AP3456
The Central Flying School (CFS) Manual of Flying

Volume 1 – The Principles of Flight

CHAPTER 1 - THE ATMOSPHERE

Introduction

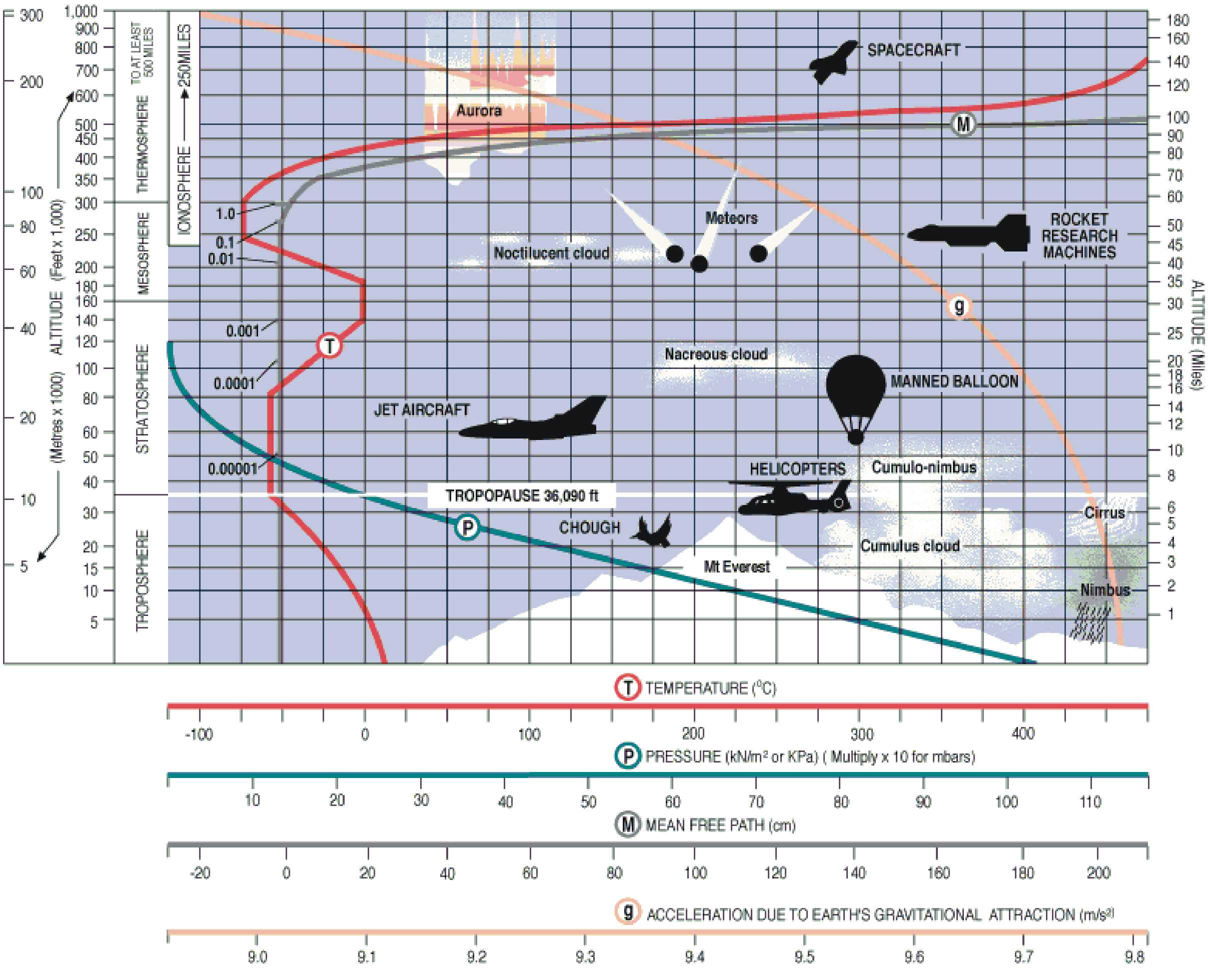
1. The atmosphere is the term given to the layer of air which surrounds the Earth and extends upwards from the surface to about 500 miles. The flight of all objects using fixed or moving wings to sustain them, or air-breathing engines to propel them, is confined to the lower layers of the atmosphere. The properties of the atmosphere are therefore of great importance to all forms of flight.

2. The Earth's atmosphere can be said to consist of four concentric gaseous layers. The layer nearest the surface is known as the troposphere, above which are the stratosphere, the mesosphere and the thermosphere. The boundary of the troposphere, known as the tropopause, is not at a constant height but varies from an average of about 25,000 ft at the poles to 54,000 ft at the equator. Above the tropopause the stratosphere extends to approximately 30 miles. At greater heights various authorities have at some time divided the remaining atmosphere into further regions but for descriptive purposes the terms mesosphere and thermosphere are used here. The ionosphere is a region of the atmosphere, extending from roughly 40 miles to 250 miles altitude, in which there is appreciable ionization. The presence of charged particles in this region, which starts in the mesosphere and runs into the thermosphere, profoundly affects the propogation of electromagnetic radiations of long wavelengths (radio and radar waves).

3. Through these layers the atmosphere undergoes a gradual transition from its characteristics at sea level to those at the fringe of the thermosphere, which merges with space. The weight of the atmosphere is about one millionth of that of the Earth, and an air column one square metre in section extending vertically through the atmosphere weighs 9,800 kg. Since air is compressible, the troposphere contains much the greater part (over three quarters in middle latitudes) of the whole mass of the atmosphere, while the remaining fraction is spread out with ever-increasing rarity over a height range of some hundred times that of the troposphere.

4. Average representative values of atmospheric characteristics are shown in Fig 1. It will be noted that the pressure falls steadily with height, but that temperature falls steadily to the tropopause, where it then remains constant through the stratosphere, but increasing for a while in the warm upper layers. Temperature falls again in the mesosphere and eventually increases rapidly in the thermosphere. The mean free path (M) in Fig 1 is an indication of the distance of one molecule of gas from its neighbours, thus, in the thermosphere, although the individual air molecules have the temperatures shown, their extremely rarefied nature results in a negligible heat transfer to any body present.

1-1 Fig 1 The Atmosphere



Physical Properties of Air

5. Air is a compressible fluid and as such it is able to flow or change its shape when subjected even to minute pressures. At normal temperatures, metals such as iron and copper are highly resistant to deformation by pressure, but in liquid form they flow readily. In solids the molecules adhere so strongly that large forces are needed to change their position with respect to other molecules. In fluids, however, the degree of cohesion of the molecules is so small that very small forces suffice to move them in relation to each other. A fluid in which there is no cohesion between the molecules, and therefore no internal friction, and which is incompressible would be an 'ideal' fluid - if it were obtainable.

6. **Fluid Pressure.** At any point in a fluid the pressure is the same in all directions, and if a body is immersed in a stationary fluid, the pressure on any point of the body acts at right angles to the surface at that point irrespective of the shape or position of the body.

7. **Composition of Air**. Since air is a fluid having a very low internal friction it can be considered, within limits, to be an ideal fluid. Air is a mixture of a number of separate gases, the proportions of which are:

1-1 Table 1 Composition of Air

Element	*By Volume %*	*By Weight %*
Nitrogen	78.08	75.5
Oxygen	20.94	23.1
Argon	0.93	1.3
Carbon dioxide	0.03	0.05
Hydrogen		
Neon		
Helium		
Krypton		Traces only
Xenon		
Ozone		
Radon		

For all practical purposes the atmosphere can be regarded as consisting of 21% oxygen and 78% nitrogen by volume. Up to a height of some five to six miles water vapour is found in varying quantities, the amount of water vapour in a given mass of air depending on the temperature and whether the air is, or has recently been, over large areas of water. The higher the temperature the greater the amount of water vapour that the air can hold.

Measurement of Temperature

8. Temperature can be measured against various scales:

a. The Celsius scale (symbol º C) is normally used for recording atmospheric temperatures and the working temperatures of engines and other equipment. On this scale, water freezes at 0º C and boils at 100º C, at sea level.

b. On the Kelvin thermodynamic scale, temperatures are measured in kelvins (symbol K - note there is no degree sign) relative to absolute zero. In the scientific measurement of temperature, 'absolute zero' has a special significance; at this temperature a body is said to have no heat whatsoever. Kelvin zero occurs at –273.15º C.

c. On the Fahrenheit scale (symbol º F), water freezes at 32º F and boils at 212º F, at sea level. This scale is still used, particularly in the USA.

9. **Conversion Factors**. A kelvin unit equates to one degree C, therefore to convert º C to kelvins, add 273.15. To convert º F to º C, subtract 32 and multiply by $\frac{5}{9}$; to convert º C to º F, multiply by $\frac{9}{5}$ and add 32.

Standard Atmosphere

10. The values of temperature, pressure and density are never constant in any given layer of the atmosphere, in fact, they are all constantly changing. Experience has shown that there is a requirement for a standard atmosphere for the comparison of aircraft performances, calibration of altimeters and other practical uses. A number of standards are in existence but Britain uses the International Standard Atmosphere (ISA) defined by the International Civil Aviation Organization (ICAO).

11. The ISA assumes a mean sea level temperature of +15º C, a pressure of 1013.25 hPa* (14.7 psi) and a density of 1.225 kg/m^3. The temperature lapse rate is assumed to be uniform at the rate of

1.98º C per 1,000 ft (6.5º C per kilometre) up to a height of 36,090 ft (11 km), above which height it remains constant at – 56.5º C (see Table 2).

1-1 Table 2 ICAO Standard Atmosphere

Altitude (ft)	*Temperature (º C)*	*Pressure (hPa / mb)*	*Pressure (psi)*	*Density (kg/m³)*	*Relative Density* (%)
0	+15.0	1013.25	14.7	1.225	100.0
5,000	+5.1	843.1	12.22	1.056	86.2
10,000	–4.8	696.8	10.11	0.905	73.8
15,000	–14.7	571.8	8.29	0.771	62.9
20,000	–24.6	465.6	6.75	0.653	53.3
25,000	–34.5	376.0	5.45	0.549	44.8
30,000	–44.4	300.9	4.36	0.458	37.4
35,000	–54.3	238.4	3.46	0.386	31.0
40,000	–56.5	187.6	2.72	0.302	24.6
45,000	–56.5	147.5	2.15	0.237	19.4
50,000	–56.5	116.0	1.68	0.186	15.2

*Note: The most commonly used unit of pressure is the hectopascal (hPa). Some references may refer to the millibar (mb) which is equivalent to the hPa.

Density

12. Density (symbol rho (ρ)) is the ratio of mass to volume, and is expressed in kilograms per cubic metre (kg/m³). The relationship of density to temperature and pressure can be expressed thus:

$$\frac{p}{T\rho} = \text{constant}$$

where p = Pressure in hectopascals

and T = Absolute temperature (i.e. measured on the Kelvin scale)

13. **Effects of Pressure on Density**. When air is compressed, a greater amount can occupy a given volume; i.e. the mass, and therefore the density, has increased. Conversely, when air is expanded less mass occupies the original volume and the density decreases. From the formula in para 12 it can be seen that, provided the temperature remains constant, density is directly proportional to pressure, ie if the pressure is halved, so is the density, and vice versa.

14. **Effect of Temperature on Density**. When air is heated it expands so that a smaller mass will occupy a given volume, therefore the density will have decreased, assuming that the pressure remains constant. The converse will also apply. Thus the density of the air will vary inversely as the absolute temperature: this is borne out by the formula in para 12. In the atmosphere, the fairly rapid drop in pressure as altitude is increased has the dominating effect on density, as against the effect of the fall in temperature which tends to increase the density.

15. **Effect of Humidity on Density**. The preceding paragraphs have assumed that the air is perfectly dry. In the atmosphere some water vapour is invariably present; this may be almost negligible in certain conditions, but in others the humidity may become an important factor in the performance of an aircraft. The density of water vapour under standard sea level conditions is 0.760 kg/m³. Therefore water vapour can be seen to weigh 0.760/1.225 as much as air, roughly ⅝ as much as air at sea level. This means that under standard sea level conditions the portion of a mass of air which holds water

vapour weighs (1 – ⅝), or ⅜ less than it would if it were dry. Therefore air is least dense when it contains a maximum amount of water vapour and most dense when it is perfectly dry.

Altitude

16. **Pressure Altitude**. Pressure altitude can be defined as the vertical distance from the 1013.25 hPa pressure level. When the term 'altitude' appears in Operating Data Manuals (ODMs) and performance charts, it refers strictly to pressure altitude. Therefore, when the sea level pressure is other than 1013.25 hPa, aerodrome and obstacle elevations must be converted to pressure altitude before use in performance calculations. ODMs normally contain a conversion graph. Pressure altitude can be obtained by setting the sub-scale of an ICAO calibrated altimeter to 1013.25 hPa and reading altitude directly from the instrument. Alternatively, the approximate pressure altitude can be calculated. At sea level, 1 hPa difference in pressure is equivalent to approximately 27 ft of height change; at 20,000 ft, 1 hPa equates to approximately 50 ft. Thus, for calculations close to sea level, it can be assumed that 1 hPa = 30 ft. Pressure altitude at any point can therefore be determined by the formula:

$$\text{Pressure altitude} \doteqdot \text{Elevation} + 30p$$

where p is 1013 minus the sea level pressure at that point.

Example: To determine the pressure altitude of an airfield, elevation 1,700 ft, if sea level pressure is 1003 hPa:

$$p = 1013 - 1003 = 10 \text{ hPa}$$

$$\therefore \text{airfield pressure altitude} \doteqdot 1{,}700 + (30 \times 10)\text{ ft} \doteqdot 2{,}000 \text{ ft}$$

17. **Density Altitude**. For aircraft operations, air density is usually expressed as a density altitude. Density altitude is the pressure altitude adjusted to take into consideration the actual temperature of the air. For ISA conditions of temperature and pressure, density altitude is the same as pressure altitude. Knowledge of the density altitude can be of particular advantage in helicopter operations and is described further in Volume 2, Chapter 8. Density altitude can be determined by the formula:

$$\text{density altitude} = \text{pressure altitude} + 120t$$

where t is the actual air temperature minus the standard (ISA) temperature for that pressure altitude. Continuing the calculation from para 16, if the actual air temperature at the airfield elevation was +13º C (ISA temp for 2,000 ft is +11º C), then the density altitude would be:

$$2{,}000 \text{ ft} + 120\,(13^\circ\text{ C} - 11^\circ\text{ C}) = 2{,}000 \text{ ft} + 120\,(2) = 2{,}240 \text{ ft}$$

Dynamic Pressure

18. Because it possesses density, air in motion must possess energy and therefore exerts a pressure on any object in its path. This dynamic pressure is proportional to the density and the square of the speed. The energy due to movement (the kinetic energy (KE)) of one cubic metre of air at a stated speed is given by the following formula:

$$KE = \tfrac{1}{2}\rho V^2 \text{ joules}$$

where ρ is the local air density in kg/m^3
and V is the speed in metres per second (m/s)

If this volume of moving air is completely trapped and brought to rest by means of an open-ended tube the total energy remains constant. In being brought to rest, the kinetic energy becomes pressure energy (small losses are incurred because air is not an ideal fluid) which, for all practical purposes, is equal to $\frac{1}{2}\rho V^2$ newtons/m^2, or if the area of the tube is S m^2, then:

$$\text{total pressure (dynamic + static)} = \tfrac{1}{2}\rho V^2 S \text{ newtons.}$$

19. The term $\frac{1}{2}\rho V^2$ is common to all aerodynamic forces and fundamentally determines the air loads imposed on an object moving through the air. It is often modified to include a correction factor or coefficient. The term stands for the dynamic pressure imposed by air of a certain density moving at a given speed, which is brought completely to rest. The abbreviation for the term $\frac{1}{2}\rho V^2$ is the symbol 'q'. Note that dynamic pressure cannot be measured on its own, as the ambient pressure of the atmosphere (known as static) is always present. The total pressure (dynamic + static) is also known as stagnation or pitot pressure (see Volume 1, Chapter 2). It can be seen that:

$$\text{total pressure} - \text{static pressure} = \text{dynamic pressure}$$

MEASUREMENT OF SPEED

Method of Measuring Air Speed

20. It is essential that an aircraft has some means of measuring the speed at which it is passing through the air. The method of doing this is by comparing the total pressure (static + dynamic) with static pressure. An instrument measures the difference between the two pressures and indicates dynamic pressure in terms of speed, the indicated speed varying approximately as the square root of the dynamic pressure. This instrument is known as an air speed indicator (ASI) and is described in detail in Volume 5, Chapter 5.

Relationships Between Air Speeds

21. The general term 'air speed' is further qualified as:

a. **Indicated Air Speed (IAS)(VI)**. Indicated air speed is the reading on the ASI, and the speed reference by which the pilot will normally fly.

b. **Calibrated Air Speed (CAS)(Vc)**. When the IAS has been corrected by the application of pressure error correction (PEC) and instrument error correction, the result is known as calibrated air speed (CAS). PEC can be obtained from the Aircrew Manual or ODM for the type. When the PEC figures for individual instruments are displayed on the cockpit, they include the instrument error correction for that particular ASI (Note: In practice, any instrument error is usually very small and can, for all practical purposes, be ignored.). CAS was once termed RAS (Rectified Air Speed), a term still found in old texts and etched on some circular slide rules.

c. **Equivalent Air Speed (EAS)(Ve)**. Equivalent air speed is obtained by adding the compressibility error correction (CEC) to CAS.

d. **True Air Speed (TAS)(V)**. True air speed is the true speed of an aircraft measured relative to the undisturbed air mass through which it is moving. TAS is obtained by dividing the EAS by the square root of the relative air density.

22. The ASI can be calibrated to read correctly for only one density/altitude. All British ASIs are calibrated for ICAO atmosphere conditions. Under these conditions and at the standard sea level density (ρ_o) the EAS (V_e) is equal to the TAS (V). At any other altitude where the density is ρ, then:

$$V_e = V\sqrt{\frac{\rho}{\rho_o}} = V\sqrt{\sigma}$$

where σ is the relative density.

Thus, at 40,000 ft where the standard density is one quarter of the sea level value, the EAS will be ½ × TAS.

23. Although all the speeds mentioned in paragraph 21 have their own importance, the two most significant, in terms of aerodynamics, are EAS and TAS. The TAS is significant because it gives a measure of the speed of a body relative to the undisturbed air and the EAS is significant because the aerodynamic forces acting on an aircraft are directly proportional to the dynamic pressure and thus the EAS.

24. The TAS may be calculated in the following ways:

a. By navigational computer (see Volume 9, Chapter 8).

b. From a graph. Some ODMs contain IAS/TAS conversion graphs and these enable TAS to be computed from a knowledge of height, indicated air speed and air temperature.

c. By mental calculation (see Volume 9, Chapter 19).

Mach Number

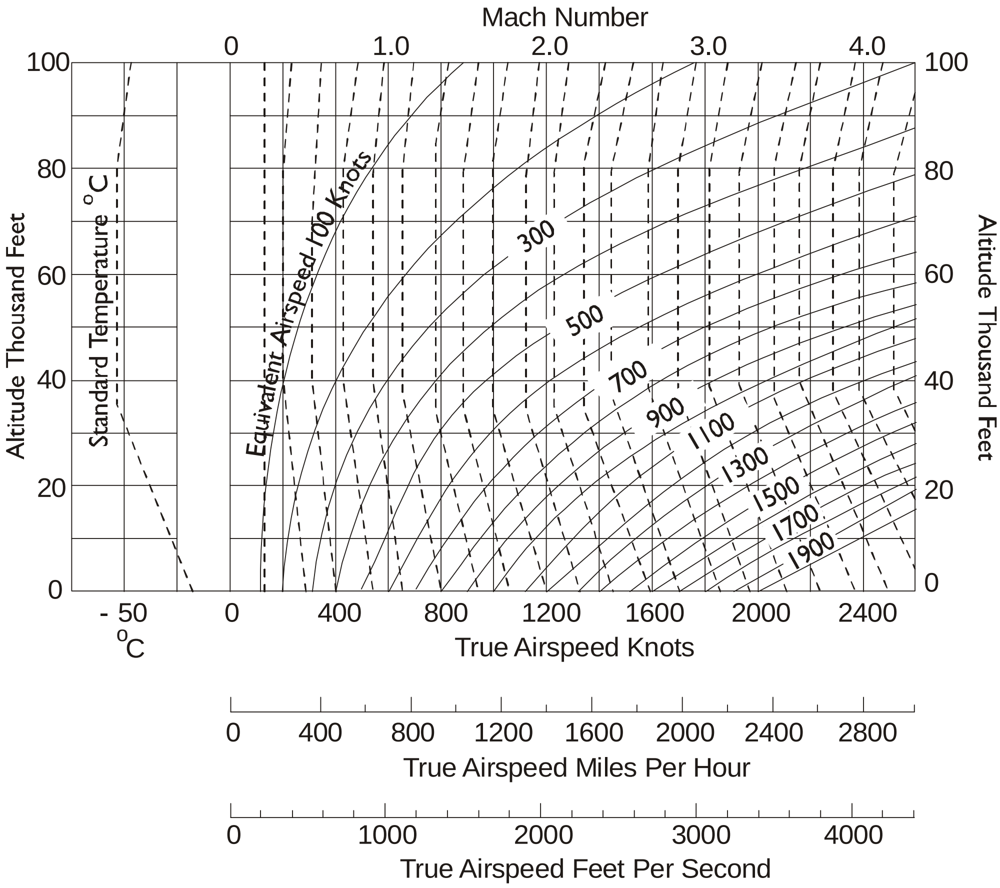
25. Small disturbances generated by a body passing through the atmosphere are transmitted as pressure waves, which are in effect sound waves, whether audible or not. In considering the motion of the body, it is frequently found convenient to express its velocity relative to the velocity of these pressure waves. This ratio is Mach number (M).

$$M = \frac{V}{a}$$

where V = TAS and a = local speed of sound in air,which varies as the square root of the absolute temperature of the local air mass.

26. Fig 2 shows the theoretical relationship between TAS, EAS and Mach number.

1-1 Fig 2 Theoretical Relationship Between TAS, EAS and Mach Number



CHAPTER 2 - AERODYNAMIC FORCE

Introduction

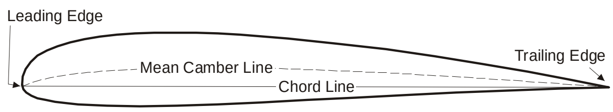
1. In aerodynamics, as in most subjects, a number of terms and conventional symbols are used. The most commonly used terms and symbols are described in the first section of this chapter. While many of these definitions are self-explanatory, some are less obvious and need to be accepted until they are explained in later chapters. The second part of this chapter discusses basic aerodynamic theory.

DEFINITIONS AND SYMBOLS

Aerofoil and Wing

2. **Aerofoil**. An aerofoil is an object which, as a result of its shape, produces an aerodynamic reaction perpendicular to its direction of motion. All aerofoils are subject to resistance in their direction of motion. This resistance is known as 'drag', while the perpendicular aerodynamic reaction is known as 'lift'. The cross-sectional profile of an aerofoil is referred to as an 'aerofoil section' (see Fig 1).

1-2 Fig 1 An Aerofoil Section



3. **Mean Camber Line or Mean Line**. As shown in Fig 1, the mean camber line is a line equidistant from the upper and lower surfaces of an aerofoil section. The points where the mean camber line meets the front and back of the section are known as the 'leading edge' (LE) and 'trailing edge' (TE) respectively.

4. **Chord Line**. The chord line is a straight line joining the leading and trailing edges of an aerofoil (see Fig 1).

5. **Chord**. The chord is the distance between the leading and trailing edge measured along the chord line.

6. **Mean Chord**. The mean chord of a wing is the average chord, ie the wing area divided by the span.

7. **Maximum Camber**. Maximum camber is usually expressed as:

$$\frac{\text{Maximum distance between the camber line and the chord line}}{\text{Chord}}$$

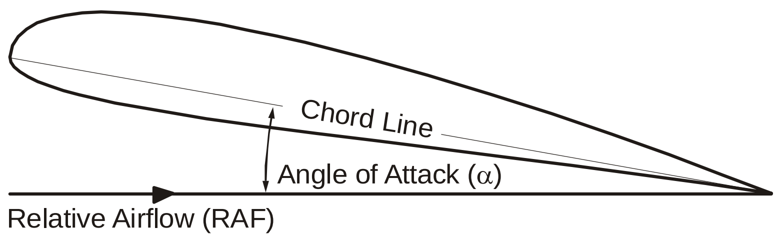
8. **Positive Camber**. Where the mean camber line lies above the chord line, the aerofoil is said to have positive camber (as in Fig 1). If the mean camber line is coincident with the chord line, the aerofoil is symmetrical.

9. **Wing Span**. Wing span is the maximum lateral dimension of a wing (wingtip to wingtip).

10. **Wing Area**. The wing area is the planform area of the wing (as opposed to surface area), including those parts of the wing where it crosses the fuselage.

11. **Angle of Attack**. The angle of attack (α) is the angle between the chord line and the flight path or relative airflow (RAF) (see Fig 2).

1-2 Fig 2 Angle of Attack



12. **Angle of Incidence**. The angle of incidence is that angle at which an aerofoil is attached to the fuselage (sometimes referred to as the 'rigger's angle of incidence'). It can be defined as the angle between the chord line and the longitudinal fuselage datum. The term is sometimes used erroneously instead of angle of attack.

13. **Thickness/Chord Ratio**. The maximum thickness or depth of an aerofoil section, expressed as a percentage of chord length, is referred to as the thickness/chord (t/c) ratio.

14. **Aspect Ratio**. Aspect ratio is the ratio of span to mean chord, ie

$$\text{Aspect Ratio} = \frac{\text{span}}{\text{mean chord}} \text{ or } \frac{\text{span}^2}{\text{wing area}}$$

15. **Wing Loading**. Wing loading is the weight of the aircraft divided by its wing area, ie the weight per unit area.

16. **Load Factor**. Load Factor (n or g) is the ratio of total lift divided by weight, ie:

$$n = \frac{\text{total lift}}{\text{weight}}$$

Note that engineers tend to use 'n' to denote load factor, whereas pilots usually refer to it as 'g'.

Lift and Drag

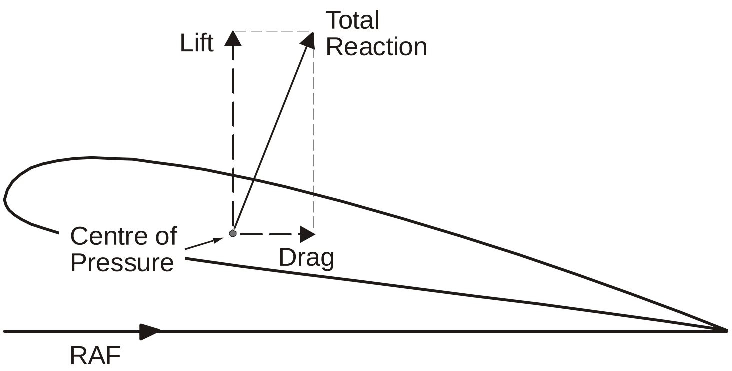
17. **Total Reaction**. The total reaction (TR) is the resultant of all the aerodynamic forces acting on the wing or aerofoil section (see Fig 3).

18. **Centre of Pressure**. The centre of pressure (CP) is the point, usually on the chord line, through which the TR may be considered to act.

19. **Lift**. Lift is the component of the TR which is perpendicular to the flight path or RAF.

