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## MATERIAL CHARACTERISTICS

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This chapter discusses some characteristics of the materials whose costs are assessed in this report toward the goal of providing useful background information to the reader in interpreting the cost results presented herein. The chapter begins by providing some background and historical perspective on aircraft material mix, and it then presents an overview of some key material properties that are important for airframe structural use. We then present a characterization of composite material; a discussion of the advantages and disadvantages of composite use in airframe structure; and an overview of metals used in airframe structures.

Many material characteristics not discussed in this chapter are also important to airframe structure design but lie beyond the scope of this report. These include thermal and electrical conductive properties and radar transparencies relevant to stealth airframes. The reader should thus be aware that this is not a comprehensive account of important material properties for airframe application.

### BACKGROUND AND HISTORICAL PERSPECTIVE

#### **What Are Composite Materials, and Why Are They Used in Producing Aircraft?**

A composite material is simply one composed of distinct kinds of material components. The composites we consider are typically made up of two components: a reinforcing material and a matrix material into which the reinforcing material is embedded. The rein-

forcing material is usually made up of discrete fibers that are distributed within the matrix, a contiguous material that envelops and supports the reinforcing fibers. Carbon fiber is the most common airframe structure reinforcing material; the most common matrices used in airframe structures are thermosets such as epoxy and BMI resins as well as some thermoplastic resins. We will say much more about manufacturing techniques in Chapter Three, but for introductory purposes we will give a brief description of a typical way in which one kind of carbon-epoxy is made.

Continuous carbon fibers are collimated (i.e., lined up in parallel) and pressed into a resin film. This material is called “preimpregnated,” or “prepreg.” The tape is cut into shapes appropriate to the aircraft part, and several layers of the tape are laid upon each other to form the part. This assembly is then cured in an autoclave, a device that heats the material under pressure. The resulting part is a black solid that is surprisingly light in comparison to an aluminum part of the same size and shape.

Although composites are used in airframe structures for several purposes, their primary advantage lies in their light weight. As we discuss in further detail below, composites have mechanical properties that are comparable to metals, such as strength and stiffness, but are lighter than metals. Composites also lend themselves to more efficient structural designs by combining several distinct parts into one (a design practice called “unitization”). Hence, when composite structures replace metal designs in an aircraft, the airframe is lighter and has higher range and payload capabilities. In addition, composites offer advantages over metal in the way they resist fatigue and corrosion and tolerate damage. Composites have other properties, such as electric and thermal conductivity and radar transparency, that make them an ideal material for stealth applications and radome construction.

### **Why Aren't All Metal Parts Replaced by Composites?**

There are four major reasons composites have not completely supplanted metal parts. First, some metals, such as titanium and steel, have mechanical and temperature properties that are crucial in some applications and cannot be matched by today's composites. Second, composites are still evolving, and new fibers and matrices are being

introduced whose properties are not as well known as those of metals. Therefore, a conservative approach has generally been taken toward introducing new composites, at least until their properties—particularly how they fail—are more completely understood. Third, some complex shapes cannot be made from composites in a cost-effective fashion. Finally, a primary focus of this study lies in the fact that composites generally cost more to produce per pound than do metals, especially aluminum. Chapter Four directly addresses this cost issue.

With this background, we introduce Figure 2.1, which shows historical trends in the use of composites in airframes. This figure shows the gradual increase in the percentage of composites used in aircraft. An interesting aspect of the history of composite use in military aircraft is that the percentage of composites in any given design typically decreases as the design matures in the course of the development process (see Figure 2.2). This phenomenon is generally

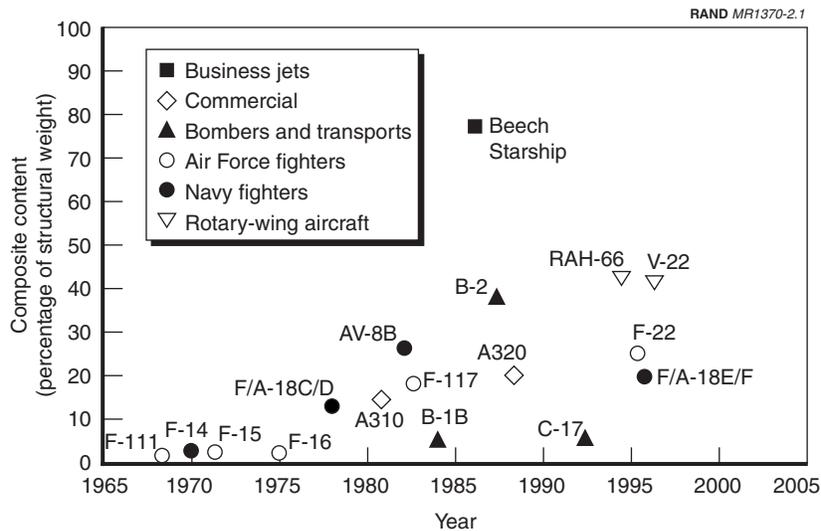
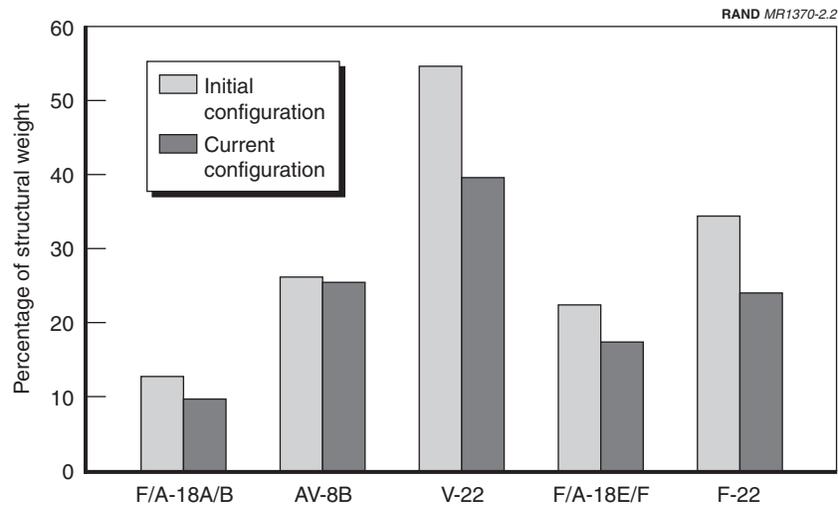


Figure 2.1—History of Aircraft Composite Use



**Figure 2.2—A Comparison of the Percentage of Composites Used in the Initial Configuration with the Current Configuration of Some Military Airframes<sup>1</sup>**

believed to occur because at the beginning of the design process, designers are thought to be overly optimistic about both the properties and the ease of fabrication of composites. As testing and early production continue, however, some of this optimism proves unfounded, at which point the initial design no longer meets cost, schedule, and weight constraints. As a result, the percentage of composites falls. (To be sure, there can be other explanations for this phenomenon, such as an increase in metal weight of an airframe with no change in composite weight and changes to the original requirements; however, the major reason is believed to be overoptimism early in the design process.)

<sup>1</sup>For all aircraft but the V-22, this figure compares the percentage of composites in the initial proposal to the percentage of composites in the current configuration. The V-22 comparison is between the airframe built during the early full-scale development (FSD) program and the current configuration.

Figure 2.3 shows the history of titanium use on military aircraft. As this figure illustrates, titanium use shows no consistent trend over time. However, it does tend to be higher in dedicated air superiority fighters, which are characterized by stringent temperature and other performance requirements.

The following figures show five recent aircraft that identify where various materials are used. Figure 2.4 illustrates three views of the F/A-18E/F. Figure 2.4a shows material use in the substructure; Figure 2.4b shows carbon-epoxy use on the surfaces, highlighting differences between the F/A-18C/D and the E/F; and Figure 2.4c shows overall material use. Figures 2.5, 2.6, 2.7, and 2.8 show material distribution on the B-2, F-117, F-22, and V-22, respectively.<sup>2</sup> Finally, Table 2.1 outlines the types of applications for which composites have been used in a variety of aircraft.

