10 CHAPTER From Introduction to Flight6e

Flight Vehicle Structures and Materials

Forces cannot be seen, but the effects of forces can be seen or understood. It is the effect of a force that is of interest in structural analysis.

Frederick K. Teichmann, Professor of
Aeronautics and Astronautics,
New York University, 1968

10.1 INTRODUCTION

In the grand scheme of flight vehicles, the consideration of structural design and analysis plays a special role. No matter how good the aerodynamics, or how powerful the propulsion, or how spectacular the flight dynamics, if the vehicle does not structurally hold together, then all is for naught. Samuel Langley learned this the hard way, as the disaster shown in Fig. 1.21 attests. Return to the road map for this book shown in Fig. 2.1, and read again the discussion surrounding this road map. In our "introduction to flight" we are almost at the end of the map—at the box labeled *Structures*.

The subject of flight vehicle structures is wide and rich. In this chapter we only scratch the surface. The purpose of this chapter is simply to give you a perspective on structures relative to the other flight disciplines shown in Fig. 2.1.

PREVIEW BOX

This chapter introduces yet a new subject—it is another fresh start. The discipline of flight structures is one of the four classic components of the flight vehicle system, the others being aerodynamics, flight dynamics, and propulsion. The latter three have been treated extensively in this book, but this chapter is much shorter and thinner in coverage. Why? Because we are reaching a length constraint for this book. This does not mean that structures are a less important subject. Indeed the study of flight structures is so important that numerous books exist about the subject, and most aerospace engineering curricula require not one but two consecutive structures courses.

That being said, we note that the first three editions of this book did not address structures at all. This chapter was added to provide a perspective, although very brief, of this important subject. Its purpose is to open your mind to the discipline.

In the early days of flight, most accidents were caused by engine failure, and more catastrophically structural failure. For example, the famous Notre Dame football coach Knute Rockne was killed in 1931 in the crash of an airliner when its wing totally collapsed in flight. Modern flight vehicles today are much safer, in part due to progress in the understanding and design of flight vehicle structures. This chapter gives you some insight about such understanding. How do you ensure that, as you maneuver your airplane, the wing does not crumple or the tail does not fall off? These are truly important, showstopping considerations. Reading this chapter will give you some basic answers.

Uninhabited combat aerial vehicles (UCAVs) are now coming on the scene, as discussed in Sec. 6.20. The design space for UCAVs is open to much higher accelerations than vehicles with humans on board, greatly increasing the load factor on the vehicle. How do you deal with such increased *g*-forces on the vehicle structure? How do stress, strain, and material fatigue enter the picture? What different materials are used for flight vehicle structures? How do you go about choosing them for a particular design? This chapter will give you some insight for the answers. It will also give you the appetite to read more about the subject. Read on, and enjoy.

This chapter will be brief; the road map shown in Fig. 10.1 has only three destinations, but they are sufficient to introduce the subject. The book by Raymer listed in the bibliography has a nice review of structures, and much of the following material is gleaned from this source.

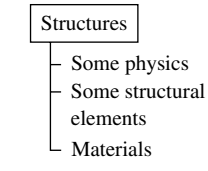


Figure 10.1 Road map for Chap. 10.

10.2 SOME PHYSICS OF SOLID MATERIALS

Our previous discussions of aerodynamics were built on the science of fluid dynamics. A study of flight vehicle structures is built on the science of solid mechanics. In this section we introduce some of the basic physics associated with forces exerted on solid materials and their response to these forces.

10.2.1 Stress

When a force is impressed on a solid material, it is transmitted through the material, much as an externally applied pressure is transmitted through a fluid. This behavior is a molecular effect. The molecules of the solid material, being closely packed together, experience strong intermolecular forces. When an external force is applied to a solid, the shape or size of the solid tends to change.

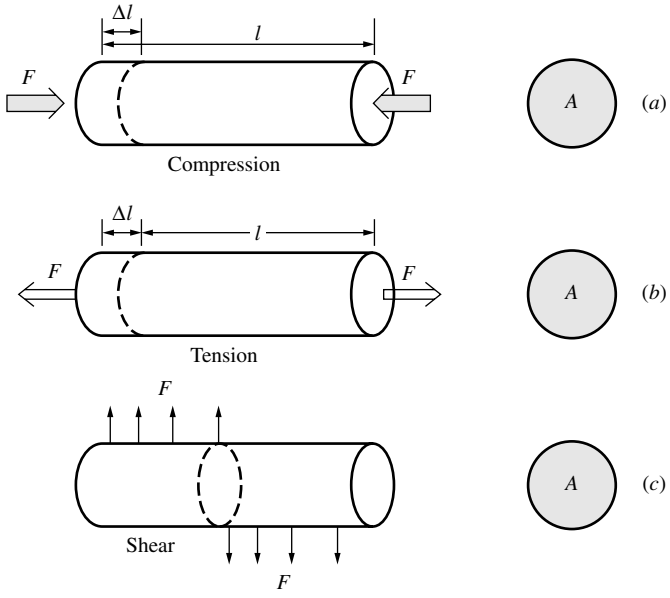


Figure 10.2 Three forms of stress.

The molecules of the solid material, being so closely locked together by the intermolecular force, resist this change in the form of an internal force. The molecules become slightly displaced, enough that the internal force builds up and reaches an equilibrium with the external force.

This internal force, per unit area, is called *stress*. There are three general classes of stress, as follows.

Compression Consider a segment of a solid rod with an external force F imposed on one end, as shown in Fig. 10.2*a*. The force acts in a direction into the rod. For the rod segment to remain in equilibrium, there must be an equal and opposite force F imposed on the other end, as shown in Fig. 10.2*a*. This force is pressing into the segment; by definition, it is a *compressive force*. The cross-sectional area of the rod is A. By definition, the stress, denoted by σ, is

$$\sigma = \frac{F}{A} \tag{10.1}$$

The dimensions of stress are force per unit area, such as pounds per square inch (psi or lb/in^2), pounds per square foot (lb/ft^2), or newtons per square meter (N/m^2). This compressive stress acts perpendicular to the cross-sectional area A.

Tension Consider the segment of a rod shown in Fig. 10.2*b*. Here the force F is acting away from the segment on both ends. This is called a *tensile force*. The stress in tension is also defined by Eq. (10.1). The tensile stress, like compressive stress, acts perpendicular to the cross-sectional area A.

Shear Consider the segment of a rod shown in Fig. 10.2*c*. Here an external force F acts in the upward direction over the left part of the rod, and an equal and opposite force acts in the downward direction over the right part of the rod. These equal and opposite forces tend to slide the right part of the rod relative to the left part. This creates a *shear stress,* denoted by τ, defined as

$$\tau = \frac{F}{A} \tag{10.2}$$

In contrast to compressive and tensile stress, which act perpendicular to the cross-sectional area A, shear stress acts *tangentially* to the cross-sectional area A. A classic example of shear is two plates fastened by a rivet, shown in Fig. 10.3. The force F acting to pull the plates apart creates a shear stress in the rivet.

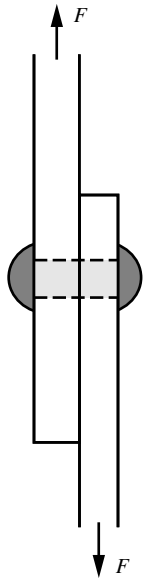


Figure 10.3 Rivet in shear.

10.2.2 Strain

When a compressive, tensile, or shear stress acts on a material, the material tends to change its shape and size. For example, return to Fig. 10.2*a* and *b*. When the compressive or tensile stress acts on the rod, the *length* of the rod changes. Let l denote the length of the rod before the stress acts on it, and let Δl be the change in length after the stress is imposed. By definition, the change in length per unit length is called *strain,* denoted by ϵ:

$$\epsilon \equiv \frac{\Delta l}{l} \tag{10.3}$$

For most materials, up to a certain limiting value of the stress (the yield stress), the stress is directly proportional to the strain. The proportionality constant is defined as the *modulus of elasticity,* or *Young's modulus,* denoted by E. Hence

$$\boxed{\sigma = E\epsilon} \tag{10.4}$$

Equation (10.4) is called *Hooke's law*. Because strain ϵ is a dimensionless quantity, the units of E are the same as the units of stress σ.

The deformation of the material due to shear is shown in Fig. 10.4, which is a greatly magnified view of the "kink" in the material that would be caused by the type of vertical shearing forces sketched in Fig. 10.2*c*. Note in Fig. 10.4 that the equal and opposite shear stresses acting on the vertical sides of the kinked segment set up a moment that acts in the counterclockwise direction on the segment. For

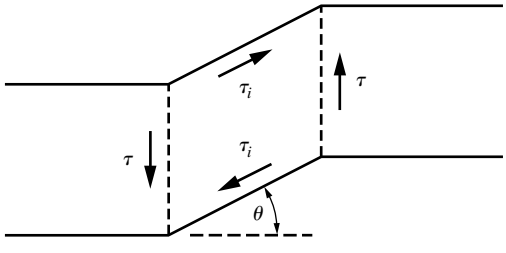


Figure 10.4 Deformation due to shear.

the material to be in equilibrium, an equal and opposite moment must be set up by the induced shear stress τ_i acting on the top and bottom sides of the segment, as shown in Fig. 10.4. The measure of the deformation of the material is the angle θ, shown in Fig. 10.4. The angle θ is called the *shearing strain* and is given in units of radians. As in the case of compression and tension, up to a certain limit, the shear stress is proportional to the strain. The proportionality constant is called the *shear modulus* or *modulus of rigidity,* denoted by G. Hence

$$\boxed{\tau = G\theta} \tag{10.5}$$

10.2.3 Other Cases

There are many types of loads exerted on a structure that cause compression, tension, and shear other than the simple cases illustrated in Fig. 10.2. For example, consider the cantilever beam shown in Fig. 10.5. In Fig. 10.5*a*, the beam is in its natural position—straight and perpendicular to the wall. On the top and bottom surfaces, there are no compressive or tensile stresses. However, when the beam is bent upward by an applied load F, as shown in Fig. 10.5*b*, the top surface experiences a compressive stress and the bottom surface a tensile stress. These stresses are transferred to the juncture with the wall. This juncture must be able to handle these stresses as well as the shear stress at the wall caused by the upward-applied load. The bending case illustrated in Fig. 10.5 is not unlike that of an airplane wing bending upward under the influence of a lift force greater than the weight of the wing.

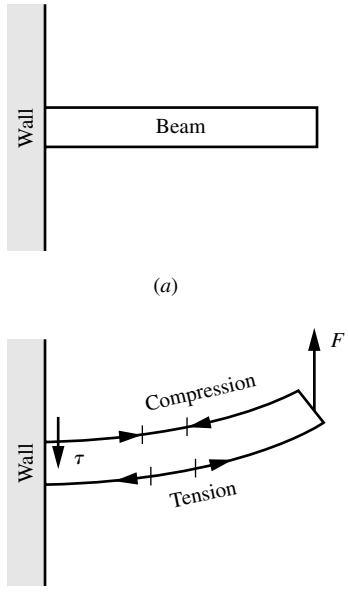


Figure 10.5 Bending of a beam.

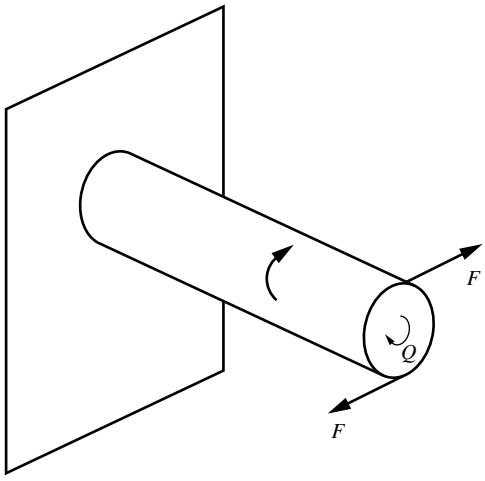


Figure 10.6 Case of torsion.

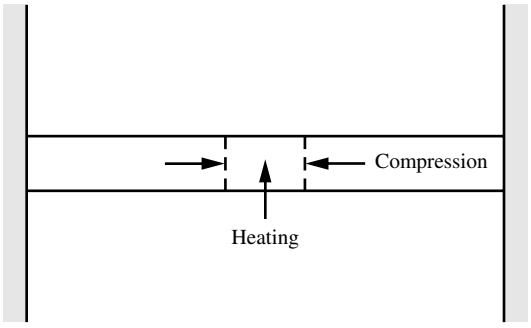


Figure 10.7 Example of thermal stress.

Another case is that of a cantilevered body being twisted by an applied torque, such as that sketched in Fig. 10.6. The torque Q is applied to the body, and this results in a tangential shear stress induced in the body, which resists the torque. This is not unlike the torsion created at the wing root of an airplane by the net aerodynamic moment exerted on the wing.

A third case is that of thermally induced stresses. Consider a body that is totally constrained, such as the beam supported between the two walls sketched in Fig. 10.7. Assume that heat is added to some location on the body. When the material gets warmer, its volume expands (thermal expansion). There is no room for it to move—it is constrained between two walls. As a result, a compressive stress is induced in the material of just the right value to produce a strain that cancels the thermal expansion. If the body is locally cooled instead, the thermal contraction is opposed by a tensile stress induced in the body. These are simple examples of *thermal stresses* induced in the material by local heating or cooling. Thermal stress is an important consideration in the design of supersonic and hypersonic vehicles, where aerodynamic heating of the surface becomes a factor.