20. **Drag**. Drag is the component of the TR which is parallel to the flight path or RAF.

1-2 Fig 3 Total Reaction, Lift and Drag

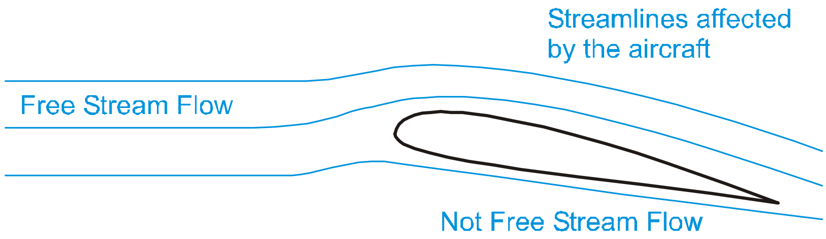


Airflow

21. **Free Stream Flow**. The free stream flow is the air in a region where pressure, temperature and relative velocity are unaffected by the passage of the aircraft through it (see Fig 4). It is also sometimes called relative airflow (RAF).

22. **Streamline**. Streamline is the term given to the path traced by a particle in a steady fluid flow (see Fig 4).

1-2 Fig 4 Free Stream Flow and Streamlines



23. **Ideal Fluid**. An ideal fluid is one with no viscosity, ie no shearing resistance to motion.

24. **Incompressible Fluid**. An incompressible fluid is one in which the density is constant.

25. **Boundary Layer**. The boundary layer is the layer of a fluid close to a solid boundary along which it is flowing. In the boundary layer, the velocity of flow is reduced from the free stream flow by the action of viscosity.

26. **Laminar or Viscous Flow**. Laminar (or viscous) flow is a type of fluid flow in which adjacent layers do not mix, except on a molecular scale.

27. **Turbulent Flow**. Turbulent flow is a type of fluid flow in which the particle motion at any point varies rapidly in both magnitude and direction. Turbulent flow gives rise to high drag, particularly in the boundary layer.

28. **Transition Point**. The transition point is that point within the boundary layer at which the flow changes from being laminar to being turbulent.

Symbols

29. The following paragraphs explain many of the common symbols used in aerodynamic theory.

30. **Density**.

a. Density at any unspecified point = ρ (rho)

b. Density at ISA mean sea level = ρ_0 = 1.2250 kg per m^3

c. Relative density = σ (sigma) = $\frac{\rho}{\rho_0}$

31. **Velocity**.

a. Equivalent air speed (EAS) = V_e

b. True air speed (TAS) = V, then $V_e = V\sqrt{\sigma}$

32. **Pressure**.

a. Static pressure at any unspecified point = p

b. Free stream static pressure = p_0

c. Dynamic pressure = q = $\frac{1}{2}\rho V^2$ or q = $\frac{1}{2}\rho_0 V_e^2$

d. Total pressure (also called total head pressure, stagnation pressure, or pitot pressure) is usually denoted by p_S.

33. **Angle of Attack of Aerofoil or Wing**.

a. Angle of attack = α (alpha)

b. Zero lift angle of attack = α_0 (alpha nought)

34. **Coefficient of Lift**.

a. *Aerofoil*.

$$C_1 = \frac{\text{lift/span}}{qc}$$

b. *Wing*.

$$C_L = \frac{\text{lift}}{qS}$$

Where S = wing area, c = chord

35. **Coefficient of Drag**.

a. *Aerofoil*. $C_d = \dfrac{\text{drag/span}}{qc}$

b. *Wing*. $C_D = \dfrac{\text{drag}}{qS}$

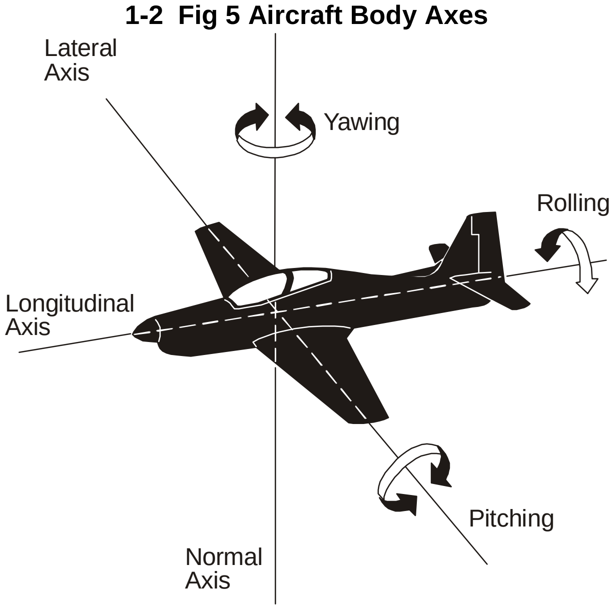
Where S = wing area, c = chord

36. **Coefficient of Pressure**.

$$\frac{\text{pressure differential}}{\text{dynamic pressure}} = \frac{p - p_0}{q}$$

37. **Aircraft Body Axes - Notation**. Aircraft body axes are used when explaining aircraft stability. They are illustrated in Fig 5; the associated notation is listed in Table 1.

1-2 Fig 5 Aircraft Body Axes



1-2 Table 1 Aircraft Body Axes - Notation

		Longitudinal	*Lateral*	*Normal*
Axis	Symbol Positive Direction	x Forward	y To Right	z Down
Force	Symbol	X	Y	Z
Moment	Symbol Designation	L Rolling	M Pitching	N Yawing
Angle of Rotation	Symbol	ϕ (phi)	θ (theta)	ψ (psi)
Velocity during a disturbance	Linear Angular	u p	v q	w r
Moment of Inertia	Symbol	A	B	C

AERODYNAMIC PRINCIPLES

General

38. Several methods or theories have been developed to predict the performance of a given wing/aerofoil shape. These can be used to explain the subtle changes in shape necessary to produce the required performance appropriate to the role of the aircraft. In practice, the appropriate wing shape is calculated from the required performance criteria.

39. The theory in this volume is confined to the Equation of Continuity and Bernoulli's Theorem, thereby explaining lift by pressure distribution. Volume 1, Chapter 3 contains other theories, namely the Momentum Theory, Circulation Theory and Dimensional Analysis.

Pressure Distribution

40. The most useful non-mathematical method is an examination of the flow pattern and pressure distribution on the surface of a wing in flight. This approach will reveal the most important factors affecting the amount of lift produced, based on experimental (wind-tunnel) data. As a qualitative method however, it has limitations and the student will sometimes be presented with facts capable only of experimental proof or mathematical proof.

41. The pattern of the airflow round an aircraft at low speeds depends mainly on the shape of the aircraft and its attitude relative to the free stream flow. Other factors are the size of the aircraft, the density and viscosity of the air and the speed of the airflow. These factors are usually combined to form a parameter known as Reynolds Number (RN), and the airflow pattern is then dependent only on shape, attitude and Reynolds Number (see also para 73 et seq).

42. The Reynolds Number (i.e. size, density viscosity and speed) and condition of the surface determine the characteristics of the boundary layer. This, in turn, modifies the pattern of the airflow and distribution of pressure around the aircraft. The effect of the boundary layer on the lift produced by the wings may be considered insignificant throughout the normal operating range of angles of attack. In later chapters it will be shown that the behaviour of the boundary layer has a profound effect on the lift produced at high angles of attack.

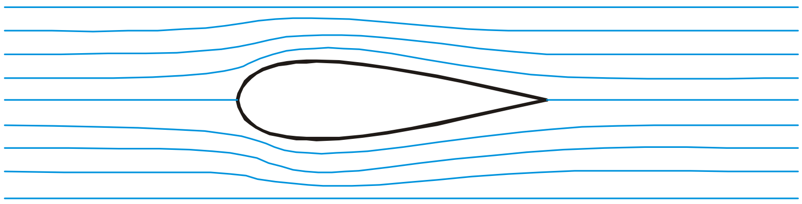
43. When considering the velocity of the airflow it does not make any difference to the pattern whether the aircraft is moving through the air or the air is flowing past the aircraft; it is the relative velocity which is the important factor.

Types of Flow

44. **Steady Streamline Flow**. In a steady streamline flow the flow parameters (eg speed, direction, pressure etc) may vary from point to point in the flow but, at any point, are constant with respect to time. This flow can be represented by streamlines and is the type of flow which it is hoped will be found over the various components of an aircraft. Steady streamline flow may be divided into two types:

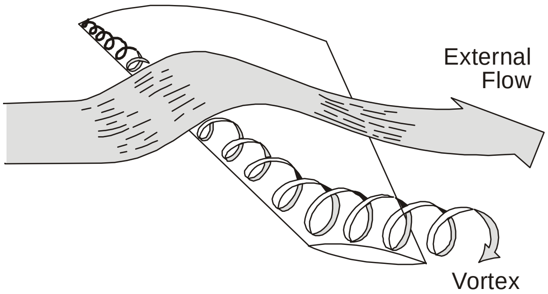
   a. **Classical Linear Flow***.* Fig 6 illustrates the flow found over a conventional aerofoil at low angle of attack, in which the streamlines all more or less follow the contour of the body, and there is no separation of the flow from the surface.

1-2 Fig 6 Classical Linear Flow



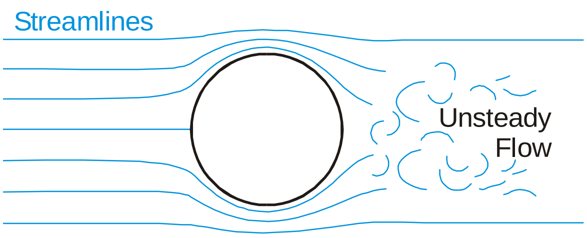
b. **Controlled Separated Flow or Leading Edge Vortex Flow.** This is a halfway stage between steady streamline flow and unsteady flow described later. Due to boundary layer effects, generally at a sharp leading edge, the flow separates from the surface, not breaking down into a turbulent chaotic condition but, instead, forming a strong vortex which, because of its stability and predictability, can be controlled and made to give a useful lift force. Such flows, illustrated in Fig 7, are found in swept and delta planforms, particularly at the higher angles of attack, and are dealt with in more detail in later chapters.

1-2 Fig 7 Leading Edge Vortex Flow



45. **Unsteady Flow**. In an unsteady flow, the flow parameters vary with time and the flow cannot be represented by streamlines (see Fig 8).

1-2 Fig 8 Unsteady Flow

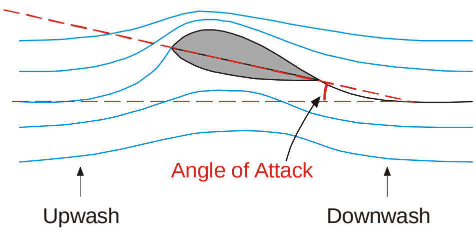


46. **Two-dimensional Flow**. If a wing is of infinite span, or, if it completely spans a wind tunnel from wall to wall, then each section of the wing will have exactly the same flow pattern round it, except near the tunnel walls. This type of flow is called two-dimensional flow since the motion in any one plane parallel to the free stream direction is identical to that in any other, and there is no cross-flow between these planes.

47. **Velocity Indication**. As the air flows round the aircraft, its speed changes. In subsonic flow, a reduction in the velocity of the streamline flow is indicated by an increased spacing of the streamlines whilst increased velocity is indicated by decreased spacing of the streamlines. Associated with the velocity changes there will be corresponding pressure changes.

48. **Pressure Differential**. As the air flows towards an aerofoil, it will be turned towards the low pressure (partial vacuum) region at the upper surface; this is termed 'upwash'. After passing over the aerofoil the airflow returns to its original position and state; this is termed 'downwash' and is shown in Fig 9. The reason for the pressure and velocity changes around an aerofoil is explained in later paragraphs. The differences in pressure between the upper and lower surfaces of an aerofoil are usually expressed as relative pressures by '–' and '+'. However, the pressure above is usually a lot lower than ambient pressure and the pressure below is usually slightly lower than ambient pressure (except at high angles of attack), i.e. both are negative.

1-2 Fig 9 Two-dimensional Flow around an Aerofoil



49. **Three-dimensional Flow**. The wing on an aircraft has a finite length and, therefore, whenever it is producing lift the pressure differential tries to equalize around the wing tip. This induces a span-wise drift of the air flowing along the wing, inwards on the upper surface and outwards on the lower surface, producing a three-dimensional flow which crosses between the planes parallel to the free stream direction.

50. **Vortices**. Because the spanwise drift is progressively less pronounced from tip to root, the amount of transverse flow is strong at the tips and reduces towards the fuselage. As the upper and lower airflows meet at the trailing edge they form vortices, small at the wing root and larger towards the tip (see Fig 10). A short distance behind the wing, these form one large vortex in the vicinity of the wing tip, rotating clockwise on the port wing and anti-clockwise on the starboard wing; viewed from the rear. Tip spillage means that an aircraft wing can never produce the same amount of lift as an infinite span wing, for the same angle of attack.

1-2 Fig 10 Three-dimensional Flow

Note:
The end 'a' of the aerofoil section abuts the wind tunnel wall and may be regarded as a wing root. The wool streamer indicates virtually no vortex.
End 'b' is a free wing tip and a marked vortex can be

51. **Vortex Influence**. The overall size of the vortex behind the trailing edge will depend on the amount of the transverse flow. It can be shown that the greater the coefficient of lift, the larger the vortex will be. The familiar pictures of wing-tip vortices, showing them as thin white streaks, only show the low pressure

central core (where atmospheric moisture is condensing) and it should be appreciated that the influence on the airflow behind the trailing edge is considerable. The number of accidents following loss of control by flying into wake vortex turbulence testifies to this. The vortices upset the balance between the upwash and downwash of two-dimensional flow, reinforcing the downwash and reducing the effective angle of attack, and inclining the total reaction slightly backwards as the effective relative airflow is now inclined downwards. The component of the total reaction parallel to the line of flight is increased; this increase is termed the induced drag. The resolved vector perpendicular to the flight direction (i.e. lift) is slightly reduced, (see Volume 1, Chapter 5).

BASIC AERODYNAMIC THEORY

General

52. The shape of the aircraft (and boundary layer) will determine the velocity changes and consequently the airflow pattern and pressure distribution. For a simplified explanation of why these changes occur it is necessary to consider:

a. The Equation of Continuity.

b. Bernoulli's Theorem.

A more detailed explanation of other aerodynamic theories is given in Volume 1, Chapter 3.

The Equation of Continuity

53. The Equation of Continuity states basically that mass can neither be created nor destroyed or, simply stated, **air mass flow rate is a constant**.

54. Consider the streamline flow of air through venturi (essentially, a tube of varying cross-sectional area). The air mass flow rate, or mass per unit time, will be the product of the cross-sectional area (A), the flow velocity (V) and the density (ρ). This product will remain a constant value at all points along the tube, i.e.

$$AV\rho = \text{constant.}$$

This is the general equation of continuity which applies to both compressible and incompressible fluids.

55. When considering compressible flow it is convenient to assume that changes in fluid density will be insignificant at Mach numbers below about 0.4 M. This is because the pressure changes are small and have little effect on the density. The equation of continuity may now be simplified to:

$$A \times V = \text{constant}, \quad \text{or} \quad V = \frac{\text{constant}}{A}$$

from which it may be seen that a reduction in the tube's cross-sectional area will result in an increase in velocity and vice versa. This, combined with Bernoulli's theorem (see para 56), is the principle of a venturi. The equation of continuity enables the velocity changes round a given shape to be predicted mathematically.

Bernoulli's Theorem

56. Consider a gas in steady motion. It possesses the following types of energy:

a. Potential energy due to height.

b. Heat energy (ie kinetic energy of molecular motion).

c. Pressure energy.

d. Kinetic energy due to bulk fluid motion.

In addition, work may be done by or on the system and heat may pass into or out of the system. However, neither occurs in the case of flow around an aerofoil.

57. Daniel Bernoulli's work demonstrated that in the **steady streamline flow outside the boundary layer**, the sum of the energies present remained constant. It is emphasised that the words in bold represent the limitations of Bernoulli's experiments. In low subsonic flow (< 0.4 M), it is convenient to regard air as being incompressible, in which case the heat energy change is insignificant. Above 0.4 M, however, this simplification would cause large errors in predicted values and is no longer permissible.

58. Bernoulli's Theorem may be simplified still further by assuming changes in potential energy to be insignificant. For practical purposes therefore, in the streamline flow of air round a wing at low speed:

$$\text{pressure energy} + \text{kinetic energy} = \text{constant}$$

It can be shown that this simplified law can be expressed in terms of pressure, thus:

$$p + \tfrac{1}{2}\rho V^2 = \text{constant},$$

where p = static pressure,
ρ = density,
V = flow velocity.

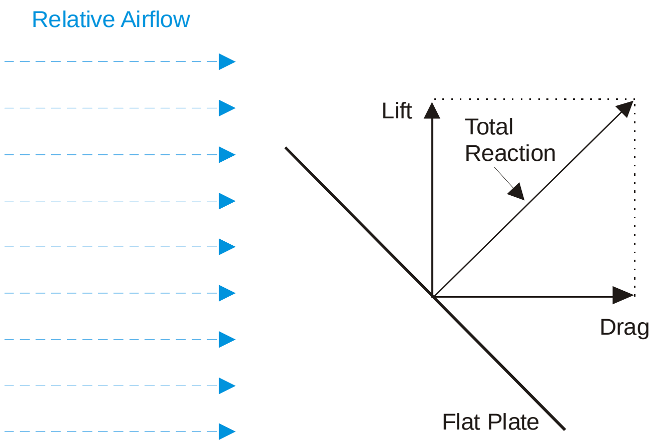
The significance of this law will be recognized if it is translated into words: static pressure + dynamic pressure is a constant. The constant is referred to as total pressure, stagnation pressure or pitot pressure. The dynamic pressure, $\tfrac{1}{2}\rho V^2$, is sometimes designated by the letter q.

59. It has already been stated that the flow velocity is governed by the shape of the aircraft. From Bernoulli's Theorem (simplified) it is evident that an increase in velocity causes a decrease in static pressure and vice versa.

The Flat Plate Effect

60. At the stall, there is a partial collapse of the low pressure on the top surface of the wing, considerably reducing the total lift. The contribution of the lower surface is relatively unchanged. The circulation theory of lift becomes invalid at any angle of attack beyond the stall; the aerofoil may then be regarded as a flat inclined plate (as in Fig 11) producing lift from the combined effect of stagnation pressure and flow deflection from the underside (change of momentum).

1-2 Fig 11 Inclined Flat Plate - Lift Acting at Centre of Area

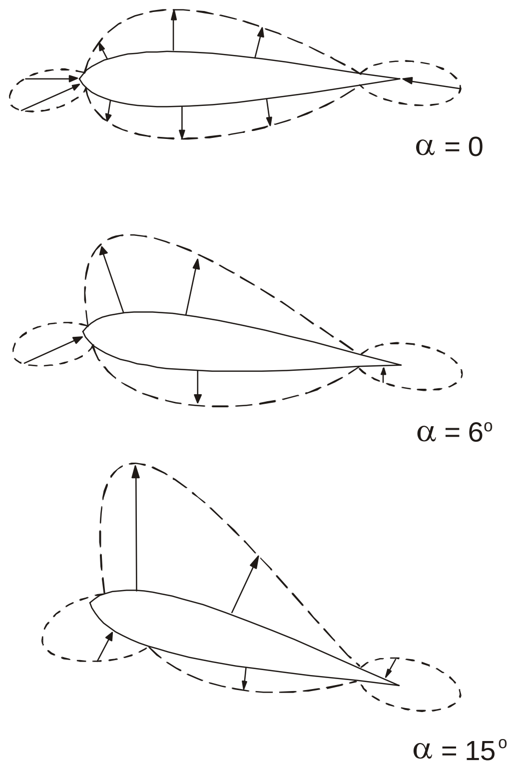


Pressure Distribution around an Aerofoil

61. Although the whole aircraft contributes towards both lift and drag, it may be assumed that the wing is specifically designed to produce the necessary lift for the whole of the aircraft. Examination of the distribution of pressure round the wing is the most convenient practical way to see how the lift is produced. Fig 12 shows the pressure distribution round a modern general-purpose aerofoil section in two-dimensional flow and how it varies with changes in angle of attack.

62. The pressure distribution around an aerofoil is usually measured with a multi-tube manometer: this consists of a series of glass tubes filled with a liquid and connected to small holes in the aerofoil surface. As air flows over the aerofoil the variations in pressure are indicated by the differing levels of liquid in the tubes.

1-2 Fig 12 Pressure Distribution around an Aerofoil



63. The pressure force at a point on the aerofoil surface may be represented by a vector arrow, perpendicular to the surface, whose length is proportional to the difference between absolute pressure at

that point and free stream static pressure p_0, i.e. proportional to $(p - p_0)$. It is usual to convert this to a non-dimensional quantity called the pressure coefficient (C_p) by comparing it to free stream dynamic pressure (q), thus:

$$C_p = \frac{(p - p_o)}{q}$$

64. The convention for plotting these pressure coefficients is as follows:

a. Measured pressure higher than ambient pressure: at these points, $(p - p_0)$ will be positive, giving a positive C_p and the pressure force vector is drawn pointing towards the surface.

b. Measured pressure lower than ambient pressure: here, $(p - p_0)$ will be negative, resulting in a negative C_p giving a force directed outwards from the surface.

65. It is useful to consider the value of C_p at the leading-edge stagnation point where the air is brought to rest. The pressure at this point is designated p_s. This pressure will be total pressure = free stream static pressure + free stream dynamic pressure. The pressure coefficient will therefore be:

$$\frac{(p_0 + q) - p_0}{q} \text{ and therefore } C_p = +1$$

C_p can never be greater than 1, since this would imply slowing the air down to a speed of less than zero, which is impossible.

66. Each of the pressure coefficient force vectors will have a component perpendicular to the free stream flow which is, by definition, a lift component. Therefore, it is possible to obtain the total lift coefficient from the pressure distribution, by subtracting all the lift components pointing down (relative to the free stream) from all those pointing up.

67. Inspection of the pressure force distribution diagrams gives an indication of the direction and magnitude of the total reaction (TR) and the position of the centre of pressure (CP). In the normal, unstalled operating range, two facts are immediately apparent:

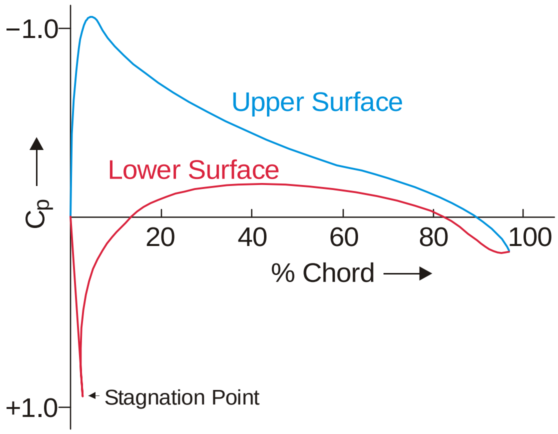
a. The lift coefficient increases with an increase in angle of attack.

b. The CP moves forward with an increase in angle of attack.

These two facts will be of considerable use in later chapters.

68. A common way to illustrate the pressure distribution round an aerofoil section is to plot it in the form of a graph (see Fig 13). Notice that, whereas in the previous diagrams the C_p force vectors are plotted perpendicular to the aerofoil surface, in the graph they are converted so that they can be plotted perpendicular to the chord line. Note also in Fig 13, the convention of plotting negative values upwards to relate the graph to lift in the natural sense.

1-2 Fig 13 Graphical Representation of Aerofoil Pressure Distribution (Angle of Attack approx 10º)



Summary

69. The airflow pattern depends on:

a. Angle of attack.

b. Shape (thickness/chord ratio, camber).

70. Lift depends on the airflow pattern. It also depends on:

a. Density of the air.

b. Size of the wing.

c. Speed.

71. The equation of continuity states that:

a. Air mass flow rate is a constant.

b. Cross-sectional area $\times$ velocity $\times$ density = constant, i.e. $AV\rho = k$. It also assumes that, at speeds below M = 0.4, pressure changes, being small, have little effect on ρ and therefore:

$$A \times V = k \text{ (a different constant)}$$

$$\therefore V = \frac{k}{A}$$

and therefore, a reduced A gives greater velocity. This (together with the Bernoulli principle) is the venturi principle.

72. Bernoulli's Theorem states that, assuming steady incompressible flow (ie M<0.4), pressure energy plus kinetic energy is constant, therefore:

$$p + \frac{1}{2}\rho V^2 = \text{a constant}$$

This constant is known as the total pressure, stagnation pressure or pitot pressure.

REYNOLDS NUMBER

Introduction

73. In the early days of flying, when aircraft speeds were of the order of 30 mph to 40 mph, testing of aerofoils was a comparatively crude operation. But as speeds increased and the design of aircraft became more sophisticated, wind tunnels were developed for the purpose of testing scale models. This led to problems in dynamic similarity; these were easier to understand than to solve.

Scale

74. If a 1/10th scale model is used, all the linear dimensions are 1/10th of those of the real aircraft, but the areas are 1/100th; and, if the model is constructed of the same materials, the mass is 1/1,000th of the real aircraft mass. So the model is to scale in some respects, but not in others. This is one of the difficulties in trying to learn from flying models of aircraft and, unless the adjustment of weights is very carefully handled, the results of tests in manoeuvres and spinning may be completely false.

Fluid Flow

75. During the 19th Century, a physicist named Reynolds was involved in experiments with the flow of fluids in pipes. He made the important discovery that the flow changed from streamlined to turbulent when the velocity reached a value which was inversely proportional to the diameter of the pipe. The larger the pipe, the lower the velocity at which the flow became turbulent, e.g. if the critical velocity in a pipe of 2.5 cm diameter was 6 m/s, then 3 m/s would be the critical velocity in a pipe of 5 cm diameter. He also discovered that the rule applied to the flow in the boundary layer around any body placed within the stream. For example, if two spheres of different sizes were placed within a flow, then almost the entire boundary layer would become turbulent when the free stream velocity reached a value which was inversely proportional to the diameter of each sphere.

Scale Correction

76. Reynolds' theorem states that the value of velocity $\times$ size must be the same for both the experimental model and for the full size aircraft it represents. This is because the nature of the boundary layer flow must be the same in both cases, if dynamic similarity is to be ensured. If it is necessary to carry out a test on a 1/10th scale model to determine what would happen on the full size aircraft at 200 kt, the wind-tunnel speed would have to be 2,000 kt. However, this is a supersonic speed, and clearly well above M = 0.4, and so compressibility would also need to be taken fully into account. In fact, not only is it a requirement that the Reynolds' principle should hold, but also that the Mach number must be equal for both aircraft and model. This is clearly not the case here.

77. The reason for Reynolds' speed-size relationship is related to the internal friction (ie viscosity) and density of the fluid. To account for all of these factors, he established that similarity of flow pattern would be achieved if the value of

$$\frac{\text{density} \times \text{velocity} \times \text{size}}{\text{viscosity}} \text{ was constant.}$$

This value is called the Reynolds Number.