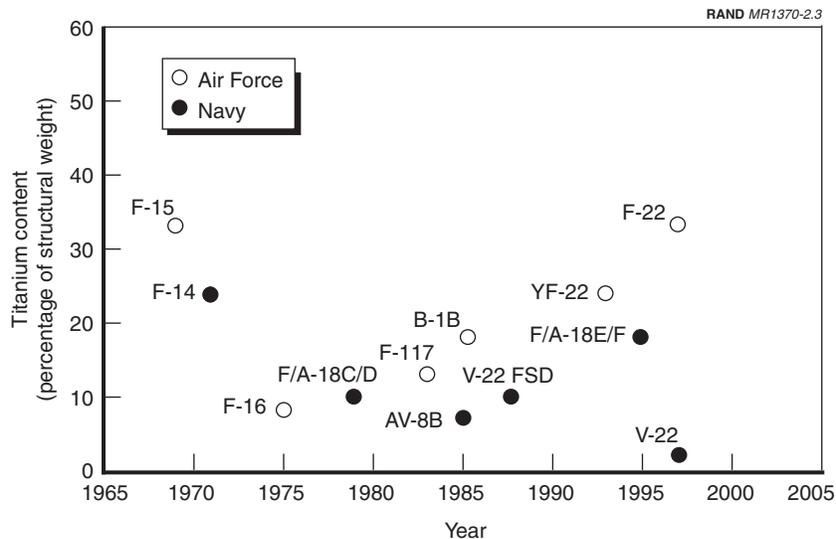


Figure 2.3—History of Titanium Use

<sup>2</sup>In Figure 2.5, the extensive use of composite skins in the B-2 might give the impression of a higher percentage of composite use than the actual figure—which, as Figure 2.1 shows, is roughly 40 percent.

RAND MR1370-2.4a

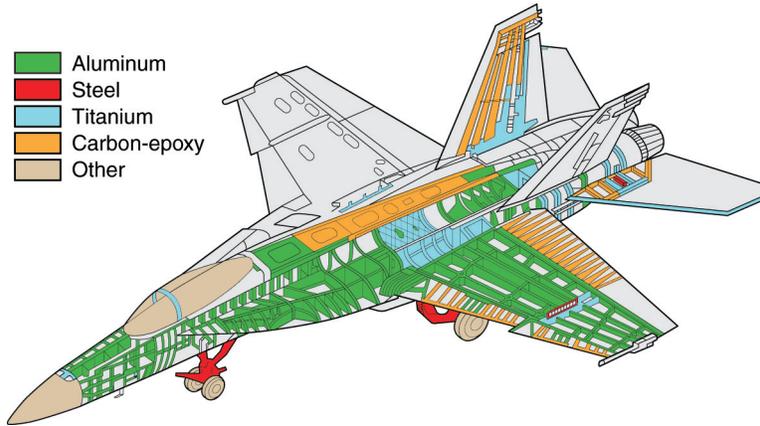


Figure 2.4a—F/A-18E/F Substructure Material Use

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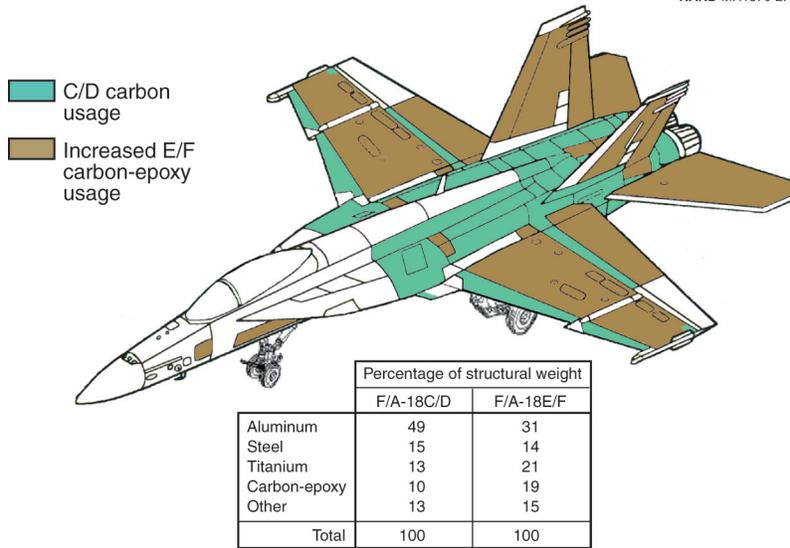


Figure 2.4b—F/A-18E/F Composite Use As Compared to the F/A-18C/D

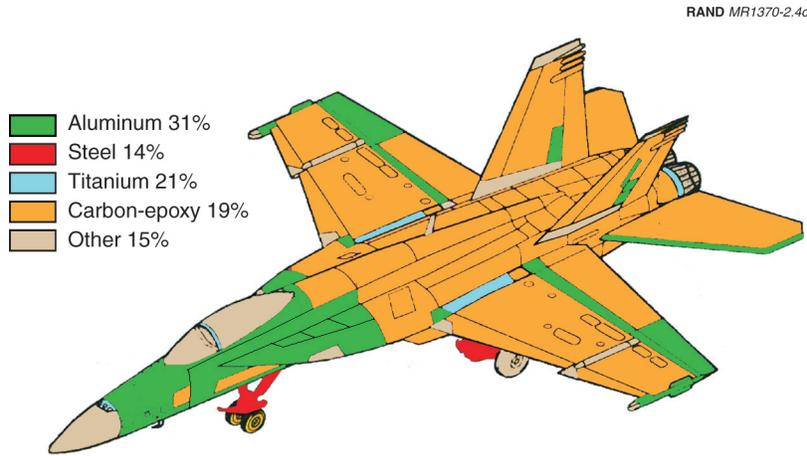


Figure 2.4c—F/A-18E/F Overall Material Use

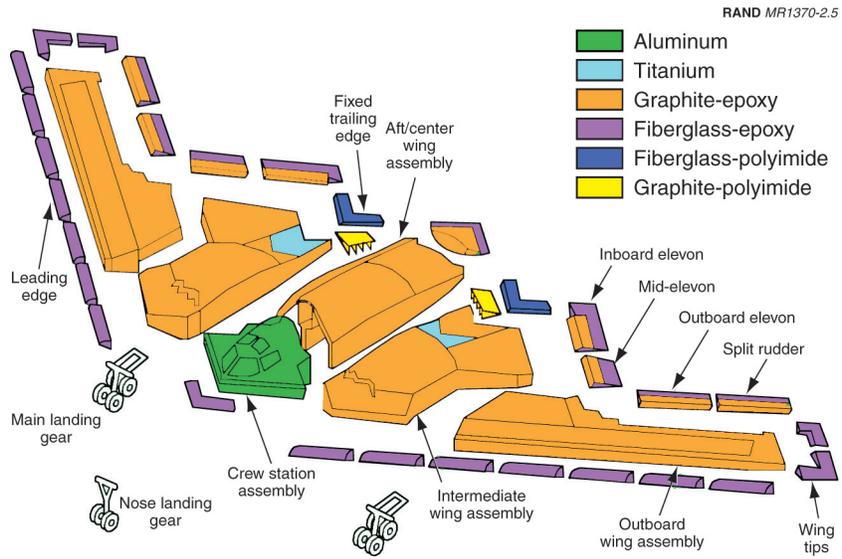
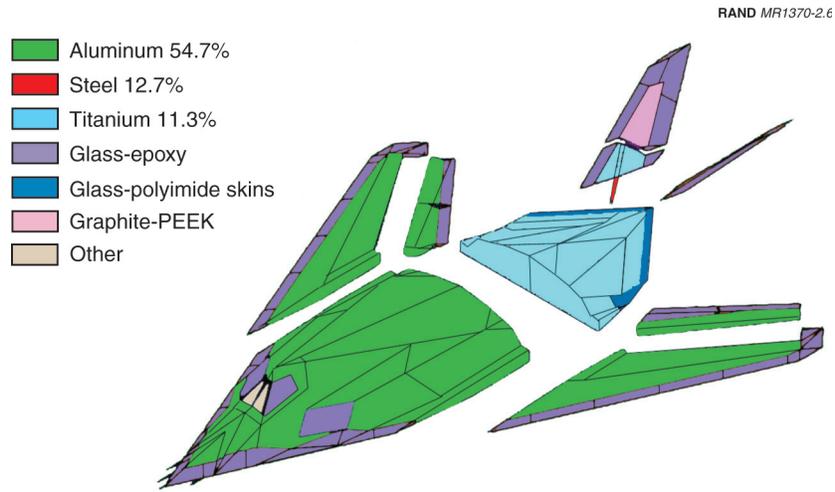


Figure 2.5—B-2 Overall Material Use



NOTE: Glass-epoxy, glass-polyimide, graphite-PEEK, and the category "other" constitute 21.3%.

