerance performance with stitched fabrics relative to conventional materials (refer to Advanced Composites Technology programme where NCF laminates are processed by resin film infusion). It is essential that if composites were to become affordable they must change their basic processes to get away from pre-preg material technology, which currently results in an expensive solution and hence product. However, autoclaved continuous fibre composites will still dominate for the high levels of structural efficiency required.

1.2 Analysis and design

Aircraft design from the 1940s has been based primarily on the use of aluminium alloys, and as such, an enormous amount of data and experience exists to facilitate the design process. With the introduction of laminated composites that exhibit anisotropic properties the methodology of design had to be reviewed and in many cases replaced. It is accepted that designs in composites should not merely replace the metallic alloy but should take advantage of exceptional composite properties if the most efficient designs are to evolve. Of course the design should account for through-thickness effects that are not encountered in the analysis of isotropic materials. For instance, in a laminated structure since the layers (laminae) are elastically connected through their faces, shear stresses are developed on the faces of each lamina. The transverse stresses (σ_z, τ_{xz}, τ_{yz}) thus produced can be quite large near a free boundary (free edge, cut-out, an open hole) and may influence the failure of the laminate [4].

The laminate stacking sequence can significantly influence the magnitude of the interlaminar normal and shear stresses, and thus the stacking sequence of plies can be important to a designer. It has been reported that the fatigue strength of a $(\pm 15/\pm 45)_s$ Boron fibre/epoxy laminate is about 175 MPa lower than a $(\pm 45/\pm 15)_s$ laminate of the same system. The interlaminar normal stress, σ_{zz}, changes from tension to compression by changing the stacking sequence, and thus accounts for the difference in strengths. In this case progressive delamination is the failure mode in fatigue. Approximate analytical methods and numerical approaches such as finite difference and finite element (FE) techniques [5] can be used to analyse the interlaminar stress distributions near free edges, open holes, bolted joints and help to identify the optimum fibre orientation and laminate stacking sequence for the given loading and kinematic boundary conditions. Generally, the determination of local stress distribution in a bolted joint is a three-dimensional problem due to bending effects and clamping of the fastener. The stress state in the vicinity of a bolted hole depends on many complex factors such as friction properties of the members, contact problem, geometry and stiffness of the joined members, joint configuration, clamping force and loading conditions. To precisely include all these factors in a stress analysis of a joint based on conventional analytical methods is extremely cumbersome [6−8]. Figure 1.1 illustrates a finite element model used to analyse a single bolt fastener in a double-lap joint and simulate the clamping force in the joint [8].

The lay-up geometry of a composite strongly affects not only crack initiation but also crack propagation, with the result that some laminates appear highly notch sensitive whereas others are totally insensitive to the presence of stress concentrators [9−11]. The selection of fibres and resins, the manner in which they are combined in the lay-up, and the quality of the manufactured composite must all be carefully controlled if optimum toughness is to be achieved. Furthermore, materials requirements for highest tensile and shear strengths of laminates are often incompatible with requirements for highest toughness. Compared with fracture in metals, research into the fracture behaviour of composites is in its infancy. Much of the necessary theoretical framework is not yet fully developed, and there is no simple recipe for predicting the toughness of all composites. We are not able yet to design with certainty the structure of any composite so as to produce the optimum combination of strength and toughness.

In metallic and plastic materials, even relatively brittle ones, energy is dissipated in nonelastic deformation mechanisms in the region of the crack tip. This energy is lost in