78. To account for the reduction in size, the density could be increased. Clearly, it is not very practical to fill a wind tunnel with oil or water and accelerate it to 150 kt or 200 kt to obtain the flight Reynolds Number. In any case, no model could be built strong enough to endure the forces that would arise without breaking or being unacceptably distorted. However, it is quite possible to increase the density of the air by using a high-pressure tunnel. Increasing the pressure has little or no effect upon the viscosity, so that with increased density it is possible to reduce the velocity and/or size and still maintain aerodynamic similarity. For example, if the air is compressed to 25 atmospheres, then the density factor is 25, with no corresponding increase in viscosity (μ). It is then possible to test the model referred to in para 76 at

$$\frac{2{,}000\text{ kt}}{25} \text{ i.e. at 80 kt.}$$

Viscosity

79. Although a change in density does not affect the viscosity of the air, the temperature does, and so it is necessary to keep the compressed air cool to prevent the viscosity from rising. Unlike liquids, which become less viscous with rise in temperature, air becomes more viscous, and any increase in viscosity would offset the benefits of increased density.

Definition of Reynolds Number

80. For every wind-tunnel test there is one Reynolds Number (RN), and it is always published with the results of the test.

$$\text{RN} = \frac{\rho \text{VL}}{\mu}$$

Where ρ is density,
V is the velocity of the test,
L is a dimension of the body (usually chord),
μ is the viscosity of the fluid.

Considering the units involved it is not surprising to see test results quoted at RN = 4×10^6, or even 12×10^6.

Aerodynamic Forces

81. Since one of the major problems in using models is the effect of the large aerodynamic force felt on a small model, it is useful to look at the effect of using high-pressure tunnels. For example, taking again a 1/10th model, a speed of 80 kt and 25 atmospheres, to test for a full-scale flight at 200 kt, the forces will be factored by:

$$\frac{1}{100}\text{th (scale)} \times \frac{80^2}{200^2}\text{(speed)} \times 25 \text{ (density factor)}$$

$$= \frac{1}{100} \times \frac{6{,}400}{40{,}000} \times \frac{25}{1} = \frac{1}{25}$$

Thus dynamic similarity is achieved, with forces that are 1/25th of the full-scale forces; even that is quite high, considering that the weight of the model is only 1/1000th of the weight of the full-scale aircraft (if made of the same materials).

Inertial and Viscous Forces

82. On geometrically similar bodies, the nature of the boundary layer, and hence the flow pattern, is completely determined as long as the density (ρ) and viscosity (μ) of the fluid, and the speed (V) and length (L) of the body, remain constant (ie the Reynolds Number is the same).

83. Furthermore, RN is a measure of the ratio between the viscous forces and the inertial forces of the fluid, and this can be verified as follows. The inertial force acting on a typical fluid particle is measured by the product of its mass and its acceleration. Now the mass per unit volume of the fluid is, by definition, the density (ρ), while the volume is proportional to the cube of the characteristic length (L3), hence the mass is proportional to ρL3. The acceleration is the rate of change of velocity; that is, the change in velocity divided by the time during which the change occurs. The change in the velocity, as the fluid accelerates and decelerates over the body, is proportional to (V). The time is proportional to the time taken by a fluid particle to travel the length of the body at the speed (V); so the time is proportional to $\frac{L}{V}$. The acceleration is therefore proportional to

$$V \div \frac{L}{V} \quad \text{ie } \frac{V^2}{L}$$

The inertial force, which is mass × acceleration, can now be expressed:

$$\text{inertial force} \propto \frac{\rho L^3}{1} \times \frac{V^2}{L} \propto \rho\, L^2\, V^2$$

84. The viscous forces are determined by the product of the viscous sheer stress and the surface area over which it acts. The area is proportional to the square of the characteristic length, that is, to L^2. The viscous sheer stress is proportional to viscosity (μ) and to the rate of change of speed with height from the bottom of the boundary layer. The latter is proportional to $\frac{V}{L}$. Hence, the shear stress is proportional to $\mu\left(\frac{V}{L}\right)$. Therefore the viscous force is proportional to:

$$L^2 \times \mu \times \frac{V}{L} = L\mu V$$

85. Comparing inertial forces with viscous forces:

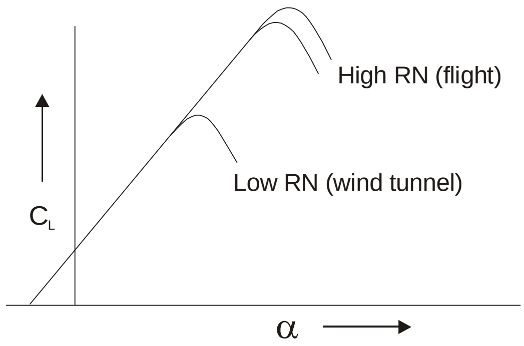
$$\frac{\text{inertial forces}}{\text{viscous forces}} \propto \frac{\rho L^2 V^2}{L \mu V} = \frac{\rho V L}{\mu}$$

which is the formula for Reynolds Number. $\frac{\mu}{\rho}$ is called the kinematic viscosity of the fluid and when this is constant the only variables are V and L.

Relevance of Reynolds Number to Pilots

86. The graph at Fig 14 shows the typical variations in the lift curve slope of an aerofoil over a range of Reynolds Numbers. There is a substantial difference between C_{Lmax} at flight RNs and wind tunnel test RNs, and an aircraft designer must take account of this. However, over the range of RN values experienced in flight, the effect on the lift curve and C_{Lmax} is negligible. Although aircrew should have an awareness of Reynolds Number, since it affects the design of the aircraft in which they fly, it is of no operational significance.

1-2 Fig 14 Scale Effect on CL and CLmax Curves



CHAPTER 3 - AERODYNAMIC THEORIES

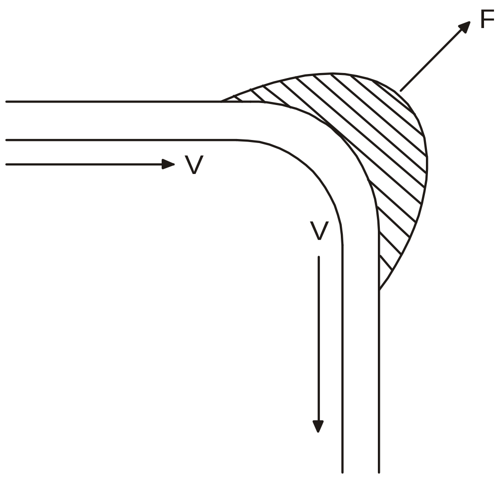
Introduction

1. Momentum Theory, Dimensional Analysis and Vortex Theory are three of the more common theories used to predict the performance of the aerofoil shape. Momentum Theory is another way of looking at Bernoulli's Theorem, and Dimensional Analysis is a mathematical derivation of aerodynamic force and takes into account the effect of Reynolds Number and Mach Number.

Momentum Theory

2. If a vane or blade deflects a jet of fluid as in Fig 1 then, assuming frictionless flow past the vane, there will be no loss of velocity in the jet.

1-3 Fig 1 Flow Past a Vane



3. From Newton's second law:

$$F = ma = \text{rate of change of momentum}$$
$$= \left(m\frac{dV}{dt}\right)$$

If A is the cross-section area of the jet and ΔV is the net change in velocity,

$$F = m\frac{\text{velocity change}}{\text{time}}$$
$$= \left(\frac{m}{\text{time}}\right) \times \text{velocity change}$$
$$= \text{mass flow} \times \Delta V$$
$$\therefore F = (\rho AV) \times \Delta V$$

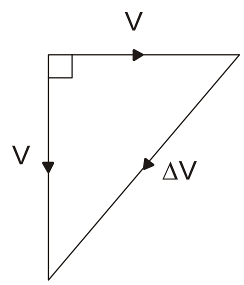
and from the vector diagram in Fig 2

$$\Delta V = \sqrt{2} \times V \approx 1.4V$$
$$\therefore F = 1.4\rho V^2 A$$

and substituting,

$$q = \tfrac{1}{2}\rho V^2$$
$$F = 2.8qA$$

1-3 Fig 2 Vector Diagram



4. Thus, by merely deflecting the stream a force of up to 2.8qA may be produced. This principle can be applied on an aerofoil which can be said to produce an aerodynamic advantage by changing the momentum of a stream of air. It can also be seen that the force produced depends on the attitude in the stream (α) and on the curvature of the aerofoil.

Dimensional Analysis

5. It can be demonstrated experimentally that the force acting on geometrically similar objects moving through the air is dependent on at least the following variables:

a. Free stream velocity.

b. Air density.

c. The size of the object.

d. The speed of sound, i.e. speed of propagation of small pressure waves.

e. The viscosity of the air.

f. Shape, attitude and smoothness of the object.

6. It is possible to derive explicit equations for lift and drag by the use of dimensional analysis. This principle relies on the fact that in any physical equation the dimensions of the left-hand side (LHS) must be the same as the dimensions of the right-hand side (RHS). The dimensions used are those of mass, length and time (MLT) which are independent of specific units. The factors listed in para 5 are repeated below together with their appropriate symbols and dimensions.

Dependent Variable	Symbol	Dimensions
velocity	V	LT^{-1}
density	ρ	ML^{-3}
area (wing)	S	L^2
sonic speed	a	LT^{-1}
viscosity	μ	$ML^{-1} T^{-1}$

There is no convenient method of measuring shape, attitude or smoothness, so these variables are included in a constant (k).

7. The dependence of aerodynamic force on V, ρ, S, a and μ may be expressed as:

$$F = k(V.\rho.S.a.\mu)$$

which theory allows to be written:

$$F = k\left(V^a \times \rho^b \times S^c \times a^d \times \mu^e\right) \qquad (1)$$

where k, a, b, c, d and e are unknown.

8. Equation (1) may now be rewritten in terms of the MLT system, remembering that

$$\text{force} = \text{mass} \times \text{acceleration} = MLT^{-2}$$

$$= k\left(LT^{-1}\right)^a\left(ML^{-3}\right)^b\left(L^2\right)^c\left(LT^{-1}\right)^d\left(ML^{-1}T^{-1}\right)^e \qquad (2)$$

9. The condition for dimensional equivalence between the two sides of equation (2) is that the sum of the indices of a given quantity on the RHS must equal the index of that quantity on the LHS. Applying this rule we get:

for time	$-a-d-e$	$=-2$
for mass	$b+e$	$=+1$
for length	$a-3b+2c+d-e$	$=+1$

10. Taking the equations for time and mass from para 9 and transposing them to make a and b the subject of the equations respectively gives:

$$a = 2-d-e$$
$$b = 1-e$$

Similarly, transposing the equation for length to make c the subject gives:

$$2c = 1-a+3b-d+e$$

substituting for a and b in the above formula gives:

$$2c = 1 - (2 - d - e) + 3(1 - e) - d + e$$

which simplifies to:

$$c = 1 - \left(\frac{e}{2}\right)$$

11. Substituting these values of a, b and c in equation (1) we get:

$$F = k\left[(V)^{2-d-e} \times (\rho)^{1-e} \times (S)^{1-\frac{e}{2}} \times a^{d} \times \mu^{e}\right]$$

and collecting terms of like indices we get:

$$F = k\rho V^2 S\left(\frac{a}{V}\right)^d \left(\frac{\mu}{\rho V S^{\frac{1}{2}}}\right)^e \qquad (3)$$

12. Dimensionally, $S^{\frac{1}{2}} = \left(L^2\right)^{\frac{1}{2}} = L$

and so the last term of equation (3) is $\left(\frac{\mu}{\rho VL}\right)$

which is $\frac{1}{\text{Reynolds Number}} = \frac{1}{RN}$,

also $\left(\frac{a}{V}\right) = \frac{1}{\text{Mach Number}} = \frac{1}{M}$

13. Equation (3) therefore becomes:

$$F = k\rho V^2 S M^{-d} RN^{-e}$$

and, noting that dynamic pressure = $\frac{1}{2}\rho V^2$ we can introduce another constant, C, such that C = 2k and obtain:

$$F = C\tfrac{1}{2}\rho V^2 S(M^{-d} RN^{-e}) \qquad (4)$$

14. It is significant that this solution includes unknown exponential functions of Mach number and Reynolds Number, a fact which is often ignored when the lift and drag equations are quoted. In practice, at the sacrifice of an exact equation, the parameters M and RN are included in the constant C. C, therefore, becomes a coefficient dependent on Mach number, Reynolds Number, shape, attitude and smoothness.

Thus $C = 2kM^{-d}RN^{-e}$

15. Equation (4) is a general equation for aerodynamic force and represents either the lift component, the drag component or their vector sum (TR), i.e.

$$\text{lift} = C_L \tfrac{1}{2}\rho V^2 S$$

$$\text{and drag} = C_D \tfrac{1}{2}\rho V^2 S$$

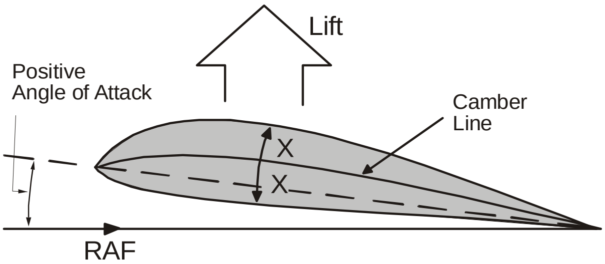
$$\text{Similarly, moment} = C_M \tfrac{1}{2}\rho V^2 S$$

where C_M is a function of M, RN, etc.

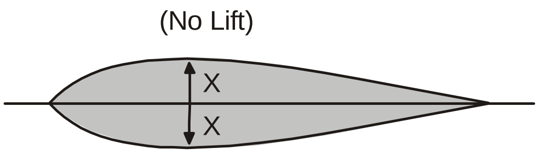
Vortex or Circulation Theory

16. If a cambered aerofoil at positive angle of attack (as in Fig 3) is compared with the aerofoil in Fig 4, it will be seen that the mean camber line in the latter has been straightened and set at zero angle of attack, and that the thickness distribution of the original aerofoil has been plotted about the straight mean camber line. This shape is known as the symmetrical fairing of an aerofoil section.

1-3 Fig 3 Aerofoil at Positive Angle of Attack

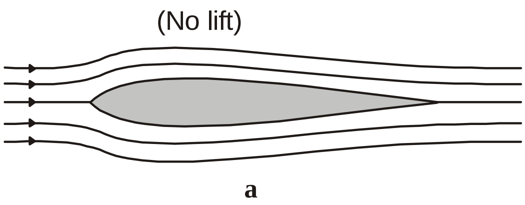


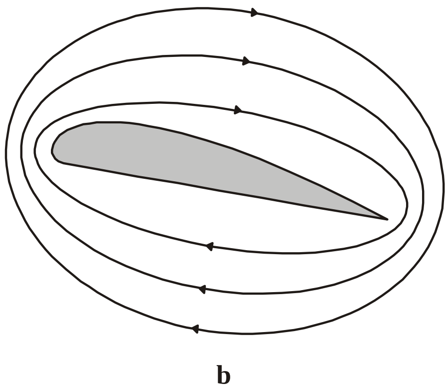
1-3 Fig 4 Symmetrical Fairing of an Aerofoil



17. The effect of angle of attack and camber on the flow pattern round the aerofoil in Fig 3 relative to the flow round the symmetrical fairing, is to accelerate the air over the top surface and decelerate the air along the bottom surface. Thus the flow over an aerofoil can be regarded as the combination of two different types of flow: one of these is the normal flow round the symmetrical fairing at zero angle of attack (Fig 5a) and the other is a flow in which air circulates around the aerofoil, towards the trailing edge over the upper surface and towards the leading edge over the lower surface (Fig 5b).

1-3 Fig 5 The Flow Round a Lifting Aerofoil Section Separated into Two Elemental Flows





The lift of the aerofoil due to camber and angle of attack must therefore be associated with the circulatory flow. A special type of flow which is the same as that of Fig 5b is that due to a vortex, that is a mass of fluid rotating in concentric circles.

18. The two important properties of vortex flow are:

a. **Velocity**. In the flow round a vortex the velocity is inversely proportional to the radius, i.e. radius × velocity = constant.

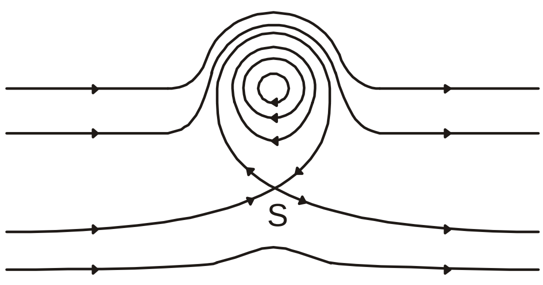
b. **Pressure.** Because the flow is steady, Bernoulli's Theorem applies and the pressure decreases as the velocity increases towards the centre of the vortex.

19. A strong vortex will have high rotational speeds associated with it and vice versa. The constant of para 18a determines the size of the rotational speeds and it is therefore used to measure vortex strength. For theoretical reasons the formula radius × velocity = constant is changed to:
$2\pi rV$ = another constant = K. The constant K is called the circulation.

20. The circulation and velocity at a point outside the vortex core are connected by the formula: $K = 2\pi rV$ with dimensions L^2T^{-1} (ft^2 per sec) where r = radius, and V = velocity.

21. If a vortex is in a stream of air and some restraint is applied to the vortex to prevent it being blown downstream, the streamlines of the flow will be as shown in Fig 6.

1-3 Fig 6 Flow Round a Vortex Rotating in a Uniform Stream



The point S is of some interest; it is a stagnation point in the stream, and the direction of flow passing through S is shown by the arrows. The air which passes through S splits into two, some going over the top of the vortex and some below it. That part of the streamline through S which passes over the vortex forms a closed loop and the air within this loop does not pass downstream but circulates continually round the vortex. The velocity at any point is the vector sum of the velocity of the uniform stream and the velocity due to the vortex in isolation. Thus it can be seen that the air above the vortex is speeded up and that below the vortex is slowed down.

22. As a result of these velocity differences between the flow above and below the vortex, there are pressure differences. These pressure differences cause a force on the vortex of magnitude ρVK per unit length which acts perpendicular to the direction of the free stream, ie it is a lift force by definition. This relationship is known as the Kutta-Zhukovsky relationship.

23. The flow round an aerofoil section can be idealized into three parts:

a. A uniform stream.

b. A distortion of the stream due to aerofoil thickness.

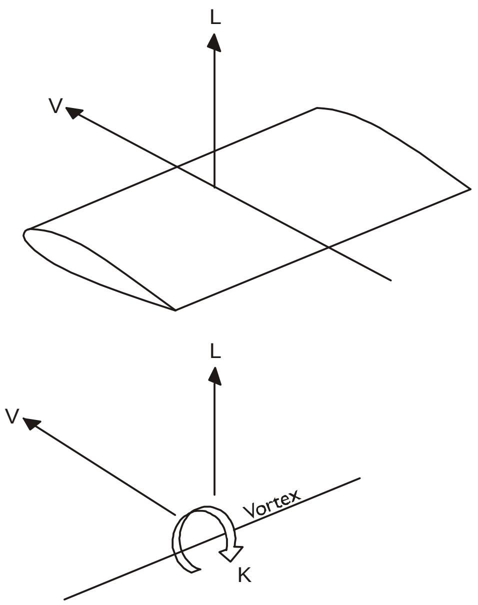
c. A flow similar to a vortex.

This is another way in which lift may be explained because in para 22 it was stated that a vortex in a uniform stream experiences a lift force.

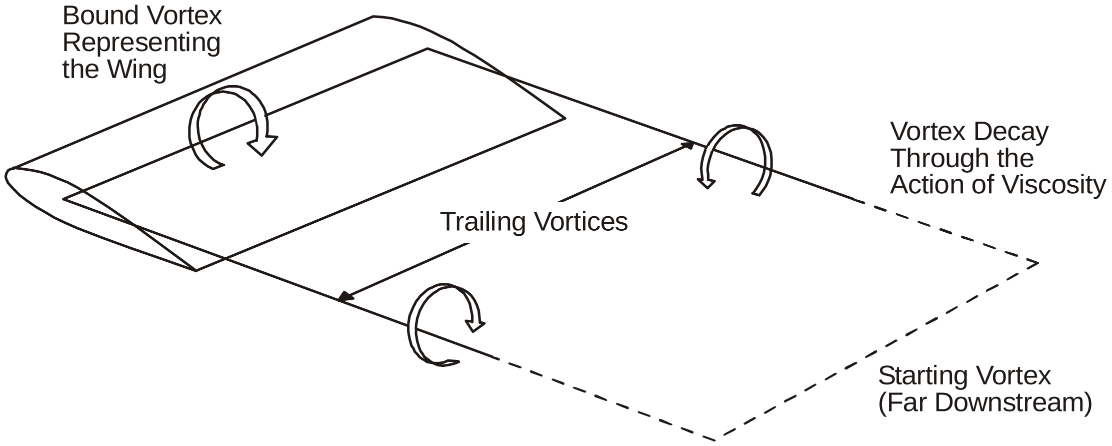
24. Assuming, for simplicity, that the lift is uniform along the span, and ignoring the effects of thickness and viscosity, a wing can be replaced by a vortex of suitable strength along the locus of the centres of pressure. This vortex is known as line or bound vortex, or just a lifting line, and is illustrated in Fig 7.

25. The arrangement in Fig 7 is physically impossible because a vortex cannot exist with open ends due to the low pressure in its core. A vortex must therefore either be re-entrant, or abut at each end on a solid boundary. The simple line vortex is therefore modified to represent the uniformly loaded wing by a horseshoe vortex as illustrated in Fig 8. To make the vortex re-entrant it is necessary to complete the fourth side of the rectangle by a suitable vortex. When a wing is accelerated from rest, the circulation around it, and therefore lift, is not produced instantaneously. As the circulation develops, a starting vortex is formed at the trailing edge and is left behind the wing. This vortex is equal in strength and opposite in sense to the circulation around the wing.

1-3 Fig 7 A Wing with Uniform Lift replaced by a Line Vortex



1-3 Fig 8 The Horseshoe Vortex



26. The starting vortex, or initial eddy, can be demonstrated by simple experiment. A flat piece of board to represent a wing is placed into water, cutting the surface at moderate angle of attack. If the board is suddenly moved forward, an eddy will be seen to leave the rear of the board.

27. The transformation of a wing into a vortex of suitable strength is a complex mathematical operation. It can be seen, however, that when the value of K is obtained, the lift can be obtained from the Kutta-Zhukovsky relationship. This theoretical model of an aircraft wing is most useful in explaining three-dimensional effects such as induced drag, aspect ratio etc (Volume 1, Chapter 5).

CHAPTER 4 – LIFT

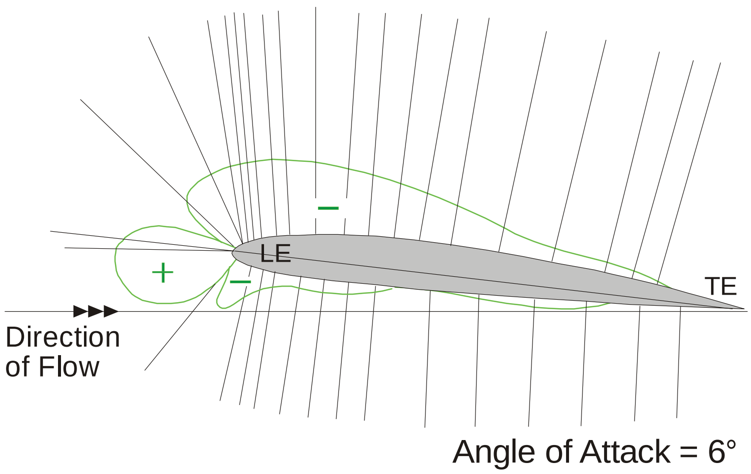
Introduction

1. Having covered the basic aerodynamic principles in Volume 1, Chapter 2, this chapter will deal in a little more detail with pressure distribution and Centre of Pressure (CP) movement, define and discuss aerodynamic centre and then move on to the factors affecting lift and the lift/drag ratio. Finally the various types of aerofoils will be discussed.

Distribution of Pressure About the Wing

2. Fig 1 illustrates an actual pressure plot around an aerofoil at 6° angle of attack. The straight lines indicate the positions from which the tappings were taken and the positive and negative pressures are those above and below free stream static pressure. Conditions at the trailing edge (TE) are difficult to plot because of the small values of pressure there and the difficulty of providing adequate tappings.

1-4 Fig 1 Pressure Plotting



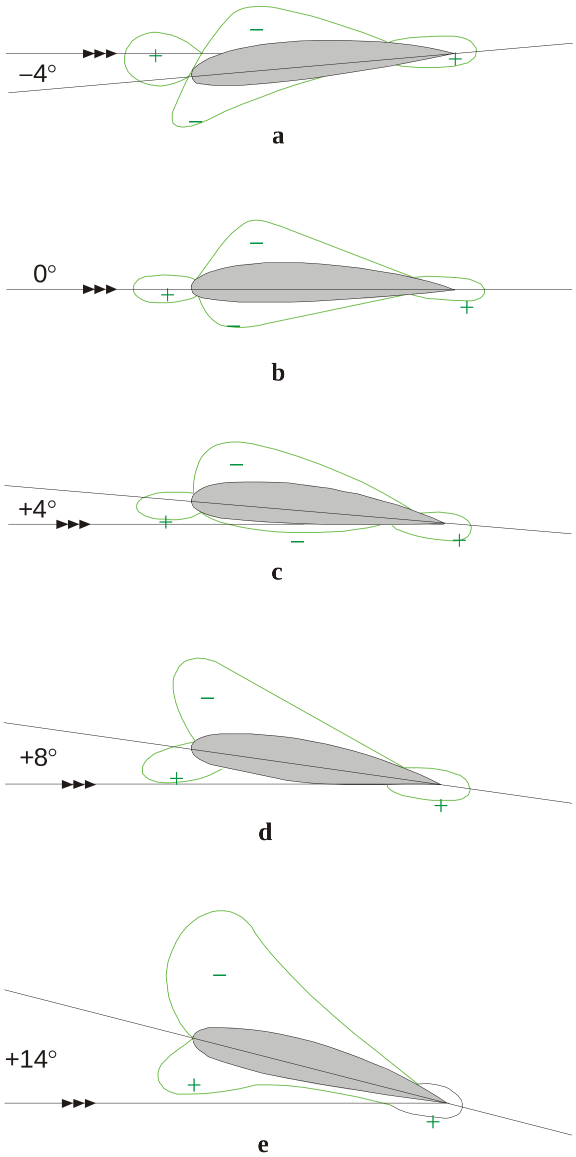
3. Although most low speed aerofoils are similar in shape, each section is intended to give certain specific aerodynamic characteristics. Therefore, there can be no such thing as a typical aerofoil section or a typical aerofoil pressure distribution and it is only possible to discuss pressure distributions around aerofoils in the broadest of general terms. So, in general, at conventional angles of attack, compared with the free stream static pressure there is a pressure decrease over much of the upper surface, a lesser decrease over much of the lower surface so that the greatest contribution to overall lift comes from the upper surface.

4. The aerofoil profile presented to the airflow determines the distribution of velocity and hence the distribution of pressure over the surface. This profile is determined by the aerofoil geometry, ie thickness distribution and camber, and by the angle of attack. The greatest positive pressures occur at stagnation points where the flow is brought to rest, at the trailing edge, and somewhere near the leading edge (LE), depending on the angle of attack. At the front stagnation point, the flow divides to pass over and under the section. At this point there must be some initial acceleration of the flow at the surface otherwise there could be no real velocity anywhere at the aerofoil surface, therefore there must be some initial reduction of pressure below the stagnation value. If the profile is such as to produce a continuous acceleration there will be a continuous pressure reduction and vice versa. Some parts of the contour will produce the first effect, other parts the latter, bearing in mind always

that a smooth contour will produce a smoothly changing pressure distribution which must finish with the stagnation value at the trailing edge.