Figure 2.6—F-117 Overall Material Use

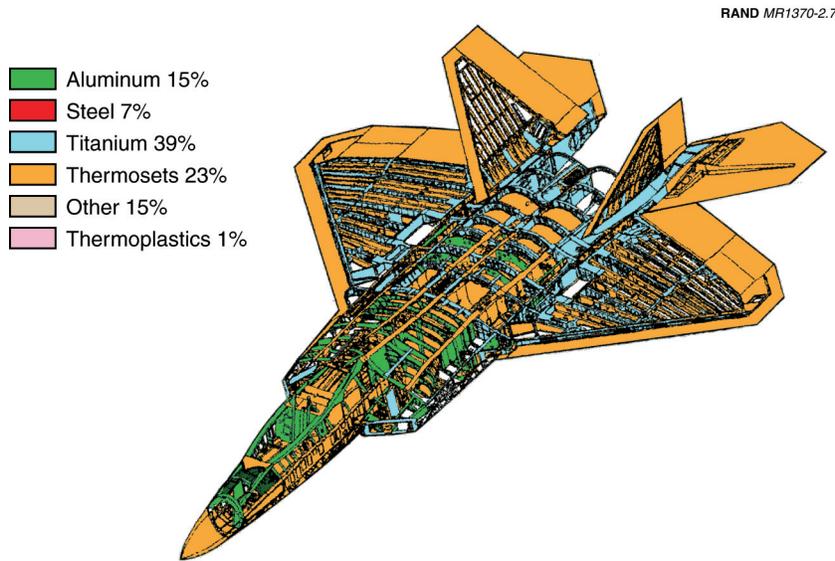


Figure 2.7—F-22 Overall Material Use

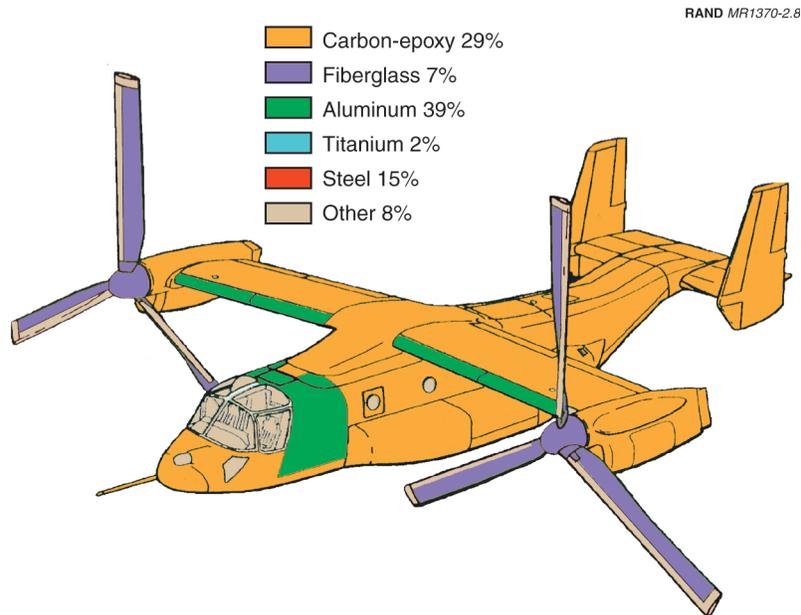


Figure 2.8—V-22 Overall Material Use

## MATERIAL PROPERTIES DEFINED

This section is a catalog of material properties that defines only those properties to which we refer later in the report.<sup>3</sup> We begin with **density**. Density is simply the weight of a material per unit volume, which is measured in pounds per cubic inch. When a material is referred to as lightweight, low density is what is technically meant.

Next, we discuss mechanical properties such as strength, modulus of elasticity, and toughness. Aircraft structural parts are subjected to forces during flight called loads. The **stress** experienced by a part is defined as the force to which it is subject divided by the area of the

<sup>3</sup>This section simply presents definitions with only minimal discussion of the materials science behind them. Readers interested in more background information are referred to Gordon (1978), which is a good nontechnical primer on these issues, or to any of the many excellent materials engineering textbooks, such as Flinn and Trojan (1975), Juvinal (1967), Van Vlack (1985), and Shigley and Mischke (1989).

**Table 2.1**  
**Composite Components in Aircraft Applications**

Composite Component	F-14	F-15	F-16	F-18A	B-1	AV-8B	F-18E	F-22	V-22	DC-10 Demo	L-1011 Demo	737 Demo	757	767	Lear Fan
Doors	✓			✓	✓	✓	✓	✓	✓				✓	✓	✓
Rudder		✓				✓	✓	✓		✓			✓	✓	✓
Elevator									✓			✓	✓	✓	✓
Vertical tail		✓	✓	✓	✓	✓	✓	✓	✓	✓	✓				✓
Horizontal tail	✓	✓	✓	✓	✓	✓	✓	✓	✓			✓			✓
Aileron						✓				✓			✓	✓	
Spoiler												✓	✓	✓	
Flap						✓	✓	✓	✓				✓		✓
Wing box						✓	✓	✓	✓				✓		✓
Body						✓	✓	✓	✓						✓
Miscellaneous	Fairings	Speed brake	Speed brake, fairings	Speed brake, fairings	Slats inlet	Fairings	Speed brake, fairings	Fairings	Fairings	Rotor blades	Fairings	Fairings	Fairings	Fairings	Propeller blades

SOURCES: J. M. Anglin (1987) and Boeing and Lockheed Martin nonproprietary information.

part facing (i.e., perpendicular to) the force. The **strength** of a material is simply the maximum stress that material can experience without failure.<sup>4</sup> It is measured in units of force per area (we will use thousand pounds per square inch, abbreviated KSI).

**Tensile strength** is the maximum pulling stress a piece of material can withstand without failing. In this report, when we refer to the numerical value of the strength of a material (as in Table 2.2), tensile strength is what is meant. **Compressive strength** is the maximum pushing stress that a material can withstand without failing. Tensile and compressive strength are not necessarily equal. Airframe materials subject to bending loads are subject to both tensile and compressive stress.

Materials lengthen (that is, stretch) when pulled, and the percentage a material changes in length when subjected to a pulling load is referred to as **strain**. At levels of stress below those at which materials begin to fail, the strain of a material is proportional to the stress put upon it.<sup>5</sup> Put another way, the amount a part of a given material stretches when subjected to a force is proportional to the level of the force.<sup>6</sup>

**Modulus of elasticity** is defined as stress divided by strain at levels of stress below the beginning of failure. Since strain is proportional to stress in that region, this quantity is well defined.<sup>7</sup> Modulus of elasticity is the measure of the **stiffness** or rigidity of a material; the higher the modulus, the less a material lengthens when subjected to a given stress. It is measured in units of force per area (we will use million pounds per square inch, abbreviated MSI). Stiffness is an

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<sup>4</sup>The definition of failure depends on the material. Some materials fail by suddenly breaking apart, while others elongate or yield so that they can bear little or no load. Strength is an inherent property of a material; therefore, the strength of a part is the same whether the part is operating in an airframe or waiting to be assembled.

<sup>5</sup>This relation is known as Hooke's law.