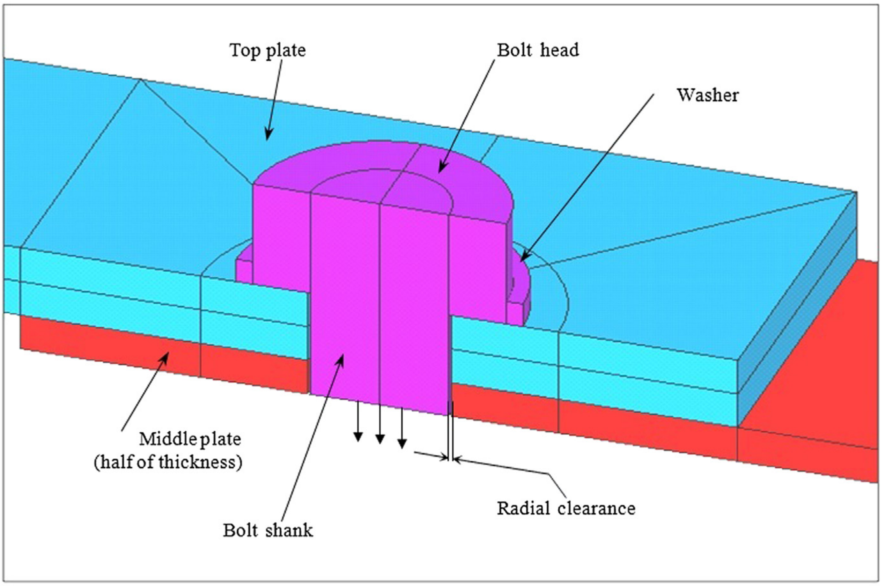

Figure 1.1 A finite element solid bolt model where the clamping force is represented by a negative displacement at bottom surface of bolt shank [8].

moving dislocations in a metal and in viscoelastic flow or craze formation in a polymer. In composites stressed parallel to the fibre axis, the fibres interfere with crack growth, but their effect depends on how strongly they are bonded to the matrix (resin). For example, if the fibre/matrix bond is strong, the crack may run through both fibres and matrix without deviation, in which case the composite toughness would be low and approximately equal to the sum of the separate component toughness. On the other hand, if the bond is weak the crack path becomes very complex, and many separate damage mechanisms may then contribute to the overall fracture work of the composite. For example, a brittle polymer or epoxy resin with a fracture energy $G \cong 0.1$ KJ/m and brittle glass fibres with $G \cong 0.01$ KJ/m can be combined together in composites some of which have fracture energies of up to 100 kJ/m. For an explanation of such a large effect we must look beyond simple addition [12,13].

If the crack is oriented parallel to the fibres and is subjected either to tensile stresses perpendicular to the fibres or shear stresses parallel to them, the fibres have little influence on crack propagation. In these circumstances a weak fibre-matrix bond can cause the crack to propagate rapidly along the fibre/matrix interface. If the interface bond is stronger than stresses required to fail the matrix in either shear or direct stress, crack growth is largely a function of matrix strength and toughness.

Fracture in composite materials seldom occurs catastrophically without warning, but tends to be progressive, with substantial damage widely dispersed through the material. Tensile loading can produce matrix cracking, fibre bridging, fibre pull-out, fibre/matrix debonding and fibre rupture, which provide extra toughness and delay failure [12]. The fracture behaviour of the composite can be reasonably well explained in terms of some summation of the contributions from these mechanisms, but as said earlier, it is not yet possible to design a laminated composite to have a given toughness.

Another important modelling issue is the fatigue life of the composite. In contrast to homogeneous metallic materials, in which fatigue failure generally occurs by the initiation and propagation of a single crack, the fatigue process in composite materials is very complex and involves several damage modes, including fibre/matrix debonding, matrix cracking, delamination and fibre fracture (tensile or compressive failure in the form of fibre microbuckling or kinking); Figure 1.2 shows a typical fibre microbuckling failure observed in currently used carbon fibre/epoxy systems [14,15]. By a combination of these processes, widespread damage develops throughout the bulk of the composite and leads to a permanent degradation in mechanical properties, notably laminate stiffness and residual strength [16,17]. Despite these complexities the in-plane fatigue strength of both glass and carbon fibre laminates is significantly superior to most metallic alloys, to such an extent that in-plane fatigue durability is no longer a design issue. As in the static loading case, it is the out of plane fatigue properties that can limit design strains.

Although these complexities (free edge effects, impact damage, joints, fatigue life prediction) lengthen the design process, they are more than compensated for by the mass savings and improvements in aerodynamic efficiency that result. The finite element analysis is also a crucial component, and the biggest time-saving strides have been in the user-friendly developments in creating the data and interpreting the results using modern sophisticated graphical user interfaces. The key is using parametric software to generate the geometry and the meshes. Apparently it used to take Boeing (Phantom Works) in St. Louis, Missouri, more than six months to perform the initial FE element stiffness and strength analysis for a complete aircraft, and this now takes less than three weeks with a handful of engineers, so composites can become more attractive.