5. Fig 2 shows the pressure distribution around a particular aerofoil section at varying angles of attack. The flow over the section accelerates rapidly around the nose and over the leading portion of the surface, the rate of acceleration increasing with increase in angle of attack. The pressure reduces continuously from the stagnation value through the free stream value to a position when a peak negative value is reached. From there onwards the flow is continuously retarded, increasing the pressure through the free stream value to a small positive value towards the trailing edge. The flow under the section is accelerated much less rapidly than that over the section, reducing the pressure much more slowly through the free stream value to some small negative value, with subsequent deceleration and increase in pressure through free stream value to a small positive value toward the trailing edge. If the slight concavity on the lower surface towards the trailing edge was carried a little further forward, it might be possible to sustain a positive pressure over the whole of the lower surface at the higher angles of attack. However, although this would increase the lifting properties of the section, it might also produce undesirable changes in the drag and pitching characteristics. Therefore, it can be seen that any pressure distribution around an aerofoil must clearly take account of the particular aerofoil contour.

1-4 Fig 2 Pressure Distribution About an Aerofoil

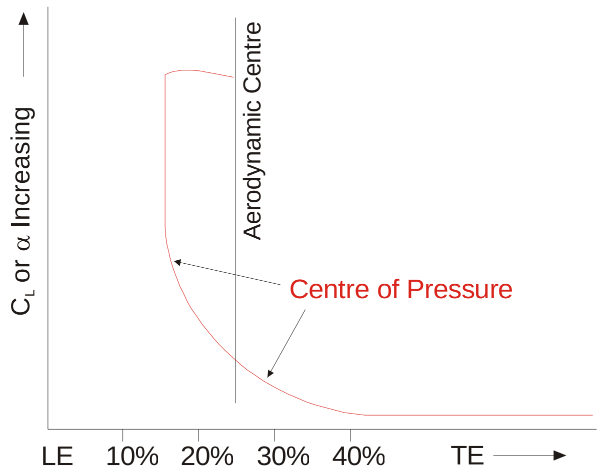


6. From examinations of Fig 2 it can be seen that at small angles of attack the lift arises from the difference between the pressure reductions on the upper and lower surfaces, whilst at the higher angles of attack the lift is due partly to the decreased pressure above the section and partly to the increased pressure on the lower surface. At a small negative angle of attack (about –4° for this aerofoil) the decrease in pressure above and below the section would be equal and the section would give no lift. At the stalling angle the low pressure area on the top of the section suddenly reduces and such lift as remains is due principally to the pressure increase on the lower surface.

Centre of Pressure (CP)

7. The overall effect of these pressure changes on the surface of the aerofoil can be represented in various simplified ways, one of which is to represent the effects by a single aerodynamic force acting at a particular point on the chord line, called the centre of pressure (CP). The location of the CP is a function of camber and section lift coefficient, both the resultant force and its position varying with angle of attack (see Fig 3). As the angle of attack is increased the magnitude of the force increases and the CP moves forward; when the stall is reached the force decreases abruptly and the CP generally moves back along the chord. With a cambered aerofoil the CP movement over the normal working range of angles of attack is between 20% and 30% of the chord aft of the leading edge. With a symmetrical aerofoil there is virtually no CP movement over the working range of angles of attack at subsonic speeds.

1-4 Fig 3 Movement of the Centre of Pressure



Aerodynamic Centre

8. An aircraft pitches about the lateral axis which passes through the centre of gravity. The wing pitching moment is the product of lift and the distance between the CG and the CP of the wing. Unfortunately, the CP moves when the angle of attack is altered so calculation of the pitching moment becomes complicated.

9. The pitching moment of a wing can be measured experimentally by direct measurement on a balance or by pressure plotting. The pitching moment coefficient (C_m) is calculated as follows:

$$C_m = \frac{\text{Pitching moment}}{qSc}$$

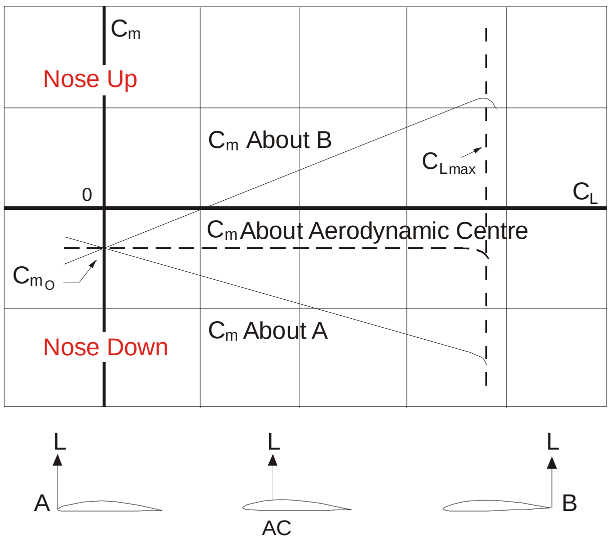
where c is the mean aerodynamic chord, q is dynamic pressure, and S is the wing area. The value of pitching moment and therefore C_m will depend upon the point about which the moment is calculated and will vary because the lift force and the position of the CP change with angle of attack.

10. If pitching moments are measured at various points along the chord for several values of C_L one particular point is found where the C_m is constant. This point occurs where the change in C_L with angle of attack is offset by the change in distance between CP and CG. This is the aerodynamic centre (AC). For a flat or curved plate in inviscid, incompressible flow the aerodynamic centre is at approximately 25% of the chord from the leading edge. Thickness of the section and viscosity tend to move it forward and compressibility moves it rearwards. Some modern low drag aerofoils have the AC a little further forward at approximately 23% chord.

11. There are two ways of considering the effects of changing angle of attack on the pitching moment of an aerofoil. One way is to consider change in lift acting through a CP which is moving with angle of attack; the other simpler way is to consider changes in lift always acting through the AC which is fixed.

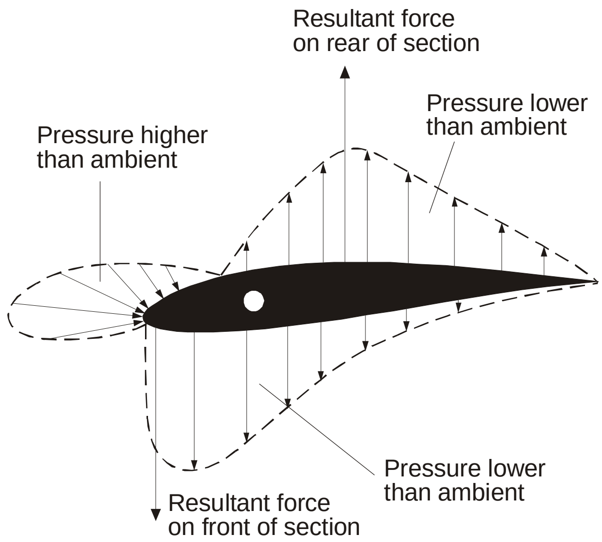
12. The rate of change of C_m with respect to C_L or angle of attack is constant through most of the angle of attack range and the value of $\frac{C_m}{C_L}$ depends on the point on the aerofoil at which C_m is measured. Curves of C_m versus C_L are shown in Fig 4. It can be seen that a residual pitching moment is present at zero lift.

1-4 Fig 4 C_m Against C_L



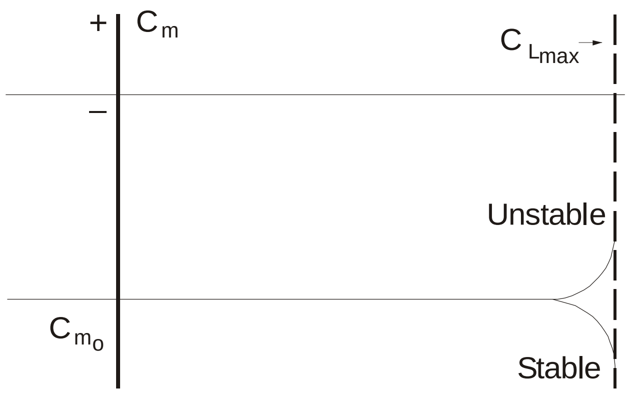
This is because an aerofoil with positive camber has a distribution of pressure as illustrated in Fig 5. It should be noted that the pressure on the upper surface towards the leading edge is higher than ambient and towards the trailing edge the pressure is lower than ambient. This results in a nose down (negative) pitching moment even though there is no net lift at this angle of attack. The C_m at zero lift angle of attack is called C_{mo} and, since the pitching moment about the AC is constant with C_L by definition, its value is equal to C_{mo}. The value of C_{mo} is determined by camber and is usually negative but is zero for a symmetrical aerofoil and can be positive when there is reflex curvature at the trailing edge. When C_m is measured at point A (Fig 4), an increase in C_L will cause an increase in lift and a larger negative (nose down) moment. When measured at point B, an increase in C_L will cause the moment to become less negative and eventually positive (nose up).

1-4 Fig 5 Pressure Pattern at Zero Lift Angle of Attack



13. Approaching C_{Lmax}, the C_m/C_L graph departs from the straight line, Fig 6. The C_L decreases and the CP moves aft. If at this stage, the C_m becomes negative; it tends to unstall the wing and is stable. If the C_m becomes positive the pitch up aggravates the stall and is unstable. This is known as pitch-up and is associated with highly swept wings.

1-4 Fig 6 C_m In The Region of C_{Lmax}



Lift

14. By definition, lift is that component of the total aerodynamic reaction which is perpendicular to the flight path of the aircraft.

15. It can be demonstrated experimentally that the total aerodynamic reaction, and therefore the lift acting on a wing moving through air, is dependent upon at least the following variables:

a. Free stream velocity (V^2).

b. Air density (ρ).

c. Wing area (S).

d. Wing shape in section and in planform.

e. Angle of attack (α).

f. Condition of the surface.

g. Viscosity of the air (μ).

h. The speed of sound. ie the speed of propagation of small pressure waves (a).

16. In Volume 1, Chapter 2, and earlier in this chapter, it was seen that lift increased when the angle of attack of a given aerofoil section was increased; and that the increase in lift was achieved mechanically by greater acceleration of the airflow over the section, with an appropriate decrease in pressure. The general and simplified equation for aerodynamic force is $\frac{1}{2}\rho V^2 S \times$ a coefficient, and the coefficient indicates the change in the force which occurs when the angle of attack is altered.

17. The equation for lift is $C_L\frac{1}{2}\rho V^2 S$, and C_L for a given aerofoil section and planform allows for angle of attack and all the unknown quantities which are not represented in the force formula. Three proofs for the lift equation are given in Volume 1, Chapter 3.

Coefficient of Lift (C_L)

18. The coefficient of lift is obtained experimentally at a quoted Reynolds Number from the equation:

$$\text{Lift} = C_L\tfrac{1}{2}\rho V^2 S$$

$$C_L = \frac{\text{Lift}}{\frac{1}{2}\rho V^2 S} = \frac{\text{lift}}{qS}$$

and the values are plotted against angle of attack. It is then possible to consider the factors affecting lift in terms of C_L and to show the effects on the C_L curve.

Factors Affecting C_L

19. The coefficient of lift is dependent upon the following factors:

a. Angle of attack.

b. Shape of the wing section and planform.

c. Condition of the wing surface.

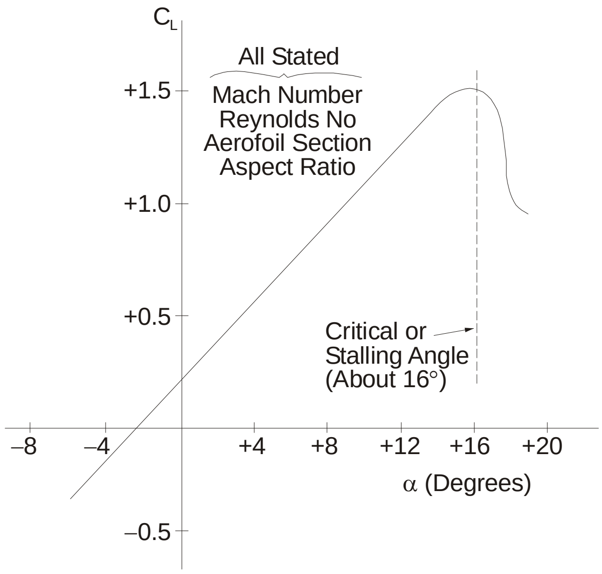
d. Reynolds Number $\left(\frac{\rho VL}{\mu}\right)$

e. Speed of sound (Mach number).

20. **Angle of Attack.** A typical lift curve is shown in Fig 7, for a wing of 13% thickness/chord (t/c) ratio and 2% camber. The greater part of the curve is linear and the airflow follows the design contour of the aerofoil almost to the trailing edge before separation. At higher angles of attack the curve begins to lean over slightly, indicating a loss of lifting effectiveness. From the point of maximum thickness to the trailing edge of the aerofoil, the flow outside the boundary layer is

decelerating, accompanied by a pressure rise (Bernoulli's theorem). This adverse pressure gradient thickens the existing boundary layer. In the boundary layer, the airflow's kinetic energy has been reduced by friction, the energy loss appearing as heat. The weakened flow, encountering the thickened layer, slows still further. With increasing angle of attack, the boundary layer separation point (Volume 1, Chapter 5, paragraph 19) moves rapidly forward, the detached flow causing a substantial reduction of C_L. The aerofoil may be considered to have changed from a streamlined body to a bluff one, with the separation point moving rapidly forwards from the region of the trailing edge. The desirable progressive stall of an actual wing is achieved by wash-out at the tips or change of aerofoil section along the span, or a combination of both.

1-4 Fig 7 Typical Lift Curve for a Moderate Thickness Cambered Section

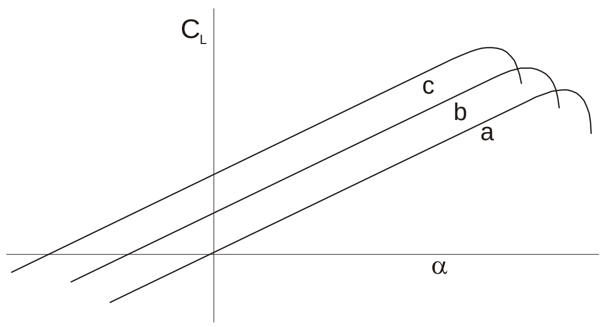


21. **Effect of Shape.** Changes in the shape of a wing may be considered under the following headings:

a. **Leading Edge Radius**. The shape of the leading edge, and the condition of its surface, largely determines the stalling characteristics of a wing. In general, a blunt leading edge with a large radius will result in a well-rounded peak to the C_L curve. A small radius, on the other hand, invariably produces an abrupt stall but this may be modified considerably by surface roughness which is discussed later.

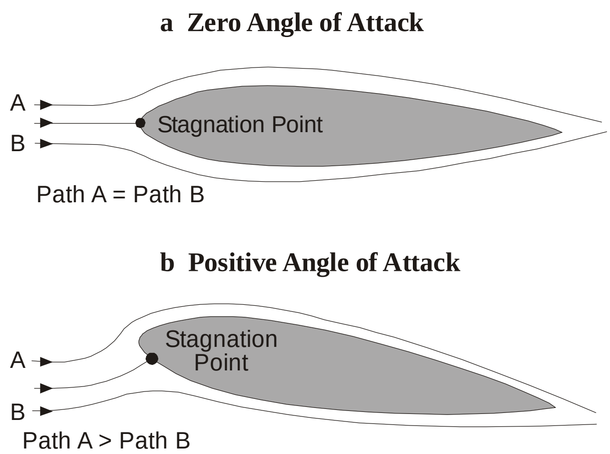
b. **Camber.** The effect of camber is illustrated in Fig 8. Line (a) represents the curve for a symmetrical section. Lines (b) and (c) are for sections of increasing camber. A symmetrical wing at zero angle of attack will have the same pressure distribution on its upper and lower surfaces, therefore it will not produce lift.

1-4 Fig 8 Effect of Camber



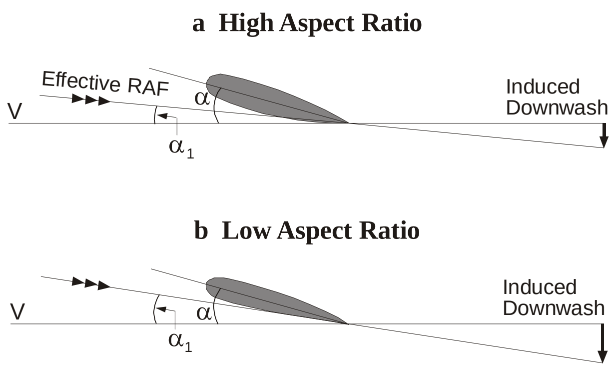
As the angle of attack is increased, the stagnation point moves from the chord line to a point below, moving slightly further backwards with increase in angle of attack. This effectively lengthens the path of the flow over the top surface and reduces it on the lower, thus changing the symmetrical section (Fig 9a) to an apparent cambered one as in Fig 9b. A positively cambered wing will produce lift at zero angle of attack because the airflow attains a higher velocity over the upper surface creating a pressure differential and lift. This gives it a lead over the symmetrical section at all normal angles of attack but pays the penalty of an earlier stalling angle as shown by the C_L versus α curve which shifts up and left in Fig 8 as the camber is increased. The angle of attack at which the C_L is zero is known as the zero-lift angle of attack (α_{Lo}) and a typical value is $-3°$ for a cambered section.

1-4 Fig 9 Symmetrical Aerofoil

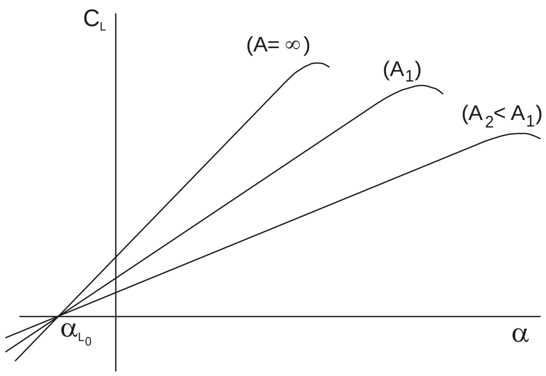


c. **Aspect ratio.** Fig 10 shows the downward component of airflow at the rear of the wing, caused by trailing edge vortices and known as induced downwash (ω). The induced downwash causes the flow over the wing to be inclined slightly downwards from the direction of the undisturbed stream (V) by the angle α_1. This reduces the effective angle of attack, which determines the airflow and the lift and drag forces acting on the wing. The effect on the C_L by change of aspect ratio (AR) will depend on how the effective angle of attack is influenced by change in AR. It has been shown in the previous chapter, that a wing of infinite span has no induced downwash. It can be demonstrated that the nearer one gets to that ideal, ie high AR, the less effect the vortices will have on the relative airflow along the semispan and therefore the least deviation from the shape of the C_L curve of the wing with infinite AR. It can be seen from Fig 11 that at any angle α, apart from the zero lift angle, the increase in C_L of the finite wing lags the infinite wing, the lag increasing with reducing AR due to increasing α_1. Theoretically the C_L peak values should not be affected, but experimental results show a slight reduction of C_{Lmax} as the aspect ratio is lowered.

1-4 Fig 10 Effect of Aspect Ratio on the Induced Downwash

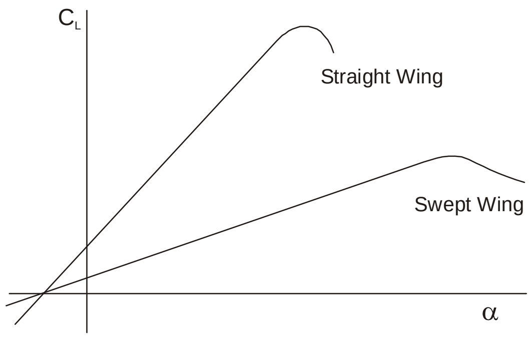


1-4 Fig 11 Influence of Aspect Ratio A on the Lift Curve



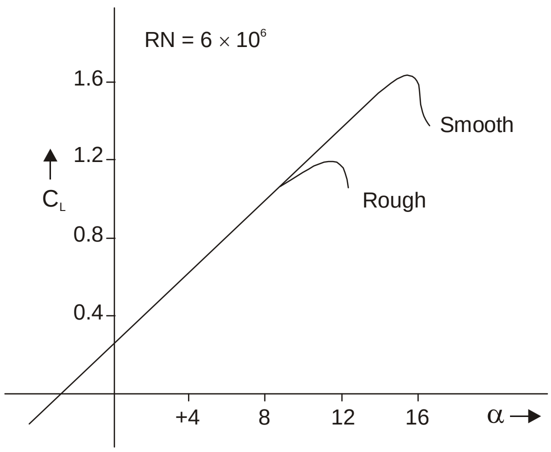
d. **Sweepback**. If an aircraft's wings are swept and the wing area remains the same, then by definition (span^2/area) the aspect ratio must be less than the AR of the equivalent straight wing. The shape of the C_L versus angle of attack curve for a swept wing, compared to a straight wing, is similar to the comparison between a low and a high aspect ratio wing. However, this does not explain the marked reduction in C_{Lmax} at sweep angles in excess of 40° to 45°, which is mainly due to earlier flow separation from the upper surface. An alternative explanation is to resolve the airflow over a swept wing into two components. The component parallel to the leading edge produces no lift. Only the component normal to the leading edge is considered to be producing lift. As this component is always less than the free stream flow at all angles of sweep, a swept wing will always produce less lift than a straight wing (see Fig 12).

1-4 Fig 12 Effect of Sweepback on C_L



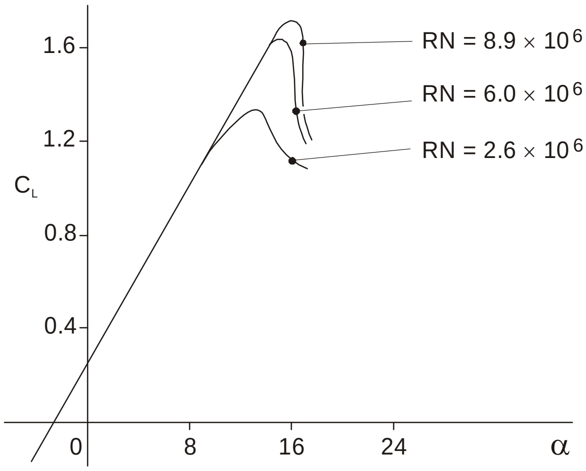
22. **Effect of Surface Condition.** It has long been known that surface roughness, especially near the leading edge, has a considerable effect on the characteristics of wing sections. The maximum lift coefficient, in particular, is sensitive to the leading edge roughness. Fig 13 illustrates the effect of a roughened leading edge compared to a smooth surface. In general, the maximum lift coefficient decreases progressively with increasing roughness of the leading edge. Roughness of the surface further downstream than about 20% chord from the leading edge has little effect on C_{Lmax} or the lift-curve slope. The standard roughness illustrated is more severe than that caused by usual manufacturing irregularities or deterioration in service, but is considerably less severe than that likely to be encountered in service as a result of the accumulation of ice, mud or combat damage. Under test, the leading edge of a model wing is artificially roughened by applying carborundum grains to the surface over a length of 8% from the leading edge of both surfaces.

1-4 Fig 13 Effect of Leading Edge Roughness



23. **Effect of Reynolds Number.** The formula for Reynolds Number is $\frac{\rho VL}{\mu}$, that is density × velocity × a mean chord length, divided by viscosity. A fuller explanation of Reynolds Number is given in Volume 1, Chapter 2. If we consider an aircraft operating at a given altitude, L is constant, ρ is constant, and at a given temperature, the viscosity is constant: the only variable is V. For all practical purposes the graph in Fig 14 shows the effect on C_L of increasing velocity on a general-purpose aerofoil section. It should be remembered than an increase in Reynolds Number, for any reason, will produce the same effect. Fig 14 shows that with increasing velocity both the maximum value of C_L and the stalling angle of attack is increased. An increase in the velocity of the airflow over a wing will produce earlier transition and an increase in the kinetic energy of the turbulent boundary layer due to mixing; the result is delayed separation. An increase in density, or a reduction in viscosity will have the same effect on the stall. The effect shown in Fig 14 is generally least for thin sections (t/c < 12%) and greatest for thick, well-cambered sections.

1-4 Fig 14 Effect of Reynolds Number on C_{Lmax}



24. **Effect of Mach Number.** The effect of Mach number is discussed in the chapter on Transonic and Supersonic Aerodynamics (Volume 1, Chapter 21).

Aerofoils

25. The performance of an aerofoil is governed by its contour. Generally, aerofoils can be divided into three classes:

a. High lift

b. General purpose

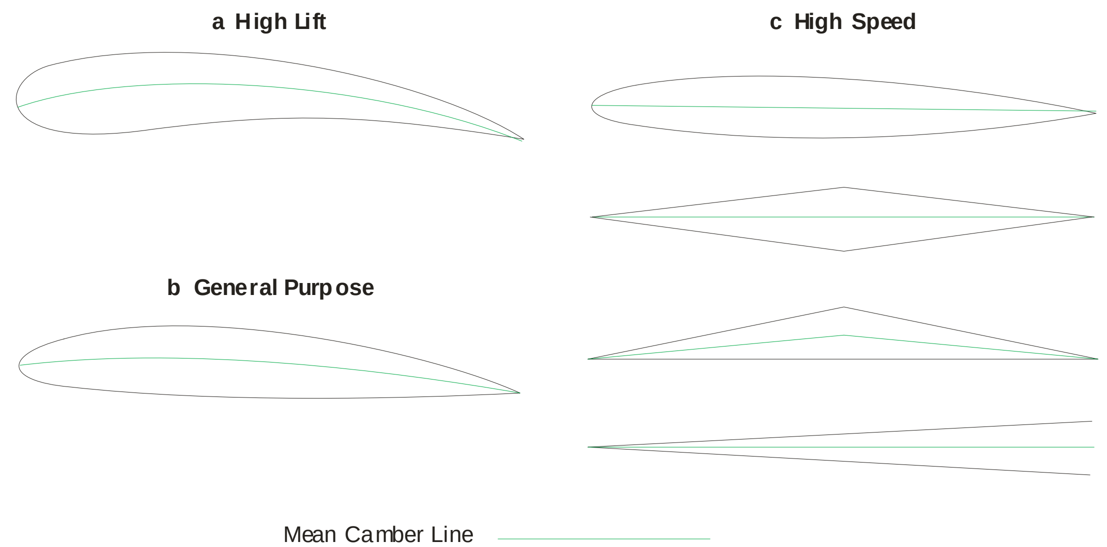
c. High speed

26. **High Lift Aerofoils**. A typical high lift section is shown in Fig 15a.

a. High lift sections employ a high t/c ratio, a pronounced camber, and a well-rounded leading edge: their maximum thickness is at about 25% to 30% of the chord aft of the leading edge.

b. The greater the camber, ie the amount of curvature of the mean camber line, the greater the shift of centre of pressure for a given change in the angle of attack. The range of movement of the CP is therefore large on a high lift section. This movement can be greatly decreased by reflexing upwards the trailing edge of the wing, but some lift is lost as a result.

1-4 Fig 15 Aerofoil Sections



c. Sections of this type are used mainly on sailplanes and other aircraft where a high C_L is all-important and speed a secondary consideration.