<sup>6</sup>The relation of tensile stress and strain is empirically determined through a tensile test, in which a test specimen is mounted in a test machine and pulled by applying increasing loads (stresses) while the percentage change in the length of the specimen is recorded. The observed stresses and strains are then plotted in a **stress-strain diagram**. The stress-strain diagram is discussed in detail in Appendix A.

<sup>7</sup>This region is referred to as the Hookean region.

important property of airframe structural materials because if airframe parts changed length substantially under loads, aerodynamic performance, which is dependent on many relative dimensions of the airframe, would suffer dramatically. Structural integrity would also be hard to maintain.

The relation of the strength of a material to its weight is very important in airframe applications, and **specific strength**—or strength divided by density—is a measure of that concept. The weight of a part that can withstand a given load pattern will be approximately proportional to its specific strength, and achieving low weight is critical to improving the performance characteristics of aircraft. It should be noted that since strength is measured in weight per area and density in weight per volume, specific strength is measured in length (thousands of inches in this report).

**Specific stiffness** is stiffness divided by density. As with specific strength, this variable is important because the weight of a part that will change dimension only a given percentage under a given load pattern will be approximately proportional to specific stiffness. Since stiffness is measured in MSI, specific stiffness is measured in millions of inches.

A part failure that results from repeated (constant or fluctuating) tensile and compressive stress is called a **fatigue failure**. A fatigue failure begins as a small crack in metals and as a delamination<sup>8</sup> in composite parts. This is important in that airframe parts undergo repeated loads during operation. As the airframe ages, structural fatigue is carefully monitored by the airframe maintainer. Failure can also be caused by **corrosion**. Corrosion in structural parts results from contact with external agents. These agents can be natural, such as the environmental agents of the atmosphere and ocean (e.g., salt), or artificial, such as solvents. Both cause chemical degradation and changes in material properties.

**Toughness** is the ability of a material to absorb energy without damage. It is defined as the energy that a material can absorb without fracturing, measured in units of energy per volume. For aircraft, this property is important because tools and the like may be dropped on

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<sup>8</sup>Delamination of composites is discussed later in this chapter.

surfaces; the tougher the material, the greater the impact that can be absorbed without causing damage. The opposite of toughness is called brittleness. A “damage-tolerant” material or part is defined as one that can withstand a reasonable level of damage or defect during the manufacturing process or while in service without jeopardizing aircraft safety. In contrast to metals, composites are relatively brittle; therefore, toughening agents are added to the matrix to create a more damage-tolerant composite part.

The **service temperature** of a material is the highest temperature that a material can withstand in operation without suffering significant loss of its structural integrity or essential mechanical properties. Although service temperature is an inherent property of metals, it is, of course, affected by alloy composition. The property is quite important in military aircraft, since skin temperature is related to speed and exhaust temperature to engine power.

## **SPECIFIC MATERIAL PROPERTIES**

We now turn to a discussion of the specific properties of materials considered in this report. We introduce it with Table 2.2, which shows the properties of some common airframe structural materials. It should be noted that material properties change with the specific designation and form of the material; these are representative values for aircraft application. Material designations generally begin as specific to a given company but often are subsequently licensed to other manufacturers. The first three columns in Table 2.2 present data for metals commonly used in aircraft construction; the next three columns present data for composites.

The implications of this table are discussed extensively below. Here we simply point out that composites show significantly higher specific strength and stiffness than do metals.

## **COMPOSITE MATERIALS**

This section presents a brief overview of composite materials that is intended to complement the overview given in Resetar, Rogers, and Hess (1991). This overview highlights changes that have occurred since that earlier report.

**Table 2.2**  
**Material Properties**

Property	Aluminum (7050- T7451) <sup>a</sup>	Titanium (6Al-4V) <sup>a</sup>	Steel (PH13- 8Mo) <sup>a</sup>	Carbon/ Epoxy (IM7/ 977-3) <sup>a</sup>	Carbon/ BMI (IM7/ 5250-4) <sup>a</sup>	Carbon/ Thermo- plastic (IM7/PEEK) <sup>a</sup>
Density (lb./sq in.)	0.102	0.160	0.279	0.057	0.056	0.058
Strength (KSI)	70	134	201	332	349	323
Stiffness (MSI)	10.3	16.0	28.3	22.2	22.2	22.7
Specific strength (K in.)	685	840	720	5825	6230	5570
Specific stiffness (M in.)	100	100	100	390	395	390
Service tempera- ture (degrees F)	250	450	1000	275	325	275

<sup>a</sup>The designations in parentheses refer to the specific alloy or fiber/matrix.

As described above, composite materials are simply those composed of two or more constituent parts. This report focuses on composites made up of a reinforcing material embedded in a binding matrix. The primary reinforcing material that we consider is carbon fiber in long-strand form.<sup>9</sup> In this context, we focus on three matrices: epoxy, BMI, and thermoplastics.

### Reinforcing Material

We concentrate herein on continuous carbon fiber strands as a reinforcing material. To put this in context, we note that reinforcing ma-

<sup>9</sup>The terms *carbon fiber* and *graphite fiber* are often used interchangeably, although technically carbon and graphite fibers differ in the temperature at which they are produced and in their carbon content. In this report, we use the term *carbon fiber* to refer to both.

terial can come in forms other than continuous fiber—for example, short or long whiskers or particulates. By continuous fiber strands we mean fibers of approximately the same length as the dimensions of the part itself or longer; whiskers and particulates tend to be short in comparison to part dimensions. The advantage of continuous fiber is that the strands can be oriented within the part so that it has different mechanical properties, such as strength and stiffness, in different directions. This is called having *anisotropic properties*. This property is advantageous in airframe applications because loads on parts tend to be highly concentrated in specific directions. In general, metals are equally strong and stiff in all directions and are thus described as having *isotropic properties*. Carbon-whisker- and particulate-reinforced composites tend to have lower strength than continuous fibers and thus have only limited airframe applications. Continuous fiber reinforcement currently dominates airframe applications.

Several other continuous-fiber-reinforcing materials are commonly used. These include glass (as in fiberglass), aramid (known commercially as Kevlar), and boron. All of these materials have had some aerospace applications, but since carbon fibers predominate, we will concentrate on them in the remainder of this report. Carbon fibers dominate because in general, glass is relatively heavy, boron is relatively expensive, and aramid has a lower tensile modulus of elasticity. Glass is transparent to radio waves and is therefore an ideal material for radome construction and low-observable (LO) applications. Aramid fibers are used primarily in products such as bulletproof vests. Boron has been used in very flat applications but currently is not used in airframe structures.

Carbon fibers are made from a precursor such as polyacrylonitrile (PAN), petroleum pitch, or rayon in a continuous, precisely controlled process. During this process, the fiber is exposed to heat and tension in a series of ovens. This heating process chemically changes the precursor, yielding high strength-to-weight and high stiffness-to-weight properties. PAN-based fibers are the most common carbon fibers used in military airframe structures.<sup>10</sup> Figure 2.9 illustrates this

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<sup>10</sup>Pitch-based fibers have higher stiffness and thermal conductivity, which make them ideal for space applications. Rayon-based carbon fibers have extremely low thermal

process. The resulting product consists of very fine carbon fiber filaments—each approximately 5 microns, or 0.0002 inch, in diameter—that feel like fine silk thread to the touch. These filaments are then arranged into various forms, described below. One example is “tow,” 0.125-inch-wide by 0.005-inch-thick tape in which all the fibers are aligned along the tape. In aerospace applications, each strip of tow contains 3000, 6000, or 12,000 filaments.