10.2.4 Stress–Strain Diagram

A generic diagram that plots stress versus strain for a given material is sketched in Fig. 10.8. As the stress is increased from zero, the strain follows in direct proportion as given by Eq. (10.4) or (10.5), up to a limiting value of stress called the *proportional limit*. This regime in which stress varies linearly with strain is called the *elastic range*. When the stress at any point in this elastic range is relieved, the material returns to its original shape. There is no permanent structural damage. For stress levels above the proportional limit, there is a permanent set

DESIGN BOX

For aluminum alloys, the most common material used in aircraft structures, the ultimate stress is approximately 1.5 times the yield stress. Return to the *V–n* diagram shown in Fig. 6.55. Note the positive and negative limit load factors in the *V–n* diagram. These correspond to stress levels equal to the yield stress. An airplane designed not to exceed the limit load factor will not encounter the ultimate stress until the load factor is 1.5 times the limit factor. This provides a *safety factor* of 1.5—a much smaller value than that for earthbound structures such as bridges, but typical for airplanes, where the saving of weight is paramount. Reflecting on the *V–n* diagram in Fig. 6.55, when an airplane exceeds its limit load factor, some elements of its structure will have a permanent deformation and must be repaired before the next flight.

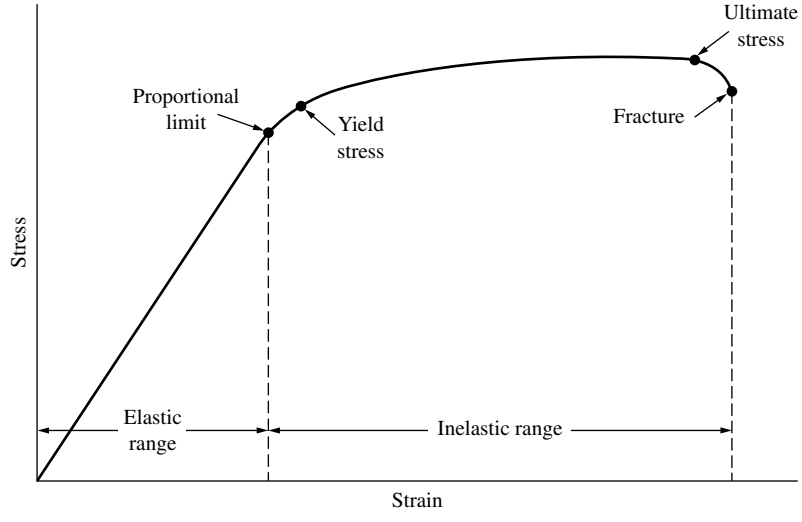


Figure 10.8 Stress–strain diagram.

(a permanent deformation); this is called the *inelastic range,* as sketched in Fig. 10.8. In the inelastic range, when the stress is relieved, the material does not return to its original shape; this constitutes permanent structural damage. The point labeled *yield stress* in Fig. 10.8 is the shear stress at which the inelastic range effectively begins. The yield stress is rather arbitrarily defined as the stress that results in a permanent set of 0.002. Hooke's law, embodied in Eqs. (10.4) and (10.5), does not apply in the inelastic range. Finally, the *ultimate stress,* labeled in Fig. 10.8, is the highest stress the material can withstand. It is associated with a large permanent set, beyond which the material actually fractures. The fracture point is also labeled in Fig. 10.8.

The yield and ultimate stress, as well as the values of the modulus of elasticity E and the shear modulus G, are tabulated here for several materials used in

aircraft structures. In this tabulation, the following symbols are used:

σ_{tu} = ultimate tensile stress
σ_{ty} = yield tensile stress
σ_{cy} = yield compressive stress
σ_{su} = ultimate shear stress

	lb/in^2 x 10^{-3}				lb/in^2 x 10^{-6}	
Material	σ_{tu}	σ_{ty}	σ_{cy}	σ_{su}	E	G
Aircraft steel (5 Cr-M-V)	260	220	240	155	30	11
Stainless steel (AM-350)	185	150	158	120	29	11
Aluminum 2024	61	45	37	37	10.7	4
Titanium (Ti-6Al-4V)	160	145	154	100	16	6.2
Inconel X-750	155	100	100	101	31	11

EXAMPLE 10.1

Consider a rod of stainless steel with a diameter of 0.75 in and a length of 100 in. The rod is in tension with a load of 12,000 lb. How much will the rod elongate under this load?

■ **Solution**

The cross-sectional area of the rod is

$$A = \frac{\pi d^2}{4} = \frac{\pi(0.75)^2}{4} = 0.442 \text{ in}^2$$

The tensile stress is

$$\sigma = \frac{F}{A} = \frac{12{,}000}{0.442} = 27{,}149 \text{ lb/in}^2$$

This is far below the tensile yield stress of 150,000 lb/in^2, as seen from the preceding tabulation. Hence Hooke's law, Eq. (10.4), holds:

$$\sigma = E\epsilon$$

or

$$\epsilon = \frac{\sigma}{E} = \frac{27{,}149}{29 \times 10^6} = 9.36 \times 10^{-4}$$

Because $\epsilon = \Delta l / l$, the elongation is

$$\Delta l = \varepsilon l = (9.36 \times 10^{-4})(100) = \boxed{0.0936 \text{ in}}$$

EXAMPLE 10.2

Consider an aluminum rod with a diameter of 1/4 in. A load of 2500 lb in tension is exerted on the rod. Will the rod experience a permanent set?

■ **Solution**

The cross-sectional area of the rod is

$$A = \frac{\pi d^2}{4} = \frac{\pi (0.25)^2}{4} = 0.049 \text{ in}^2$$

The tensile stress in the rod is

$$\sigma = \frac{F}{A} = \frac{2500}{0.049} = 51{,}020 \text{ lb/in}^2$$

Comparing this stress with the preceding tabulation for aluminum 2024, we find that $\sigma_{ty} = 45{,}000$ lb/in^2 and $\sigma_{tu} = 61{,}000$ lb/in^2. The applied stress of 51,020 lb/in^2 is greater than the yield stress but less than the ultimate stress. Hence the rod will experience a permanent set, but it will not fracture.

10.3 SOME ELEMENTS OF AN AIRCRAFT STRUCTURE

The structural design of an airplane is an intricate arrangement of various structural elements, such as wing spars, fuselage longerons, stringers, and stiffeners. This is certainly highlighted by the cutaway drawing of the B-17 shown in Fig. 2.17. Although a World War II vintage aircraft, the B-17 is a good example of a classical aircraft structure. Examine Fig. 2.17 closely to get a feel for what is meant by the term *aircraft structures*.

In this section we identify the most basic elements of a conventional aircraft structure. Some of these are illustrated in Fig. 10.9, which shows a side view and

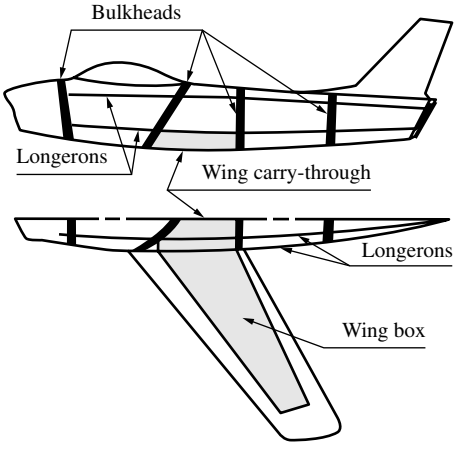


Figure 10.9 Some main structural elements for an airplane.

top view of a generic airplane. The fuselage structure is characterized by *bulkheads,* which form the cross-sectional shape of the fuselage, and *longerons,* which are heavy strips that run the length of the fuselage and are attached to the outer edge of the bulkheads. The fuselage skin is attached to the longerons. The longerons carry to the wings the major loads exerted on the fuselage. They are usually I-shaped or H-shaped extrusions and are heavy elements of the fuselage structure. The primary wing structure is the *wing box,* shown as the shaded area in Fig. 10.9. The large moment created by the lift on the wing is transmitted by the wing box to the juncture with the fuselage, and this moment is carried across the fuselage by the *wing carry-through structure*.

The main element of the wing box is illustrated in Fig. 10.10: the *wing spars,* which are large I-beams that run most of the span of the wing, with heights that reach from the bottom to the top surface of the wing. The spars are basically cantilever beams extending from the fuselage carry-through structure.

Three types of fuselage members are illustrated in Fig. 10.11. A *keelson* is shown in Fig. 10.11*a*; this is a strong beam placed at the bottom of the fuselage, much like the keel on a boat. The keelson is designed to carry the fuselage bending loads and is frequently used in military fighter design. Another type of fuselage structural element is a *stringer,* illustrated in Fig. 10.11*b*. A stringer is a relatively light strip of small cross section that runs the length of the fuselage, and it is attached to the outer edge of the bulkhead. A large number of stringers are distributed over the circumference of the fuselage. Yet another fuselage element is a *longeron,* illustrated in Fig. 10.11*c* and mentioned earlier in conjunction with Fig. 10.9. Longerons are heavier than stringers but lighter than keelsons.

Figure 10.12 shows a scale drawing of a cross section of the Concorde supersonic transport, illustrating one of the bulkheads, a section of the wing spar, and the carry-through structure. Figure 10.13 shows a detail of the wing structure for a small, general aviation airplane, the Robin Hr100. Note the spars that run along the span of the wing, and the airfoil sections that help to form the shape of the wing; these airfoil sections are called the *wing ribs*. Figures 10.12 and 10.13

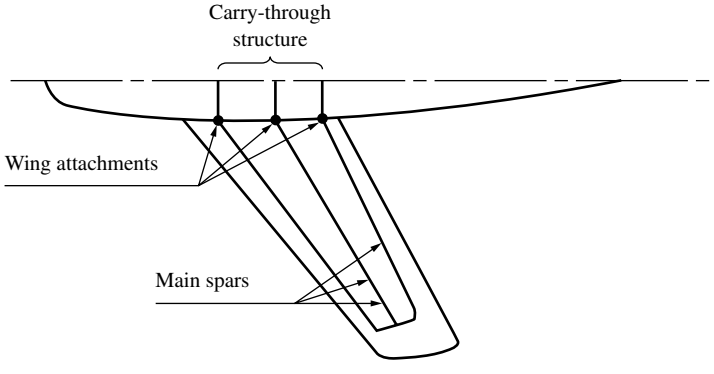


Figure 10.10 The wing box and carry-through structure.

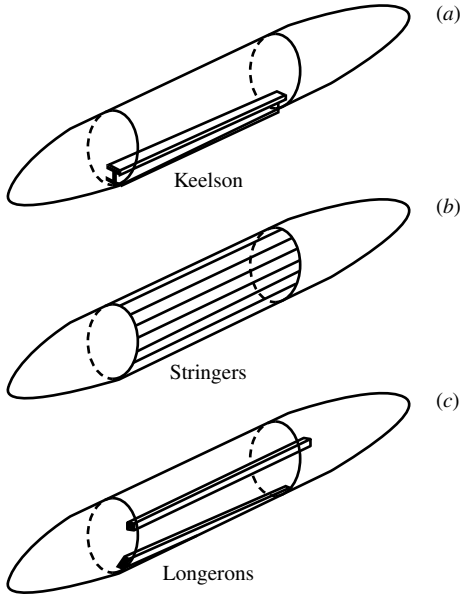


Figure 10.11 Structural elements of the fuselage.

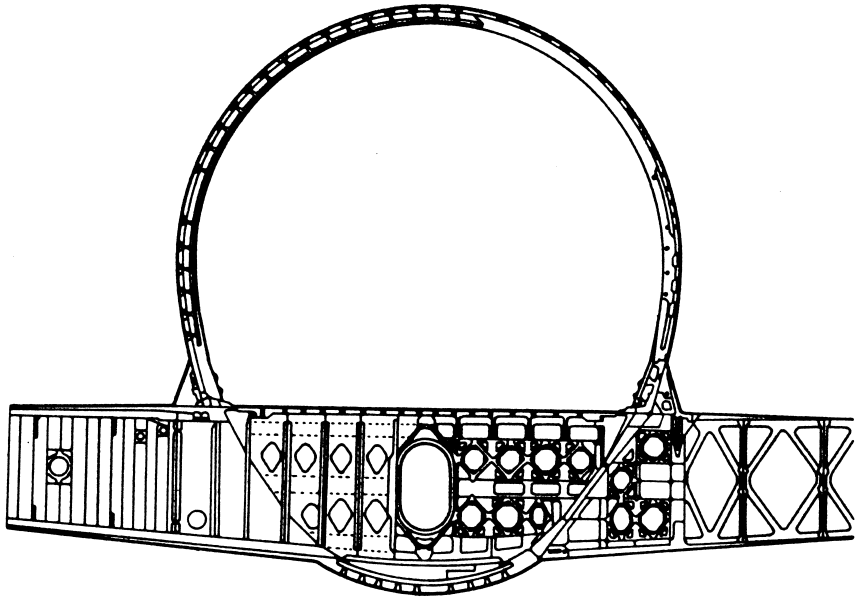


Figure 10.12 A fuselage bulkhead of the Concorde SST.
(Source: *From* The International Encyclopedia of Aviation, *ed. by David Mondey, Crown Publishers, New York, 1977.*)

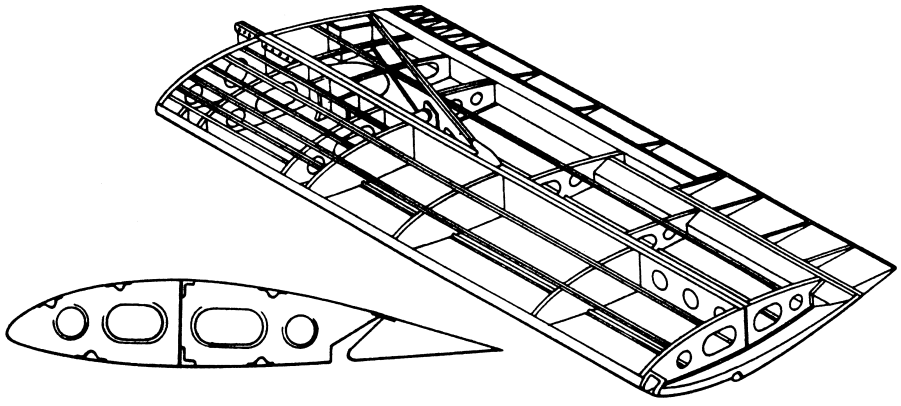


Figure 10.13 Wing structure of the Robin Hr 100 general aviation airplane. (Source: *From* The International Encyclopedia of Aviation, *ed. by David Mondey, Crown Publishers, New York, 1977.*)

give you an idea of what some of these structural members actually look like. Note the cutout regions in some of these structures, such as the holes in the wing ribs—an effort to save weight.

