Cambridge University Press & Assessment
978-1-107-07577-1 — Mechanics of Aero-structures
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Excerpt
More Information

1 Aircraft Structural Components and Loads

The structural components comprising an aircraft may be grouped into three categories: Fuselage, wings, and tail. These three groups interact with each other through mechanical connections and aerodynamic coupling. Their overall shape can be viewed as metal cages wrapped in an aluminum or a composite skin. Fig. 1.1 shows a sketch of a plane with the groups of structures marked. An additional component not considered in this book is the landing gear. The landing gear design is intrinsically connected to shock absorbers and hydraulics.

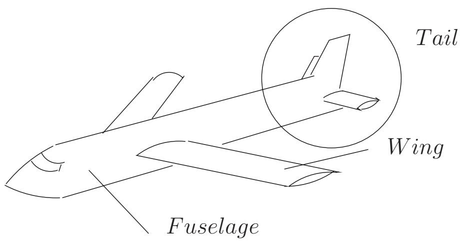

Figure 1.1 The three structural groups of an aircraft.

The details of the tail section are similar to the wing section, but on a smaller scale. That reduces the three groups to two.

From the early days of aviation when wood, cloths, and cables (remember the Wright brothers' biplane) were in use, most of the modern components originated in France and we use many French words, such as *fuselage*, *empennage* (tail section), *longerons* (longitudinal bars), *ailerons* (control surfaces on wings), and so on, to describe structural parts.

1.0.1 Fuselage

Fuselages have circular or oval cross-sections with their dimensions varying along the length of the fuselage. A number of circular (or oval) metal rings called bulkheads are arranged parallel to each other to maintain the circular shape and to form support points for the wings. These bulkheads are connected by longerons (with "Z," "L," or channel shaped cross-sections) to form the basic metal cage. Aluminum sheets cover this cage and form the skin of the plane. The skin is capable of withstanding the pressurization of the cabin, shear stresses due to fuselage torsion, and minor impacts. As a first approximation, the fuselage may be viewed as a beam with concentrated forces acting on it through the points where the wings are connected and through the points where the tail section is joined. Its own weight and aerodynamic forces provide additional distributed loads. Due to asymmetrical loading of the wings and tail arising from control surface movements and wind conditions, there is also a torque acting on the fuselage. Fuselage bending stresses are carried by the longerons. The bulkheads redistribute the forces transmitted from the wings to the fuselage through the longerons. Fig. 1.2 shows the components of a fuselage.

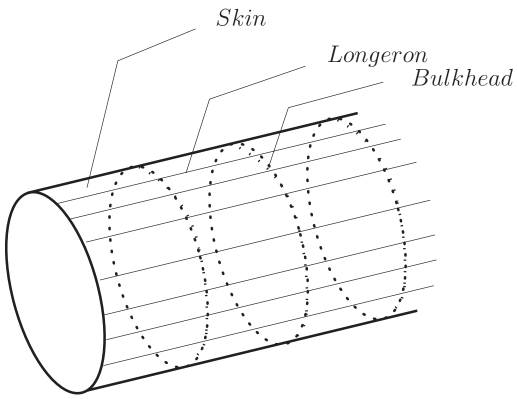

Figure 1.2 The basic structural components of a fuselage.

1.0.2 Wing

Wings support the plane through the lift generated by their airfoil shape. They are, essentially, cantilever beams undergoing bending and torsion. Fig. 1.3 shows the components of a typical wing. Depending on the size of the plane, there

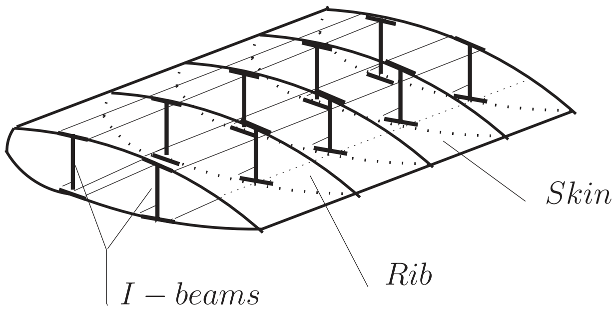

Figure 1.3 The structural components of a wing.

may be one or more I-beams running from the fuselage to the tip of the wing with the needed taper to conform to the airfoil shape. In small planes, instead of the I-beams, metal tubes are used. There are also longerons on the top and bottom surfaces parallel to the beams. The ribs (diaphragms) are used to divide the length of the wing into compartments and to maintain the airfoil shape. The wing is subjected to a distributed aerodynamic loading producing shear force, bending moment, and torque. Concentrated forces may be present if the engines are mounted on the wing. The bending stresses are carried by the beams and the longerons. The skin carries shear stresses due to torsion and the wing volume is used as a fuel tank. Flaps and ailerons are not shown in this sketch.