Carbon fibers are classified primarily according to their stiffness or modulus of elasticity. Most carbon fiber used in aerospace structural applications today is intermediate modulus (IM), which is characterized by a stiffness of 40–50 MSI. IM-7, the type of reinforcement used in the materials shown in Table 2.2, is such a fiber. Note that the stiffness of the composite materials listed in this table is only about 22 MSI. This is because the stiffness (as well as the strength) of a composite is typically lower than that of the reinforcing material.<sup>11</sup>

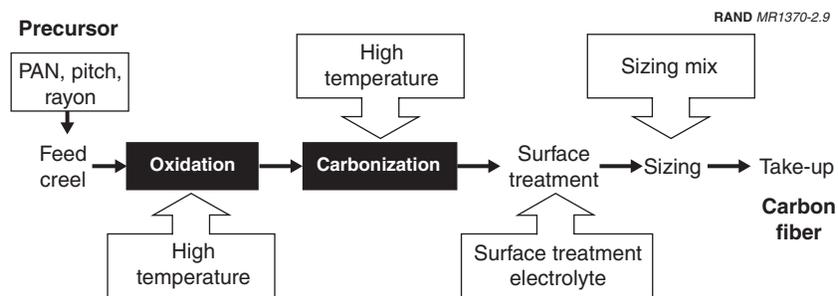


Figure 2.9—Carbon Fiber Fabrication Process

conductivity, which makes them ideal for use in extremely high temperature applications (Traceski, 1999).

<sup>11</sup>Carbon fiber also comes in standard modulus (SM) (33–35 MSI), high modulus (HM) (50–70 MSI), and ultrahigh modulus (UHM) (70–140 MSI). SM fiber is used in aerospace applications in which lower mechanical properties can be accepted. A part made of SM fiber and untoughened epoxy has about 20 percent less strength and 10 percent less stiffness than a part made of IM fiber and toughened epoxy. HM and UHM do not have widespread use in airframes owing to their poor compression properties, although there is limited HM use now.

## Matrix Materials

In this report we concentrate on polymer matrices, which are by far the most commonly used matrices in aerospace applications. However, three other kinds of matrix materials are used in advanced composites: metal, ceramic, and carbon. Metal matrices are currently too expensive and difficult to work with for widespread application, and ceramic matrices have insufficient toughness (i.e., they are brittle). Both have limited use in aircraft applications but are used in some applications involving high-temperature operating conditions (e.g., engines). Carbon matrix composites are costly and difficult to produce, and their primary airframe application is in aircraft brakes (they are also used in racing-car brakes). Carbon matrix composites are also used in space launch and reentry vehicles.

## Thermoset Matrices

Polymer matrices, which are also called resins, are characterized as either thermosets or thermoplastics. Thermosets are the most widely used polymer matrices in airframe structures. In this study, we consider two kinds of thermosets: epoxy, which is most commonly used, and BMI, the next most common thermoset. No other thermoset currently has widespread aircraft structural application.<sup>12</sup>

After a composite part with a thermoset matrix has been formed at room temperature (a process described in Chapter Three), it must be cured under temperature and pressure, typically in an autoclave. During this process, chemical reactions occur in the resin in which molecules are cross-linked, forming a three-dimensional network of strong covalent bonds. (A cured thermoset composite piece is hard to the touch, much like shellacked wood.) Once cured, a part cannot be reprocessed. Most thermoset materials must be stored frozen; after a limited time out of the freezer at room temperature, they start curing, thereby losing properties, becoming difficult to form, and

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<sup>12</sup>Other thermosets include phenolics, cyanate esters, and polyimides. Phenolics are used in aircraft interior features for their flame resistance. Cyanate esters are used in rocket motor cases for their low moisture absorption and electrical properties. Polyimides have limited high-temperature applications.

eventually becoming useless. These out-of-freezer times are between one and four weeks. Thermosets also have a shelf life in the freezer of only 6 to 18 months.

Epoxies are the most commonly used thermoset resins in aircraft structures, with toughened epoxies used for high-performance applications. For example, the epoxy 977-3, shown in Table 2.2, is toughened. Toughening agents such as thermoplastics and rubber are added to counteract brittleness.

BMIs are used in higher-temperature and toughness applications; the resin 5250-4 in Table 2.2 is a toughened BMI. BMIs available in the 1980s were harder to work with than were epoxies and were thus associated with manufacturing labor penalties. As illustrated in Chapter Four, today's BMIs have improved handling qualities, and their cost differential has thus diminished.

### **Thermoplastic Materials**

Thermoplastics are the other type of polymer matrix used in aircraft composites. One important potential advantage of thermoplastics is that after a thermoplastic part is formed, it can be reformed through the application of heat and pressure. Thus, if parts are defective on first try or are later damaged, they can be repaired rather than scrapped. This is because thermoplastics do not undergo any permanent chemical transformation akin to the molecular cross-linking that occurs in thermosets during autoclave processing. PEEK (polyetheretherketone) in Table 2.2 is a thermoplastic, as are PEKK (polyetherketoneketone), PEI (polyetherimide), and LCPs (liquid crystal polymers).

Another advantage of thermoplastics is that they are solid at room temperature and can be stored without refrigeration, thus offering a virtually indefinite shelf life. Moreover, thermoplastics offer high toughness and impact resistance. However, this advantage was greatly eroded with the advent of increased-toughness thermosets such as 977-3 epoxy and 5250-4 BMI, whose toughness is similar to that of thermoplastics.

The primary drawback of thermoplastics lies in their high cost, which is due primarily to difficulties in manufacture. Thermoplastics have

poor handling qualities at room temperature, which makes laying them up difficult and therefore time-consuming.<sup>13</sup> In addition, some thermoplastics require solvents to allow the material to be worked into the desired shape. During the autoclave process, these solvents must be removed from the part. Sometimes, however, solvents are not completely removed, causing porosity that greatly compromises the mechanical properties and reliability of the part. Porosity is currently a major problem in thermoplastic part manufacturing and inspection.

Autoclave processing temperatures for most thermoplastics are between 500°F and 700°F.<sup>14</sup> Such temperatures require that production tooling be made from materials with a low coefficient of thermal expansion (CTE) that can withstand the wear from repeated high-temperature autoclave cycles. Tools with these qualities—such as Invar, a high-quality nickel alloy with a desirable CTE—are very expensive. Finally, thermoplastic raw material costs are relatively high compared with those of thermosets.<sup>15</sup>

Both because these manufacturing difficulties have not been overcome since the early 1980s and because some thermoplastic advantages have eroded, thermoplastic applications in airframe structures remain limited. (The relatively low rate of new-aircraft development and production since the end of the Cold War no doubt made the introduction of new materials even less attractive to industry.) Boeing,

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<sup>13</sup>These poor qualities include low drapability and low tack. Drapability is ease of conforming to a complex surface; thus, a ply made of low-drapability material can be laid up on a complex tool only with difficulty, which means that more labor hours are required for manufacturing. Tack is stickiness, so laid-up plies made of a low-tack material tend to separate more easily, leading to voids that later lead in turn to higher susceptibility to delamination. To compensate for these factors, more time must be taken in manufacturing to ensure that plies appropriately adjoin. This extra time includes additional debulking and compaction steps.

<sup>14</sup>Technically, thermoplastics do not “cure” during autoclave processing; they “form.” We will use the term *autoclave processing* when referring specifically to thermoplastics. We will use the term *cure* generically when referring to composites in general.

<sup>15</sup>There may well be a “vicious circle” effect relating high thermoplastic raw material prices to low levels of use. For many industrial products, an expansion of demand leads to incentives for manufacturers to invest in more efficient large-scale processes, which ultimately leads to lower prices. Industry experts often project that such an effect would occur for thermoplastics, although this cannot be known with certainty before the fact.

for example, originally intended its Joint Strike Fighter (JSF) candidate to involve substantial thermoplastic use (e.g., in its wing skins), but as the development program continued, it was decided not to use thermoplastics. This illustrates not only the limited penetration thermoplastics have made but also the tendency to use proven materials in an aircraft as development proceeds (see Figure 2.2). Examples of aircraft that use thermoplastics are the F-117 and F-22 doors and panels (see Figures 2.6 and 2.7), which are susceptible to runway debris (Harper-Tervet et al., 1997).

### **Combining Reinforcement and the Matrix: The Composite Material**

Composite material for aircraft use comes in two basic forms; unidirectional tape and fabric. In unidirectional tape (hereafter simply called tape), all the fibers (i.e., filaments) are aligned lengthwise in the same direction along the tape. A roll of tape is typically 0.005 inch thick, between 0.125 inch and 60 inches wide, and on the order of 1000 feet long. Narrow tape is often referred to as “slit tape.”