Why does the wing structure shown in Fig. 10.13 look the way it does—a structure built up from spars and ribs? And why does the fuselage structure schematically shown in Figs. 10.9, 10.11, and 10.12, and that shown in detail in Fig. 2.17 for the Boeing B-17 fuselage, look the way it does? In the following subsections we progressively build up answers to these questions.

10.3.1 *Beams*

Consider the stationary beam sketched in Fig. 10.14. The beam experiences a download F shown in Fig. 10.14*a*. It is supported at both ends so that the download is resisted by the upward reaction forces R_1 and R_2 at each end, shown in Fig. 10.14*b*. For the stationary beam, clearly $R_1 + R_2 = F$. Under these loads the beam is bent downward in the middle as sketched in Fig. 10.14*a*. Now consider the face A, which is any cross-section of the beam as sketched in Fig. 10.14*a*. An edge view of face A is shown in Fig. 10.14*c*. Because of the combined effect of the applied download F and the reaction forces R_1 and R_2, there exists a shear stress τ acting parallel to face A as shown in Fig. 10.14*c*. This shear stress tries to deform the beam in the manner shown in Fig. 10.4; that is, if the material of the beam in Fig. 10.14 would not resist this shearing action, the face A would tend to slip as illustrated in Fig. 10.14*d,* as if the beam hypothetically was cut parallel to face A. The shear stress shown acting parallel to the face A is, however, not the only shear stress induced by the applied force F. The beam bends in the manner shown in Fig. 10.14*a,* analogous to the bending action shown in Fig. 10.5*b*. As discussed in Section 10.2.3, the bending beam in Fig. 10.14*a* will experience compression on the top and tension on the bottom. Consider face B inside the beam parallel to the top and bottom surfaces of

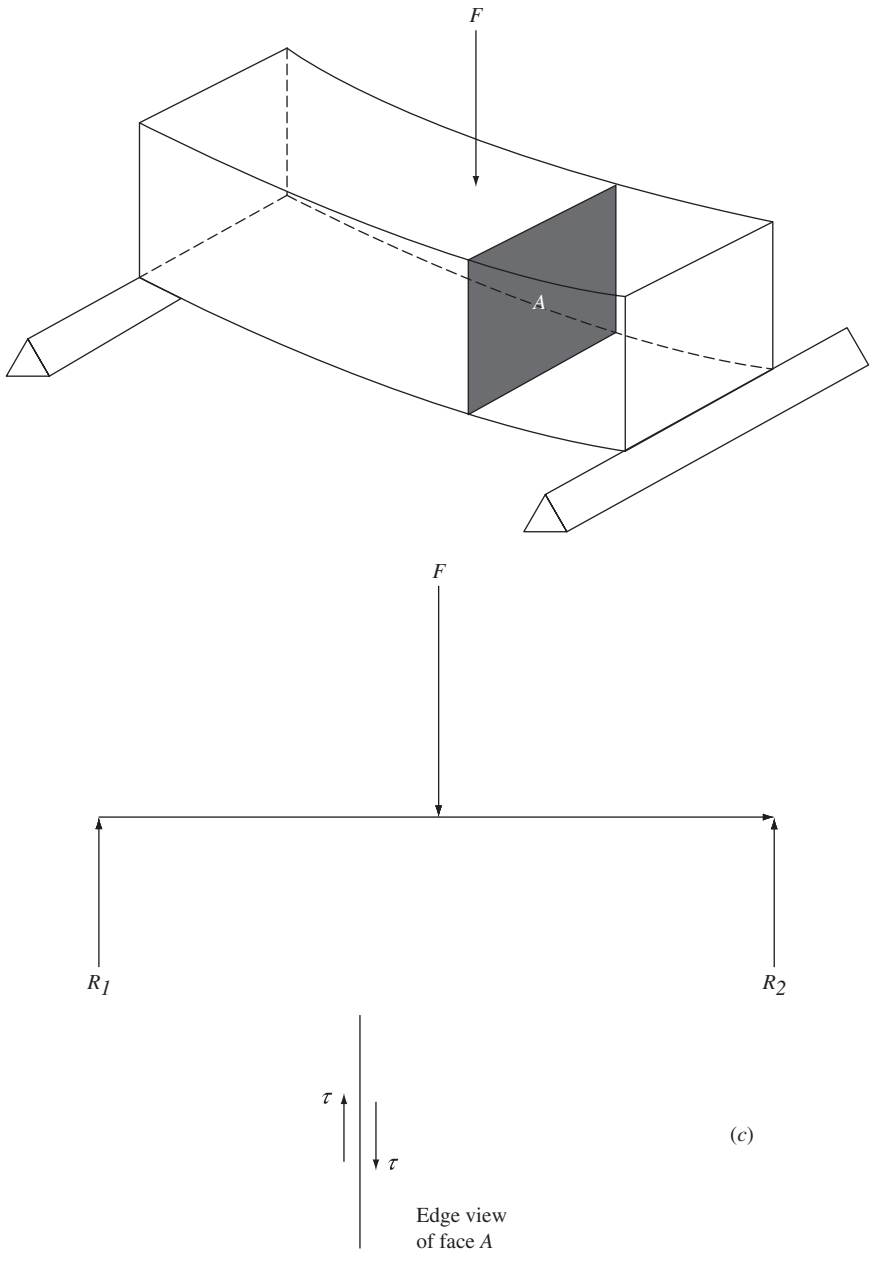


Figure 10.14 Aspects of beam deflection under load. (*a*) The applied load *F*. (*b*) Reactions to *F*. (*c*) Shear stress on face *A*. (*d*) Hypothetical displacement of *A* due to shear stress. (*e*) Face *B* between the top and bottom surfaces of the beam. (*f*) Hypothetical displacement at face *B* due to compression on the top surface and tension on the bottom surface. (*continued*)

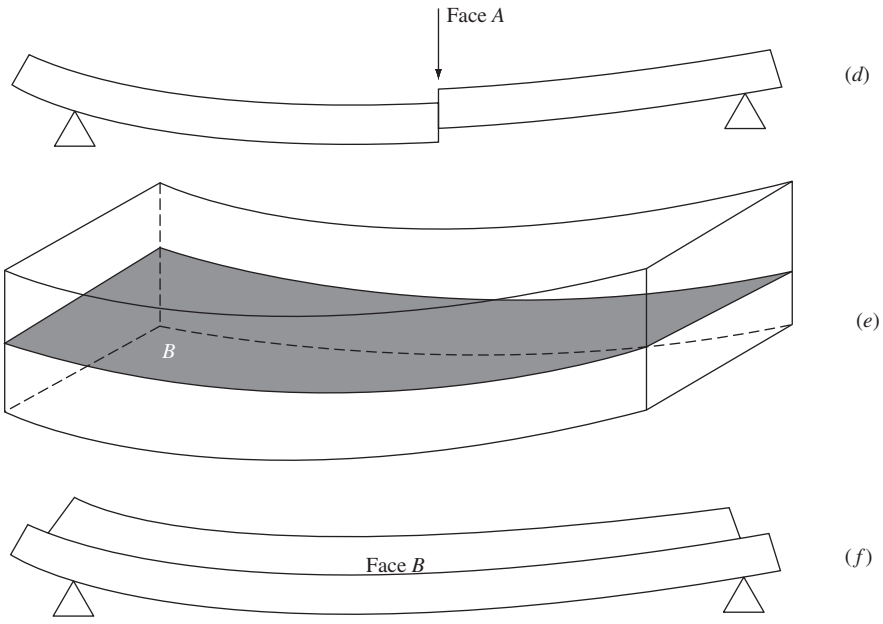


Figure 10.14 (*continued*)

the beam, as sketched in Fig. 10.14*e*. We have compressive stress acting on the top, changing to a tensile stress acting at the bottom, inducing a shear across face *B* parallel to face *B*. If the material of the beam would not resist this shearing action, face *B* would tend to slip as illustrated in Fig. 10.14*f,* as if the beam hypothetically was cut parallel to face *B*. In summary, looking at the beam bending under the influence of the applied load *F* in Fig. 10.14*a,* the top of the beam is stressed in compression and the bottom is stressed in tension. However, the applied load also induces shear stresses inside the beam; these internal shear stresses have a lot to do with the design of the wing structure in Fig. 10.13 and with the structural design of the aircraft in general, as we will see.

The beam shown in Fig. 10.14*a* has a rectangular cross section. Such a beam would be unnecessarily heavy when used as a spar for the wing structure shown in Fig. 10.13, for the following reason. A schematic of the variation of the compressive and tensile stresses across face *A* is shown in Fig. 10.15. The maximum stress occurs at the top and bottom of the beam. Therefore, the cross-sectional area of the beam needs to be larger at the top and bottom but can be smaller in the middle, thus saving weight. The cross section of an I-beam shown in Fig. 10.16*a* is one such example. Furthermore, in the quest to save weight in aircraft structures, structural designers want to make the web as thin as possible, as sketched in Fig. 10.17*a*. Recall, however, that shear stresses are induced in the beam by the applied load, and the thin web carries such shear stresses. Under the applied load, the thin web tends to buckle, and the shear stresses create a diagonal

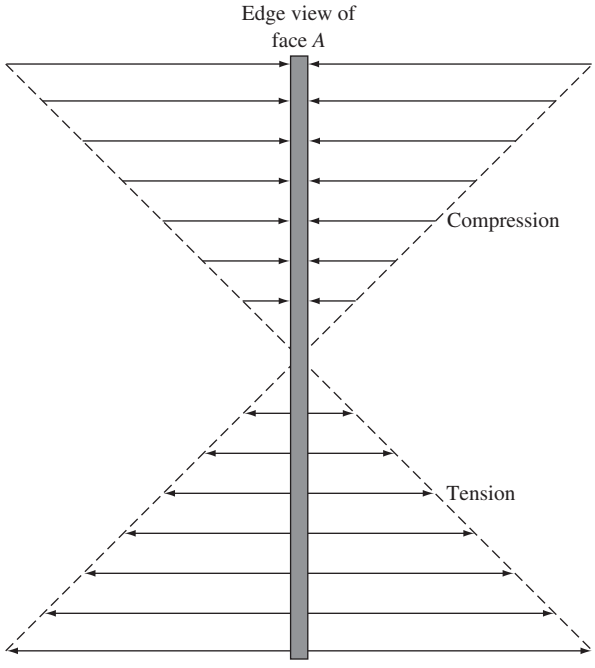


Figure 10.15 Schematic of the compressive and tensile stress across face *A*.

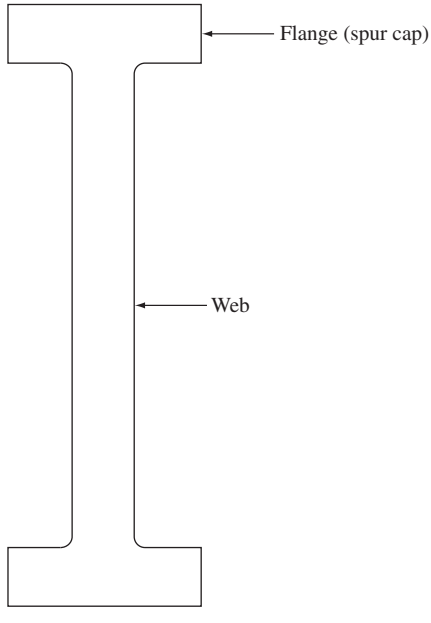


Figure 10.16 I-beam cross section.