1.1 Elements of Aerodynamic Forces

A major part of the forces and moments acting on a wing in flight is due to aerodynamics. A specific airfoil shape according to the NACA (forerunner of NASA) classification is usually selected for the wing cross-section. Depending on the design speed of the plane, wings are given dihedral angles, taper, and sweep-back. The aerodynamic forces are computed for various altitudes and airplane speeds. They are then reduced to a reference state known as the "standard level flight" condition. The word "standard" here refers to sea-level values of air density, pressure, and temperature. Sea level is taken as the datum for measuring altitudes. As we know, density, pressure, and temperature strongly depend on the altitude. Fig. 1.4 shows an airfoil at an angle of attack.

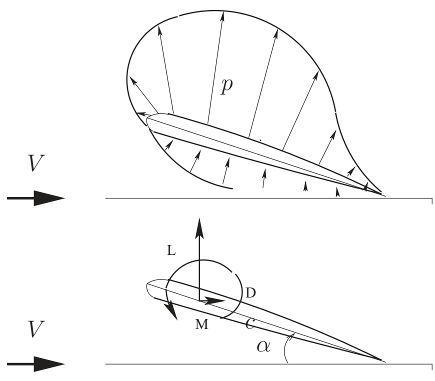

Figure 1.4 An airfoil at an angle of attack α with a pressure distribution and its equivalent forces, L and D, and moment, M, at the quarter chord.

The pressure distribution on the airfoil surfaces is integrated to obtain the wing loading in the form of the lift L, the drag D, and the moment M about the quarter chord of the wing. These can be expressed as

$$L = \frac{1}{2}\rho V^2 S C_L, \quad D = \frac{1}{2}\rho V^2 S C_D, \quad M = \frac{1}{2}\rho V^2 S\bar{c} C_M, \tag{1.1}$$

where ρ is the air density; V is the relative wind speed; S is the plan-form area of the wing; C_L, C_D, C_M are the lift, drag, and the moment coefficients; and $\bar{c}$ is the average chord. We can express the coefficients as

$$C_L = C_{L\alpha}\alpha, \quad C_D = C_0 + C_{D\alpha}\alpha, \quad C_M = C_{M\alpha}\alpha, \tag{1.2}$$

where $C_{L\alpha}$, $C_{D\alpha}$, and $C_{M\alpha}$ are the slopes of the lift, drag, and moment coefficients plotted against the angle of attack α and C_0 is a constant. Generally, within operating ranges of α, $C_{L\alpha}$ and $C_{M\alpha}$ may be taken as constants and $C_{D\alpha}$ may be approximated using a straight line. As the angle of attack, α, increases, C_L

increases and reaches a maximum. The lift suddenly drops beyond the maximum point. This phenomenon is known as stalling. The value of α at stall is called the stall angle of attack. During takeoff and landing, the speed of the plane is low and to obtain sufficient lift to balance the weight, the maximum available lift coefficient is augmented by extended flaps.

The theoretical value for $C_{L\alpha}$, for an airfoil in the idealized shape of a flat plate, is 2π per radian of the angle of attack, which is useful in remembering the approximate upper bound for this constant. Fig. 1.5 shows the typical shapes of the lift, drag, and moment coefficients.

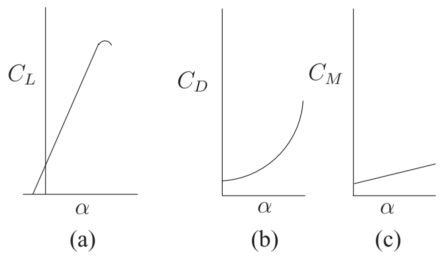

Figure 1.5 Sketches of (a) lift, (b) drag, and (c) moment coefficients as functions of the angle of attack α.