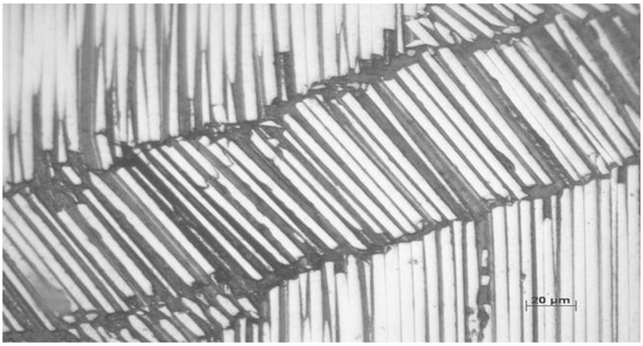

Figure 1.2 Fibre kinking induced by fibre instability or fibre microbuckling observed in a carbon fibre/epoxy laminate; fibre diameter is about 6 μm [14].

Figure 1.3 V22-Osprey tiltrotor plane [3].

The majority of aircraft control-lift surfaces produced have a single degree of curvature due to limitation of metal fabrication techniques. Improvements in aerodynamic efficiency can be obtained by moving to double curvature allowing, for example, the production of variable camber, twisted wings. Composites and modern mould tools allow the shape to be tailored to meet the required performance targets at various points in the flying envelope. A further benefit is the ability to tailor the aeroelasticity of the surface to further improve the aerodynamic performance. This tailoring can involve adopting laminate configurations that allow the cross-coupling of flexure and torsion such that wing twist can result from bending and vice versa. Finite element analysis allows this process of aeroelastic tailoring, along with strength and dynamic stiffness (flutter) requirements to be performed automatically with a minimum of post-analysis engineering yielding a minimum mass solution.

Early composite designs were replicas of those, which employed metallic materials, and as a result the high material cost and man-hour-intensive laminate production jeopardised their acceptance. This was compounded by the increase in assembly costs due to initial difficulties of machining and hole production. The cost is directly proportional to the number of parts in the assembly, and, as a consequence, designs and manufacture techniques had to be modified to integrate parts, thereby reducing the number of associated fasteners. A number of avenues are available for reducing the parts count, amongst which are the use of integrally stiffened structures, co-curing or co-bonding of substructures onto lift surfaces such as wings and stabilisers and the use of honeycomb sandwich panels. Hand lay-up techniques and conventional assembly results in manufacturing costs 60% higher than the datum, and only with the progressive introduction of automated lay-up and advanced assembly techniques composites compete with their metallic counterparts. Also, the introduction of virtual reality and virtual manufacturing will play an enormous role in further reducing the overall cost. The use of virtual reality models in engineering prior to manufacture to identify potential problems is relatively new but has already demonstrated great potential. Bell Textron in the United States made a significant use of IT during the product definition phase (for the V22 Osprey Tilt-rotor, Figure 1.3) to ensure 'right first-time' approach. Other manufacturing tools that can reduce production cost and make composites more attractive are Virtual Fabrication (creating parts from raw materials), Virtual Assembly (creation of assembly from parts) and Virtual Factory (evaluation of the shop floor). Virtual manufacturing validates the product definition and optimises the product cost; it reduces rework and improves learning.

1.3 Manufacturing techniques

The largest proportion of carbon fibre composites used on primary class-one structures is fabricated by placing layer upon layer of unidirectional (UD) material to the designer's requirement in terms of ply profile and fibre orientation. On less critical items, woven fabrics very often replace the prime unidirectional form. A number of techniques have been developed for the accurate placement of the material, ranging from labour-intensive hand lay-up techniques to those requiring high capital investment in automatic tape layers (ATLs). Tape-laying machines operating under numerical control are currently limited in production applications to flat lay-up, and significant effort is being directed by machine manufacturers at overcoming these problems associated with laying on contoured surfaces. The width of UD tape applied varies considerably from about 150 mm down to a single tow for complex structures. The cost of machinery is high and deposition rates low. In 1988 the first Cincinnati tape layer was installed in the Phantom Works, and in 1995 a 7-axis Ingersol fibre placement machine was installed. This gave the capability to steer fibres within an envelope of 40 × 20 ft with a 32-tow capability. An over-wing panel had been manufactured where it was able to steer around cut-outs. Collaboration with DASA on global optimisation software was to be completed at the end of 1998. This software is claimed to have produced a 13% weight saving. Other applications include an engine cowling door, ducting with a complex structure, FA18 E/F and T45 horizontal stabiliser skins. Its capacity was extended to take a 6-in-wide tape and Boeing 777 has been converted from hand lay-up to fibre placed (back to back then split) spars with a saving of $5000 per set. Bell Textron has a 10-axis Ingersol, contoured automatic tape laying machine for the B609 skin lay-up, which is placing a 6-in wide T300 tape onto an inner mould line Invar tool with pre-installed hat stringers. Fibre placement and filament winding technologies are also being used to manufacture components for the V22 [3].