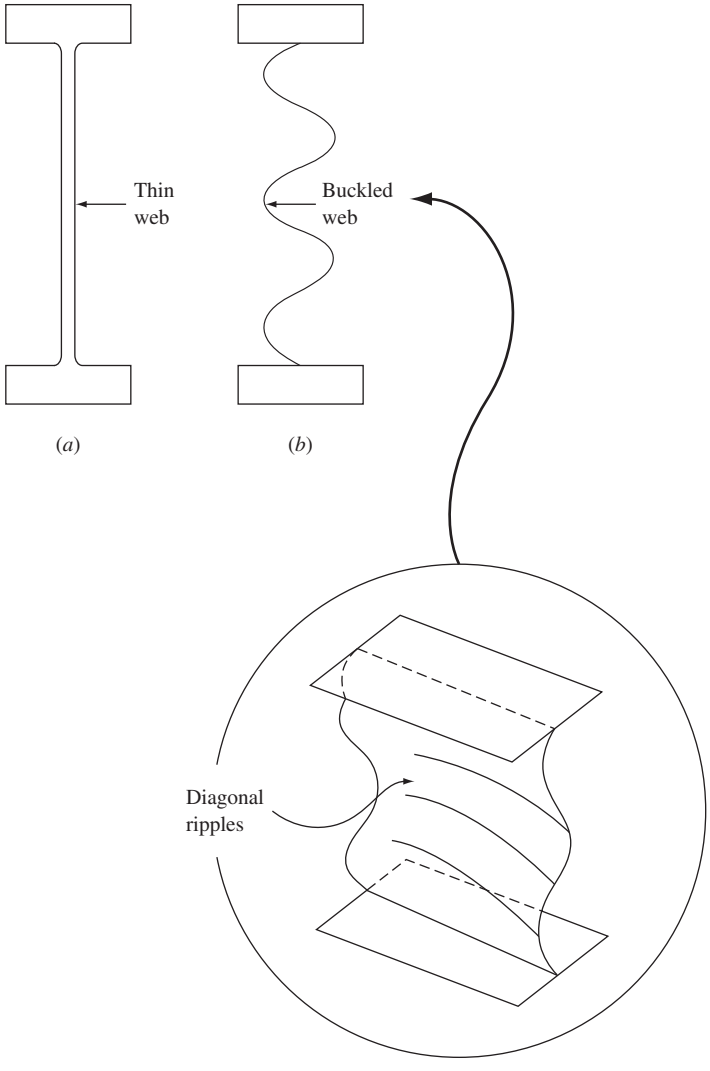


Figure 10.17 Thin web beam and buckling.

wavelike buckling pattern in the web, such as sketched in Fig. 10.17*b*. In the early days of airplane design, before the 1930s, such buckling was considered a structural failure that was to be avoided. Amazingly, however, even after the thin web is in this postbuckling situation, it can still carry a load. This phenomenon was studied by Dr. Herbert Wagner in Germany in 1928, who worked out a diagonal tension field beam theory to explain the behavior of large sheets of thin metal held by the edges. Since the early 1930s the Wagner beam has found much use in airplane structural design, and web buckling is an accepted phenomenon.

Examining the wing structure in Fig. 10.13, we see that the airfoil-shaped ribs, in addition to forming the streamlined aerodynamic shape of the wing, also serve as attachment points at the end of each section of the web of the spar, effectively reducing the spanwise length of each section of web. The Wagner beam is an example of the use of thin *stressed skin*, a concept that revolutionized aircraft structures in the 1930s. We will return to the concept of stressed skin design in later sections.

10.3.2 *Box Structures*

The wing box has been identified in Figs. 10.9 and 10.10, and the essential structural elements of a wing box are shown in Fig. 10.13. Figs. 10.18 through 10.20 progressively build a rationale for the structural design shown in Fig. 10.13. First consider a simple box with sides made of thin sheets of material as sketched in Fig. 10.18. Imagine that the box is a wing with the aerodynamic lift *L,* drag *D*, and moment *M* applied to the box as shown. (Of course the lift, drag, and moments are due to *distributed* loads over the wing surface as emphasized in Chapters 4 and 5. For example, Fig. 5.45 illustrates the *lift distribution* over the span of the wing. For our discussion here, however, we will represent these distributed loads by distinct lift and drag forces and moments

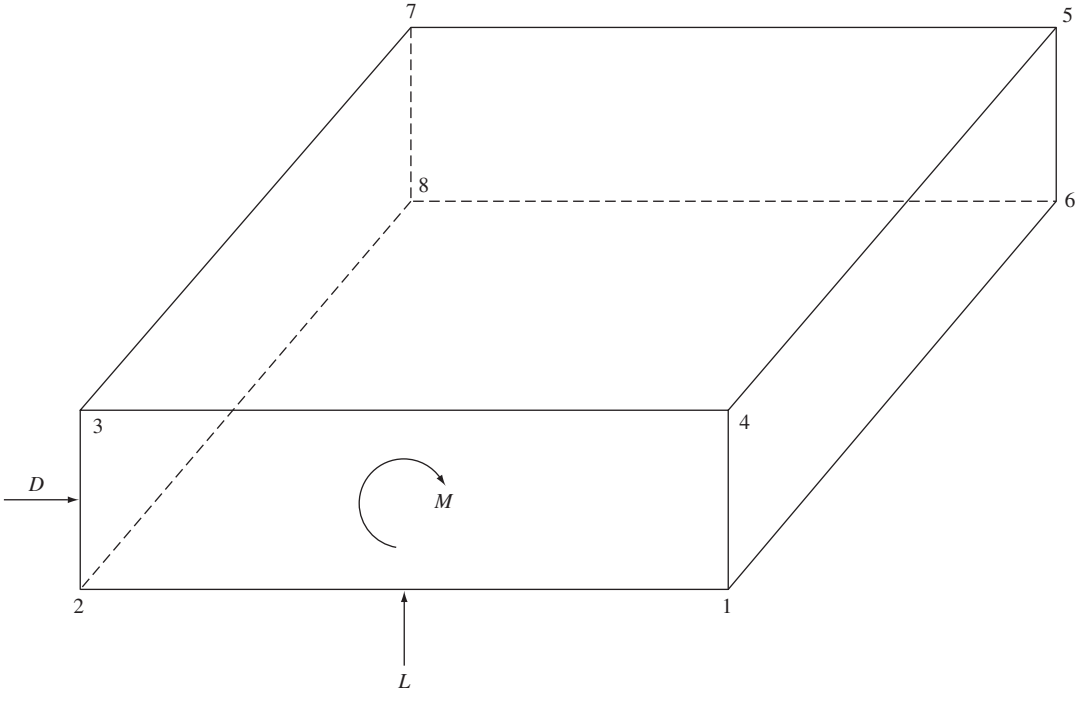


Figure 10.18 Lift, drag, and moments exerted on a simple box structure.

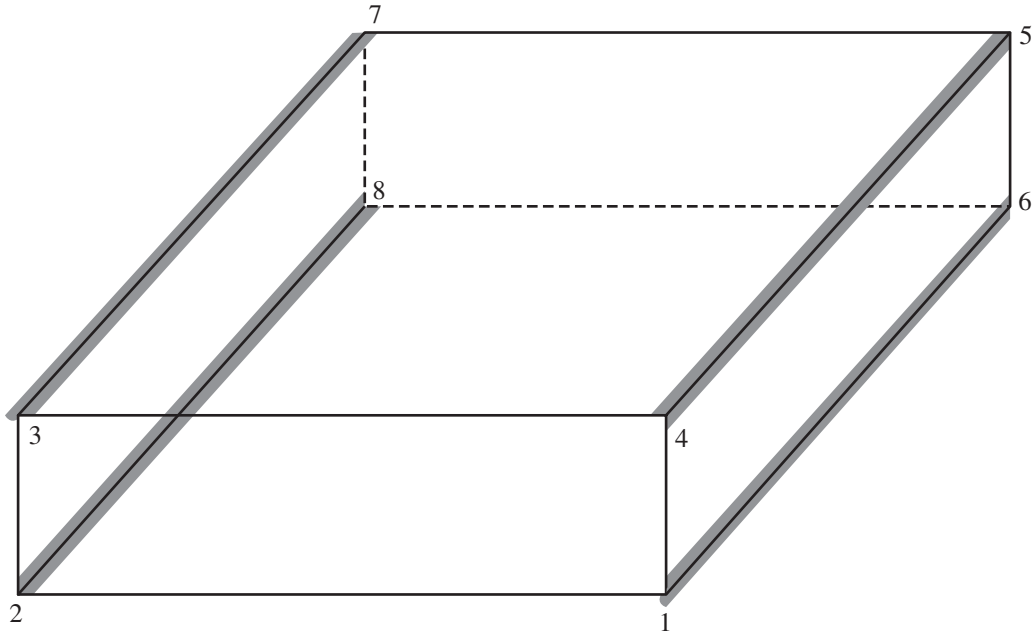


Figure 10.19 Box structure with joints along the long corners.

applied at a point on the wing.) Furthermore, imagine that face 6875 is on the centerline of the airplane, and there is an identical wing box on the other side. For a symmetrical loading over both sides, the position of face 6578 can be assumed to be fixed, just as if it were screwed into a wall. Compression, tension, and shear stresses will be induced on the sides of the box due to the applied loads. For example, consider the effect of lift L. From our discussion in Section 10.3.1, L will induce compression in sheet 4375, tension in sheet 1286, and shear in sheets 1456 and 2378. Similarly, drag D will induce compression in sheet 1456, tension in sheet 2378, and shear in sheets 1234 and 8756. To help carry the compression and tension stresses, extra thickness is added at the corners 1–6, 2–8, 3–7, and 4–5; this can be in the form of joints along the corners, as illustrated in Fig. 10.19.

At this stage in our structural buildup, the box structure in Fig. 10.19 contains some large thin sheets of material: 4378, 1286, 1456, and 2378. These represent the skin of the wing, and they are under stress; that is, Fig. 10.19 represents a *stressed-skin* structure. These large panels will tend to buckle under load. To minimize buckling, stringers can be added in the spanwise direction and ribs in the chordwise direction, as sketched in Fig. 10.20*a*. The stringers and ribs effectively divide the stressed skin sheets into smaller panels that buckle at higher loads, hence effectively delaying the onset of buckling. Also, if the box structure is a wing, the ribs can be airfoil shapes that help to form the airfoil contours of the wing, as sketched in Fig. 10.20*b*.

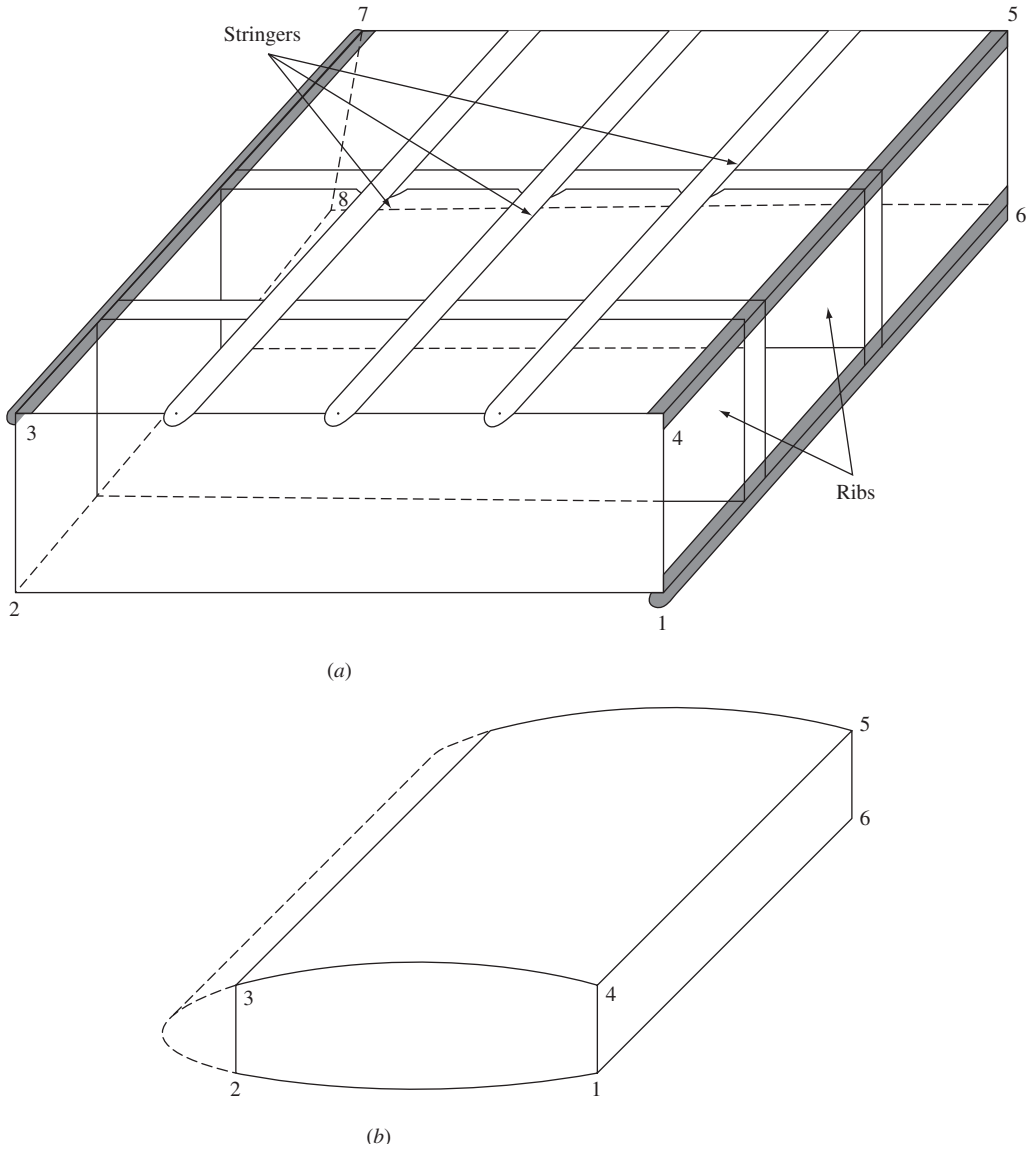


Figure 10.20 (*a*) Ribs and stringers added to the box structure. (*b*) The ribs can have an airfoil shape, as in the case of a wing box structure.

With Fig. 10.20, we have finished our elementary structural buildup of the wing box. Returning to the wing structure in Fig. 10.13, we can easily identify the spar, struts, and airfoil-shaped ribs. Moreover, the whole wing structure is covered with a thin metal skin that carries a substantial stress (a stressed skin). From the previous discussion, we now have a qualitative understanding of the role played by the structural elements shown in Fig. 10.13.