1.2 Level Flight

Under standard sea level conditions, a plane in level flight is in equilibrium with the following forces and moments: wing lift L; drag D; moment M acting at the aerodynamic center, which has been standardized as the quarter chord point of the wing; tail lift L_T; weight W; and engine thrust T. The tail and fuselage drags are incorporated into D and the aerodynamic moment around the quarter chord of the tail is neglected in the first round of calculations. Fig. 1.6 shows these

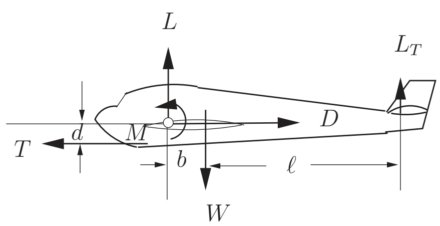

Figure 1.6 Forces and moments on a plane in level flight.

forces for a plane with its engines mounted under the wing. The wing is set at angle of incidence α_i with respect to the fuselage to generate lift during a level flight. For the equilibrium of the plane, we require

$$L + L_T - W = 0, \quad T - D = 0 \tag{1.3}$$

$$Td - M - L_T\ell + Wb = 0. \tag{1.4}$$

The last equation shows the important role of tail lift in balancing the plane.

1.3 Load Factor

Structural members experience amplified loads when a plane undergoes acceleration. The basic loads experienced during level flight are multiplied by a factor known as the load factor to take this amplification into account. We may illustrate

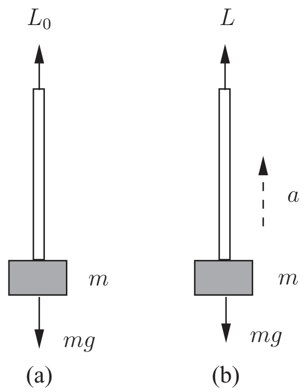

Figure 1.7 Load on a bar under (a) static and (b) dynamic conditions.

this using a simple bar with a cross-sectional area A supporting a mass m. As shown in Fig. 1.7, under static conditions, L_0 balances the weight of the mass, mg, and the stress in the bar is

$$\sigma_0 = \frac{L_0}{A} = \frac{mg}{A} = \frac{W}{A}. \tag{1.5}$$

If we give a vertical acceleration a to this system, for dynamic equilibrium, Newton's second law requires

$$L = m(g + a), \quad \sigma = \frac{mg}{A}\left(1 + \frac{a}{g}\right). \tag{1.6}$$

Then the load factor n is obtained as

$$n = \frac{\sigma}{\sigma_0} = \frac{L}{W} = 1 + \frac{a}{g}. \tag{1.7}$$

Of course, under static conditions $n = 1$.

We use n as a load factor for accelerations perpendicular to the fuselage. There are cases where the plane may be pitching and the acceleration is centrifugal (parallel to the fuselage). The load factor in that case is distinguished using the notation n_x.

1.4 Maneuvers

Even if the plane is not designed for aerobatics, there are certain maneuvers all planes have to undergo. Pullout (or pull-up) from a dive (or descent) and banked

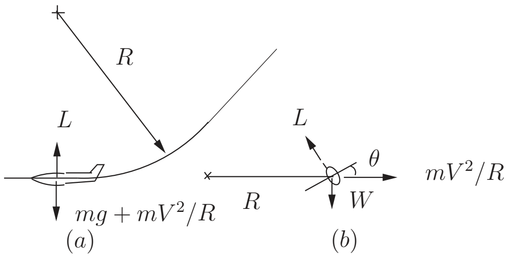

Figure 1.8 Two typical airplane maneuvers: (a) Pullout from a dive and (b) banked turn.

turns are two of these maneuvers. Fig. 1.8 shows the forces acting on the plane during these maneuvers. In addition to the weight and the lift forces, we also have centripetal forces due to the circular motions.