Once the component is laid up on the mould, it is enclosed in a flexible bag tailored approximately to the desired shape and the assembly is enclosed usually in an autoclave, a pressure vessel designed to contain a gas at pressures generally up to 1.5 MPa and fitted with a means of raising the internal temperature to that required to cure the resin. The flexible bag is first evacuated, thereby removing trapped air and organic vapours from the composite, after which the chamber is pressurised to provide additional consolidation during cure. The process produces structures of low

Figure 1.4 A typical aircraft composite wing rib element produced by RTM.
Courtesy of Polyworx, RTM-Worx.

porosity, less than 1%, and high mechanical integrity. Large autoclaves have been installed in the aircraft industry capable of housing complete wing or tail sections.

Alternatively, low cost non-autoclave processing methods can be used like the vacuum moulding (VM), RTM, Figure 1.4, vacuum-assisted RTM (VARTM) and resin film infusion (RFI). The vacuum moulding process makes use of atmospheric pressure to consolidate the material while curing, thereby obviating the need for an autoclave or a hydraulic press. The laminate in the form of pre-impregnated fibres or fabric is placed on a single mould surface and overlaid by a flexible membrane, which is sealed around the edges of the mould by a suitable clamping arrangement. The space between the mould and the membrane is then evacuated, and the vacuum is maintained until the resin has cured. Quite large, thin shell mouldings can be made in this way at low cost. The majority of systems suitable for vacuum-only processing are cured at 60−120 °C and then post-cured typically at 180 °C to fully develop properties. In 1991 the evaluation of this method started at the Phantom Works using the resin system LTM10 (low temperature moulding) and they created a small allowables database for their X36 fighter research aircraft study. In 1996, McDonnell Douglas characterised LTM45 EL for the Joint Strike Force (JSF) prototype and generated design-allowable data. In 1998, Boeing also produced LTM45 EL data. LTM10 applications demonstrated for complex parts with a 140 °F cure under vacuum include a serpent inlet duct. A box using LTM10 was shown at the 1998 Farnborough Airshow. A research programme at NASA Langley is looking at the development of 180 °C material properties using low temperature curing resins. The main advantages of LTM systems are the potential to use autoclave-free cures, the use of cheaper tooling and reduced spring back of parts.

RTM and RFI are the predominant curing processes being developed today of which there are several variations. In traditional pre-preg technology, the resin has already infiltrated the fibres, and processing mainly removes air and volatiles, consolidates and cure. RTM in its simplest form involves a fabric preform being placed in an enclosed cavity and resin forced into the mould to fill the gaps under pressure and cure. The RFI method utilises precast resin tiles with thickness ranging from 0.125 in to 0.25 in. This approach reduces the number of consumables used but is very process sensitive, relying on the resin being of sufficiently low permeability to fully impregnate the fabric before cure advances too far. The use of an autoclave or press to apply pressure varies. The RFI process is being applied within the Advanced Composites Technology (ACT) Programme in conjunction with traditional autoclave processing. Heat is the energy source to activate the resin cure, but some resin systems can be activated by radiation. Wright Patterson claim that thermal oven processing could save 90% of autoclave processing time and energy and hence 50% cost. There is also a radiation curing process developed jointly by NASA and ACG (Advanced Composites Group) and of innovative electron beam-cured structures being developed by Foster Miller, Lockheed Martin and Oakridge National Laboratories in the United States [3].

The VARTM is a liquid resin infusion process and is currently considered by the aircraft industry to be the favoured low cost manufacturing process for the future. It is an autoclave-free process that has been identified as reducing the cost of component processing. It is reported that dimensional tolerance and mass measurements are comparable with stitched RFI autoclave panels. A conventional blade stiffened test panel (3 × 2 ft with 4-in-high blades 0.5 in thick) has been manufactured recently at NASA by using the VARTM method, achieving a reasonable quality.