The box structure sketched in Figs. 10.18 through 10.20*a* can also represent a fuselage, in which case L represents the lift force on the horizontal tail, D represents the side force on the vertical tail, and M is the moment induced by D because the side force is usually located above the line 34. Our previous discussion applies directly to the fuselage structure shown in Figs. 10.11 and 10.12—but now the ribs are the fuselage bulkheads (formers), the spar is the keelson, and the stringers and corner supports are the stringers and longerons in Fig. 10.11. This fuselage structure is covered with a thin stressed skin just as in the case of the wing, helping to form the streamlined shape of the fuselage. The Douglas DC-3 shown in Fig. 6.76 is a beautiful example of the early use of thin stressed skin in airplane design.

Note: The level of discussion in Secs. 10.3.1 and 10.3.2 is based on and motivated by the discussion by John Cutler in his excellent and readily understandable description of aircraft structures in *Understanding Aircraft Structures* (see the bibliography at the end of this chapter). Cutler's book is an excellent introduction to aircraft structures; indeed it is the only such reference at this level, to this author's knowledge. Flight structures are a vitally important discipline within the general field of aerospace engineering, and you are encouraged to read Cutler's book for a broader horizon in this discipline before starting a more advanced study of the subject.

10.4 MATERIALS

We are almost at the end of our brief journey through structures, as you can see from the short road map in Fig. 10.1. In this last section we list and discuss some materials commonly used in flight structures.

Aluminum Aluminum is the most widely used material in aircraft structures. Modern commercial transports such as the Boeing 747 use aluminum for about 80 percent of the structure. Aluminum is readily formed and machined, has reasonable cost, is corrosion-resistant, and has an excellent strength-to-weight ratio. In its pure form, aluminum is too soft for aircraft use. Therefore alloys of aluminum are used, the most common being aluminum 2024, an alloy consisting of 93.5 percent aluminum, 4.4 percent copper, 1.5 percent manganese, and 0.6 percent magnesium. This alloy is also called *duralumin*. (*Historical note:* The first metal-covered airplanes were designed by Hugo Junkers, a professor at Aachen University in Germany, beginning in 1914. He first used all steel, which proved to be too heavy. In 1915 he turned to the use of duralumin and established a tradition that prevails to the present day.)

Steel For a typical commercial transport, steel makes up about 17 percent of the structure. It is used in areas requiring very high strength, such as wing attachment fittings, landing gears, engine fittings, and flap tracks. Steel is an alloy of iron and carbon; typical steel alloys have about 1 percent carbon. Stainless steel is an alloy of steel and chromium that has good corrosion-resistant properties.

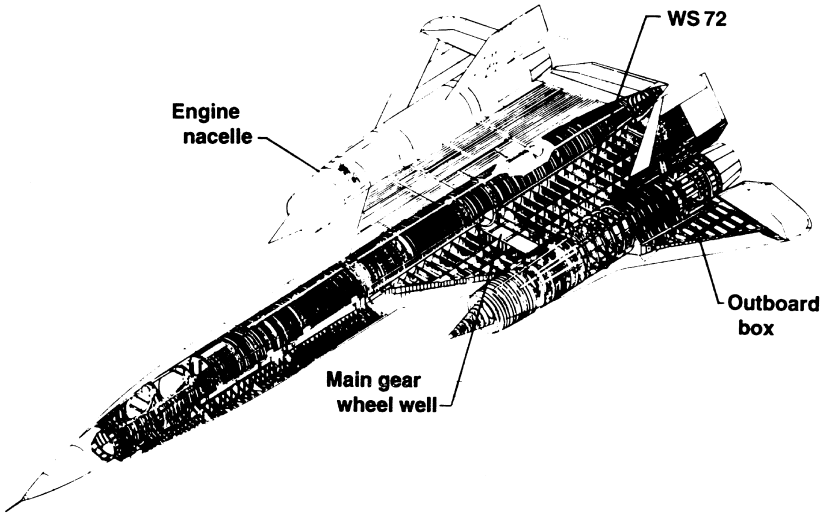


Figure 10.21 The structure of the Lockheed YF-12.

Titanium Titanium has a better strength-to-weight ratio than aluminum and retains its strength at higher temperatures. However, it is hard to form and machine and is expensive, costing about 5 to 10 times more than aluminum. Titanium is rarely mined in the United States; most titanium comes from Russia. But some supersonic aircraft have to use titanium because of the high skin temperatures due to aerodynamic heating. The SR-71 is such a case. A cutaway drawing of the YF-12, an intercepter version of the SR-71, is shown in Fig. 10.21. This airplane cruises at Mach 3 and above; hence it was the first airplane to make extensive use of titanium. How to deal with titanium and how to obtain a sufficient supply were part of the challenge faced by the Lockheed Skunk Works engineers who designed the SR-71 and the YF-12. (See the historical case study of the SR-71 and YF-12 in the book by Anderson listed in the bibliography at the end of this chapter.) Today titanium is still a major consideration in the decision about the design Mach number of a second-generation supersonic transport. For an SST designed to cruise at Mach 2.2 or lower, aluminum can be used. But at a cruise Mach number of 2.4, the skin temperatures cross a threshold value requiring titanium.

High-Temperature Nickel Alloys We note that hypersonic airplanes require advanced, high-temperature materials to withstand the high rates of aerodynamic heating at hypersonic speeds. Some nickel-based alloys are capable of withstanding the temperatures associated with moderate hypersonic speeds. The X-15, shown in Fig. 5.89, is such a case. This aircraft was designed to fly as fast as Mach 7; hence its structure made extensive use of inconel, a nickel-based alloy.

Composites Composite materials are bringing about a revolution in aircraft structures because, for the same loads, a composite structure can yield at least a

25 percent reduction in weight. Composites are quite different from metals in both composition and physical properties.

From the *American Heritage Dictionary of the English Language,* the definition of the word *composite* is "made up of distinct components." This is an apt description of composite materials, which consist of a reinforcing material suspended in a matrix binder. A commonly used composite is graphite epoxy, which uses carbon fibers as the reinforcing materials and an epoxy as the matrix binder. A schematic of a generic composite material is shown in Fig. 10.22, where the fibers are black and the matrix material is white. A structural part made of a composite material is usually molded, and then it can be cured at room or elevated temperatures and pressures to increase its strength. The arrangement of the fibers within the matrix can be of several types. Shown in Fig. 10.22 are alternate layers of fibers at 90° to each other. Some composites have the fibers running in just one direction, which is usually aligned with one of the principal axes to provide maximum strength in that direction. Other composites are made of alternate layers with the fibers oriented at 0°, 45°, and 90°; these alternate layers are called *plies*.

Composite materials usually follow a linear stress–strain relationship up to the proportional limit stress, as sketched in Fig. 10.23. However, composites have essentially no inelastic range compared to a metal, also shown in Fig. 10.23 for comparison. Rather, the composite will dramatically fracture at a stress very near the proportional stress. Thus the factor of safety of 1.5 between yield stress and fracture that is built into a metal such as aluminum is not there for a composite material. Composite structures must have a safety factor explicitly included in the structural design.

Modern airplanes are making extensive use of composite materials, and this trend will grow in the future. For example, the Lockheed-Martin F-22, shown in Fig. 10.24, has 28 percent of its structure made up of composite material in the actual production aircraft, with the remainder being 33 percent aluminum, 24

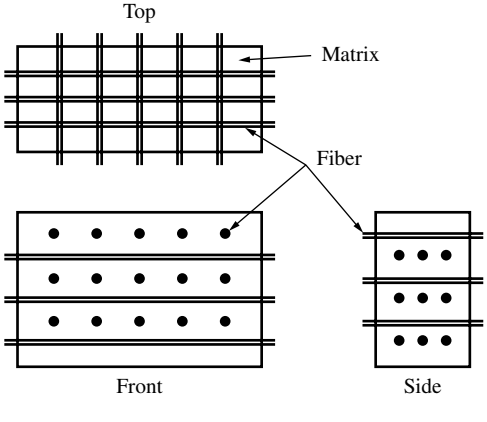


Figure 10.22 Generic sketch of a composite material.

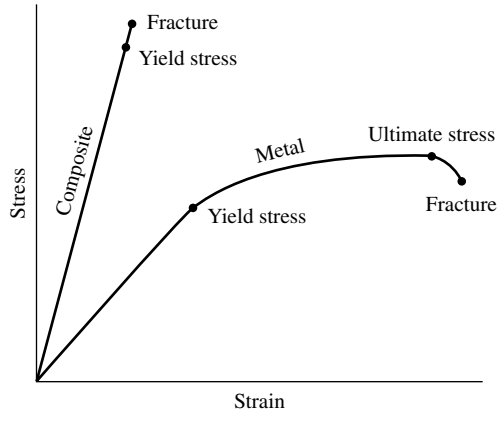


Figure 10.23 Stress–strain diagram comparing the behavior of composites with metals.

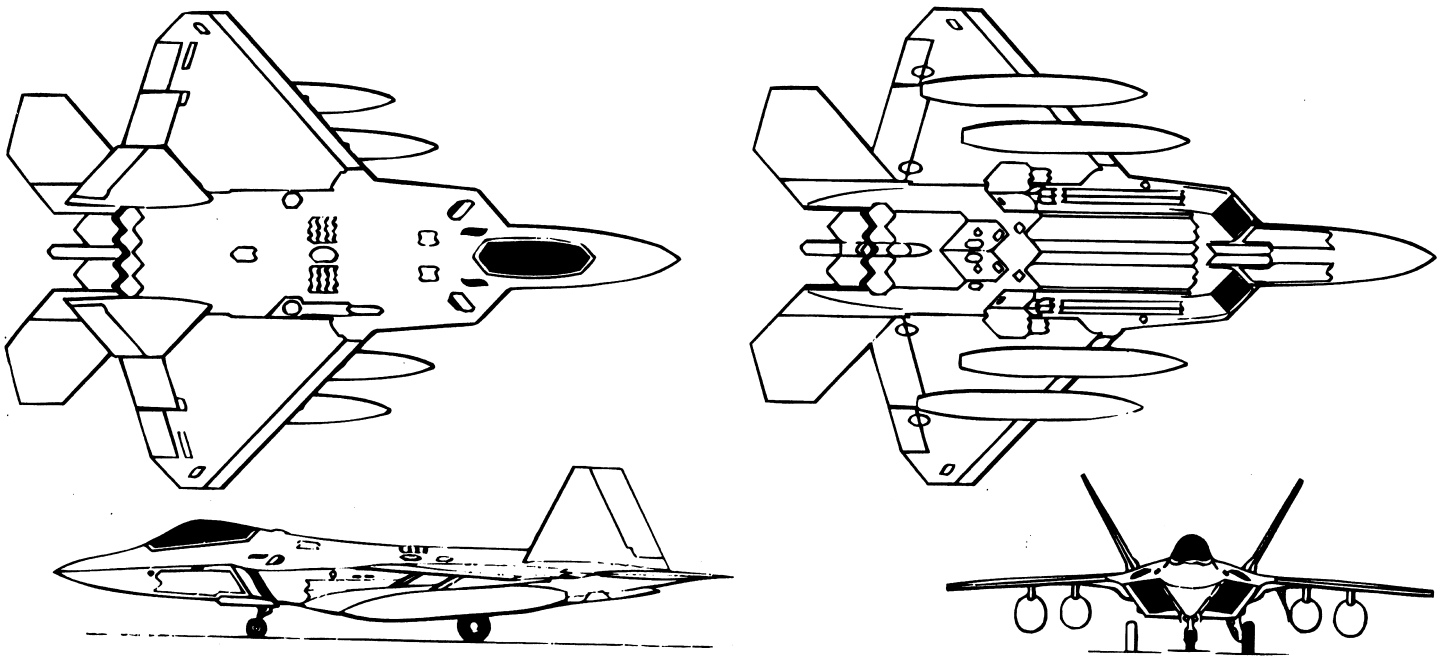


Figure 10.24 The Lockheed-Martin F-22.

percent titanium, 5 percent steel, and 10 percent miscellaneous. The Boeing 777, shown in Fig. 6.27, has 10 percent of its structural weight in composite material incorporated in such elements as the horizontal and vertical stabilizer torsion boxes; cabin floor beams; control surfaces such as the rudder, elevator, and ailerons; and the flaps, engine nacelles, landing gear doors, wing leading edge, and wing root fairing.

10.5 FATIGUE

Common experience shows that when you bend a piece of metal back and forth enough times, it will break. This is an example of *fatigue*. Various elements of an airplane structure are repeatedly being bent back and forth by the changing loads on the structure. The wing is bent upward by the high lift load at takeoff and is then bent downward upon landing by the weight of the wing after the lift is reduced to essentially zero. The structure goes through numerous load cycles in turbulent air. The fuselage experiences load cycles when the cabin pressure is increased and decreased. Military aircraft experience extreme load cycles during maneuvering in combat. So metal fatigue is a major concern in flight vehicles.