A composite part made of tape is typically fabricated by cutting a series of plies from the tape, stacking (“laying up”) the plies on a tool to form the shape of the part, and then curing the part with heat and pressure in an autoclave. For narrower tape, the plies are typically just lengths of tape; for wider tape, they are often cut out with complex shapes. (More detail on the fabrication process is provided in Chapter Three.) If the plies are laid so that all the fibers in the part are aligned in the same direction, the part will have maximum strength and stiffness in that direction and substantially less in others owing to the anisotropic nature of composite material properties discussed earlier. A quasi-isotropic part can also be made with tape by stacking the plies such that one-quarter of the fibers are aligned in one direction—say,  $0^\circ$ —and the other three-quarters are aligned in directions  $45^\circ$ ,  $90^\circ$ , and  $-45^\circ$ , respectively. Figure 2.10 illustrates both a quasi-isotropic and a unidirectional laminate. Since a part is typically made by laying up between 4 and 80 plies, a wide variety of strength/stiffness differentials can be achieved. As will be discussed further, this property is one of the critical advantages of composites. Fiber can be aligned so that the directional strength/stiffness properties of a part can best meet the loads the part is expected to experi-

ence in flight. Additional layers can be added to specific areas of a part to increase strength locally as well in a process commonly referred to as “planking.” The three composite examples given in Table 2.2 are all unidirectional tape, and the strength and stiffness numbers are maximum values—that is, they are the values that apply if force is applied in the same direction as that in which the fibers are aligned.

The strength and stiffness of a quasi-isotropic part made with tape for given constituent materials (see Figure 2.10) is on the order of one-third to one-half the values unidirectional tape has in the maximum direction. (Of course, for quasi-isotropic fabric, the strength and stiffness values apply for loads coming from all directions, not just two.) Actual relative values vary by material type.

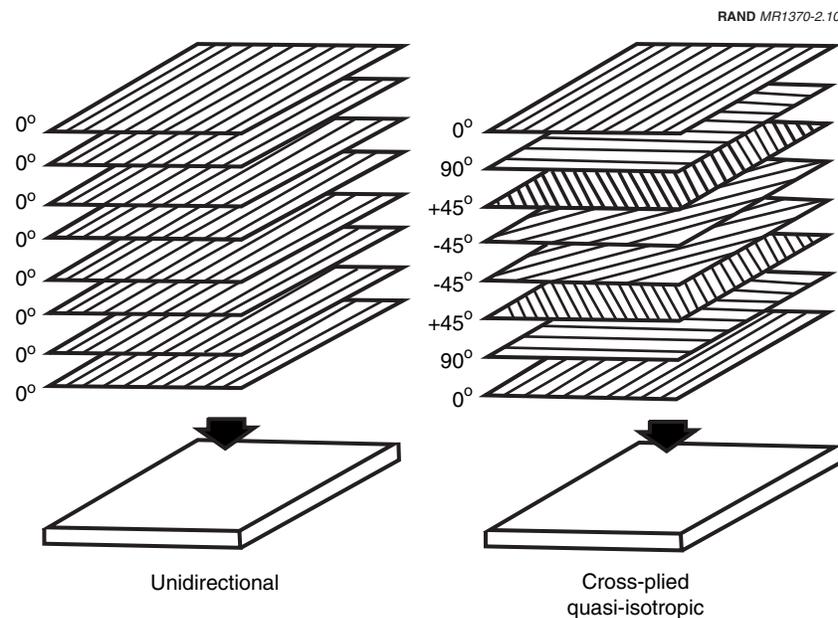


Figure 2.10—Composition of a Quasi-Isotropic and a Unidirectional Laminate

In fabric composite materials, fibers are woven together into a pattern. Fabric can be made in a wide variety of patterns and forms, and there can also be complex three-dimensional braids of fiber in the fabric. Various weave designs can be chosen to achieve different patterns of directional properties, and these designs affect other properties of the part, such as toughness. Typical weave patterns are shown in Figure 2.11.

The decision to use tape or fabric rests on several factors. As discussed above, unidirectional tape leads to structurally efficient part design, which can provide the lowest possible weight for a given part geometry and directional load pattern. It is most often used for parts with mild contours and for larger parts. Fabric has excellent contour capability and is most often used in lightly loaded parts that are small and complex. Three-dimensional carbon fiber braids are often used as preforms for the RTM process described below. Fibers such as glass or aramid can be combined and woven with different types of carbon fibers to improve damage tolerance and to optimize electrical

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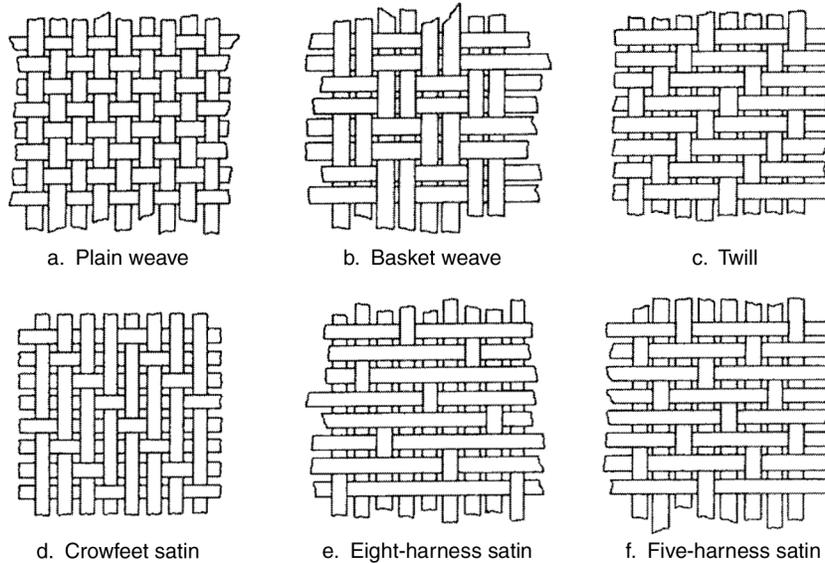


Figure 2.11—Typical Composite Fabric Weave Patterns

conductivity. In both tape and fabric form, fiber is usually about 60 to 65 percent of the volume of the composite material and the matrix about 35 to 40 percent.

The fiber and resin of a composite material can be put together in two ways. In the first, dry carbon fiber is “wetted” (i.e., enveloped in resin) just as the part is being fabricated, shortly before cure. More commonly, prepreg is produced. This is done by applying resin to the fiber, resulting in a combined resin/fiber tape or fabric product (prepreg) that is uncured and must be stored frozen until it is fabricated into parts (with the exception of thermoplastics, which can be stored at room temperature). This tape or fabric is called prepreg because it is impregnated with resin well before it is used to make parts. When prepreg is made, the resin can be applied to the fiber through either a solution or a hot-melt process. In the solution process, the fibers are pulled through a resin bath. In the hot-melt process, a thin resin film is applied to the fiber form and melted onto it.

## **ADVANTAGES OF COMPOSITE MATERIALS IN AIRFRAME APPLICATIONS**

One of the primary purposes of this research project is to estimate the cost of using composite materials in airframe production. As further background for analysts who may use the cost-estimating factors and methods presented in later chapters, we present this account of the considerations that go into deciding whether to use composite materials in an airframe. We begin with the disclaimer that this is nowhere near a complete account of such a complex decision process but is simply an overview of the primary issues.

### **Weight Versus Strength and Stiffness**

As already discussed, the great advantage of composites lies in the fact that their weight is relatively low compared to their strength and stiffness. Table 2.2 clearly shows this advantage. The specific strengths of composites are on the order of eight times higher than those of metals; specific stiffnesses are on the order of four times larger.