27. **General Purpose Aerofoils**. A typical general purpose section is shown in Fig 15b.

a. General purpose sections employ a lower t/c ratio, less camber and a sharper leading edge than those of the high lift type, but their maximum thickness is still at about 25% to 30% of the chord aft of the leading edge. The lower t/c ratio results in less drag and a lower C_L than those of a high lift aerofoil.

b. Sections of this type are used on aircraft whose duties require speeds which, although higher than those mentioned in para 26, are not high enough to subject the aerofoil to the effects of compressibility.

28. **High Speed Aerofoils**. Typical high speed sections are shown in Fig 15c.

a. Most high speed sections employ a very low t/c ratio, no camber and a sharp leading edge. Their maximum thickness is at about the 50% chord point.

b. Most of these sections lie in the 5% to 10% t/c ratio band, but even thinner sections have been used on research aircraft. The reason for this is the overriding requirement for low drag; naturally the thinner sections have low maximum lift coefficients.

c. High speed aerofoils are usually symmetrical about the chord line; some sections are wedge-shaped whilst others consist of arcs of a circle placed symmetrically about the chord line.

The behaviour and aerodynamics of these sections at supersonic speeds are dealt with in detail in Volume 1, Chapter 21.

Performance

29. The performance of all aerofoils is sensitive to small changes in contour. Increasing or decreasing the thickness by as little as 1% of the chord, or moving the point of maximum camber an inch or so in either direction, will alter the characteristics. In particular, changes in the shape of the leading edge have a marked effect on the maximum lift and drag obtained and the behaviour at the stall - a sharp leading edge stalling more readily than one that is well rounded. Also it is important that the finish of the wing surfaces be carefully preserved if the aircraft is expected to attain its maximum performance. Any dents or scratches in the surface bring about a deterioration in the general performance. These points are of particular importance on high performance aircraft when a poor finish can result in a drastic reduction not only in performance, but also in control at high Mach numbers.

Summary

30. The pressure distribution around an aerofoil varies considerably with shape and angle of attack, however, at conventional angles of attack, the greatest contribution to overall lift comes from the upper surface.

31. There are two points through which the lift force may be considered to act. The first is a moving point called the centre of pressure, and the second is a fixed point called the aerodynamic centre. The aerodynamic centre is used in most work on stability.

32. Lift is dependent upon the following variables:

a. Free stream velocity.

b. Air density.

c. Wing area.

d. Wing shape in section and planform.

e. Angle of attack.

f. Condition of the surface.

g. Viscosity of the air.

h. The speed of sound.

33. The coefficient of lift is dependent upon the following factors:

a. Angle of attack.

b. Shape of the wing section and planform.

c. Condition of the wing surface.

d. Speed of sound.

e. Reynolds Number $\frac{\rho VL}{\mu}$

34. Aerofoils are generally divided into three classes:

a. High lift.

b. General purpose.

c. High speed.

CHAPTER 5- DRAG

Introduction

1. Each part of an aircraft in flight produces an aerodynamic force. Total drag is the sum of all the component of the aerodynamic forces which act parallel and opposite to the direction of flight. Each part of total drag represents a value of resistance of the aircraft's movement, that is, lost energy.

Components of Total Drag

2. Some textbooks still break down Total Drag into the old terms Profile Drag and Induced Drag. These latter terms are now more widely known as Zero Lift Drag and Lift Dependent Drag respectively and are described later. There are three points to be borne in mind when considering total drag:

a. The causes of subsonic drag have changed very little over the years but the balance of values has changed e.g. parasite drag is such a small part of the whole that it is no longer considered separately, except when describing helicopter power requirements (Volume 12, Chapter 6).

b. An aircraft in flight will have drag even when it is not producing lift.

c. In producing lift the whole aircraft produce additional drag and some of this will be increments in those components which make up zero lift drag.

3. **Zero Lift Drag**. When an aircraft is flying at zero lift angle of attack the resultant of all the aerodynamic forces acts parallel and opposite to the direction of flight. This is known as Zero Lift Drag (but Profile Drag or Boundary Layer Drag in some textbooks) and is composed of:

a. Surface friction drag.

b. Form drag (boundary layer normal pressure drag).

c. Interference drag.

4. **Lift Dependent Drag**. In producing lift the whole aircraft will produce additional drag composed of:

a. Induced drag (vortex drag).

b. Increments of:

(1) Form drag.

(2) Surface friction drag.

(3) Interference drag.

ZERO LIFT DRAG

The Boundary Layer

5. Although it is convenient to ignore the effects of viscosity whenever possible, certain aspects of aerodynamics cannot be explained if viscosity is disregarded.

6. Because air is viscous, any object moving through it collects a group of air particles which it pulls along. A particle directly adjacent to the object's surface will, because of viscous adhesion, be pulled along at approximately the speed of the object. A particle slightly further away from the surface will also be pulled along; however its velocity will be slightly less than the object's velocity. As we move further and further away from the surface the particles of air are affected less and less, until a point is reached where the movement of the body does not cause any parallel motion of air particles whatsoever.

7. The layer of air extending from the surface to the point where no dragging effect is discernable is known as the boundary layer. In flight, the nature of the boundary layer determines the maximum lift coefficient, the stalling characteristics of a wing, the value of form drag, and to some extent the high-speed characteristics of an aircraft.

8. The viscous drag force within the boundary layer is not sensitive to pressure and density variations and is therefore unaffected by the variations in pressures at right angles (normal) to the surface of the body. The coefficient of viscosity of air changes in a similar manner to temperature and therefore decreases with altitude.

Surface Friction Drag

9. The surface friction drag is determined by:

a. The total surface area of the aircraft.

b. The coefficient of viscosity of the air.

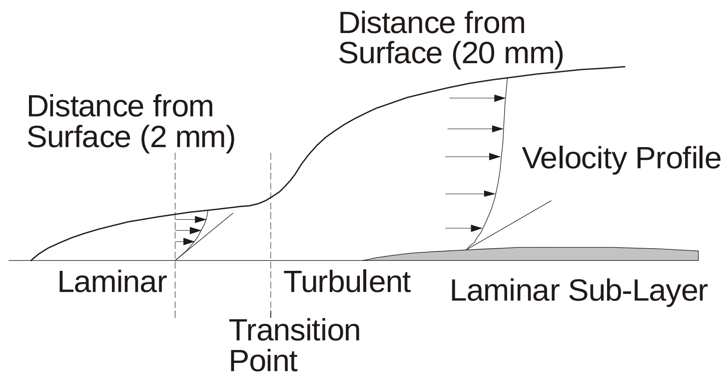
c. The rate of change of velocity across the flow.

10. **Surface Area**. The whole surface area of the aircraft has a boundary layer and therefore has surface friction drag.

11. **Coefficient of Viscosity**. The absolute coefficient of viscosity (μ) is a direct measure of the viscosity of a fluid, ie it is the means by which a value can be allotted to this property of a fluid. The greater the viscosity of the air, the greater the dragging effect on the aircraft's surface.

12. **Rate of Change of Velocity**. Consider the flow of air moving across a thin flat plate, as in Fig 1. The boundary layer is normally defined as that region of flow in which the speed is less than 99% of the free stream flow, and usually exists in two forms, laminar and turbulent. In general, the flow at the front of a body is laminar and becomes turbulent at a point some distance along the surface, known as the transition point From Fig 1 it can be seen that the rate of change of velocity is greater at the surface in the turbulent flow than in the laminar. This higher rate of change of velocity results in greater surface friction drag. The velocity profile for the turbulent layer shows the effect of mixing with the faster moving air above the boundary layer. This is an important characteristic of the turbulent flow since it indicates a higher level of kinetic energy. Even when the boundary layer is turbulent however, a very thin layer exists immediately adjacent to the surface in which random velocities are smoothed out. This very thin layer (perhaps 1% of the total thickness of the turbulent layer) remains in the laminar state and is called the laminar sub-layer. Though extremely thin, the presence of this layer is important when considering the surface friction drag of a body and the reduction that can be obtained in the drag of a body as a result of smoothing the surface.

1-5 Fig 1 Boundary Layer



Transition to Turbulence

13. It follows from para 12 that forward movement of the transition point increases the surface friction drag. The position of the transition point depends upon:

a. Surface condition.

b. Speed of the flow.

c. Size of the object.

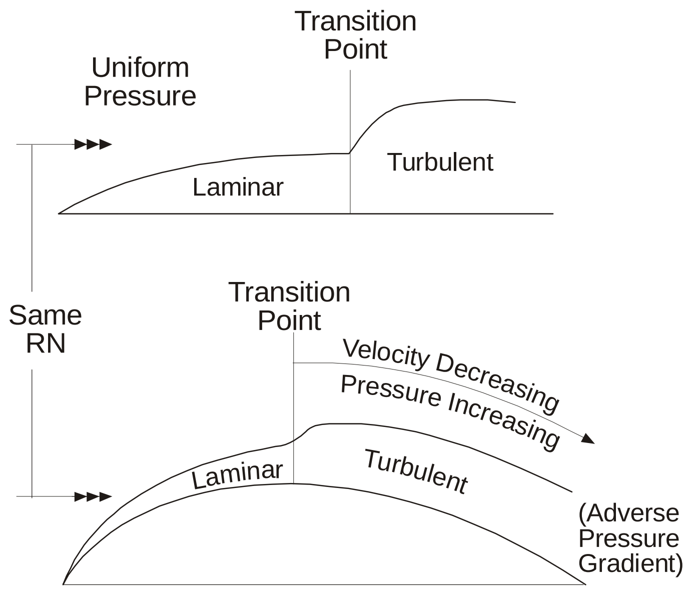
d. Adverse pressure gradient.

14. **Surface Condition**. Both the laminar and turbulent boundary layers thicken downstream, the average thickness varying from approximately 2 mm at a point 1 m downstream of the leading edge, to 20 mm at a point 1 m downstream of the transition point. Generally, the turbulent layer is about ten times thicker than the laminar layer but exact values vary from surface to surface. The thin laminar layer is extremely sensitive to surface irregularities. Any roughness which can be felt by the hand, on the skin of the aircraft, will cause transition to turbulence at that point, and the thickening boundary layer will spread out fanwise down-stream causing a marked increase in surface friction drag.

15. **Speed and Size**. The nineteenth century physicist, Reynolds, discovered that in a fluid flow of given density and viscosity the flow changed from streamline to turbulent when the velocity reached a value that was inversely proportional to the thickness of a body in the flow. That is, the thicker the body, the lower the speed at which transition occurred. Applied to an aerofoil of given thickness it follows that an increase of flow velocity will cause the transition point to move forward towards the leading edge. (A fuller explanation of Reynolds Number (RN) is given in Volume 1, Chapter 2.) Earlier transition means that a greater part of the surface is covered by a turbulent boundary layer, creating greater surface friction drag. However, it should be remembered that the turbulent layer has greater kinetic energy than the laminar, the effect of which is to delay separation, thereby increasing the maximum value of C_L (see Volume 1, Chapter 4).

16. **Adverse Pressure Gradient**. It has been found that a laminar boundary layer cannot be maintained, without mechanical assistance, when the pressure is rising in the direction of flow, ie in an adverse pressure gradient. Thus on the curved surfaces of an aircraft the transition point is usually beneath, or near to, the point of minimum pressure and this is normally found to be at the point of maximum thickness. Fig 2 illustrates the comparison between a flat plate and a curved surface.

1-5 Fig 2 Effect of Adverse Pressure Gradient

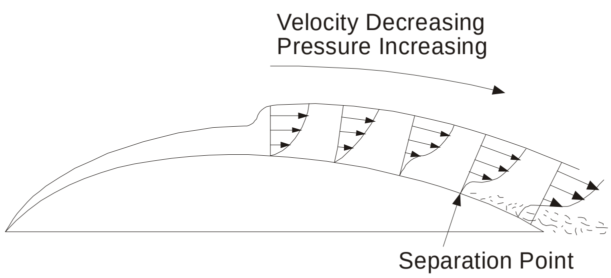


Form Drag (Boundary Layer Normal Pressure Drag)

17. The difference between surface friction and form drag can be easily appreciated if a flat plate is considered in two attitudes, first at zero angle of attack when all the drag is friction drag, and second at 90º angle of attack when all the drag is form drag due to the separation.

18. **Separation Point**. The effect of surface friction is to reduce the velocity, and therefore the kinetic energy, of the air within the boundary layer. On a curved surface the effect of the adverse pressure gradient is to reduce further the kinetic energy of the boundary layer. Eventually, at a point close to the trailing edge of the surface, a finite amount of the boundary layer stops moving, resulting in eddies within the turbulent wake. Fig 3 shows the separation point and the flow reversal which occurs behind that point. Aft of the transition point, the faster moving air above mixes with the turbulent boundary layer and therefore has greater kinetic energy than the laminar layer. The turbulent layer will now separate as readily under the influence of the adverse pressure as would the laminar layer.

1-5 Fig 3 Boundary Layer Separation



Streamlining

19. When a boundary layer separates some distance forward of the trailing edge, the pressure existing at the separation point will be something less than at the forward stagnation point. Each part of the aircraft will therefore be subject to a drag force due to the differences in pressure between its fore and aft surface areas. This pressure drag (boundary layer normal pressure drag) can be a large part of the total drag and it is therefore necessary to delay separation for as long as possible. The

streamlining of any object is a means of increasing the fineness ratio to reduce the curvature of the surfaces and thus the adverse pressure gradient (fineness ratio of an aerofoil is $\frac{\text{chord}}{\text{thickness}}$).

Interference Drag

20. On a complete aircraft the total drag is greater than the sum of the values of drag for the separate parts of the aircraft. The additional drag is the result of flow interference at wing/fuselage, wing/nacelle and other such junctions leading to modification of the boundary layers. Further turbulence in the wake causes a greater pressure difference between fore and aft surface areas and therefore additional resistance to movement. For subsonic flight this component of total drag can be reduced by the addition of fairings at the junctions, e.g. at the trailing edge wing roots.

Summary

21. The drag created when an aircraft is not producing lift, e.g. in a truly vertical flight path, is called zero lift drag; it comprises:

a. Surface friction drag.

b. Form drag (boundary layer normal pressure drag).

c. Interference drag.

22. Surface friction drag is dependent upon:

a. Total wetted area.

b. Viscosity of the air.

c. Rate of change of velocity across the flow:

(1) Transition point.

(2) Surface condition.

(3) Speed and size.

(4) Adverse pressure gradient.

23. Form drag is dependent upon:

a. Separation point:

(1) Transition point.

(2) Adverse pressure gradient.

b. Streamlining.

24. Interference drag is caused by the mixing of airflows at airframe junctions.

25. Zero lift drag varies as the square of the equivalent air speed (EAS).

LIFT DEPENDENT DRAG

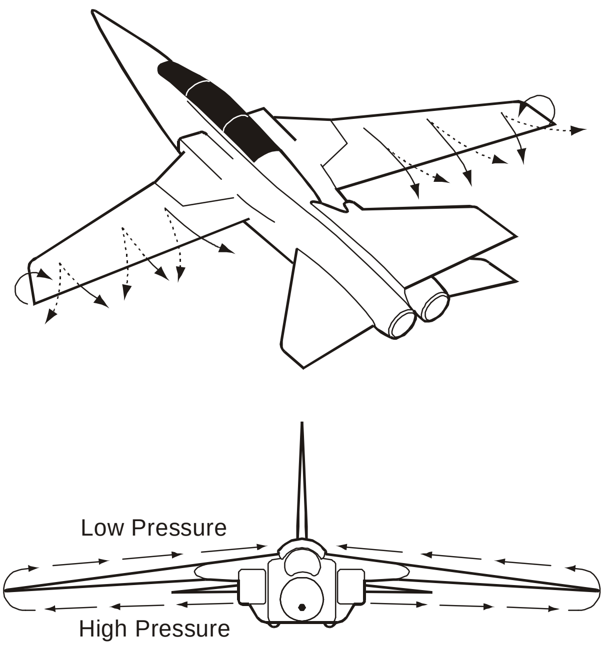
General

26. All of the drag which arises because the aircraft is producing lift is called lift dependent drag. It comprises in the main induced drag, but also contains increments of the types of drag which make up zero lift drag. The latter are more apparent at high angles of attack.

Induced Drag (Vortex Drag)

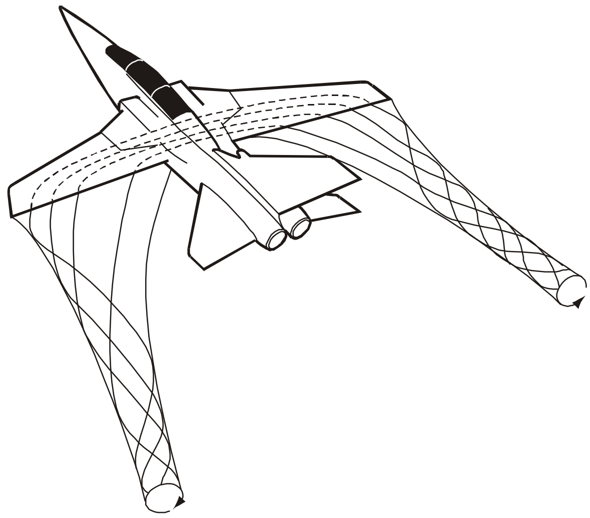
27. If a finite rectangular wing at a positive angle of attack is considered, the spanwise pressure distribution will be as shown in Fig 4. On the underside of the wing, the pressure is higher than that of the surrounding atmosphere so the air spills around the wing tips, causing on outward airflow towards them. On the upper surface, the pressure is low, and the air flows inwards. This pattern of airflow results in a twisting motion in the air as it leaves the trailing edge. Viewed from just downstream of the wing, the air rotates and forms a series of vortices along the trailing edge, and near the wing tips the air forms into a concentrated vortex.

1-5 Fig 4 Spanwise Flow



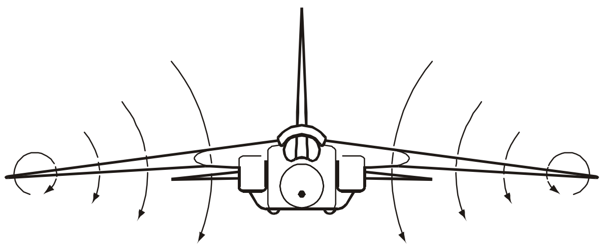
Further downstream all of the vorticity collects into trailing vortices as seen in Fig 5. The wing tip vortices intensify under high lift conditions, e.g. during manoeuvre, and the drop in pressure at the core may be sufficient to cause vapour trails to form.

1-5 Fig 5 Trailing Vortices



28. **Induced Downwash**. The trailing vortices modify the whole flow pattern. In particular, they alter the flow direction and speed in the vicinity of the wing and tail surfaces. The trailing vortices therefore have a strong influence on the lift, drag and handling qualities of an aircraft. Fig 6 shows how the airflow behind the wing is being drawn downwards. This effect is known as downwash which also influences the flow over the wing itself, with important consequences. Firstly the angle of attack relative to the modified total airstream direction is reduced. This in effect is a reduction in angle of attack, and means that less lift will be generated unless the angle of attack is increased. The second and more important consequence is that what was previously the lift force vector is now tilted backward relative to the free stream flow. There is therefore a rearward component of the force which is induced drag or vortex drag. A further serious consequence of downwash is that the airflow approaching the tailplane is deflected downwards so that the effective angle of attack of the tailplane is reduced.

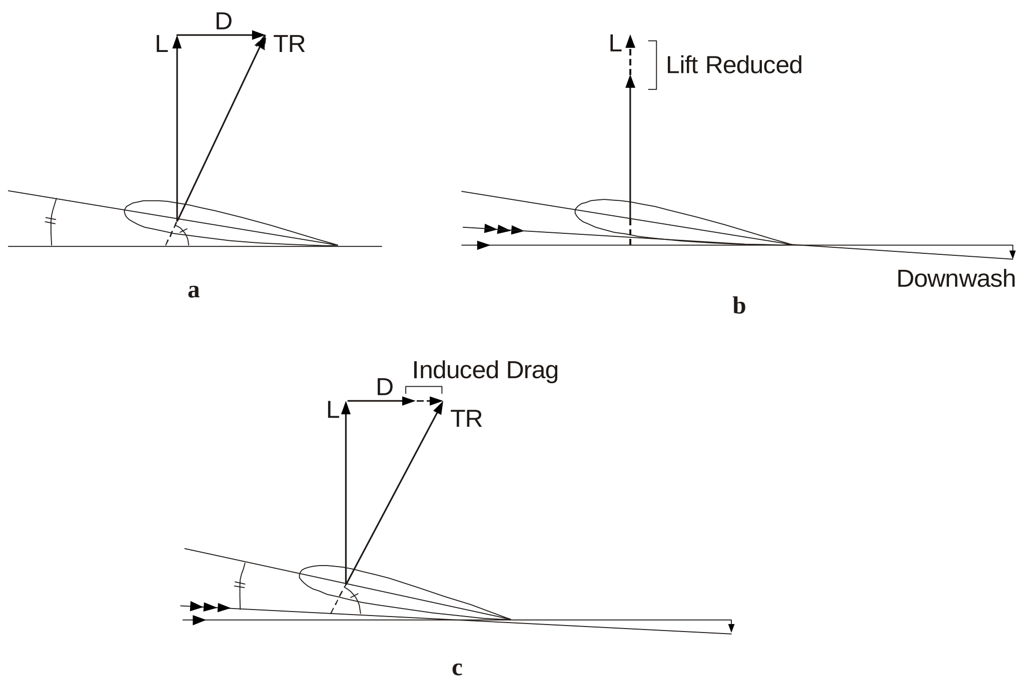
1-5 Fig 6 Downwash



Diagrammatic Explanation

29. Consider a section of a wing of infinite span which is producing lift but has no trailing edge vortices. Fig 7a shows a total aerodynamic reaction (TR) which is divided into lift (L) and drag (D). The lift component, being equal and opposite to weight, is at right angles to the direction of flight, and the drag component is parallel and opposite to the direction of flight. The angle at which the total reaction lies to the relative airflow is determined only by the angle of attack of the aerofoil.

1-5 Fig 7 Effect of Downwash on Lift and Drag



30. Fig 7b shows the same section, but of a wing of finite span and therefore having trailing edge vortices. The effect of the induced downwash (due to the vortices) is to tilt downwards the effective relative airflow, thereby reducing the effective angle of attack. To regain the consequent loss of lift the aerofoil must be raised until the original lift value is restored (see Fig 7c). The total reaction now lies at the original angle, but relative to the effective airflow, the component parallel to the direction of flight is longer. This additional value of the drag - resulting from the presence of wing vortices is known as induced drag or vortex drag.

Factors Affecting Induced Drag

31. The main factors affecting vortex formation and therefore induced drag are:

a. Planform.

b. Aspect ratio.

c. Lift and weight.

d. Speed.

For an elliptical planform, which gives the minimum induced drag at any aspect ratio:

$$C_{Di} = \frac{C_L^2}{\pi A} \text{ where } C_{Di} = \text{coefficient of induced drag}$$

$$A = \text{aspect ratio}$$

A correction factor k is required for other planforms.

32. **Planform**. Induced drag is greatest where the vortices are greatest, that is, at the wing tips; and so, to reduce the induced drag the aim must be to achieve an even spanwise pressure distribution. An elliptical planform has unique properties due to elliptic spanwise lift distribution, viz:

a. The downwash is constant along the span.

b. For a given lift, span and velocity, this planform creates the minimum induced drag.

An elliptical wing poses manufacturing difficulties, fortunately a careful combination of taper and washout, or section change, at the tips, can approximate to the elliptic ideal.

33. **Aspect Ratio (AR)**. It has been previously stated that, if the AR is infinite, then the induced drag is zero. The nearer we can get to this impossible configuration, then the less induced drag is produced. The wing tip vortices are aggravated by the increased tip spillage, caused by the transverse flow over the longer chord of a low AR wing and the enhanced induced downwash affects a greater proportion of the shorter span. Induced drag is inversely proportional to the AR, e.g. if the AR is doubled, then the induced drag is halved.

34. **Effect of Lift and Weight.** The induced downwash angle (α_1) (Volume 1, Chapter 4, Fig 10), and therefore the induced drag, depends upon the difference in pressure between the upper and lower surfaces of the wing, and this pressure difference is the lift produced by the wing. It follows then that an increase in C_L (e.g. during manoeuvres or with increased weight) will increase the induced drag at that speed. In fact, the induced drag varies as C_L^2 and, therefore, as weight^2, at a given speed.

35. **Effect of Speed**. If, while maintaining level flight, the speed is reduced to, say, half the original, then the dynamic pressure producing the lift ($\frac{1}{2} \rho V^2$) is reduced four times. To restore the lift to its original value, the value of C_L must be increased four-fold. The increased angle of attack necessary to do this inclines the lift vector to the rear, increasing the contribution to the induced drag. In addition, the vortices are affected, since the top and bottom aerofoil pressures are altered with the angle of attack.

36. **High Angles of Attack**. Induced drag at high angles of attack, such as occur at take-off, can account for nearly three-quarters of the total drag, falling to an almost insignificant figure at high speed.

Increments of Zero Lift Drag Resulting from Lift Production

37. **Effect of Lift.** Remembering that two of the factors affecting zero lift drag are transition point and adverse pressure gradient, consider the effect of increasing lift from zero to the maximum for manoeuvre. Forward movement of the peak of the low-pressure envelope will cause earlier transition of the boundary layer to turbulent flow, and the increasing adverse pressure gradient will cause earlier separation. Earlier transition increases the surface friction drag and earlier separation increases the form drag.

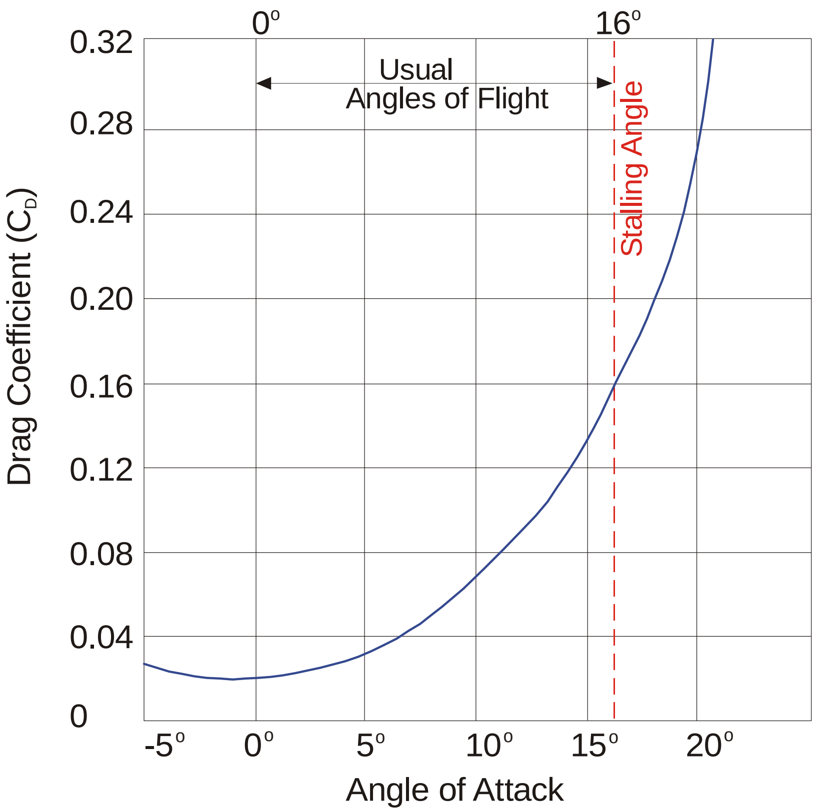
38. **Frontal Area**. As an aircraft changes its angle of attack, either because of a speed change or to manoeuvre, the frontal area presented to the airflow is changed and consequently the amount of form drag is changed.