The nature of fatigue is shown in Fig. 10.25, which is a plot of the number of cycles that a metal structural element can experience before failure, as a function of the maximum stress imposed on it. Such information is determined empirically by *fatigue testing*. Figure 10.25 applies to fatigue tests of joints and splices using aluminum 7075-76. At the extreme left of Fig. 10.25, we see that a maximum stress of 80,000 lb/in^2 will break the material in just one cycle—this stress is, of course, the ultimate stress as defined in Sec. 10.2. Fig. 10.25 shows that as the maximum stress is made smaller than the ultimate stress, the metal can go through more load cycles before breaking. Note that by limiting the maximum

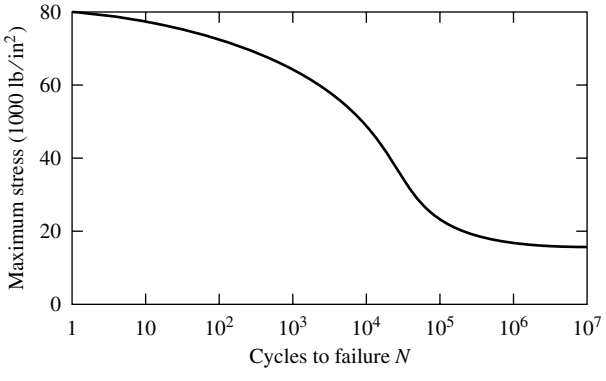


Figure 10.25 Fatigue test of joint and splices; 7075-76 aluminum alloy.
(Source: *After R. S. Shevell,* Fundamentals of Flight, *2d ed., Prentice Hall, New Jersey, 1989.*)

stress levels in a given airplane structural design to much less than the ultimate stress, many load cycles can be endured without structural failure. In airplane design, the structure is designed for certain extreme loads to be anticipated at the very edge of the airplane's operational envelope. Because these extreme loads are rarely encountered during routine operation of the airplane, the fatigue life of the structure is quite long.

The prediction of the fatigue life of various components of a flight vehicle structure is vital in the design and operation of an aircraft. This, in concert with regular airframe inspection and maintenance, will usually result in replacement of components before they fail.

10.6 SOME COMMENTS

In this chapter we have just scratched the surface of the subjects of flight vehicle structural design and material selection. From our discussion, however, you can appreciate that an aircraft structure can be divided into different components, all integrated and working together. In a sense, an airplane fuselage is essentially a stiffened shell, and the wings are essentially a combination of stiffened plates and beams. In more advanced studies of flight vehicle structures, you will examine in depth the structural nature of shells, plates, and beams. In further studies of materials, as applied to flight vehicles, you will search for materials with such favorable properties as high strength-to-weight ratio, ease of manufacture, low cost, good high-temperature performance, and long fatigue life.

And finally we mention the big picture. A flight vehicle is a finely tuned system in which aerodynamics, propulsion, flight dynamics, and structures are rolled into one "ball of wax," so to speak. The proper design and performance of the vehicle depend on all these disciplines working together. An airplane is analogous to a good sound system, where the quality of the sound is no better than the lowest quality of any one of its components. The quality of an airplane is no better than the lowest quality of its aerodynamics, propulsion, flight dynamics, and structures.

10.7 HISTORICAL NOTE: EVOLUTION OF FLIGHT STRUCTURES

Take a look at the *Wright Flyer* in Figs. 1.1 and 1.2. This airplane established the first phase of aircraft structures: a strut-and-wire braced biplane with thin wings, which prevailed through the era of the strut-and-wire biplane discussed in Section 6.26 and identified in Figs. 6.73 and 6.74. For most aircraft in this period the wings were so thin that the thin spars along the span, along with the fabric covering of the wings, carried very little of the applied aerodynamic loads exerted on the wings. Instead these loads were carried for the most part by the struts and wires between the two wings (see for example the SPAD XIII in Fig. 6.75), which, together with the two wings, made a sturdy boxlike structure. The primary disadvantage of this

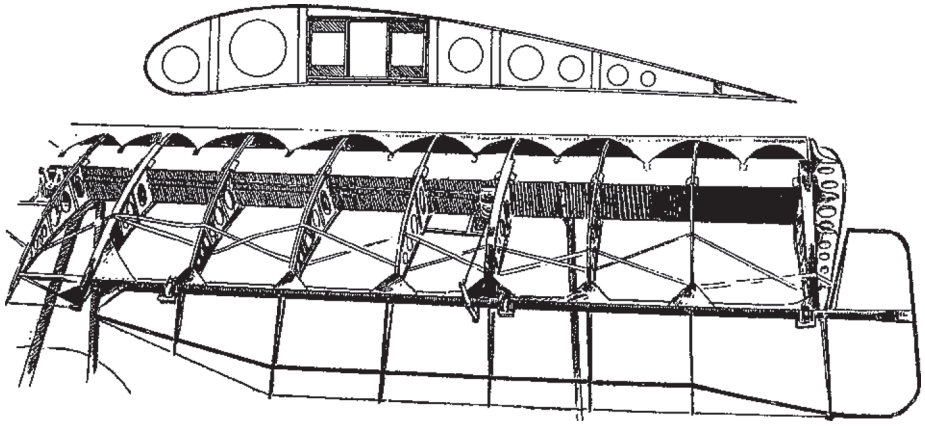


Figure 10.26 Wing and rib structural design for the Fokker Dr.1 triplane and the Fokker D-VII. Both had cantilevered wings with thick airfoil sections.
(Source: S. T. G. Andrews and S. F. Benson, *The Theory and Practice of Airplane Design*, E. P. Dutton and Co., New York, 1920.)

structural arrangement was aerodynamic: The struts and wires created high drag, and the two superimposed wings set up an unfavorable aerodynamic interaction with each other that decreased lift and increased drag.

The advent in Germany toward the end of World War I of the cantilevered wing with thick airfoils changed not only the aerodynamic characteristics of such airplanes but the wing structural design as well. A drawing of the basic wing design used for the Fokker triplane and the D-VII is shown in Fig. 10.26. Here, in place of the conventional isolated front and rear spars, we see a massive central support system, which is essentially two thick spars placed very close to each other, united by sheets of plywood. This type of cantilevered, box spar wing construction was used by Fokker well into the 1930s. It completely eliminated the need for struts and wires, and in fact eventually led to the demise of the biplane configuration in favor of the single-wing monoplane configuration from the 1930s onward.

After 1908 airplanes started to have identifiable fuselages. Three types of fuselage structure were used. The girder type was the most common and is illustrated in Fig. 10.27. Here we see a boxlike structure made up of several longitudinal rods, called *longerons*, running the entire length of the fuselage. These are supported by struts and wires, and the whole system forms a box lattice girder, which can withstand vertical and side loads and torsion (twisting) about the longitudinal axis. The girder structure was usually constructed of wood and wire. However, the chief designer for Fokker, Reinhold Platz, came from a welding background, and the later Fokker aircraft such as the Dr.1 triplane and D-VII had welded steel tube girder fuselages.

The second type of fuselage structure, developed as early as 1912, was the *monocoque*, a French word meaning "single shell." First developed by the Swiss engineer Ruchonnet, the monocoque structure was adopted by Louis Bechereau

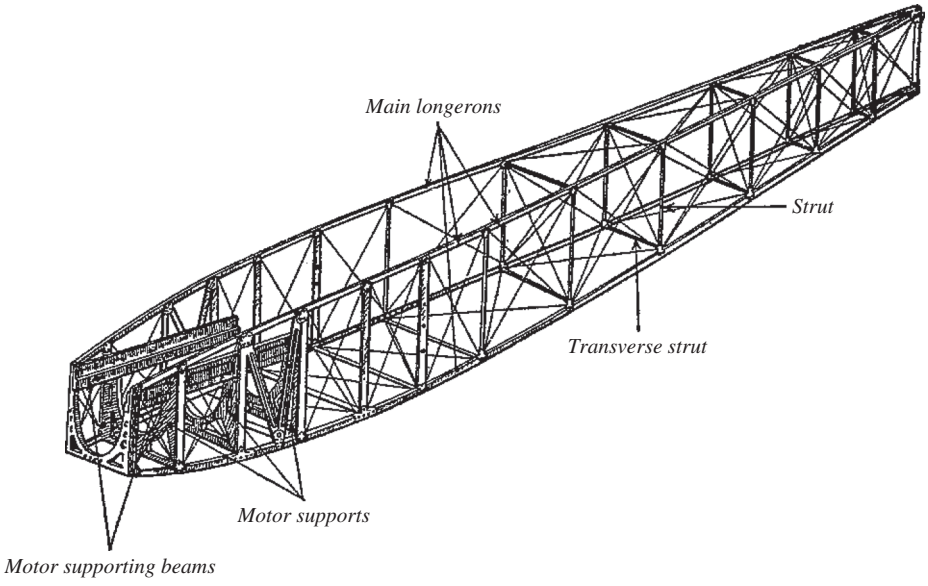


Figure 10.27 Generic girder fuselage design.
(Source: O. Pomilio, *Airplane Design and Construction*, McGraw-Hill, New York, 1919.)

for the design of the highly streamlined (for its day) Deperdussin airplane. A Deperdussin racer was the winning airplane in the 1913 Gordon Bennett Cup at Reims, with a phenomenal speed of 124.5 miles per hour. A Deperdussin equipped with pontoons also won the Schneider Cup race in the same year. A monocoque fuselage, such as that shown in Fig. 10.28, was made by laying thin strips of wood over a mold contoured to the desired fuselage shape. (Tulip wood was used for the Deperdussins.) Usually three layers of wood were used in this fashion, with the strips of wood running at right angles to the layers underneath. Each layer was glued to the other. Then two layers of fabric were glued to the outside of the shell and one layer of fabric on the inside. The fuselage was molded in two half-shells, which were subsequently glued together to form one single shell. The monocoque shells were amazingly thin, from 3 to 4 millimeters, and were both lightweight and strong. All of the stress in the monocoque fuselage was carried in the skin, so this monocoque design was the forerunner of stressed-skin aircraft structures. A clear advantage of the monocoque design was that fuselages could be made aerodynamically clean (streamlined). Disadvantages were that battle damage was more difficult to repair, and the cost of manufacture was considerably more than for the conventional girder type. For these reasons only a few airplane designs during World War I used monocoque fuselages.

The third type of fuselage construction, which became popular toward the end of World War I, was a combination of the first two. Called the *monocoque–girder type*, or more commonly the *veneer type*, it consisted of a boxlike structure made up of four longitudinal longerons and internal bulkheads to which wood panels were

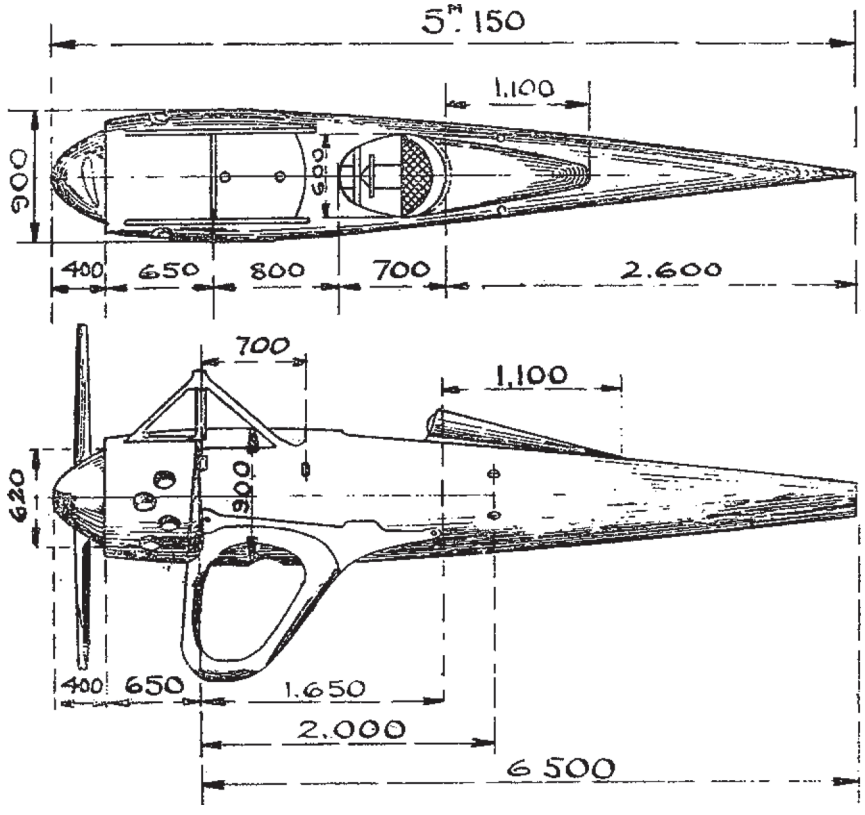


Figure 10.28 A generic monocoque fuselage.
(Source: A. W. Judge, *The Design of Aeroplanes*, Pitman and Sons, Ltd., London, 1916.)

glued, as shown in Fig. 10.29. Compare the (then conventional) girder construction shown in Fig. 10.27 with the veneer construction in Fig. 10.29. Missing in Fig. 10.29 are all of the internal struts and bracing seen in Fig. 10.27; the wood panels, glued or attached by nails or screws, provided the required stiffening.

Not all fuselages made with longerons were boxlike; nor did they fall in any of the described categories. Shown in Fig. 10.30 is the structure used by Pfalz in Germany, where a number of longerons were attached to rounded formers. The structure was then covered with fabric, forming a rounded, aerodynamically streamlined fuselage.