To be sure, these composite numbers are for unidirectional tape and thus represent strength and stiffness in the direction for which they are highest. As we have seen, however, quasi-isotropic composites have strength and stiffness in all directions of about one-third to one-half that of the maximum for unidirectional tape, which would still represent an advantage of roughly 3 for specific strength and 1.5 for specific stiffness over metals. All this implies that composite airframe structures can match the performance of metal structures with less weight, which will in turn increase the range, payload, and maneuverability of the airframe.

### **Directionality of Strength and Stiffness**

Another significant advantage of composites is that they can be designed with properties that differ by direction. Many parts are subject to much greater loads in some directions than in others. Thus, to balance the performance of the part, one would want a part that had the differential ability to withstand loads by direction. That is, a reasonable way to characterize a well-designed part would be one for which the ratio of part strength to maximum load encountered would be the same in all directions.

A part made of isotropic material will by its very nature have equal strength in all directions.<sup>16</sup> Most metals are isotropic.<sup>17</sup> A composite part, however, can readily be made to have different strengths in different directions simply by changing the pattern of orientation of the fibers. Thus, a designer who knows the pattern of loads likely to be encountered can design a part so that its directional strength is roughly proportional to the directional stresses expected. If the part is made of unidirectional tape, for example, differential directional strength can be achieved by adjusting the orientation of the fibers that are laid down to form the part. This is often characterized as “aligning the fibers with the load paths.” The more concentrated the

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<sup>16</sup>This does not mean that the part can withstand equal loads in all directions. The maximum load that can be borne is strength times the cross-sectional area perpendicular to the load, and in any part geometry that is not spherical, this cross-sectional area will differ by direction.

<sup>17</sup>Some alloys are not isotropic. Some aluminum-lithium alloys, for example, are not isotropic, posing a disadvantage because the directional properties are not controllable.

directionality of load patterns on the part, the larger the relative advantage of composites because of the ability to directionally tailor their strength. (Everything just said for strength applies to stiffness as well.)

### **Composite Part Design Issues**

The aforementioned design flexibility advantage of composites, however, has an associated disadvantage. Composite parts are inherently more difficult to design because each “part” is really a layered amalgam of several plies cured together into one part. The designer must therefore analyze how different ply alignment patterns will perform under different load patterns and must then choose the number, shape, and alignment of the plies to achieve a best outcome. Not surprisingly, the increase in degrees of freedom in design afforded by composites has led to a complicated and time-consuming design task. (This is true at least if one wants to take advantage of the increased capabilities that composites offer; designing composite parts in exactly the same way one would design metal parts has been disparagingly characterized as treating composites as “black aluminum.” This practice generally results in parts with additional weight and no significant increase in performance.) Instructions must also be prepared for the manufacturing department on how to cut and assemble plies. Creating such instructions is a complex process that lies well beyond the basic specification of part geometry.

Finally, additional airframe design complications arise when metal and composite parts are in proximity, since there are some incompatibility problems between materials. For example, galvanic corrosion problems can arise if aluminum and carbon fiber composite parts touch in the presence of an electrolyte. In such cases, the designer must either ensure that a protective coat of some material is between these parts or employ some other design practice to resolve the problem. These issues also add to the time required to develop airframe designs with composites.

### **Part Complexity and Design Automation**

Over the past 20 years, the process of aircraft design has undergone a process revolution in which automated design tools have almost

completely replaced older manual techniques. Parts and whole aircraft are now represented three-dimensionally in aircraft design computer tools such as Unigraphics and CATIA. Among other things, these design tools reduce the need for physical mock-ups to ensure that individual subsystems fit together. Such tools can be used to send digital instructions to many kinds of machines on the factory floor, including numerically controlled milling machines, composite ply cutters, and optical laser ply alignment systems (see Chapter Three). The whole set of such computer systems is referred to as computer-aided design and computer-aided manufacturing (CAD/CAM). It has been argued that this increase in automated design capability should decrease the *relative* effort required to design composite parts—that is, the effort that must be expended in relation to that required to design metal parts.

We will specifically address this issue when we discuss our cost results in Chapter Four. Here, however, we make one simple observation. One would also have expected these new technologies to have decreased all absolute costs of aircraft design and development. However, there is virtually no evidence that aircraft development costs are falling; in fact, there is substantial evidence to the contrary. Yet credible reports indicate that the CAD/CAM revolution has reduced the cost of certain specific tasks.<sup>18</sup>

How can these two observations be reconciled? Our discussions with industry indicate that rather than being used to design aircraft more cheaply, CAD/CAM tools are being used to design aircraft better. In effect, far more analyses are being done for new aircraft than was the case in the past, including more analyses of loads encountered under varying flight conditions (“load cases”) and more detailed and accurate modeling of underlying structural phenomena in aircraft, such as load paths. In addition, these tools allow for the integration of manufacturability and supportability in the design process, thereby adding more steps and time to the design phase. In short, the same number of engineers or more are being used to design an aircraft,

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<sup>18</sup>For example, the elimination of hardware mock-ups was estimated to have saved the V-22 program 150,000 man-hours in engineering and manufacturing development (EMD) (Dougherty and Liiva, 1997). Some industry estimates indicate that up to 60 percent savings in design-to-build information release time has been attained as a result of CAD/CAM systems.

and each is more productive owing to CAD/CAM. Thus, as a result of CAD/CAM, we are getting safer, more efficient, and higher-performance aircraft today than was the case 20 years ago. In addition, aircraft should in theory be more producible and supportable at the end of development, thus lowering the need for design changes during production. The military aircraft industry has therefore derived gains from CAD/CAM in the form of better aircraft rather than cheaper aircraft of unchanged quality.

### **Composite Unitization**

Another advantage of composites in design is that they lend themselves to unitization—that is, to the substitution of a larger, more complex integrated part for several smaller ones that must be fastened together into a subassembly. Unitization saves on the weight of fasteners and on the time required to assemble the subassembly, including the time needed to shim, attach fasteners, and inspect connections. In addition, the holes associated with fastening parts together are inherently weaker than integral structure and more susceptible to cracks and other damage.

Unitization can be achieved with composites in four ways. First, composite layup techniques can form relatively complex shaped parts that in standard metal design practice would have been made of several subparts. Inlet ducts are a good example of this. Second, unitized parts can be made by “cocuring”—a process in which two or more newly formed parts are cured together at the same time and are held together under pressure during the cure such that they physically join to become one part. Third, composites can be cobonded. When a newly formed part is about to be cured, it can be held with pressure against an already fully or partially cured part while in the autoclave, causing the parts to be chemically bonded together while the first one cures. Finally, composite parts can be adhesively bonded to each other more readily than can metal parts. (Whether this is really unitization is debatable, but it does do away with fasteners and holes.) An example is adhesively bonded honeycomb sandwich parts.

The unitization advantages of composites have been eroded somewhat, however, by the advent of high-speed machining of metals (described in Chapter Three). This technique lowers the cost of uni-

tization of metal parts and makes possible more extensive unitized metal structures than can be made with conventional machining techniques.

## **OTHER CONSIDERATIONS IN USING COMPOSITES**

### **Knowledge Base for Composite Materials**

One factor to be considered in using composite materials in airframes is their relative newness. (And newer materials are continuously being introduced.) Knowledge about the properties of composites is not as complete as is knowledge about metals. Thus, a more conservative approach is taken in generating “design allowables.”

A part is designed as though its strength is only some percentage of its actual estimated strength to provide a margin of safety against uncertainties in both material properties and loads encountered by the part. For example, if a part is designed as though its strength is only 50 percent of its actual estimated strength, it is said to be designed with an “allowable” of 50 percent. In aerospace applications, these allowables are between 40 and 90 percent of the strength of the part, depending on the nature of the loads expected to be encountered. The newer a material, the less confidence one has in its properties, and hence the lower (more conservative) the allowable. The same principles apply to other mechanical properties, such as stiffness: The more conservative (i.e., the lower) the allowable, the heavier and thus more expensive the part must be for any given load pattern expected.