39. **Interference Drag**. Interference drag arises from the mixing of the boundary layers at junctions on the airframe. When the aircraft is producing lift, the boundary layers are thicker and more turbulent and therefore create greater energy losses where they mix. The increments of surface friction, form and interference drag arise because the aircraft is producing lift, and these values of additional drag may be included in lift dependent drag. It follows that the greater the lift, the greater the drag increments, and, in fact, this additional drag is only really noticeable at high angles of attack.

Variation of Drag with Angle of Attack

40. Fig 8 shows that total drag varies steadily with change of angle of attack, being least at small negative angles and increasing on either side. The rate of increase becomes marked at angles of attack above about 8º and after the stall it increases at a greater rate. The sudden rise at the stall is caused by the turbulence resulting from the breakdown of steady flow.

1-5 Fig 8 Typical Drag Curve for a Cambered Section



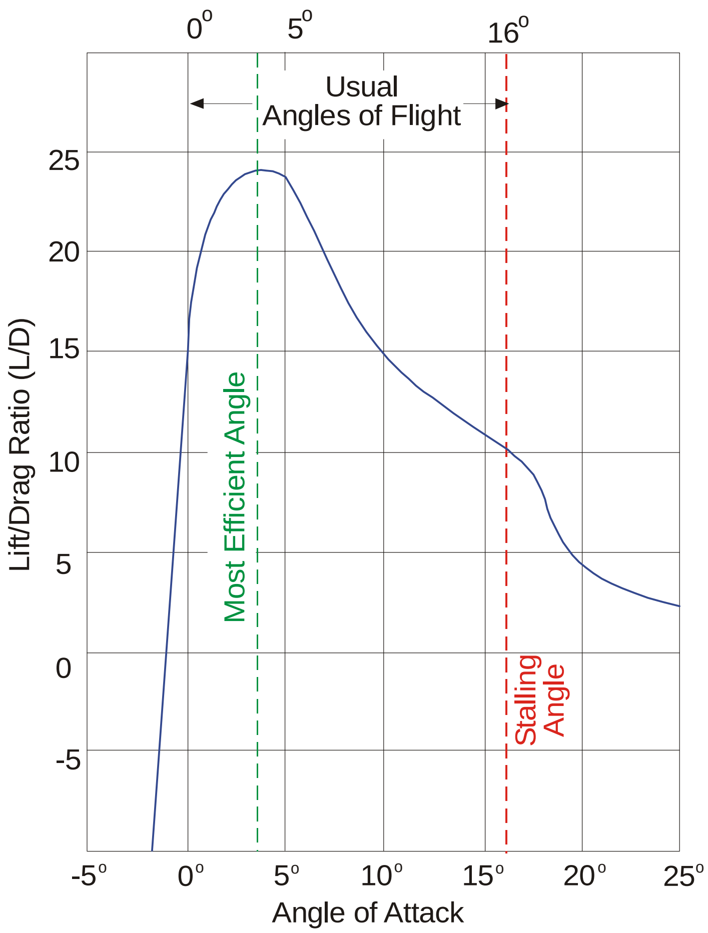
Variation of Lift/Drag Ratio with Angle of Attack

41. It is apparent that for a given amount of lift it is desirable to have the least possible drag from the aerofoil. Typically, the greatest lifting effort is obtained at an angle of attack of about 16º and least drag occurs at an angle of attack of about –2º. Neither of these angles is satisfactory, as the ratio of lift to drag at these extreme figures is low. What is required is the maximum lifting effort compared with the drag at the same angle, ie the highest lift/drag ratio (L/D ratio).

42. The L/D ratio for an aerofoil at any selected angle of attack can be calculated by dividing the C_L at that angle of attack by the corresponding C_D. In practice the same result is obtained irrespective of whether the lift and drag or their coefficients are used for the calculation:

$$\frac{L}{D} = \frac{C_L \frac{1}{2}\rho V^2 S}{C_D \frac{1}{2}\rho V^2 S} = \frac{C_L}{C_D}$$

1-5 Fig 9 Variation of L/D Ratio with Angle of Attack



43. Fig 9 shows that the lift/drag ratio increases rapidly up to an angle of attack of about 4º at which point the lift may be between 12 to 25 times the drag, the exact figure depending on the aerofoil used. At larger angles the L/D ratio decreases steadily, because even though the lift itself is still increasing the proportion of drag is rising at a faster rate. One important feature of this graph is the indication of the angle of attack for the highest L/D ratio; this angle is one at which the aerofoil gives its best all-round performance. At a higher angle the required lift is obtained at a lower, and hence, uneconomical speed; at a lower angle it is obtained at a higher, and also uneconomical, speed.

Summary

44. Lift dependent drag comprises:

a. Induced drag (vortex drag).

b. Increments of:

(1) Surface friction drag.

(2) Form drag.

(3) Interference drag.

45. Induced drag varies as:

a. C_L

b. $\frac{1}{V^2}$

c. $(\text{Weight})^2$

d. $\frac{1}{\text{aspect ratio}}$

TOTAL DRAG

General

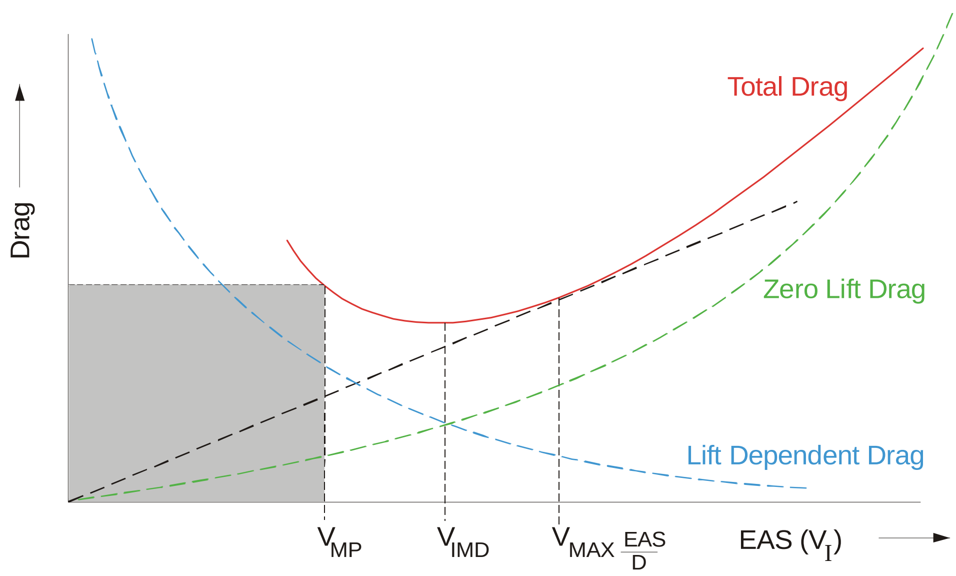
46. A typical total drag curve is shown in Fig 10. This curve is valid for one particular aircraft **for one weight only** in level flight. Speeds of particular interest are indicated on the graph:

a. The shaded area is the minimum product of drag and velocity and therefore gives the minimum power speed (V_{MP}).

b. V_{IMD} is the speed at which total drag is lowest. It coincides with the best lift/drag ratio speed.

c. Maximum EAS/drag ratio speed is the main aerodynamic consideration for best range. It has the value of 1.32 × V_{IMD}.

1-5 Fig 10 Total Drag Curve



CHAPTER 6 - STALLING

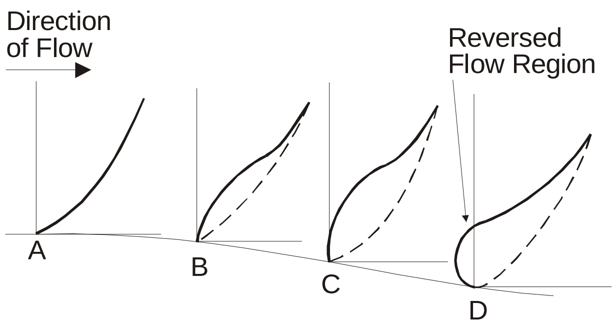
Introduction

1. In Volume 1, Chapter 5, it was stated that the nature of the boundary layer determined the stalling characteristics of a wing. In particular the phenomenon of boundary layer separation is extremely important. This chapter will first discuss what happens when a wing stalls, it will then look at the aerodynamic symptoms and the variations in the basic stalling speed and finally consider autorotation.

Boundary Layer Separation

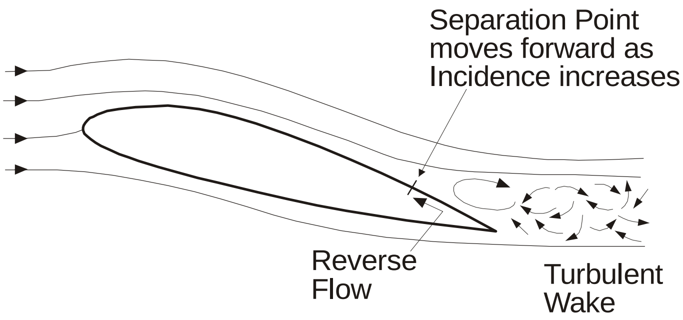
2. Boundary layer separation is produced as a result of the adverse pressure gradient developed round the body. The low energy air close to the surface is unable to move in the opposite direction to the pressure gradient and the flow close to the surface flows in the reverse direction to the free stream. The development of separation is shown in Fig 1. A normal typical velocity profile is shown corresponding to point A, while a little further down the surface, at point B, the adverse pressure gradient will have modified the velocity profile as shown. At point C, the velocity profile has been modified to such an extent that, at the surface, flow has ceased. Further down the surface, at point D, the flow close to the surface has reversed and the flow is said to have separated. Point C is defined as the separation point. In the reversed flow region aft of this point the flow is eddying and turbulent, with a mean velocity of motion in the opposite direction to the free stream.

1-6 Fig 1 Velocity Profile



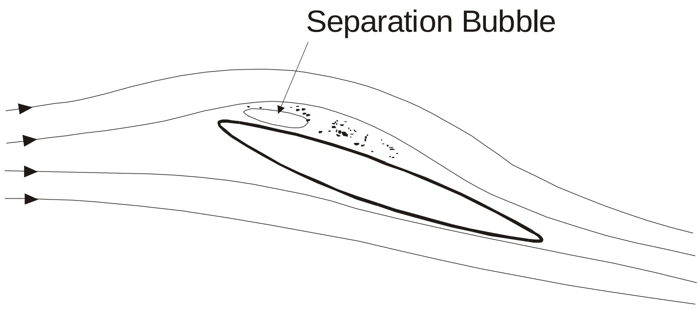
3. **Trailing Edge Separation**. On a normal subsonic section virtually no flow separation occurs before the trailing edge at low angles of attack, the flow being attached over the rear part of the surface in the form of a turbulent boundary layer. As the angles of attack increases, so the adverse pressure gradient is increased as explained in Volume 1, Chapter 5, and the boundary layer will begin to separate from the surface near the trailing edge of the wing (see Fig 2). As the angle of attack is further increased the separation point will move forwards along the surface of the wing towards the leading edge. As the separation point moves forward the slope of the lift/angle of attack curve decreases and eventually an angle of attack is reached at which the wing is said to stall. The flow over the upper surface of the wing is then completely broken down and the lift produced by the wing decreases.

1-6 Fig 2 Trailing Edge Separation



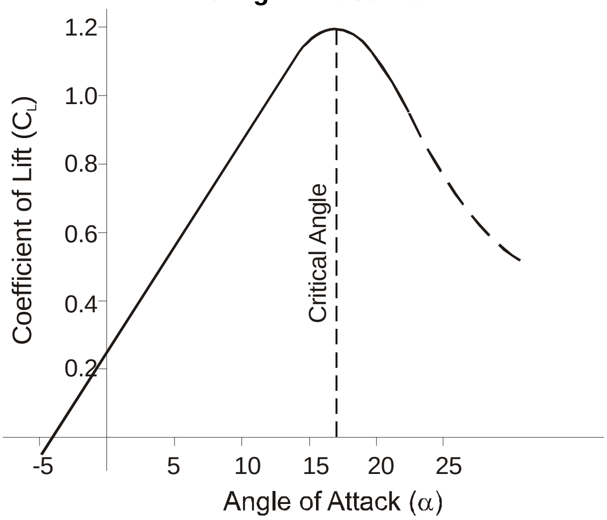
4. **Leading Edge Separation**. Turbulent trailing edge separation is usually found on conventional low speed wing sections; another type of separation sometimes occurs on thin wings with sharp leading edges. This is laminar flow separation; the flow separating at the thin nose of the aerofoil before becoming turbulent. Transition may then occur in the separated boundary layer and the layer may then re-attach further down the body in the form of a turbulent boundary layer. Underneath the separated layer a stationary vortex is formed and this is often termed a bubble (Fig 3). The size of these bubbles depends on the aerofoil shape and has been found to vary from a very small fraction of the chord (short bubbles) to a length comparable to that of the chord itself (long bubbles). The long bubbles affect the pressure distribution even at small angles of attack, reducing the lift-curve slope. The eventual stall on this type of wing is more gradual. However, the short bubbles have little effect on the pressure distribution, and hence on the lift curve slope, but when the bubble eventually bursts the corresponding stall is an abrupt one.

1-6 Fig 3 Leading Edge Separation



5. **Critical Angle.** The marked reduction in the lift coefficient which accompanies the breakdown of airflow over the wing occurs at the critical angle of attack for a particular wing. In subsonic flight an aircraft will always stall at the same critical angle of attack except at high Reynolds Numbers (Volume 1, Chapter 2). A typical lift curve showing the critical angle is seen in Fig 4; it should be noted that not all lift is lost at the critical angle, in fact the aerofoil will give a certain amount of lift up to 90º.

1-6 Fig 4 Lift Curve



Aerodynamic Symptoms

6. The most consistent symptom, or stall warning, arises from the separated flow behind the wing passing over the tail surfaces. The turbulent wake causes buffeting of the control surfaces which can usually be felt at the control column and rudder pedals. As the separation point starts to move forward, to within a few degrees of the critical angle of attack, the buffeting will usually give adequate warning of the stall. On some aircraft separation may also occur over the cockpit canopy to give additional audible warning. The amount of pre-stall buffet depends on the position of the tail surfaces with respect to the turbulent wake. When the trailing edge flap is lowered the increased downwash angle behind the (inboard) flaps may reduce the amount of buffet warning of the stall.

Pitching Moments

7. As angle of attack is increased through the critical angle of attack the wing pitching moment changes. Changes in the downwash angle behind the wing also cause the tail pitching moment to change. The overall effect varies with aircraft type and may be masked by the rate at which the elevator is deflected to increase the angle of attack. Most aircraft, however, are designed to produce a nose-down pitching moment at the critical angle of attack.

Tip Stalling

8. The wing of an aircraft is designed to stall progressively from the root to the tip. The reasons for this are threefold:

a. To induce early buffet symptoms over the tail surface.

b. To retain aileron effectiveness up to the critical angle of attack.

c. To avoid a large rolling moment which would arise if the tip of one wing stalled before the other (wing drop).

9. A rectangular straight wing will usually stall from the root because of the reduction in effective angle of attack at the tips caused by the wing tip vortex. If washout is incorporated to reduce vortex drag, it also assists in delaying tip stall. A tapered wing on the other hand will aggravate the tip stall due to the lower Reynolds Number (smaller chord) at the wing tip.

10. The most common features designed to prevent wing tip stalling are:

a. **Washout.** A reduction in incidence at the tips will result in the wing root reaching its critical angle of attack before the wing tip.

b. **Root Spoilers**. By making the leading edge of the root sharper, the airflow has more difficulty in following the contour of the leading edge and an early stall is induced.

c. **Change of Section**. An aerofoil section with more gradual stalling characteristics may be employed towards the wing tips (increased camber).

d. **Slats and Slots**. The use of slats and/or slots on the outer portion of the wing increases the stalling angle of that part of the wing. Slats and slots are dealt with more fully in Volume 1, Chapter 10.

STALLING SPEED

General

11. In level flight the weight of the aircraft is balanced by the lift, and from the lift formula (Lift = $C_L \frac{1}{2} \rho V^2 S$) it can be seen that lift is reduced whenever any of the other factors in the formula are reduced. For all practical purposes density (ρ) and wing area (S) can be considered constant (for a particular altitude and configuration). If the engine is throttled back the drag will reduce the speed, and, from the formula, lift will be reduced. To keep the lift constant and so maintain level flight, the only factor that is readily variable is the lift coefficient (C_L).

12. As has been shown, the C_L can be made larger by increasing the angle of attack, and by so doing the lift can be restored to its original value so that level flight is maintained at the reduced speed. Any further reduction in speed necessitates a further increase in the angle of attack, each succeeding lower IAS corresponding to each succeeding higher angle of attack. Eventually, at a certain IAS, the wing reaches its stalling angle, beyond which point any further increase in angle of attack, in an attempt to maintain the lift, will precipitate a stall.

13. The speed corresponding to a given angle of attack is obtained by transposing the lift formula, thus:

$$L = C_L \tfrac{1}{2} \rho_0 V_I^2 S$$

$$\therefore \quad V_I^2 = \frac{L}{C_L \frac{1}{2} \rho_0 S}$$

$$\therefore \quad V_I = \sqrt{\frac{L}{C_L \frac{1}{2} \rho_0 S}}$$

If it is required to know the speed corresponding to the critical angle of attack, the value of C_L in the above formula is C_{Lmax} and:

$$V_{Istall} = \sqrt{\frac{L}{C_{Lmax} \frac{1}{2} \rho_0 S}}$$

Inspection of this formula shows that the only two variables (clean aircraft) are V, and L, ie:

$$V_{Istall} \propto \sqrt{\text{lift}}$$

14. This relationship is readily demonstrated by the following examples:

a. **Steep Dive**. When pulling out of a dive, if the angle of attack is increased to the critical angle, separation will occur and buffet will be felt at a high air speed.

b. **Vertical Climb**. In true vertical flight lift is zero and no buffet symptoms will be produced even at zero IAS.

Basic Stalling Speed

15. The most useful stalling speed to remember is the stalling speed corresponding to the critical angle of attack in straight and level flight. It may be defined as the speed below which a clean aircraft of stated weight, with the engines throttled back, can no longer maintain straight and level flight. This speed is listed in the Aircrew Manual for a number of different weights.

16. Applying these qualifications to the formulae in para 13, it can be seen that level flight requires a particular value of lift, ie L=W, and:

$$V_B = \sqrt{\frac{W}{C_{L\max}\frac{1}{2}\rho_0 S}}$$

where V_B = basic stalling speed

17. If the conditions in para 15 are not met, the stalling speed will differ from the basic stalling speed. The factors which change V_B are therefore:

a. Change in weight.

b. Manoeuvre (load factor).

c. Configuration (changes in C_{Lmax}).

d. Power and slipstream.

Weight Change

18. The relationship between the basic stalling speeds at two different weights can be obtained from the formula in para 16:

$$\text{ie the ratio of } V_{B1}:V_{B2} = \left(\sqrt{\frac{W_1}{C_{L\max}\frac{1}{2}\rho_0 S}}\right):\left(\sqrt{\frac{W_2}{C_{L\max}\frac{1}{2}\rho_0 S}}\right)$$

and, as the two denominators on the right-hand side are identical:

$$V_{B2}:V_{B1} = \sqrt{W_2}:\sqrt{W_1}$$

$$\text{or} \quad \frac{V_{B2}}{V_{B1}} = \sqrt{\frac{W_2}{W_1}}$$

$$\text{from which} \quad V_{B2} = V_{B1}\sqrt{\frac{W_2}{W_1}}$$

where V_{B1} and V_{B2} are the basic stalling speeds at weights W_1 and W_2 respectively.

19. This relationship is true for any given angle of attack provided that the appropriate value of C_L is not affected by speed. The reason is that, to maintain a given angle of attack in level flight, it is necessary to reduce the dynamic pressure (IAS) if the weight is reduced.

Manoeuvre

20. The relationship between the basic stalling speed and the stalling speed in any other manoeuvre (V_M) can be obtained in a similar way by comparing the general formula in para 13 to the level flight formula in para 16. Thus:

$$\text{the ratio } V_M : V_B = \left(\sqrt{\frac{L}{C_{L\max}\,\frac{1}{2}\rho_0 S}}\right) : \left(\sqrt{\frac{W}{C_{L\max}\,\frac{1}{2}\rho_0 S}}\right)$$

Again, the denominators on the right-hand side are identical and so:

$$V_M : V_B = \sqrt{L} : \sqrt{W}$$

or

$$\frac{V_M}{V_B} = \sqrt{\frac{L}{W}}$$

from which,

$$V_M = V_B\sqrt{\frac{L}{W}}$$

21. The relationship $\left(\frac{L}{W}\right)$ is the load factor, n, and is indicated on the accelerometer (if fitted). Thus $V_M = V_B\sqrt{n}$ and, in a 4g manoeuvre, the stalling speed is twice the basic stalling speed.

22. In the absence of an accelerometer the artificial horizon may be used as a guide to the increased stalling speed in a level turn.

$$\text{From Fig 5, } \cos\varphi = \frac{W^1}{L} = \frac{W}{L}$$

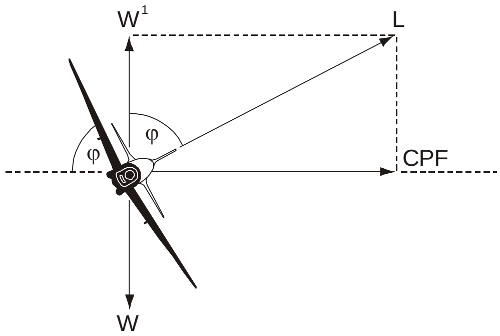
$$\text{and therefore, } \frac{L}{W} = \frac{1}{\cos\varphi}$$

substituting in the formula in para 20:

$$V_M = V_B\sqrt{\frac{1}{\cos\varphi}}$$

and, in a 60º bank turn the stalling speed is $\sqrt{2}$ or 1.4 times the basic stalling speed.

1-6 Fig 5 Forces on an Aircraft in a Level Turn



φ = Angle of Bank
W1 = Vertical Component of Lift = W
CPF = Horizontal Component of Lift = Centripetal Force

Configuration

23. From the formula in para 16 it can be seen that the stalling speed is inversely proportional to C_{Lmax} i.e. in level flight

$$V_B \propto \frac{1}{\sqrt{C_{L\,max}}}$$

24. Any change in C_{Lmax} due to the operation of high lift devices or due to compressibility effects will affect the stalling speed. In particular, lowering flaps or extending slats will result in a new basic stalling speed. These changes are usually listed in the Aircrew Manual.

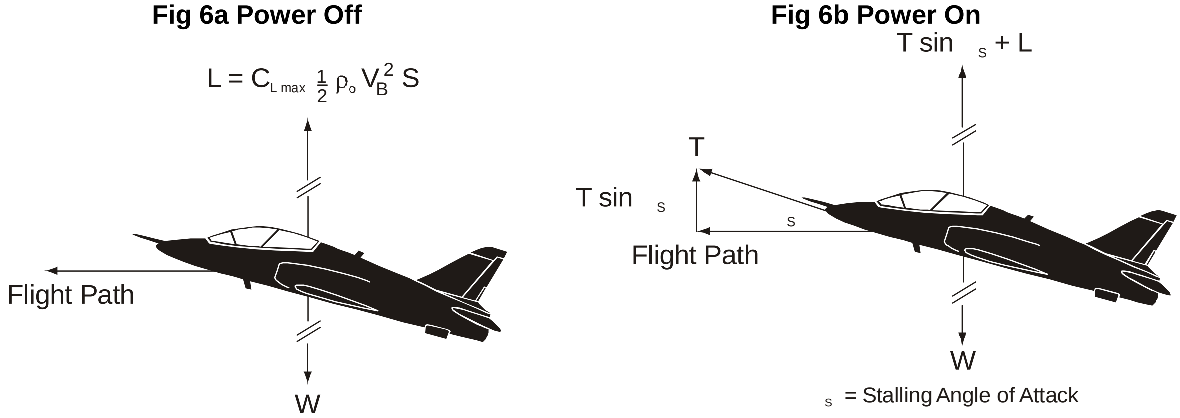
Power

25. At the basic stalling speed, the engines are throttled back and it is assumed that the weight of the aircraft is entirely supported by the wings. If power is applied at the stall the high nose attitude produces a vertical component of thrust which assists in supporting the weight and less force is required from the wings. This reduction in lift is achieved at the same angle of attack (C_{Lmax}) by reducing the dynamic pressure (IAS) and results in a lower stalling speed.

26. From Fig 6 it can be seen that L = W – T sin α_S which is less than the power-off case; and, as $V_M \propto \sqrt{L}$, V_M with power on < V_B.

1-6 Fig 6 Comparison of Power Off and Power On Stalling

Fig 6a Power Off **Fig 6b Power On**



s = Stalling Angle of Attack
T sin s = Vertical Component of Thrust

27. It should be noted that, for simplicity, the load on the tailplane has been ignored and the engine thrust line assumed parallel to the wing chord line.

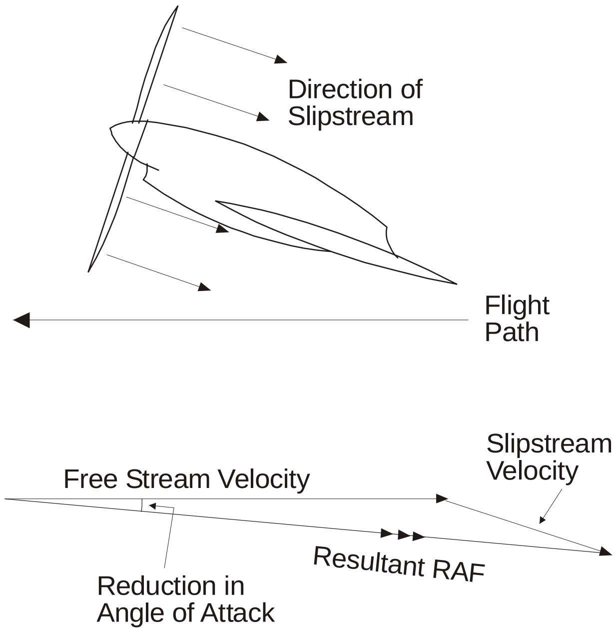
28. **Slipstream Effect**. For a propeller-powered aircraft an additional effect is caused by the velocity of the slipstream behind the propellers.

29. **Vector Change to RAF**. In Fig 7 the vector addition of the free stream and slipstream velocities results in a change in the RAF over that part of the wing affected by the propellers.

a. **Increase in Velocity**. The local increase in dynamic pressure will result in more lift behind the propellers relative to the power-off case. Thus, at a stalling angle of attack, a lower IAS is required to support the weight, ie the stalling speed is reduced.

b. **Decrease in Angle of Attack.** The reduction in angle of attack partially offsets the increase in dynamic pressure but does not materially alter the stalling speed unless $C_{L\ max}$ is increased. The only effect will be to increase the attitude at which the stall occurs. Unless the propeller slipstream covers the entire wing, the probability of a wing drop will increase because the wings will stall progressively from the tips which are at a higher angle of attack than the wings behind the propellers.