Further cost reduction when manufacturing with composites will be achieved by reducing the assembly cost, by moving away from fastening (drilling of thousands of holes followed by fastener insertion and sealing) toward bonding and to assembly with less or no expensive jigging. Bell Textron among others are building and developing a number of structures (for the V22 and B609) where they are applying state-of-the-art composites technology/processes to achieve a unitised approach to manufacturing and assembly. There are of course significant certification challenges with an adhesively bonded joint without fasteners for a primary aircraft structure application that need to be addressed.

1.4 Applications in aircraft construction

In the pioneering days of flight, aircraft structures were composite being fabricated largely of wood (natural composite), wire and fabric. Aluminium alloys took over in the 1930s and have dominated the industry since then. Wooden structures did, however, persist until World War II, and the de Havilland Mosquito aircraft (DH98) constructed of a plywood-balsa-plywood sandwich laminate probably represents the high point of engineering design with wood. The DH91 Albatross airliner in 1937 was moulded as a ply-balsa-ply sandwich construction, and the Spitfire fuselage in 1940 was designed and built of Gordon Aerolite material that was a phenolic resin incorporating untwisted flax fibres that could be regarded as the precursor of modern fibre reinforced plastics.

Current civil aircraft applications have concentrated on replacing the secondary structure with fibrous composites where the reinforcement media have either been carbon, glass, Kevlar or hybrids of these. The matrix material, a thermosetting epoxy system, is either a 125 °C or 180 °C curing system with the latter becoming dominant because of its greater tolerance to environmental degradation. Typical examples of the extensive application of composites in this manner are the Boeing 757, 767 and 777 and from Europe the Airbus A310, A320, A330 and A340 airliners. The A310 carries a vertical stabiliser (8.3 m high by 7.8 m wide at the base), a primary aerodynamic and structural member fabricated in its entirety from carbon composite (now £10−20/kg for large tow HS fibre) with a total weight saving of almost 400 kg when compared with the Al alloy unit previously used. In addition, the CFRP fin box comprises only 95 parts excluding fasteners, compared with 2076 parts in the metal unit, thus making it easier to produce. The A320 has extended the use of composites to the horizontal stabiliser in addition to the plethora of panels and secondary control surfaces, leading to a weight saving of 800 kg over Al alloy skin construction. As an indication of the benefit of such weight saving, it has been estimated that 1 kg weight reduction saves over 2900 L of fuel per year. Larger amounts of FRPs are used in the bigger A330, A340 models and of course in the A380 super jumbo airliner built by the European Airbus consortium. The wing-trailing edge panels are made of glass and carbon fibre-reinforced plastics using a new RFI method in which resin film, interleaved between glass and carbon fabric layers, when the laminate is laid up, melts when heat is applied. Melted low viscosity resin migrates easily through the thickness of the laminate where it cures to form the final component. A hybrid aluminium/glass reinforced plastic system (GLARE) is used for the A380 fuselage crown, Figure 1.5, that results in reduced weight, increased damage tolerance and improved fatigue life.

Figure 1.5 A380 Glare fuselage crown.
Courtesy of Airbus S.A.S.

Figure 1.6 A typical modern carbon fibre/epoxy fuselage section.

The new Boeing 787 Dreamliner structure including the fuselage, wings, tail, doors and interior is made of over 50% by weight composite materials (80% by volume). The all-composite fuselage, Figure 1.6, makes it the first composite airliner in production. Each fuselage barrel is manufactured in one piece (about 45 ft long) eliminating the need of more than 50,000 fasteners used in conventional aircraft building. However, major assembly issues with the composite fuselage sections were encountered that caused long delays in delivering the aircraft to the customer, and there were problems with what was coming out of the autoclave, which is certainly a costly experience. Other technical issues that had to be resolved were on the safety side with electromagnetic hazards like lightning strikes, since the polymer material does not conduct away electric energy. Another major worry, but mostly for the operator, will be damage detection, which as it was said earlier occurs mainly internally and becomes difficult to detect. Soutis and co-workers [18−22] have demonstrated the possibility of using a linear array of piezoelectric transducers for the detection of delaminations and other modes of damage in composite plates. In order to detect low velocity impact damage in multidirectional laminates, they employed the fundamental antisymmetric A_0 Lamb mode at frequencies of 15−20 kHz; Figure 1.7 illustrates an ultrasonic C-scan image of the undamaged and damaged configuration together with time-of-flight method used to detect the location of damage; its location was calculated with an error of just 2.3% from the actual position [20]. Some success, but further research and development in this area, is required, especially on repair [23,24] and structural health monitoring of repaired configurations [25,26]. Figure 1.8 illustrates a curved fuselage composite panel where bonded patch repairs have been performed successfully, but for them to be certified by the Civil Aviation Authority a real-time damage monitoring system will be required.