Instead of covering the structure shown in Fig. 10.30 with fabric, the German designers of the Albatros family of fighters covered it with wood. This gave the outward appearance of a monocoque fuselage; but because the fuselage was also internally strengthened with bulkheads and longerons, this type of construction was called *semimonocoque*. The Albatros D-Va, with its semimonocoque fuselage, is shown in Fig. 10.31. Here we see another example of the aerodynamic streamlining afforded by such a construction technique. The semimonocoque

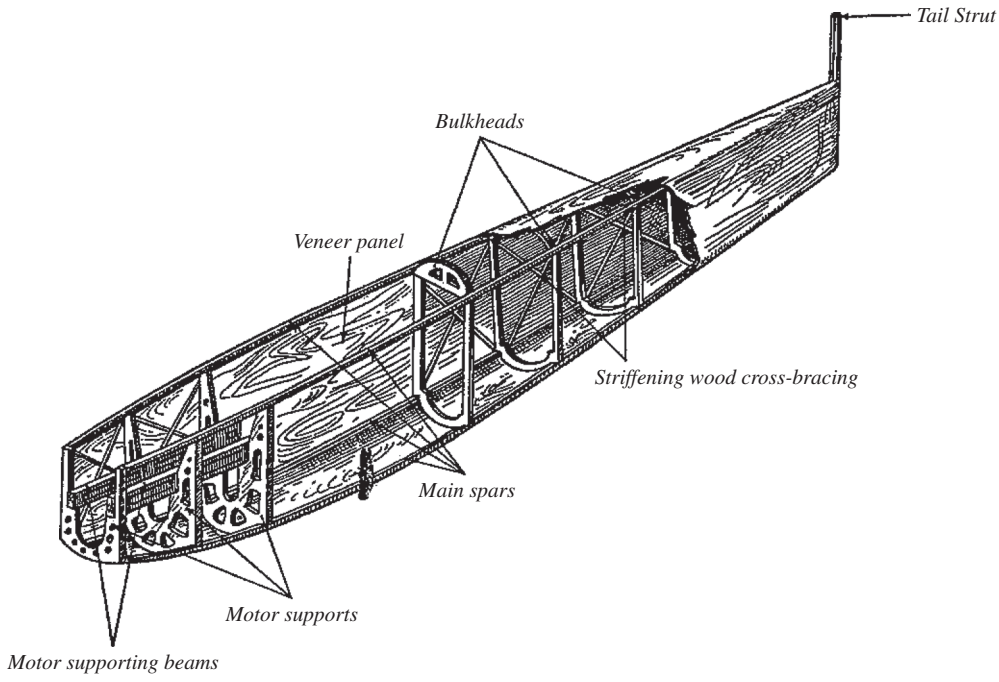


Figure 10.29 A generic veneer or monocoque–girder fuselage.
(Source: O. Pomilio, *Airplane Design and Construction*, McGraw-Hill, New York, 1919.)

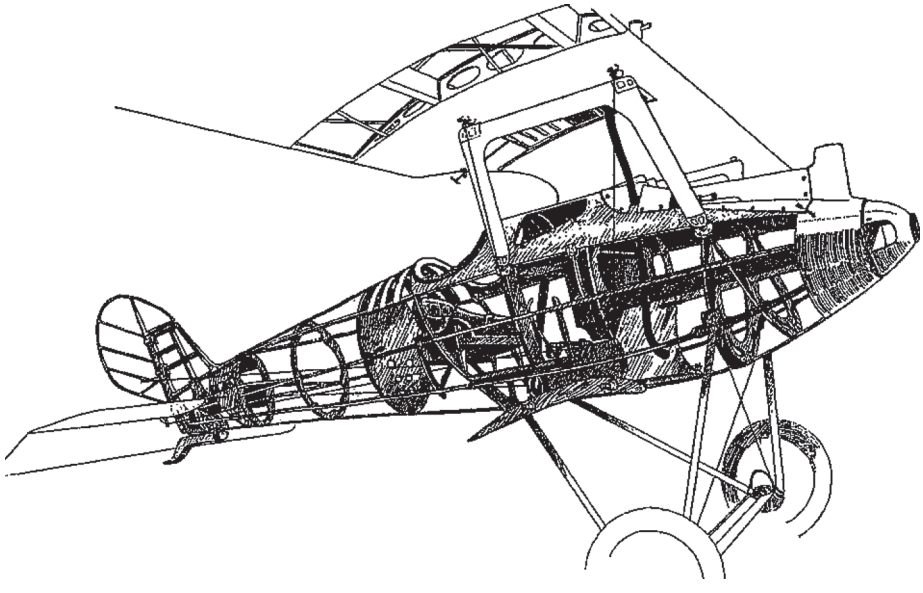


Figure 10.30 Fuselage of the German Pfalz D-III fighter.
(Source: S. T. G. Andrews and S. F. Benson, *The Theory and Practice of Airplane Design*, E. P. Dutton and Co., New York, 1920.)

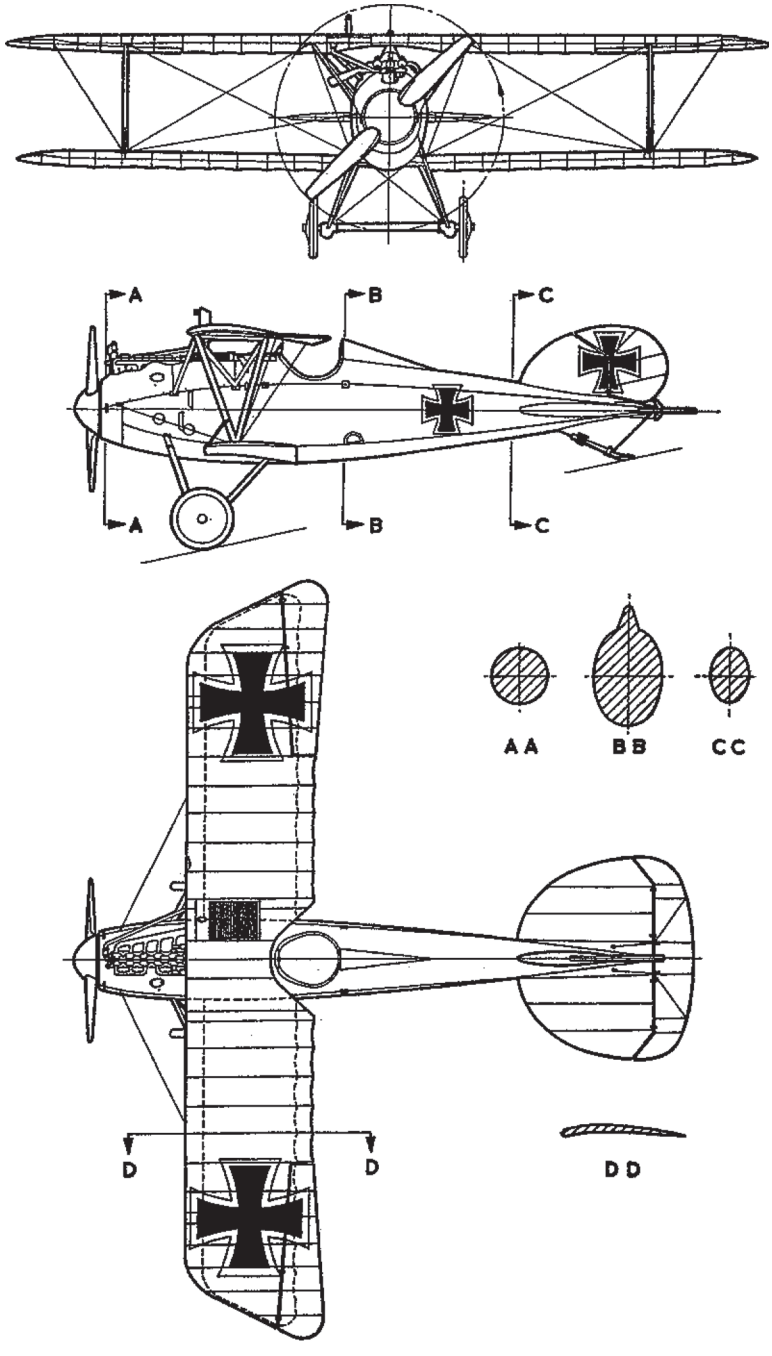


Figure 10.31 Three-view of the German Albatros D-Va, 1917.

structure would become the gold standard of fuselage design by the 1930s—so much so that the term *semimonocoque* became redundant. The term *stressed-skin construction* has basically replaced it.

The first all-metal airplane was designed and built in Germany by Hugo Junkers in 1915. Designated the J.1 and shown in Fig. 10.32, this airplane was

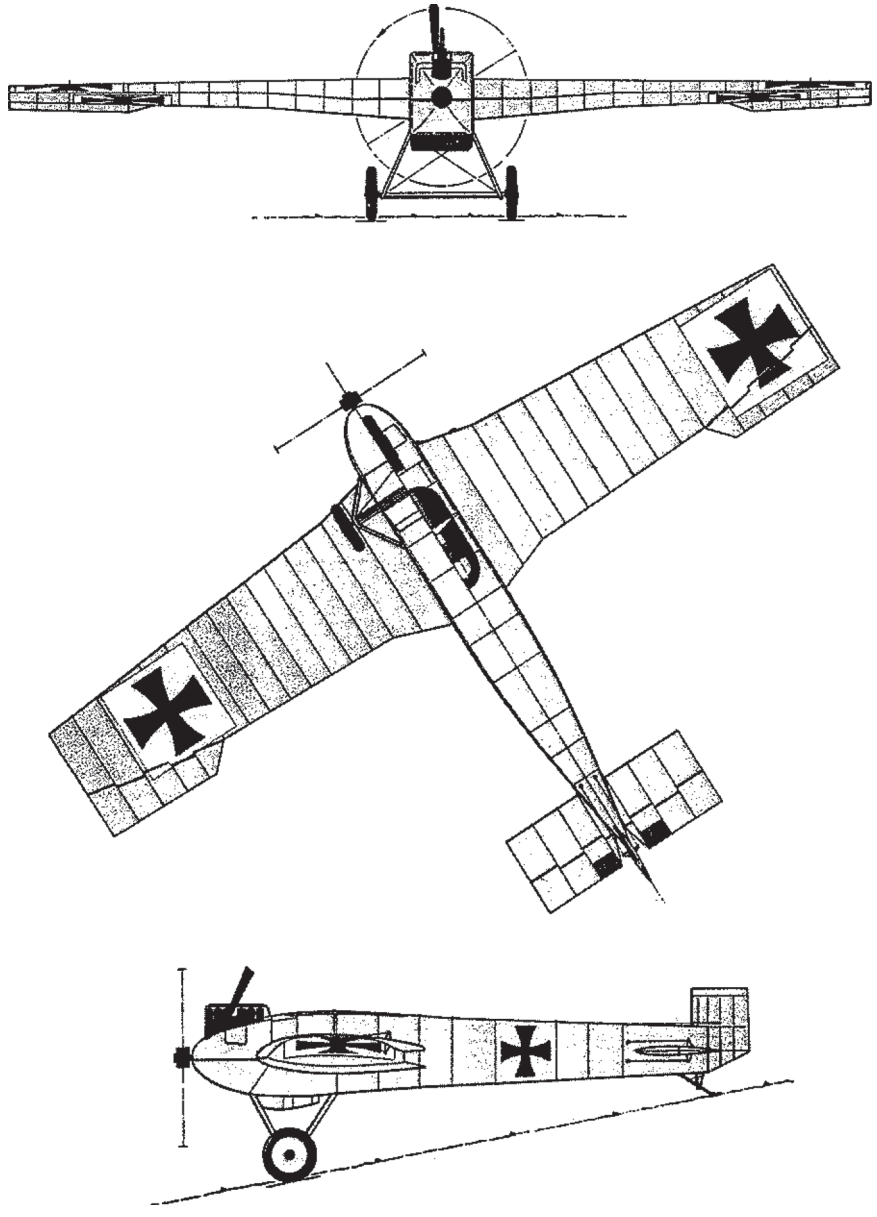


Figure 10.32 Three-view of the Junkers J.1, the first all-metal airplane, 1915.

a monoplane with a thick wing, allowing a large, strong spar cantilevered from the fuselage and hence requiring no external struts and wires for support. Although the J.1 flew successfully, and Junkers continued to design and produce subsequent all-metal airplanes, in many respects it was ahead of its time because the all-metal airplane did not come into its own until the 1930s. Nevertheless the J.1 was designed, built, and flown in an era when virtually every other airplane was a vegetable airplane—made from wood, fabric, and glue. The J.1 was built with iron. Although iron is a heavy metal, for the J.1 the sheets were very thin, varying from 0.1 to 0.5 millimeters—smooth on the outside and reinforced by welded corrugations on the inside. As a result the weight of the J.1 was a respectable 2288 pounds, which resulted in a reasonable wing loading and power loading. The J.1, however, was Junkers's only iron airplane. He shifted to the use of a special form of aluminum called *duralumin*, an exceptionally strong and hard aluminum alloy containing copper, manganese, magnesium, iron, and silicon. The development in Germany of duralumin in the autumn of 1906 would change the course of aircraft structural development in the 20th century and beyond. Junkers used sheets of duralumin that were corrugated to improve stiffness. These sheets carried some of the stress, and the corrugations helped prevent buckling. In a sense this was a stressed skin, but not of the precise nature to be developed later. Hugo Junkers's development of the all-metal airplane during World War I and in the decade after was *revolutionary*, especially when compared to other aircraft at that time. (For an in-depth discussion of Junkers's work on all-metal airplanes as well as the story of how duralumin was discovered and developed, see Anderson, *The Airplane: A History of Its Technology*, AIAA, Reston, VA, 2002.)