Initial estimates of the mechanical properties of a material are made through a series of tests conducted on many samples of material. As many as 10,000 test samples are required to develop an initial database to support the design of airframe structural parts. These samples are called “coupons,” and the tests are called “coupon tests.” These tests are also used to determine accept/reject criteria for part inspection after manufacture. For composites, this is done through “effects of defects” analyses in which defects—for example, voids or delaminations—are deliberately introduced into coupons. The resulting changes in material properties are then assessed.

Introducing new materials thus increases costs for two reasons, regardless of the cost of manufacturing or using the material itself. The first is the cost of the testing process, and the second is the weight penalties that must be accepted in the early stages of material use, when allowables are very conservative (i.e., low). As time goes on, these costs fall as the initial large testing effort is completed and data are compiled that can be used for later designs. In addition, allowables become less conservative as test data and production and flight experience increase confidence both in the average properties of the material and in the limits on variability of that material from part to part.

An extreme case of adverse cost impacts early in a material's development occurs when, as sometimes is the case, a dual-aircraft design path is chosen—that is, when aircraft sections are designed both with and without the new material in the event that the expected (or hoped-for) performance of the material is not attained. This obviously increases certain design costs.

These knowledge base problems are not of major importance with carbon-epoxy because the industry has accumulated a great deal of experience with that composite. Carbon-thermoplastic composites, on the other hand, have had little flight experience, especially in safety-of-flight-related applications, and are still relatively new in this respect. Carbon-BMI falls in between and is just now becoming widely used on aircraft.

### **Failure Modes**

One part of generating information about materials lies in the analysis and understanding of how and why parts fail. In airframe structures, corrosion, fatigue, and in-service damage cause failures. The analysis of these causes of failure in metals has been occurring for decades and, while certainly not completely understood, is ahead of similar analysis for composites.

Properly designed composite parts resist fatigue and corrosion better than do aluminum and steel, but composites fail in one way that metals do not: delamination. Since composites are laminates—that is, are produced by the curing together of a stack of plies (laminae)—they can fail through the separation, or peeling apart, of the plies.

Even small amounts of such delamination can substantially reduce mechanical properties. More will be said below about techniques for testing for minute delaminations and other kinds of voids in a part, which can eventually lead to larger and more serious delamination. One basic issue, however, is that these small initial delaminations or voids can be difficult to detect, and unless inspection techniques are very good, the first sign of a problem could thus be complete failure of the part. Both battle damage and peacetime accidents (such as dropped tools) can also induce delamination. Analysis of the causes of, indicators for, and propagation processes of delamination thus continues.

### **Tooling**

Composite parts are typically laid up on a tool that helps form the part and hold its shape while curing. This tool must therefore withstand repeated heat and pressure cycles of the autoclave cure and must not lose its often stringent dimensional tolerances as a result of these cycles.<sup>19</sup> It must also have thermal expansion properties that do not lead to distortion of part shapes.<sup>20</sup> These two qualities often require very expensive tools for two reasons. First, the raw materials for the tools are sometimes very expensive, especially Invar. Second, they are often quite difficult to work with, leading to high tool fabrication and maintenance costs. Depending on the specific application, production tools used to make composite parts can be made of metals, ceramics, carbon, or high-quality composites. Chapter Four presents our estimates of relative tooling costs for various materials.

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<sup>19</sup>This criterion is relevant for “production,” or “hard,” tooling, which is typically designed for aircraft production runs of 100 or more. “Soft” tooling, which is associated with much less durability, is sometimes used in low-production-run programs such as experimental, concept demonstrator, or prototype aircraft. It is much less expensive and often uses less durable material, such as aluminum or composites.

<sup>20</sup>This means that the CTE of the tool and the CTE of the part must be close enough that differential expansion during autoclave heating and cooling does not distort the part.

## **Nondestructive Inspection and Test (NDI/T)**

One of the critical steps in composite part fabrication is nondestructive inspection for defects. Composite part defects include porosity, ply delamination, cracks, and foreign object inclusion. NDI/T methods are also used to verify composite subassembly joint and bond integrity. The two main techniques of inspection are noninstrumented and instrumented.

Noninstrumented techniques include visual inspection and the coin-tap method. The coin-tap method literally involves tapping a coin or a special hammer on a laminate and listening for variations in sound, which indicate a void or other material nonuniformity. These two methods are effective and inexpensive but are limited in terms of the types of defects they can detect.

A number of sophisticated instrumented means of inspecting composites have also been developed. These methods include ultrasonics, X-ray, infrared (IR) thermography, laser shearography, and laser ultrasonics. The primary method used today is ultrasonics with either the through-transmission ultrasonic (TTU) or pulse-echo technique. In TTU, sound pulses are passed through the part, and the signal received is compared to that received when the same test is performed on a part known to be defect free. This technique requires access to both sides of the part. The pulse-echo method uses the same principles but requires access to only one side of the part, since the reflected echo of the transmitted sound is the signal received. The X-ray inspection method has limited application in detecting foreign objects in a part. It can be performed relatively quickly but requires expensive equipment, including lead shielding.

The other three methods, which have been developed more recently, are still being refined and are thus not yet in widespread use. In IR thermography, the part is briefly heated, and the resulting temperature differences across the part are interpreted to indicate defect areas (McDonnell Douglas, 1997). In laser shearography, stress is applied to the part, and illumination by a laser produces an image that reveals flaws (McDonnell Douglas, 1997). In laser ultrasound, a laser is used to introduce a sound pulse into the part, which is then analyzed using the ultrasound methods described above. If successful,

this method will reduce NDI time, especially for complex and large skins (Drake, 1998).

## **METALS**

This section offers a brief overview of metal properties as a complement to that on composites. The three primary metals used in aircraft are aluminum, titanium, and steel. Chapter Four discusses their cost implications.

### **Aluminum**

Conventional aircraft-grade aluminum is used where strength requirements are moderate. Historically, aluminum has found extensive use in the airframe industry owing to its comparatively low weight, low raw material cost, good thermal properties, extensive manufacturing experience, and extensive database, which lead to high-confidence design allowables.

Although aluminum-lithium alloys have not yet made any significant penetration in airframe structural applications, newly developed alloys are much more promising. Current aluminum-lithium alloys can provide improvements in specific stiffness and strength as well as in fatigue and corrosion resistance over conventional aluminum alloys. Moreover, the mechanical and processing properties of current aluminum-lithium alloys are considerably superior to those of previous versions. (Older aluminum-lithium alloys often had such high directional property variation that the average weight and strength advantages were negated. Another problem with older aluminum-lithium alloys was raw material batch-to-batch inconsistency.) However, aluminum-lithium machining chips must be segregated from normal aluminum chips and more carefully disposed of. Aluminum is normally recycled into products such as beverage cans, but aluminum-lithium cannot be used for such purposes owing to environmental and health concerns about lithium. This raises the costs of using aluminum-lithium.

## Titanium

Titanium has excellent heat and corrosion resistance and is stronger than aluminum. Its primary drawback is cost; the raw metal itself is five to seven times as expensive as aluminum, and more labor hours are required per pound to fabricate it (see Chapter Four). Titanium is used extensively in military airframe aft fuselages by virtue of the need to withstand engine exhaust temperatures, and it is also used where strength is a key property. Titanium's mechanical properties, such as strength and stiffness, are more compatible with those of composites than are aluminum's, so titanium is used for applications in airframe substructures that are part metal and part composite. Titanium prices have risen in recent years owing to commercial uses in sporting goods such as golf clubs, bicycle frames, and the like. The producer price index (PPI) for titanium rose 56 percent between 1987 and 1999, while the PPI for all metals rose only 16 percent over that time period.

Titanium alloys with small percentages of aluminum, vanadium, and other metals are commonly used in place of pure titanium in airframe structures. Titanium 6-4 (Ti-6Al-4V) is a common alloy used in airframe structures.

## Steel

Steel application in military aircraft structure is limited. It is the material of choice where very high strength is required—for example, in fasteners,<sup>21</sup> landing gear, arresting hooks, and spindles for horizontal stabilizers. It has superior strength but very high density, so steel parts are very heavy.

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<sup>21</sup>Sixty percent of the steel used in the F-16 airframe structure is in fasteners.