As shown Fig. 1.8(a), in a pullout, balance of forces gives

$$L = mg + \frac{mV^2}{R} = mg\left(1 + \frac{V^2}{gR}\right). \tag{1.8}$$

The load factor for this case is

$$n = \frac{L}{W} = 1 + \frac{V^2}{gR}. \tag{1.9}$$

Fig. 1.8(b) shows a banked turn, with a bank angle θ. To balance the forces,

$$L^2 = W^2 + (mV^2/R)^2, \quad L = W\sqrt{1 + \frac{V^4}{g^2R^2}}. \tag{1.10}$$

Then,

$$n = \sqrt{1 + \frac{V^4}{g^2R^2}}. \tag{1.11}$$

We also have

$$L\cos\theta = W, \quad n = \frac{L}{W} = \frac{1}{\cos\theta}. \tag{1.12}$$

For example, when $\theta = 60^\circ$, we get $n = 2$.

The load factor n is also known as the *g-factor* in everyday language.

1.5 Gust Load

Consider a plane in level flight with a speed V hit with an upward gust with speed v. Some times the gust speed builds up slowly, and we use the term “graded gust” to describe that situation. The situation we have is a “sharp-edged” gust.

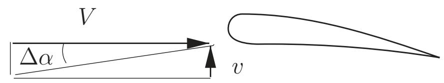

Figure 1.9 Effective increase in angle of attack due to a vertical gust.

As shown in Fig. 1.9, the direction of the net airflow has changed and the effective angle of attack of the airfoil has increased. If $v \ll V$, the change in the angle of attack is

$$\Delta\alpha = \frac{v}{V}. \tag{1.13}$$

Using

$$L = \frac{1}{2}\rho V^2 S C_{L\alpha}\alpha, \tag{1.14}$$

the change in the lift is

$$\Delta L = \frac{1}{2}\rho V^2 S C_{L\alpha}\Delta\alpha. \tag{1.15}$$

If $L/W = n$, the load factor will change by the amount

$$\Delta n = \frac{1}{2}\frac{\rho V^2 S C_{L\alpha}}{W}\frac{v}{V}. \tag{1.16}$$

Of course, this change would result in added loads on the structures.

1.6 *V–n* Diagram

A diagram with the speed V of the plane on one axis and the load factor n on the other axis is known as the V–n diagram. This diagram shows a restricted corridor in which a plane is designed to operate. The maximum values of n for normal flight and for inverted flights are given as part of the design specification. Military aircraft require much larger values of n compared with civil aircraft. At takeoff, with the flaps extended, the lift coefficient reaches a maximum value and there is a minimum speed known as the takeoff speed to lift the plane from the ground. At level flight, the load factor is unity. Maximum values of n may occur during a pullout from a dive or during a banked turn. Fig. 1.10 shows a sketch of the V–n diagram.

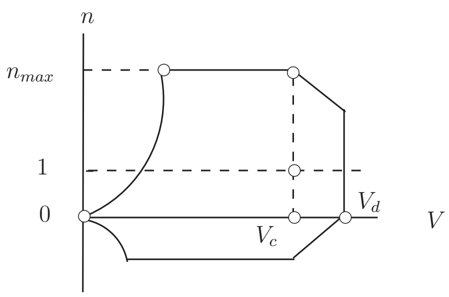

Figure 1.10 Typical shape of the V–n diagram.

The area below $n = 0$ represents inverted flights and the area above normal flights. The speed V_c with $n = 1$ is a point inside the diagram representing cruise condition. A possible dive speed is indicated by V_d. For a plane to lift off, the flaps are deflected to attain the maximum lift curve slope C_{Lmax} and the speed is increased to V_{stall}. Balancing the lift and weight, we find

$$n = 1 = \frac{1}{2}\frac{\rho S C_{Lmax}}{W} V_{stall}^2. \tag{1.17}$$

When the speed is higher than V_{stall}, n increases with the speed V parabolically as

$$n = \frac{1}{2}\frac{\rho S C_{Lmax}}{W} V^2. \tag{1.18}$$

For a semi-aerobatic plane, the load factor is bounded as $-1.8 < n < 4.5$ and for a fully aerobatic plane, $-3 < n < 6$. Generally, these bounds are specified by the customers.