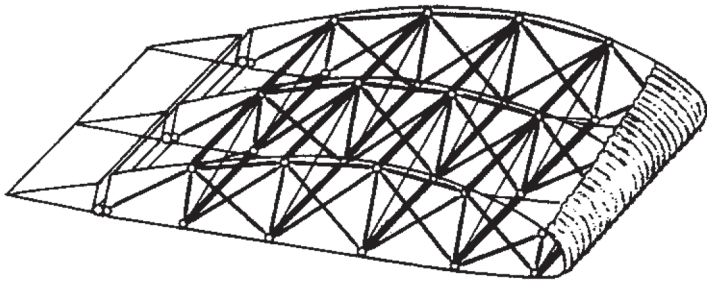
The era of the mature propeller-driven airplane was identified and discussed in Section 6.26. It was during this era, beginning in the late 1920s, that the all-metal airplane came into its own. Hugh Junkers's conviction in 1915 that the all-metal airplane was the airplane of the future finally came to fruition. Although other all-metal airplanes were designed in the decade following Junkers's pioneering J.1 in 1915, this type was not generally accepted by the airplane design community at the time. However, three advances—one in materials and two in aircraft structures—changed this situation during the 1920s. Let us take a look.

The advance in materials involved the protection of aluminum alloys from corrosion. Concern about corrosion was a major impediment to the adoption of Duralumin in both Britain and the United States, despite the fact that Junkers forged ahead with his successful all-metal airplanes after World War I. This objection to aluminum was gradually removed when G. D. Bengough and H. Sutton, working for the National Physical Laboratory in the United Kingdom in the mid-1920s, developed a technique of anodizing aluminum alloys with a protective oxide coating, and when E. H. Dix in the United States in 1927 discovered a method of bonding pure, corrosion-resistant aluminum to the external surface of duralumin.

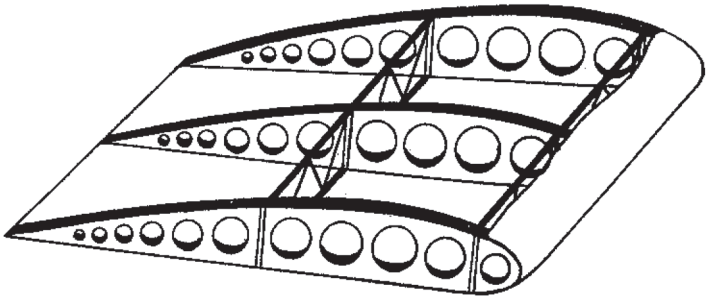
The latter manufactured form is called Alclad. The NACA, after carrying out corrosion tests on Alclad, gave its approval; and Alclad was accepted by the U.S. Army and U.S. Navy for new all-metal aircraft. The first major civilian airplane to use corrosion-resistant aluminum was the Boeing 247, which was fabricated from Alclad 17-ST.

The two major advances in structural design took place in Germany. The first was the practical introduction of the stressed-skin concept. Junkers had designed his all-metal airplanes with the idea that the metal skin could carry loads. In a 1923 paper given to the Royal Aeronautical Society, Junkers stated, "The theoretically best design appeared to be the system of the so-called supporting cover; that is, all tensile, compressive, and shear forces are taken up by the wing cover." Except for a few exceptions, however, Junkers used corrugated metal to obtain stiffness of the skin. The corrugations added extra surface area, thus increasing the skin friction drag on the airplane; moreover, the corrugations interfered with the smooth aerodynamic flow over the surface. The practical use of a smooth stressed-skin structure is largely due to Adolf Rohrbach, who is also responsible for coining the term *stressed skin* in 1924 during a paper delivered to the Royal Aeronautical Society. Born in 1889 in Gotha, Germany, Rohrbach received his engineering diploma from the Technische Hochschule Darmstadt and went on to earn his doctorate in engineering from the Technische Hochschule Berlin-Charlottenberg in 1921. Afterward he established the Rochbach Metal Airplane Company in Copenhagen, Denmark—a location chosen to circumvent the strict restrictions imposed on the German aircraft industry by the Treaty of Versailles. He concentrated on the design and construction of all-metal flying boats. In contrast to Junkers's airplanes of corrugated aluminum. Rohrbach's airplanes had smooth metal skins and hence lower overall friction drag. Moreover, the internal structure of Rohrbach's wings involved a strong, metal box beam with a rounded nose in front and a tapered section in back, all made from aluminum. This design is shown schematically in Fig. 10.33, which contrasts the early evolution of metal wing structures. Rohrbach's wing design was considered revolutionary at the time; later it served as a model for the stressed-skin wing structures that have dominated airplane design from 1930 to the present.

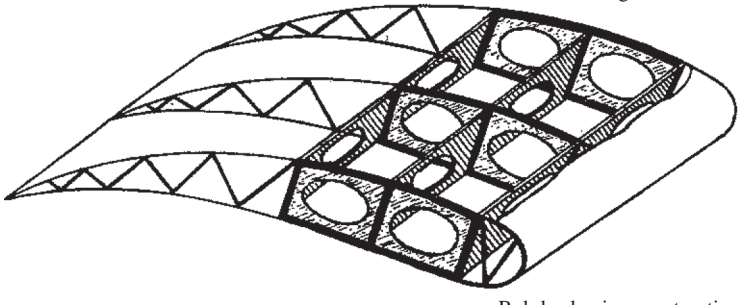
Rohrbach's rectangular skin panels were fastened to a frame that supported the panels on all four sides. Prevailing practice at the time dictated that the frame plus panel combination should be strong enough to prevent the panel from buckling because buckling was viewed as structural failure. This laid the groundwork for the second major advance in metal aircraft structures: the discovery that a structure of mutually perpendicular members covered with a thin skin did not fail if the skin buckled. Herbert Wagner, an engineer working for Rohrbach, made this discovery in 1925, although he did not publish his findings until 1929. When the skin is allowed to carry the maximum possible load, the spacing between frames can be increased, resulting in lighter structures.



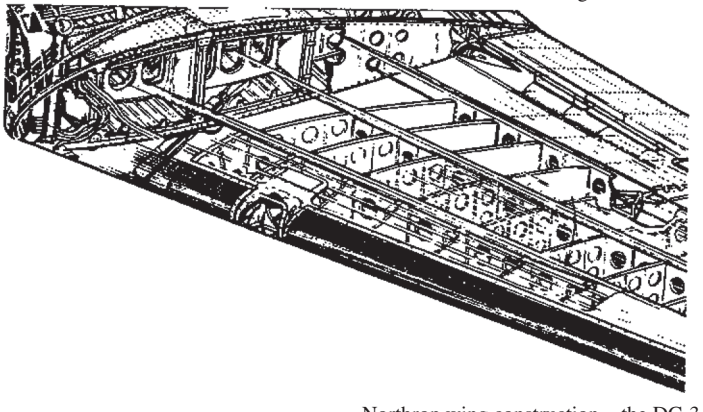
Junkers wing construction



Dornier wing construction



Rohrbach wing construction



Northrop wing construction—the DC-3

Figure 10.33 Four different structural designs for wings.
(Source: R. Miller and D. Sawers, *The Technical Development of Modern Aviation*, Praeger Publishers, New York, 1970.)

Without knowledge of Rohrbach's or Wagner's work, Jack Northrop in the United States pioneered a multicellular wing structure of spars, ribs, and stringers covered with smooth, stressed-skin sheets. He independently discovered that the sheets did not fail even after they began to buckle. The Northrop wing was adopted for several aircraft, including the DC-1-2-3 series. The DC-3 wing structure is shown in Fig. 10.33. This wing structure proved to be exceptionally long-lasting. Each of the many load-carrying elements of the wing carried low stress, which greatly increased the fatigue life of the DC-3. (Some people have criticized the DC-3 structure as being "overdesigned" and therefore not an optimal structure in terms of weight. That may be true intellectually, but the structural toughness of the DC-3 was a boon to its user.)

In short, the all-metal, mostly aluminum, semimonocoque, stressed-skin aircraft structure became the norm during the era of the mature propeller-driven airplane. The all-vegetable airplane became a thing of the past.

The basic structural elements of modern airplanes are essentially the same as those developed during the era of the mature propeller-driven airplane: the use of metal ribs, stringers, spars, stressed-skin construction, and so forth. The design and manufacturing of these elements, however, have been revolutionized by the use of modern high-speed digital computers. Structural designers now use a computational method called *finite element analysis* to design complex structures. In this analysis, the entire structure is divided into many interconnected tiny elements, and the stress and strain on these elements are computed; such elaborate computations sometimes take days on mainframe computers. A common finite element program in wide use for structural analysis is NASTRAN, developed in the 1960s by NASA. In addition, manufacturing processes are now heavily based on computers, with numerically controlled machine tools, large and small, being controlled by computers. A modern example is the automated span assembly tool (ASAT), the gigantic tool used to manufacture the wing spar for the Boeing 777. ASAT is designed to fasten up to 30 metal parts ranging from 2 to 100 feet in length. When the ASAT was laid out on the floor of Boeing's hangarlike assembly building, its C-shaped metal frames stood 30 feet high, and the whole machine stretched off into the distance. The parts of ASAT were transported to Boeing from its manufacturers in Wisconsin by 90 trucks. The building up and fastening of the elements of the 777 wing spars by ASAT are carefully measured and controlled by computers.

Another revolution involved the materials used for aircraft. All-metal airplanes were made from aluminum in the era of the mature propeller-driven airplane and in the early part of the current era of the jet-propelled airplane. As the speeds of jet-propelled airplanes increased, aerodynamic heating of the airplane skin became more of a problem; such heating is caused by intense frictional dissipation in the boundary layer adjacent to the surface. For example, at sustained cruise at Mach 2, the skin temperature can reach 300°F. Aluminum begins to lose its strength at this temperature. Steel has better high-temperature properties; so beginning in the 1950s, the parts of Mach 2 aircraft exposed to such temperatures were made from stainless steel honeycomb sandwich material, as illustrated in

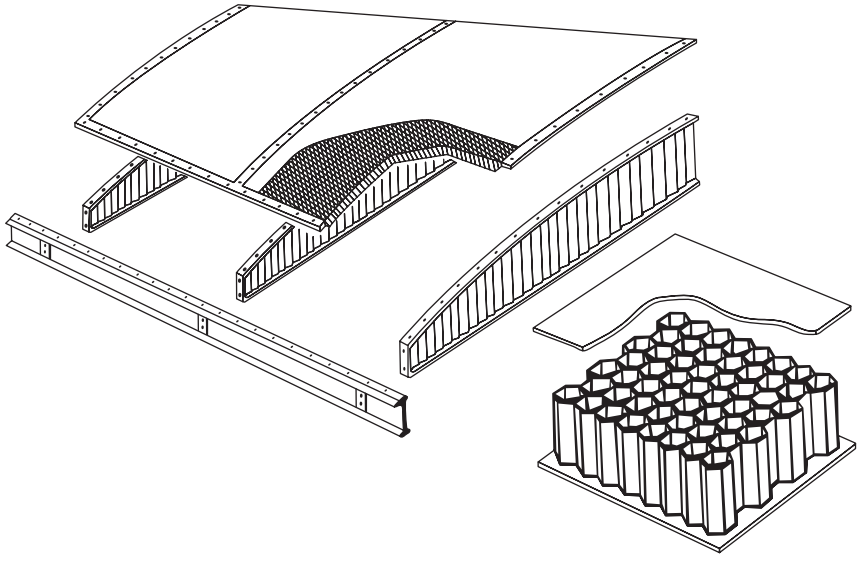


Figure 10.34 Brazed stainless steel honeycomb sandwich panel.

Fig. 10.34. The skin of America's first supersonic bomber, the Convair B-58, was mainly stainless steel honeycomb. Aerodynamic heating increases almost as the cube of the flight velocity, so the materials problem for even higher-speed aircraft becomes more severe. Not only is the high-temperature strength of the material a problem; thermal stresses induced within the metal due to thermal expansion of the material also become important. Higher-speed aircraft have to use more exotic materials to withstand aerodynamic heating. The Lockheed SR-71 Blackbird, which can fly above Mach 3, is made mainly from titanium, which is much lighter than steel, with good high-temperature properties. The Mach 7 North American X-15, thus far the only powered, piloted hypersonic aircraft, is made from Inconel-X, a nickel-alloy steel capable of withstanding temperatures of up to 1200°F. How far we have come from the primitive all-metal Junkers J-1 of 1915.

Finally, we note the growing use of composite materials in modern airplane design, as described in Section 10.4. Because composites are still in the process of making history, this seems to be an appropriate point at which to end this historical note about flight structures.

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Problems

10.1 Consider two rods with identical length and diameter of 10 ft and 1 in, respectively. One rod is AM-350 stainless steel, and the other is 2024 aluminum. Both rods are under the same tension load of 15,000 lb. Which rod will elongate the most, and by what amount compared to the other?

10.2 Consider a rod made from 2024 aluminum with a diameter of 0.5 in. What is the maximum load in tension that can be carried by this rod before it yields?

10.3 Consider an airplane with a tricycle landing gear stationary on the ground. The weight of the airplane is 5158 lb. Assume the nose wheel strut and the two main gear struts are perpendicular to the ground. The diameter of the nose wheel strut is 1 in; that of each main gear strut is 3 in. The nose wheel strut is located 5.62 ft ahead of the center of gravity of the airplane. The two main gear struts are located 1.12 ft behind the center of gravity. Calculate the compressive stresses in the nose wheel strut and the main gear struts.

10.4 Consider a flying kite supported and restrained by a cord from the ground. The lift on the kite is 20 lb, and the cord makes an angle of 60° with the ground. The cord is taut and straight. The diameter of the cord is 0.1 in. Calculate the tensile stress in the cord. Ignore the weight of the cord.

10.5 A 1-ft square plate made of 2024 aluminum is subjected to a shear load along its edges. The displacement of one side relative to the opposite side is 0.1 in. Calculate the shear stress acting on the edges.