1-6 Fig 7 Slipstream Effect

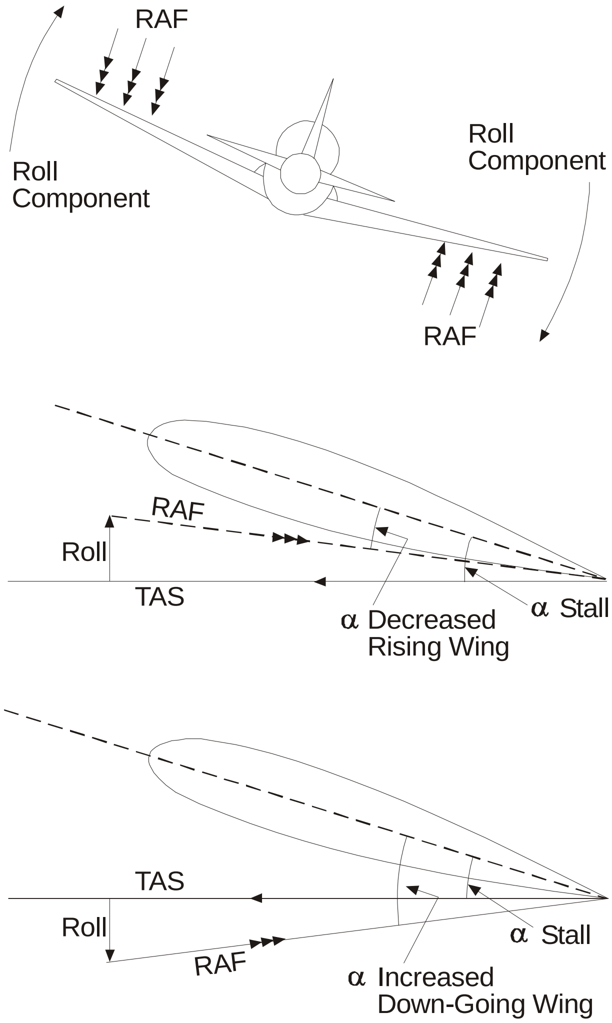


Autorotation

30. The autorotational properties of a wing are due to the negative slope of the C_L / α curve when α is greater than the stalling angle of attack.

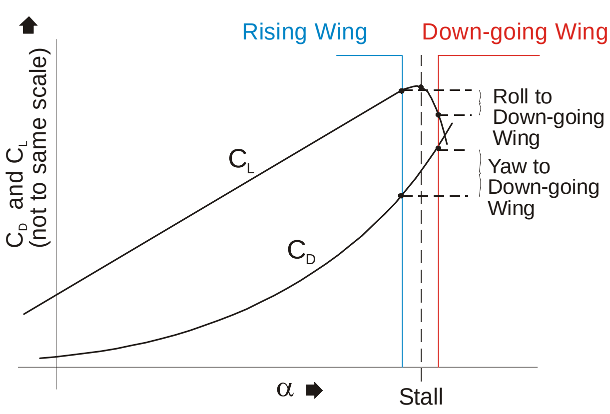
31. With reference to the stalled aircraft illustrated at Fig 8, if the aircraft starts to roll there will be a component of flow induced tending to increase the angle of attack of the down-going wing and decrease the angle of attack of the up-going wing. The cause of the roll may be either accidental (wing-drop), deliberate (further effects of applied rudder) or use of aileron at the stall.

1-6 Fig 8 Vector Addition of Roll and Forward Velocities



32. The effect of this change in angle of attack on the C_L and C_D curves *(*see Fig 9) is that the 'damping in roll' effect normally produced at low α is now reversed. The increase in angle of attack of the down-going wing decreases the C_L and increases the C_D. Conversely, the decrease in angle of attack of the up-going wing slightly decreases the C_L and decreases the C_D. The difference in lift produces a rolling moment towards the down-going wing tending to increase the angular velocity. This angular acceleration is further increased by the roll induced by the yawing motion due to the large difference in drag.

1-6 Fig 9 Autorotation



33. The cycle is automatic in the sense that the increasing rolling velocity sustains or even increases the difference in angle of attack. It should be noted, however, that at higher angles, the slope of the C_L curve may recover again to zero. This may impose a limit on the α at which autorotation is possible.

34. It is emphasized that autorotation, like the stall, is an aerodynamic event which is dependent on angle of attack. It is therefore possible to autorotate the aircraft in any attitude and at speeds higher than the basic stalling speed. This principle is the basis of many of the more advanced aerobatic manoeuvres.

Summary

35. Boundary layer separation is produced as a result of the adverse pressure gradient developed round the body.

36. In subsonic flight an aircraft will always stall at the same critical angle of attack.

37. The wing of an aircraft is designed to stall progressively from the root to the tip. The reasons for this are:

a. To induce early buffet symptoms over the tail surface.

b. To retain aileron effectiveness up to the critical angle of attack.

c. To avoid a large rolling moment which would arise if the tip of one wing stalled before the other.

38. The most common design features for preventing tip stalling are:

a. Washout.

b. Root spoilers.

c. Change of section.

d. Slats and slots.

39. Basic stalling speed is the speed below which a clean aircraft of stated weight, with engines throttled back, can no longer maintain straight and level flight.

40. The formula for the basic stalling speed is:

$$V_B = \sqrt{\frac{W}{C_{L\max}\frac{1}{2}\rho_0 S}}$$

Factors which affect V_B are:

a. Change in weight: $V_{B2} = V_{B1}\sqrt{\frac{W_2}{W_1}}$

b. Manoeuvre (load factor): $V_M = V_B\sqrt{n}$ or $V_M = V_B\sqrt{\frac{1}{\cos\varphi}}$

c. Configuration (changes in): $C_{L\max}$ $V_B \propto \frac{1}{\sqrt{C_{L\max}}}$

d. Power and Slip-stream: V_M power on < V_B

CHAPTER 7 - SPINNING

Introduction

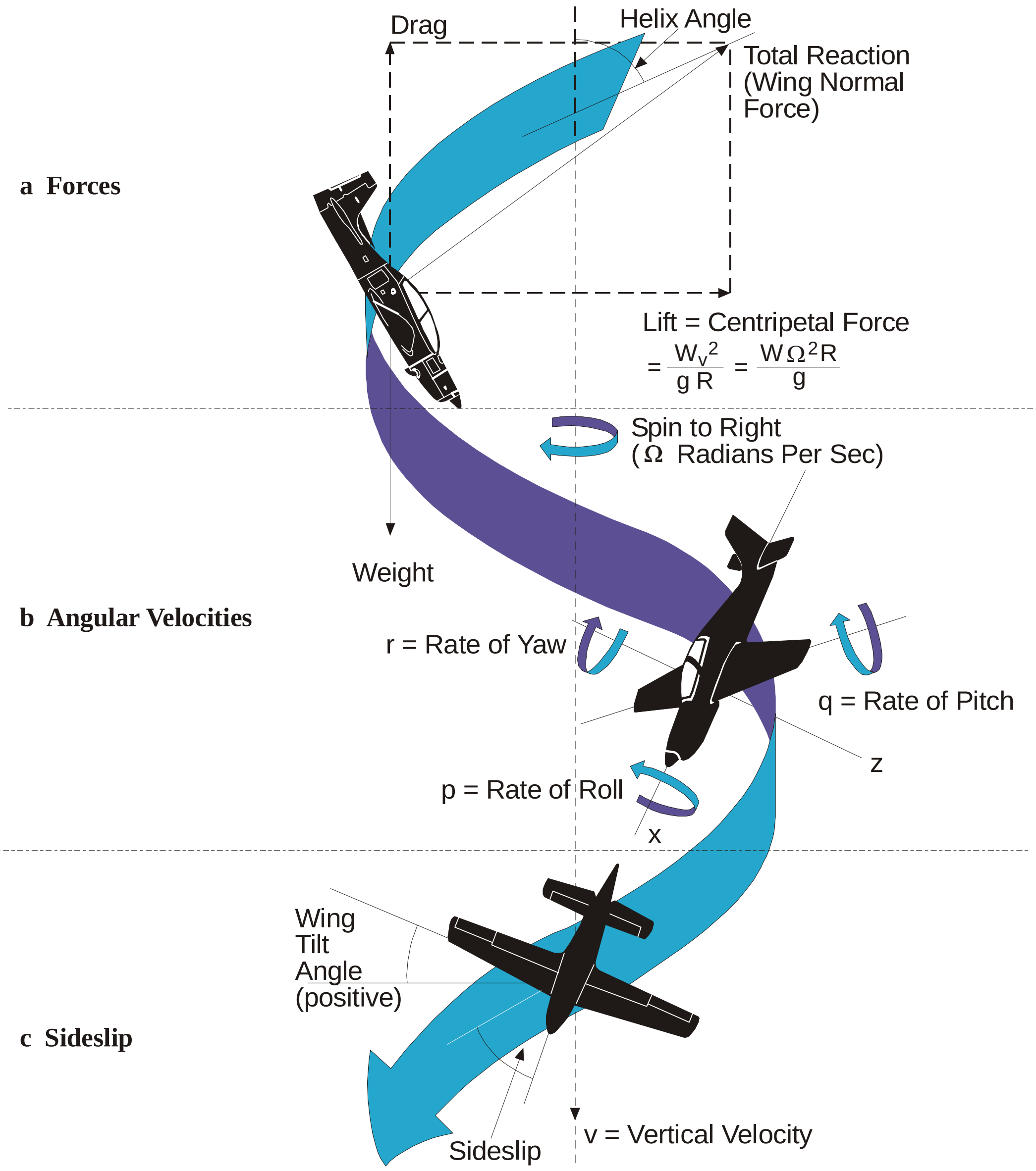
1. Spinning is a complicated subject to analyse in detail. It is also a subject about which it is difficult to make generalizations which are true for all aircraft. One type of aircraft may behave in a certain manner in a spin whereas another type will behave completely differently under the same conditions. This chapter is based on a deliberately-induced, erect spin to the right although inverted and oscillatory spins are discussed in later paragraphs.

2. The accepted sign conventions applicable to this chapter are given in Table 1, together with Fig 1.

1-7 Table 1 Sign Conventions Used in this Chapter

AXIS (Symbol)	**LONGITUDINAL (x)**	**LATERAL (y)**	**NORMAL (z)**
Positive Direction	Forwards	To right	Downwards
MOMENTS OF INERTIA	A	B	C
ANGULAR VELOCITY			
Designation	Roll	Pitch	Yaw
Symbol	p	q	r
Positive Direction	to right	nose-up	to right
MOMENTS			
Designation	rolling moment	pitching moment	yawing moment
Symbol	L	M	N
Positive Direction	to right	nose-up	to right

1-7 Fig 1 The Motion of an Aircraft in an Erect Spin to the Right



Phases of the Spin

3. The spin manoeuvre can be divided into three phases:

 a. The incipient spin.

 b. The fully-developed spin.

 c. The recovery.

4. **The Incipient Spin.** A necessary ingredient of a spin is the aerodynamic phenomenon known as autorotation. This leads to an unsteady manoeuvre which is a combination of:

 a. The ballistic path of the aircraft, which is itself dependent on the entry attitude.

 b. Increasing angular velocity generated by the autorotative rolling moment and drag-induced yawing moment.

5. **The Steady Spin**. The incipient stage may continue for some 2 to 6 turns, after which the aircraft will settle into a steady, stable, spin. There will be some sideslip and the aircraft will be rotating about all three axes. For simplicity, but without suggesting that it is possible for all aircraft to achieve this stable condition, the steady spin is qualified by a steady rate of rotation and a steady rate of descent.

6. **The Recovery**. The recovery is initiated by the pilot by actions aimed at first opposing the autorotation and then reducing the angle of attack (α) so as to unstall the wings. The aircraft may then be recovered from the ensuing steep dive.

The Steady Erect Spin

7. While rotating, the aircraft will describe some sort of ballistic trajectory dependent on the entry manoeuvre. To the pilot this will appear as an unsteady, oscillatory phase until the aircraft settles down into a stable spin with steady rate of descent and rotation about the spin axis. This will occur if the aerodynamic and inertia forces and moments can achieve a state of equilibrium. The attitude of the aircraft at this stage will depend on the aerodynamic shape of the aircraft, the position of the controls and the distribution of mass throughout the aircraft.

Motion of the Aircraft

8. The motion of the centre of gravity in a spin has two components:

a. A vertical linear velocity (rate of descent = V fps).

b. An angular velocity (Ω radians per sec) about a vertical axis, called the spin axis. The distance between the CG and the spin axis is the radius of the spin (R) and is normally small (about a wing semi-span).

The combination of these motions results in a vertical spiral or helix. The helix angle is small, usually less than 10°. Fig 1 shows the motion of the aircraft in spin.

9. As the aircraft always presents the same face to the spin axis, it follows that it must be rotating about a vertical axis passing through the centre of gravity at the same rate as the CG about the spin axis. This angular velocity may be resolved into components of roll, pitch and yaw with respect to the aircraft body axes. In the spin illustrated in Fig 1b, the aircraft is rolling right, pitching up and yawing right. For convenience the direction of the spin is defined by the direction of yaw.

10. In order to understand the relationship between these angular velocities and aircraft attitude it is useful to consider three limiting cases:

a. **Longitudinal Axis Vertical**. When the longitudinal axis is vertical the angular motion will be all roll.

b. **Lateral Axis Vertical**. For the aircraft to present the same face (pilot's head) to the spin axis, the aircraft must rotate about the lateral axis. The angular motion is all pitch.

c. **Normal Axis Vertical**. For the aircraft to present the same face (inner wing tip) to the axis of rotation, the aircraft must rotate about the normal axis at the same rate as the aircraft rotates about the axis of rotation. Thus the angular motion is all yaw.

11. Although these are hypothetical examples which may not be possible in practice, they illustrate the relationship between aircraft attitude and angular velocities. Between the extremes quoted in the previous paragraph, the motion will be a combination of roll, pitch and yaw, and depends on:

a. The rate of rotation of the aircraft about the spin axis.

b. The attitude of the aircraft, which is usually defined in terms of the pitch angle and the wing tilt angle. Wing tilt angle, often confused with bank angle, involves displacement about the normal and the longitudinal axes.

12. The aircraft's attitude in the spin also has an important effect on the sideslip present (see Fig 1c). If the wings are level, there will be outward sideslip; that is, the relative airflow will be from the direction of the outside wing (to port in the diagram). If the attitude of the aircraft is changed such that the outer wing is raised relative to the horizontal, the sideslip is reduced. This attitude change can only be due to a rotation of the aircraft about the normal axis. The angle through which the aircraft is rotated, in the plane containing the lateral and longitudinal axes, is known as the wing tilt angle and is positive with the outer wing up. If the wing tilt can be increased sufficiently to reduce the sideslip significantly, the pro-spin aerodynamic rolling moment will be reduced.

Balance of Forces in the Spin

13. Only two forces are acting on the centre of gravity while it is moving along its helical path (see Fig 1a).

a. Weight (W).

b. The aerodynamic force (N) coming mainly from the wings.

The resultant of these two forces is the centripetal force necessary to produce the angular motion.

14. Since the weight and the centripetal force act in a vertical plane containing the spin axis and the CG, the aerodynamic force must also act in this plane, ie it passes through the spin axis. It can be shown that, when the wing is stalled, the resultant aerodynamic force acts approximately perpendicular to the wing. For this reason it is sometimes called the wing normal force.

15. If the wings are level (lateral axis horizontal), from the balance of forces in Fig 1a:

a. Weight = Drag = $C_D \frac{1}{2}\rho V^2 S$

Therefore,

$$V = \sqrt{\frac{W}{C_D \frac{1}{2} \rho S}}$$

b. Lift = Centripetal force

$$C_L \frac{1}{2}\rho V^2 S = \frac{W\Omega^2 R}{g}$$

Therefore,

$$R = \frac{g C_L \frac{1}{2} \rho V^2 S}{W\Omega^2}$$

Where: R = spin radius
S = area,
V = rate of descent
W = weight

If the wings are not level, it has been seen that the departure from the level condition can be regarded as a rotation of the aircraft about the longitudinal and normal axes. Usually this angle, the wing tilt angle, is small and does not affect the following reasoning.

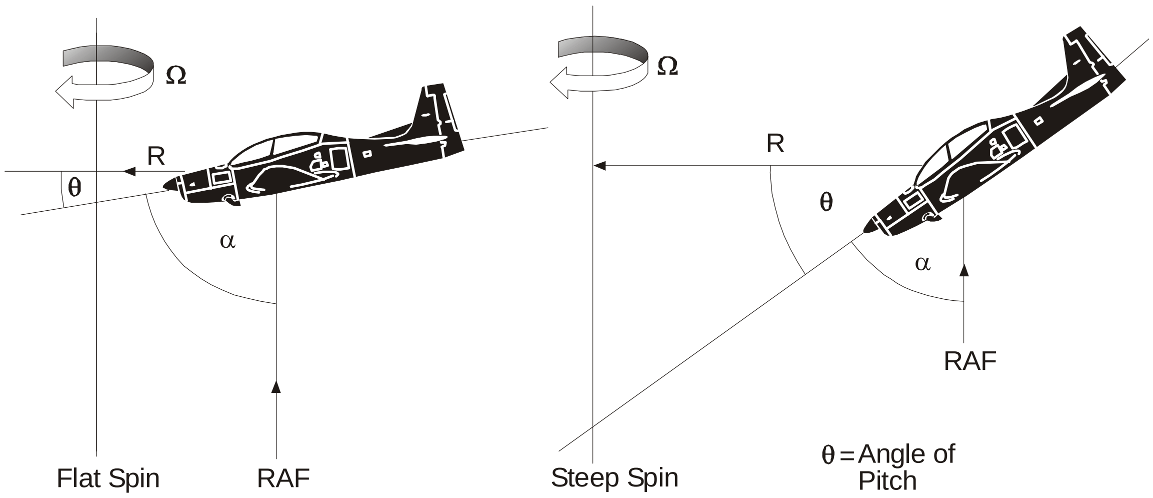
Effect of Attitude on Spin Radius

16. If for some reason the angle of attack is increased by a nose-up change in the aircraft's attitude, the vertical rate of descent (V) will decrease because of the higher C_D (para 15a). The increased alpha

on the other hand, will decrease C_L which, together with the lower rate of descent, results in a decrease in spin radius, (para 15b). It can also be shown that an increase in pitch increases the rate of spin, which will decrease R still further.

17. The two extremes of aircraft attitude possible in the spin are shown in Fig 2. The actual attitude adopted by an aircraft will depend on the balance of moments.

1-7 Fig 2 Simplified Diagram of Pitch Attitude



18. The effects of pitch attitude are summarized below: an increase in pitch (eg flat spin) will:

a. Decrease the rate of descent.

b. Decrease the spin radius.

c. Increase the spin rate.

It can also be shown that an increase in pitch will decrease the helix angle.

Angular Momentum

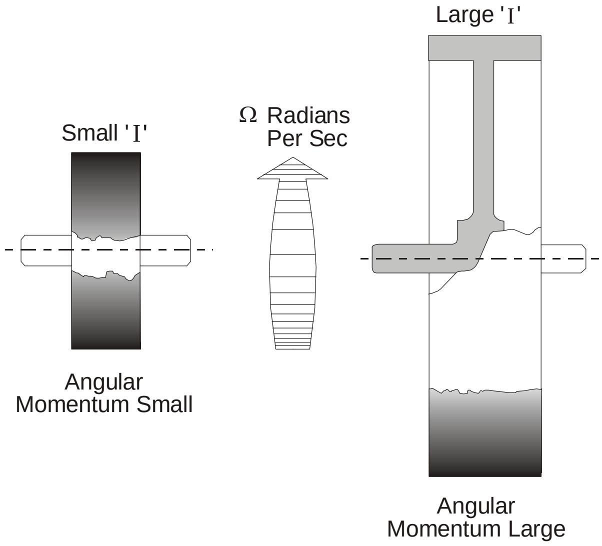
19. In a steady spin, equilibrium is achieved by a balance of aerodynamic and inertia moments. The inertia moments result from a change in angular momentum due to the inertia cross-coupling between the three axes.

20. The angular momentum about an axis depends on the distribution of mass and the rate of rotation. It is important to get a clear understanding of the significance of the spinning characteristics of different aircraft and the effect of controls in recovering from the spin.

Moment of Inertia (I)

21. A concept necessary to predict the behaviour of a rotating system is that of moment of inertia. This quantity not only expresses the amount of mass but also its distribution about the axis of rotation. It is used in the same way that mass is used in linear motion. For example, the product of mass and linear velocity measures the momentum or resistance to movement of a body moving in a straight line. Similarly, the product of moment of inertia (mass distribution) and angular velocity measures the angular momentum of a rotating body. Fig 3 illustrates how the distribution of mass affects angular momentum.

1-7 Fig 3 Two Rotors of the Same Weight and Angular Velocity



22. The concept of moment of inertia may be applied to an aircraft by measuring the distribution of mass about each of the body axes in the following way:

a. **Longitudinal Axis**. The distribution of mass about the longitudinal axis determines the moment of inertia in the rolling plane which is denoted by A. An aircraft with fuel stored in the wings and in external tanks will have a large value of A, particularly if the tanks are close to the wing tips. The tendency in modern high speed aircraft towards thinner wings has necessitated the stowage of fuel elsewhere and this, combined with lower aspect ratios, has resulted in a reduction in the value of A for those modern high performance fighter and training aircraft.

b. **Lateral Axis**. The distribution of mass about the lateral axis determines the moment of inertia in the pitching plane which is denoted by B. The increasing complexity of modern aircraft has resulted in an increase in the density of the fuselage with the mass being distributed along the whole length of the fuselage and a consequent increase in the value of B.

c. **Normal Axis**. The distribution of mass about the normal axis determines the moment of inertia in the yawing plane which is denoted by C. This quantity will be approximately equal to the sum of the moments of inertia in the rolling and pitching planes. C, therefore, will always be larger than A or B.

23. These moments of inertia measure the mass distribution about the body axes and are decided by the design of the aircraft. It will be seen that the values of A, B and C for a particular aircraft may be changed by altering the disposition of equipment, freight and fuel.

Inertia Moments in a Spin

24. The inertia moments generated in a spin are described below by assessing the effect of the concentrated masses involved. Another explanation, using a gyroscopic analogy, is given in Volume 1, Chapter 8.

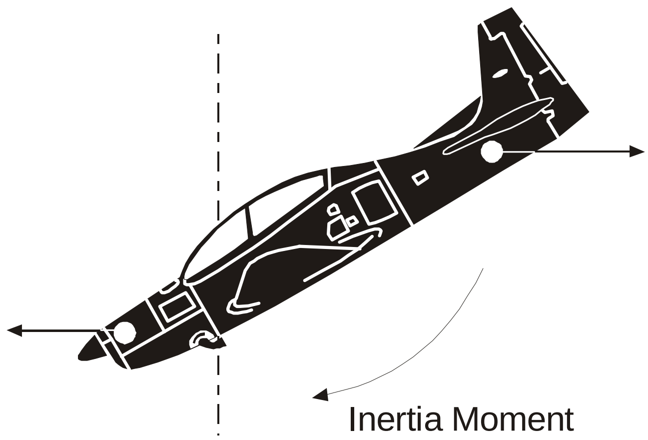
a. **Roll**. It is difficult to represent the rolling moments using concentrated masses, as is done for the other axes. For an aircraft in the spinning attitude under consideration (inner wing down pitching nose up), the inertia moment is anti-spin, ie tending to roll the aircraft out of the spin. The equation for the inertia rolling moment is:

$$L = -(C - B)rq$$

b. **Pitch**. The imaginary concentrated masses of the fuselage, as shown in Fig 4, tend to flatten the spin. The equation for the moment is:

$$M = (C - A)rp$$

1-7 Fig 4 Inertia Pitching Moment

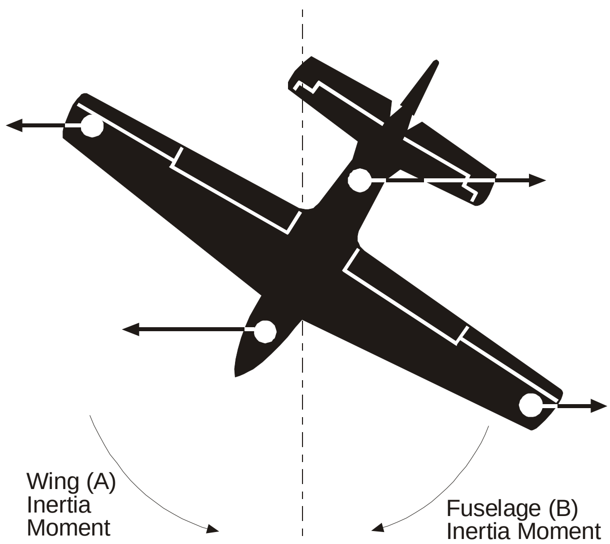


c. **Yaw**. The inertia couple is complicated by the fact that it comprises two opposing couples caused by the wings and the fuselage, see Fig 5. Depending on the dominant component, the couple can be of either sign and of varying magnitude. The inertia yawing moment can be expressed as:

$$N = (A - B)pq$$

25. This is negative and thus anti-spin when B > A; positive and pro-spin when A > B.

1-7 Fig 5 Inertia Yawing Moments



26. The B/A ratio has a profound effect on the spinning characteristics of an aircraft.

Aerodynamic Moments

27. It is now necessary to examine the contributions made by the aerodynamic factors in the balance of moments in roll, pitch and yaw. These are discussed separately below.

28. **Aerodynamic Rolling Moments**. The aerodynamic contributions to the balance of moments about the longitudinal axis to produce a steady rate of roll are as follows:

a. **Rolling Moment due to Sideslip**. The design features of the aircraft which contribute towards positive lateral stability produce an aerodynamic rolling moment as a result of sideslip. It can be shown that when an aircraft is in a spin, dihedral, sweepback, and high mounted wings all reverse the effect of positive lateral stability, and induce a rolling moment in the same sense as the sideslip. In a spin, the relative airflow is from the direction of the outer wing and, therefore, the aircraft will slideslip in the opposite direction to the spin. As a result, a rolling moment is produced in the opposite direction to the spin; this contribution is therefore anti-spin.

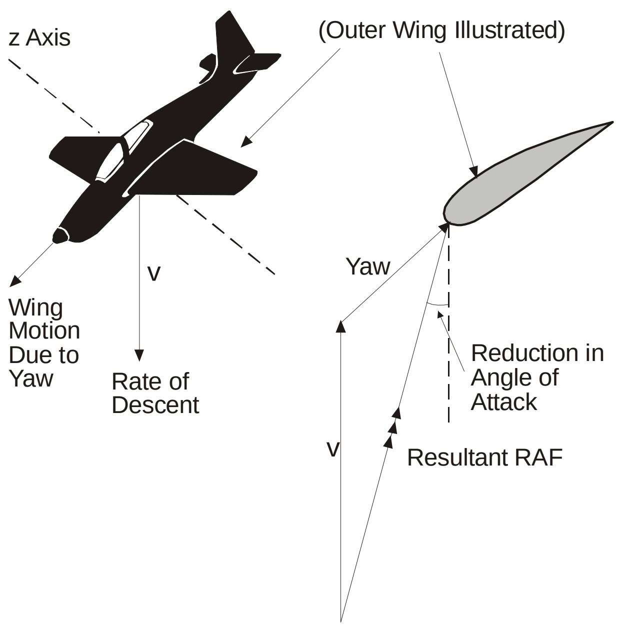
b. **Autorotative Rolling Moment.** In Volume 1, Chapter 11, it is shown that the normal damping in roll effect is reversed at angles of attack above the stall. This contribution is therefore pro-spin.

c. **Rolling Moment due to Yaw**. The yawing velocity in the spin induces a rolling moment for two reasons:

(1) **Difference in Speed of the Wings**. Lift of the outside wing is increased and that of the inner wing decreased inducing a pro-spin rolling moment.