1.7 Proof Load and Ultimate Load

With the expected maximum load factor n_{max}, we have to use a factor safety in the actual strength of any designed component. This factor is considered in two steps. In the first step, we multiply n_{max} by a factor of 1.25 to obtain the proof load factor. The meaning of the term "proof load" is that an actual manufactured structural component can be subjected to the proof load in a laboratory without it sustaining any permanent damage. In the second step, we multiply n_{max} by a factor of 1.5 to get the ultimate load. The factors 1.25 and 1.5 are the recommended values from the cumulative experience of aircraft design over the years. A component is expected to survive up to the ultimate load. Compared to mechanical and civil engineering structures, the demand for weight reduction in aero-structures limits the margin of safety in our field. In addition, we also have to take into account the statistical variability in the mechanical properties of materials. Modern manufacturing processes endeavor to minimize property variations by controlling the chemical compositions of the materials, the heat treatment, and the machining.

1.8 Optimization

Aircraft design is a cyclic process. In the overall takeoff weight calculation, we proceed with an assumed empty weight as a fraction of the total weight. This fraction is called the *structure factor*. It varies from about 0.5 for large commercial airplanes to 0.7 for for small personal planes. From this step, the performance criteria, the aerodynamic surfaces, and engine capacity are obtained. Then, we go back to the design of structures – with the objective of minimum weight, which leads to minimum cost. Often, in aircraft design, we face multiple solutions for structural configurations. Structural optimization to attain minimum weight is a computationally intensive process which is worthwhile considering the reduction

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in weight and the savings in fuel consumption. Material selection is affected by the cost and the reduction in weight. Detailed design also uses extensive finite element computations of stresses and displacements to obtain accurate values for expected stresses.

FURTHER READING

Abbott, I. and Von Doenhoff, A., *Theory of Wing Sections*, Dover (1949).
Arora, J. S., *Introduction to Optimum Design*, McGraw-Hill (1989).
Corke, T. C., *Design of Aircraft*, Prentice Hall (2003).
Dassault Systemes, ABAQUS Inc., Finite Element Software for Stress Analysis.
Haftka, R. T. and Kamat, M. P., *Elements of Structural Optimization*, Martinus Nijhoff (1985).
Nicolai, L., *Fundamentals of Aircraft Design*, University of Dayton (1975).
Perkins, C. D. and Hage, R. E., *Airplane Performance Stability and Control*, John Wiley (1949).
Zienkiewicz, O. C., *The Finite Element Method*, McGraw-Hill (1979).

EXERCISES

1.1 A plane is modeled as a uniform beam of length 15 m. The fuselage weighs 3 MN per meter. Measuring x from the nose, the weight of the two wings and engines, 30 MN, is applied at $x = 7$ m. The wing lift acts at $x = 6$ m and the tail lift acts at $x = 14$ m. Balancing the plane, obtain the values of the wing and tail lifts. Draw the bending moment and shear force diagrams for level flight. If this plane experiences an upward acceleration of 3 g, what are the maximum bending moment and shear force?

1.2 A vertical bar of length 2 m in an airplane carries a mass of 1000 kg at its lower end. The anticipated load factor for this structure is 3.5. What are the limit load, the proof load, and the ultimate load for this bar based on the factors given in Section 1.7? Design the minimum diameter of the bar if the material fails at 450 MPa. Neglect the weight of the bar in the first attempt. If the weight is included, assuming the bar is made of an aluminum alloy with density 2.78 g/cm^3, what is the new diameter?

1.3 A wing is modeled as a cantilever beam generating a total lift of 32 MN. It has a length of 8 m. An engine weighing 10 MN is mounted at the mid-point. Draw the shear force and bending moment diagrams and obtain their maximum values.

1.4 A plane of mass 3000 kg is in level flight at a speed of 300 km/hr with the wings set at an angle of incidence of 3°. If a sharp-edged vertical gust hits the plane with a speed of 30 km/hr, what is the load factor for the plane?

1.5 A wing has a chord of 3 m and it is set at an incident angle of 4° at level flight at sea level. The lift curve slope for the wing, $C_{L\alpha}$, is 4 per radian. Obtain the wing span for a rectangular wing to support a plane of mass 3000 kg flying at a speed of 300 km/hr. The air density at sea level is 1.226 kg/m^3. Neglect the lift contribution from the tail.

1.6 A plane diving at an angle of 15° with the horizon with a speed of 200 km/hr is leveled by pulling up a curve of radius R. If the maximum load factor allowed is 3, compute the minimum value of R.

1.7 A plane weighing 30 MN is executing a banked turn at an angle of 40°. Compute the load factor for this maneuver if there is no slipping.