Composites have been used in Bell helicopters (Dallas Fort Worth, Texas) since the 1980s following their advanced composites airframe programme when they were able

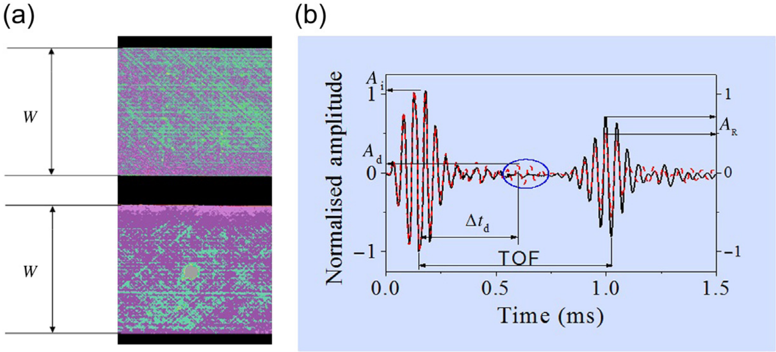

Figure 1.7 (a) An ultrasonic C-scan image (across the width, *W*) of undamaged and 2 J impact damaged laminate and (b) response of pristine and damaged composite plate [20].

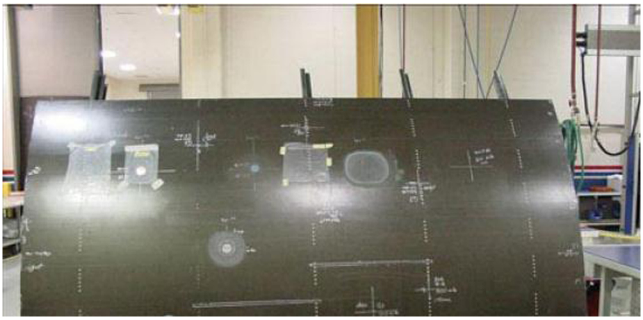

Figure 1.8 A composite fuselage panel with bonded patch repairs.
Courtesy of AMRC with Boeing, Sheffield, UK.

to achieve a 20% reduction in weight on metallic airframes. All blades on their newer vehicles (412, 407, 427, 214, 609, OH58D, V22) are all composite. The V22 Osprey tiltrotor has a composite fuselage stiffened skin, Figure 1.9, and an all-composite wing, chosen for its stiffness critical design, which was only possible in composites at low enough weight. The skins of the V22 wing are I-stiffened with co-bonded spars and bolted on ribs (the civil 609 version will use bonded ribs). It should be noted that buckling can generate severe stresses in the bond line between skin and stiffeners. Early demonstrators (from 1960s onwards) did not meet expectations until composites were available. The pylon support spindle is currently filament wound, but it is planned to be manufactured by employing the advanced fibre placement method. Over 60% of the whole vehicle weight is carbon composite, plus a further 12% in glass reinforced plastic. The V22 uses tape laying, hand lay-up and filament winding for most of the composite construction but is moving to fibre placement for the 609 civil version [3]. Mechanical fastening features heavily in the composite structure, some 3000

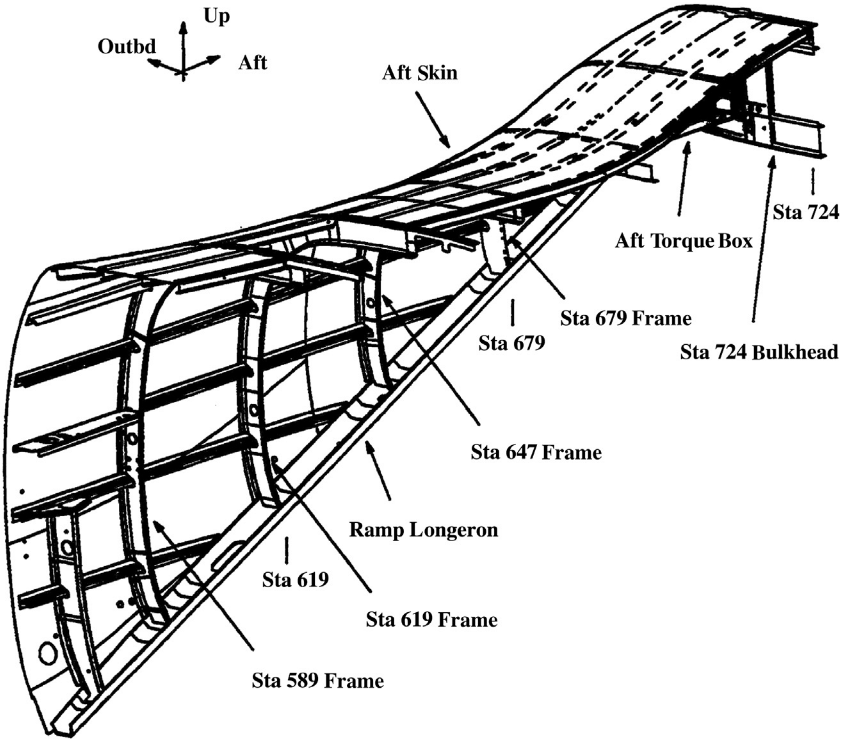

Figure 1.9 Composite fuselage stiffened skin of V-22 Osprey tiltrotor aircraft.
Courtesy of Ronald Krueger, NASA Langley, USA.

each side of the wing, is introduced by manual drilling with templates, but they are looking toward the use of automated drilling, probably involving water jet cutting.

Other examples where composites will be extensively applied are the new military cargo Airbus A400M and the tail of the C17 (USA). A 62-ft C-17 tail demonstrator has been successfully completed yielding 4300 fewer parts (including fasteners), weight reduced by 20% (260 kg) and cost by 50% compared with the existing metal tail. Without exception all agile fighter aircraft currently being designed or built in the United States and Europe (e.g., JSF, EFA, Figure 1.10) contain in the region of 40% composites in the structural mass, covering some 70% of the surface area of the aircraft. The essential agility of the a/c would be lost if this amount of composite material was not used because of the consequential mass increase.

Many of the materials, processes and manufacturing methods discussed earlier in the chapter have been implemented in their construction. Another interesting relatively new field of development in the military aircraft sphere is that of 'stealth', a concept that requires the designer to achieve the smallest possible radar cross-section to reduce the chances of early detection by defending radar sets. The essential compound curvature of the airframe with constant change of radius is much easier to form in

Figure 1.10 The Typhoon jet fighter, Eurofighter.
Photo by C. Soutis.

composites than in metal, while radar absorbent material can be effectively produced in composites.

1.5 Conclusion

The application of carbon fibre has developed from small-scale technology demonstrators in the 1970s to large structures today. From being a very expensive exotic material when first developed relatively few years ago, the price of carbon fibre has dropped to about £10/kg, which has increased applications such that the aerospace market accounts for only 20% of all production [27,28]. The main advantages provided by CFRP include mass and part reduction, complex shape manufacture, reduced scrap, improved fatigue life, design optimisation and generally improved corrosion resistance. The main challenges restricting their use are material and processing costs, damage tolerance, repair and inspection, dimensional tolerance and conservatism associated with uncertainties about relatively new and sometimes variable materials. Claims of 100% improvement in damage tolerance performance with stitched fabrics (3D woven fabrics) relative to conventional pre-preg materials are made; however, autoclaved continuous fibre composites will still dominate for the high levels of structural efficiency required in primary aircraft structures.

Carbon fibre composites are here to stay in terms of future aircraft construction since significant weight savings can be achieved. For secondary structures, weight savings approaching 40% are feasible by using composites instead of light metal alloys, while for primary structures such as wings and fuselages 20% is more realistic. These figures can always improve but innovation is the key to making composites more affordable. Some recent progress on composites failure analysis and design, where modern finite elements techniques have been employed to simulate blast behaviour of fibre metal laminates and model resin cracking and delamination developed during an impact event or a bolted joint are reported in references [29−31]. The reader will be informed in the following chapters of this book of the latest developments in new materials, 2D and 3D woven architectures, fabrication techniques, environmental effects (temperature and humidity), fracture and fatigue, design and analysis of impact, crash (at laminate and component level) and blast energy absorption, joints, in addition to repair, nondestructive evaluation and real-time structural health monitoring.

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