(2) **Difference in Angle of Attack of the Wings.** In a spin, the direction of the free stream is practically vertical whereas the direction of the wing motion due to yaw is parallel to the longitudinal axis. The yawing velocity not only changes the speed but also the angle of attack of the wings. Fig 6 illustrates the vector addition of the yawing velocity to the vertical velocity of the outer wing. The effect is to reduce the angle of attack of the outer wing and increase that of the inner wing. Because the wings are stalled (slope of C_L curve is negative), the C_L of the outer wing is increased and the C_L of the inner wing decreased thus producing another pro-spin rolling moment.

1-7 Fig 6 Change in Angle of Attack due to Yaw (Outer Wing)



d. **Aileron Response**. Experience has shown that the ailerons produce a rolling moment in the conventional sense even though the wing is stalled.

29. **Aerodynamic Pitching Moments**. The aerodynamic contributions to the balance of moments about the lateral axis to produce a steady rate of pitch are as follows:

a. **Positive Longitudinal Static Stability**. In a spin the aircraft is at a high angle of attack and therefore disturbed in a nose-up sense from the trimmed condition. The positive longitudinal stability responds to this disturbance to produce a nose-down aerodynamic moment. This effect may be considerably reduced if the tailplane lies in the wing wake.

b. **Damping in Pitch Effect.** When the aircraft is pitching nose-up the tailplane is moving down and its angle of attack is increased (the principle is the same as the damping in roll effect). The pitching velocity therefore produces a pitching moment in a nose-down sense. The rate of pitch in a spin is usually very low and consequently the damping in pitch contribution is small.

c. **Elevator Response.** The elevators act in the conventional sense. Down-elevator increases the nose-down aerodynamic moment whereas up-elevator produces a nose-up aerodynamic moment. It should be noted, however, that down-elevator usually increases the shielded area of the fin and rudder.

30. **Aerodynamic Yawing Moments**. The overall aerodynamic yawing moment is made up of a large number of separate parts, some arising out of the yawing motion of the aircraft and some arising out of the sideslipping motion. The main contributions to the balance of moments about the normal axis to produce a steady rate of yaw are as follows:

a. **Positive Directional Static Stability**. When sideslip is present keel surfaces aft of the CG produce an aerodynamic yawing moment tending to turn the aircraft into line with the sideslip vector (ie directional static stability or 'weathercock effect'). This is an anti-spin effect, the greatest contribution to which is from the vertical fin. Vertical surfaces forward of the CG will tend to yaw the aircraft further into the spin, ie they have a pro-spin effect. In a spin outward sideslip is present which, usually produces a net yawing moment towards the outer wing; ie in an anti-spin sense. Because of possible shielding effects from the tailplane and elevator and also because the fin may be stalled, the directional stability is considerably reduced and this anti-spin contribution is usually small.

b. **Damping in Yaw Effect**. Applying the principle of the damping in roll effect to the yawing velocity, it has been seen that the keel surfaces produce an aerodynamic yawing moment to oppose the yaw. The greatest contribution to this damping moment is from the rear fuselage and fin. In this respect the cross-sectional shape of the fuselage is critical and has a profound effect on the damping moment. The following figures give some indication of the importance of cross-section:

Cross-Section	**Damping Effect (anti-spin)**
Circular	1
Rectangular	2.5
Elliptical	3.5
Round top/flat bottom	1.8
Round bottom/flat top	4.2
Round bottom/flat top with strakes	5.8

$$\text{Damping effect} = \frac{\text{Damping from body of given cross-section}}{\text{Damping from circular cyclinder}}$$

Fuselage strakes are useful devices for improving the spinning characteristics of prototype aircraft. The anti-spin damping moment is very dependent on the design of the tailplane/fin combination. Shielding of the fin by the tailplane can considerably reduce the effectiveness of the fin. In extreme cases a low-set tailplane may even change the anti-spin effect into pro-spin.

c. **Rudder Response**. The rudder acts in the conventional sense, ie the in-spin rudder produces pro-spin yawing moment and out-spin rudder produces anti-spin yawing moment. Because of the shielding effect of the elevator (para 29c), it is usual during recovery to pause after applying out-spin rudder so that the anti-spin yawing moment may take effect before down-elevator is applied.

Balance of Moments

31. In para 16 it was seen that the balance of forces in the spin has a strong influence on the rate of descent. It does not, however, determine the rate of rotation, wing tilt or incidence at which the spin occurs: the balance of moments is much more critical in this respect. The actual attitude, rate of descent, sideslip, rate of rotation and radius of a spinning aircraft can only be determined by applying specific numerical values of the aircraft's aerodynamic and inertia data to the general relationships discussed below.

32. **Rolling Moments**. The balance of rolling moments in an erect spin is:

a. **Pro-spin**: The following aerodynamic rolling moments in an erect spin is:

(1) Autorotative rolling moment.

(2) Rolling moment due to sideslip.

(3) Rolling moment due to yaw.

b. **Anti-spin**. The inertia rolling moment, $-(C-B)rq$ is anti-spin. These factors show that autorotation is usually necessary to achieve a stable spin. A small autorotative rolling moment would necessitate larger sideslip to increase the effect of rolling moment due to sideslip. This, in turn, would reduce the amount of wing tilt and make the balance of moments in yaw more difficult to achieve, however the balance of moments in this axis is not as important as in the other two.

33. **Pitching Moments**. In para 25 it was seen that the inertia pitching moment, (C – A) rp, of the aircraft is always nose-up in an erect spin. This is balanced by the nose-down aerodynamic pitching moment. The balance between these two moments is the main factor relating angle of attack to rate of rotation in any given case and equilibrium can usually be achieved over a wide range. It can be shown that an increase in pitch will cause an increase in the rate of rotation (spin rate). This, in turn, will decrease the spin radius (para 16).

34. **Yawing Moments.** The balance of yawing moments in an erect spin is:

a. **Pro-spin.**

(1) Yawing moment due to applied rudder.

(2) A small contribution from the wing, due to yaw, is possible at large angles of attack.

(3) Yawing moment due to sideslip (vertical surfaces forward of CG).

(4) Inertia yawing moment, (A – B)pq, if A > B.

b. **Anti-spin**.

(1) Inertia yawing moment, (A – B)pq, if B > A.

(2) Yawing moment due to sideslip (vertical surfaces aft of the CG).

(3) Damping in yaw effect.

It can be seen that in-spin rudder is usually necessary to achieve balance of the yawing moments and hold the aircraft in a spin.

35. **Normal Axis.** For conventional aircraft (A and B nearly equal), it is relatively easy to achieve balance about the normal axis and the spin tends to be limited to a single set of conditions (angle of attack, spin rate, attitude). For aircraft in which B is much larger than A, the inertia yawing moment can be large and, thus difficult to balance. This is probably the cause of the oscillatory spin exhibited by these types of aircraft.

36. **Yaw and Roll Axis**. The requirements of balance about the yaw and roll axes greatly limit the range of incidences in which spinning can occur and determine the amount of sideslip and wing tilt involved. The final balance of the yawing moments is achieved by the aircraft taking up the appropriate angle of attack at which the inertia moments just balance the aerodynamic moments. This particular angle of attack also has to be associated with the appropriate rate of spin required to balance the pitching moments and the appropriate angle of sideslip required to balance the rolling moments.

SPIN RECOVERY

Effect of Controls in Recovery from a Spin

37. The relative effectiveness of the three controls in recovering from a spin will now be considered. Recovery is aimed at stopping the rotation by reducing the pro-spin rolling moment and/or increasing the anti-spin yawing moment. The yawing moment is the more important but, because of the strong cross-coupling between motions about the three axes through the inertia moments, the rudder is not the only means by which yawing may be induced by the pilot. Once the rotation has stopped the angle of attack is reduced and the aircraft recovered.

38. The control movements which experience has shown are generally most favourable to the recovery from the spin have been known and in use for a long time, ie apply full opposite rudder and then move the stick forward until the spin stops, maintaining the ailerons neutral. The rudder is normally the primary control but, because the inertia moments are generally large in modern aircraft, aileron deflection is also important. Where the response of the aircraft to rudder is reduced in the spin the aileron may even be the primary control although in the final analysis it is its effect on the yawing moment which makes it work.

39. The initial effect of applying a control deflection will be to change the aerodynamic moment about one or more axes. This will cause a change in aircraft attitude and a change in the rates of rotation about all the axes. These changes will, in turn, change the inertia moments.

Effect of Ailerons

40. Even at the high angle of attack in the spin the ailerons act in the normal sense. Application of aileron in the same direction as the aircraft is rolling will therefore increase the aerodynamic rolling moment. This will increase the roll rate (p) and affect the inertia yawing moment, (A – B)pq. The effect of an increase in p on the inertia yawing moment depends on the mass distribution or B/A ratio:

a. **B/A > 1.** In an aircraft where B/A > 1, the inertia yawing moment is anti-spin (negative) and an increase in p will decrease it still further, ie make it more anti-spin. The increase in anti-spin inertia yawing moment will tend to raise the outer wing (increase wing tilt) which will decrease the outward sideslip. This will restore the balance of rolling moments by decreasing the pro-spin aerodynamic moment due to lateral stability. The increase in wing tilt will also cause the rate of pitch, q, to increase, which, in turn:

(1) Causes a small increase in the anti-spin inertia rolling moment, – (C – B)rq, (C > B) and thus helps to restore balance about the roll axis (para 32).

(2) Further increases the anti-spin inertia yawing moment.

b. **B/A < 1**. A low B/A ratio will reverse the effects described above. The inertia yawing moment will be pro-spin (positive) and will increase with an increase in p.

41. Due to secondary effects associated with directional stability, the reversal point actually occurs at a B/A ratio of 1.3. Thus:

a. **B/A > 1.3.** Aileron with roll (in-spin) has an anti-spin effect.

b. **B/A < 1.3**. Aileron with roll (in-spin) has a pro-spin effect.

42. Some aircraft change their B/A ratio in flight as stores and fuel are consumed. The pilot has no accurate indication of the value of B/A ratio and, where this value may vary either side of 1.3, it is desirable to maintain ailerons neutral to avoid an unfavourable response which may delay or even prohibit recovery.

43. An additional effect of aileron applied with roll is to increase the anti-spin yawing moments due to aileron drag.

Effect of Elevators

44. In para 29 it was seen that down-elevator produces a nose-down aerodynamic pitching moment. This will initially reduce the nose-up pitching velocity (q). Although this will tend to reduce alpha, the effect on the inertia yawing and rolling moments is as follows:

a. **Inertia Yawing Moment (A – B)pq**. If B > A, the inertia yawing moment is anti-spin. A reduction in q will make the inertia yawing moment less anti-spin, ie a pro-spin change. When A > B, however, down-elevator will cause a change in inertia yawing moment in the anti-spin sense.

b. **Inertia Rolling Moment – (C – B)rq.** The inertia rolling moment is always anti-spin because C > B. A reduction in q will therefore make it less anti-spin which is again a change in the pro-spin sense.

The result of these pro-spin changes in the inertia yawing and rolling moments is to decrease the wing tilt thus increasing the sideslip angle (Fig 1) and rate of roll. It can also be shown that the rate of rotation about the spin axis will increase.

45. Although the change in the inertia yawing moment is unfavourable, the increased sideslip may produce an anti-spin aerodynamic yawing moment if the directional stability is positive. This contribution will be reduced if the down elevator seriously increases the shielding of the fin and rudder.

46. The overall effect of down-elevator on the yawing moments therefore depends on:

a. The pro-spin inertia moment when B > A.

b. The anti-spin moment due to directional stability.

c. The loss of rudder effectiveness due to shielding.

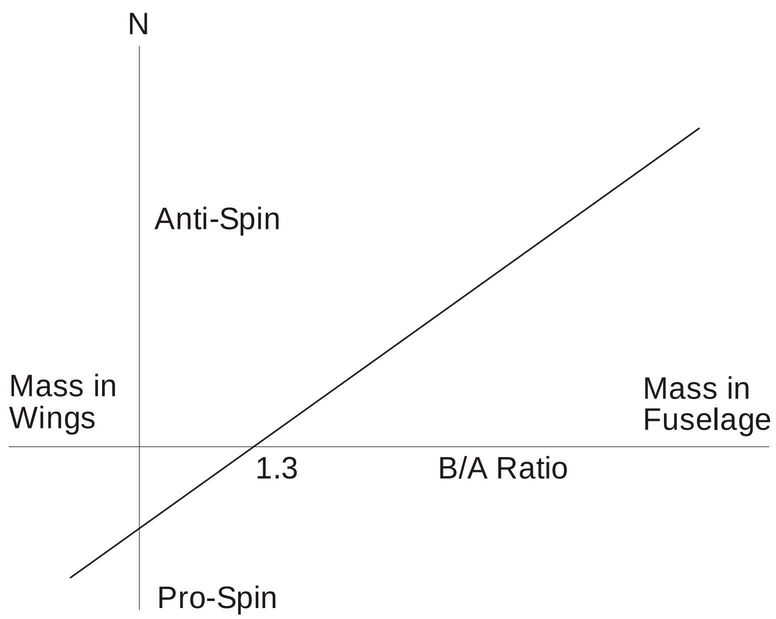
In general, the net result of moving the elevators down is beneficial when A > B and rather less so when B > A, assuming that the elevator movement does not significantly increase the shielding of the fin and rudder.

Effect of Rudder

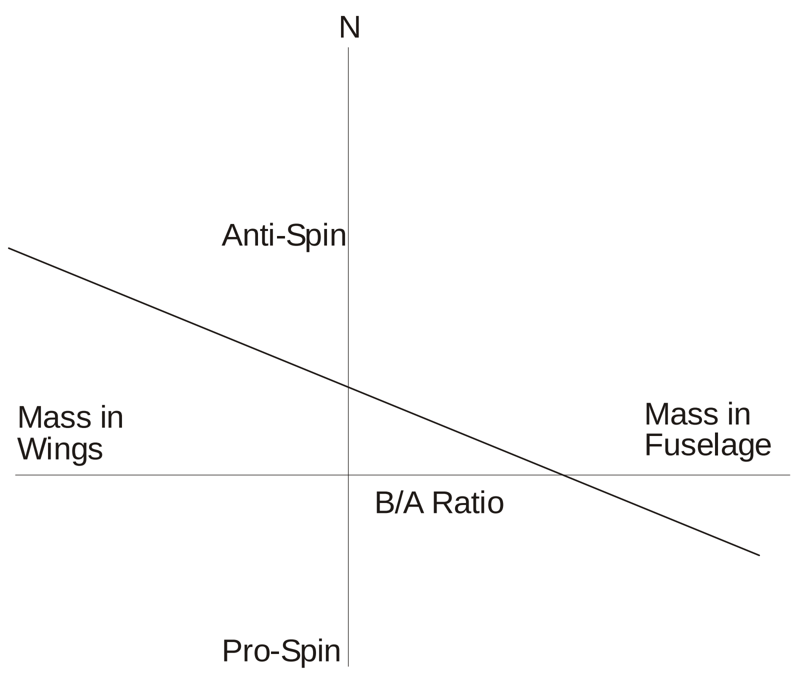
47. The rudder is nearly always effective in producing an anti-spin aerodynamic yawing moment though the effectiveness may be greatly reduced when the rudder lies in the wake of the wing or tailplane. The resulting increase in the wing tilt angle will increase the anti-spin inertia yawing moment (when B > A) through an increase in pitching velocity. The overall effect of applying anti-spin rudder is always beneficial and is enhanced when the B/A ratio is increased.

48. The effect of the three controls on the yawing moment is illustrated in Figs 7, 8 and 9.

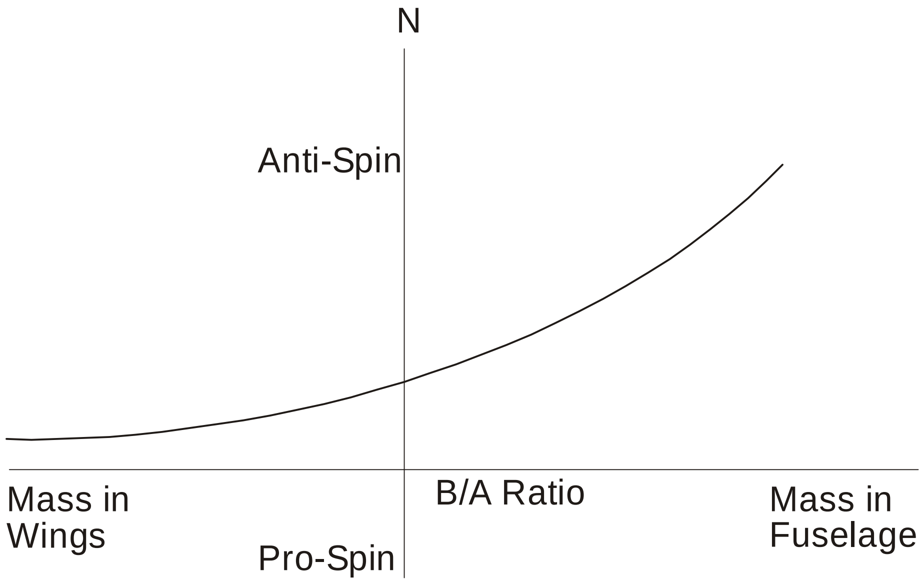
1-7 Fig 7 Yawing Moment (N) per degree of Aileron



1-7 Fig 8 Yawing Moment (N) per Degree of Down Elevator



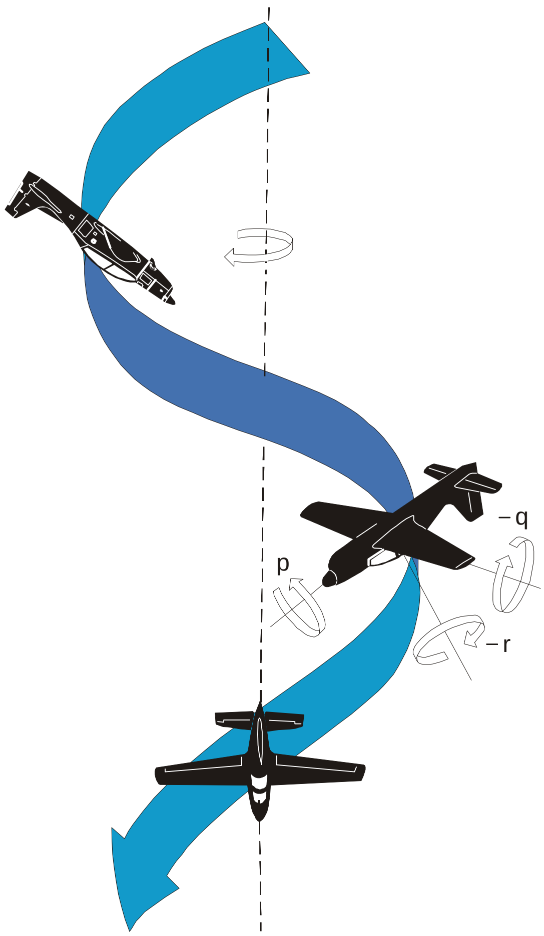
1-7 Fig 9 Yawing Moment (N) per Degree of Anti-spin Rudder



Inverted Spin

49. Fig 10 shows an aircraft in an inverted spin but following the same flight path as in Fig 1. Relative to the pilot the motion is now compounded of a pitching velocity in the nose-down sense, a rolling velocity to the right and a yawing velocity to the left. Thus roll and yaw are in opposite directions, a fact which affects the recovery actions, particularly if the aircraft has a high B/A ratio.

1-7 Fig 10 The Inverted Spin



50. The inverted spin is fundamentally similar to the erect spin and the principles of moment balance discussed in previous paragraphs are equally valid for the inverted spin. The values of the aerodynamic moments, however, are unlikely to be the same since, in the inverted attitude, the shielding effect of the wing and tail may change markedly.

51. The main difference will be caused by the change in relative positions of the fin and rudder and the tailplane. An aircraft with a low-mounted tailplane will tend to have a flatter erect spin and recovery will be made more difficult due to shielding of the rudder. The same aircraft inverted will respond much better to recovery rudder since it is unshielded and the effectiveness of the rudder increased by the position of the tailplane. The converse is true for an aircraft with a high tailplane.

52. The control deflections required for recovery are dictated by the direction of roll, pitch and yaw, and the aircraft's B/A ratio. These are:

a. Rudder to oppose yaw as indicated by the turn needle.

b. Aileron in the same direction as the observed roll, if the B/A ratio is high.

c. Elevator up is generally the case for conventional aircraft but, if the aircraft has a high B/A ratio and suffers from the shielding problems previously discussed, this control may be less favourable and may even become pro-spin.

Oscillatory Spin

53. A combination of high wing loading and high B/A ratio makes it difficult for a spinning aircraft to achieve equilibrium about the yaw axis. This is thought to be the most probable reason for the oscillatory spin. In this type of spin the rates of roll and pitch are changing during each oscillation. In a mild form it appears to the pilot as a continuously changing angle of wing tilt, from outer wing well above the horizon back to the horizontal once each turn; the aircraft seems to wallow in the spin.

54. In a fully-developed oscillatory spin the oscillations in the rates of roll and pitch can be quite violent. The rate of roll during each turn can vary from zero to about 200 degrees per second. At the maximum rate of roll the rising wing is unstalled which probably accounts for the violence of this type of spin. Large changes in attitude usually take place from fully nose-down at the peak rate of roll, to nose-up at the minimum rate of roll.

55. The use of the controls to effect a change in attitude can change the characteristics of an oscillatory spin quite markedly. In particular:

a. Anything which increases the wing tilt will increase the violence of the oscillations, eg in-spin aileron or anti-spin rudder.

b. A decrease in the wing tilt angle will reduce the violence of the oscillations, eg out-spin aileron or down-elevator.

The recovery from this type of spin has been found to be relatively easy, although the shortest recovery times are obtained if recovery is initiated when the nose of the aircraft is falling relative to the horizon.

Conclusions

56. The foregoing paragraphs make it quite clear that the characteristics of the spin and the effect of controls in recovery are specific to type. In general the aerodynamic factors are determined by the geometry of the aircraft and the inertial factors by the distribution of the mass.

57. The aspects of airmanship applicable to the spin manoeuvre are discussed in detail in Volume 8, Chapter 15, but in the final analysis the only correct recovery procedure is laid down in the Aircrew Manual for the specific aircraft.

CHAPTER 8- GYROSCOPIC CROSS-COUPLING BETWEEN AXES

Introduction

1. In the preceding chapter, the effects of the inertia moments have been explained by considering the masses of fuselage and wings acting either side of a centreline. The effect of these concentrated masses when rotating can be visualized as acting rather in the manner of the bob-weights of a governor.

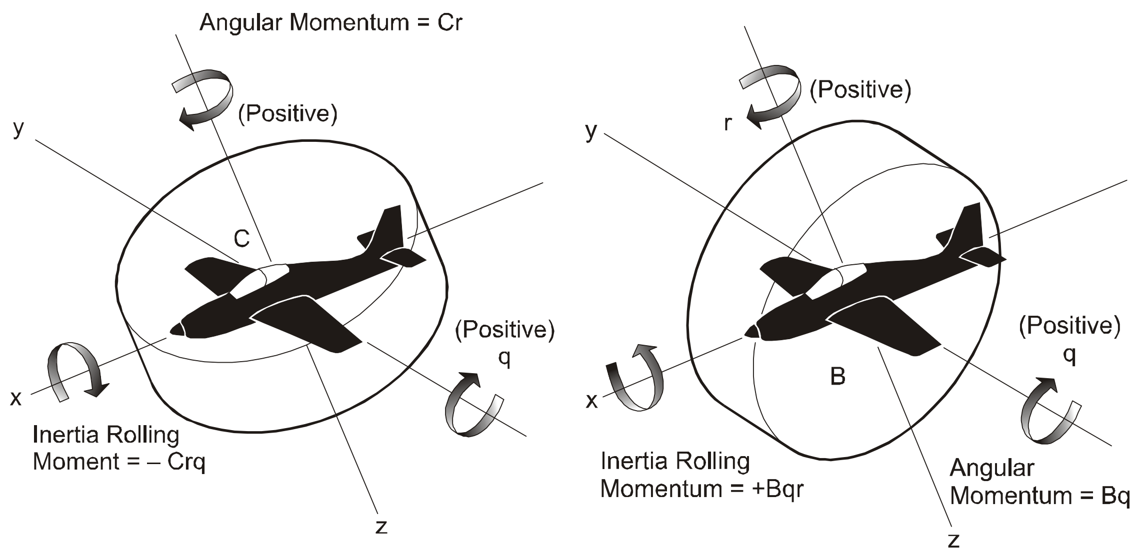
2. Another, and more versatile, explanation of the cross-coupling effects can be made using a gyroscopic analogy, regarding the aircraft as a rotor.

Inertia Moments in a Spin

3. The inertia moments generated in a spin are essentially the same as the torque exerted by a precessing gyroscope. Figs 1, 2 and 3 illustrate the inertia or gyroscopic moments about the body axes. These effects are described as follows:

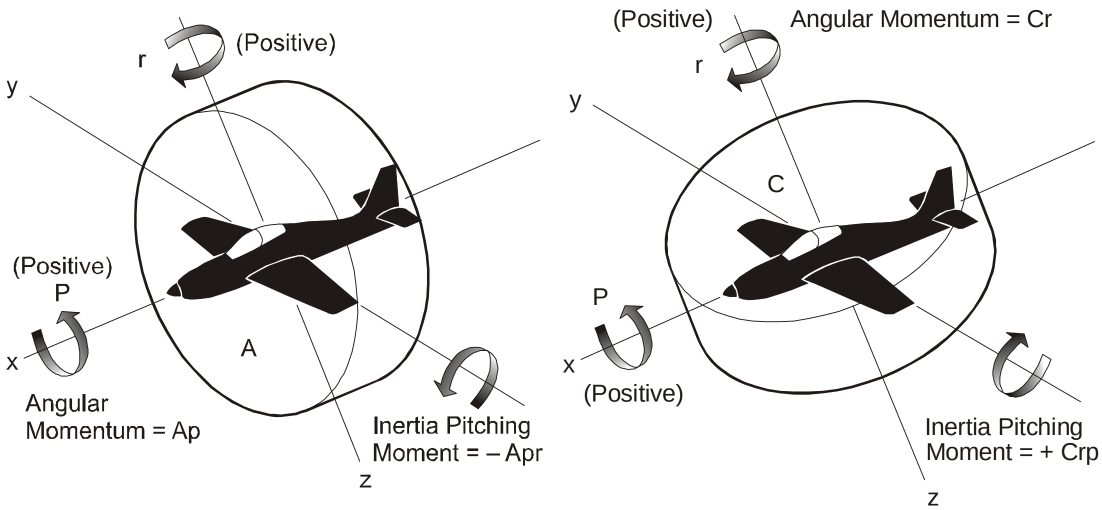
a. **Inertia Rolling Moments (Fig 1)**. The angular momentum in the yawing plane is Cr, and by imposing on it a pitching velocity of q, an inertia rolling moment is generated equal to –Crq, ie in the opposite sense to the direction of roll in an erect spin. The inertia rolling moment due to imposing the yawing velocity on the angular momentum in the pitching plane is in a pro-spin sense equal to +Brq. The total inertia rolling moment is therefore equal to (B – C)rq, or since C > B: – (C – B) rq.

1-8 Fig 1 Total Inertia Rolling Moment



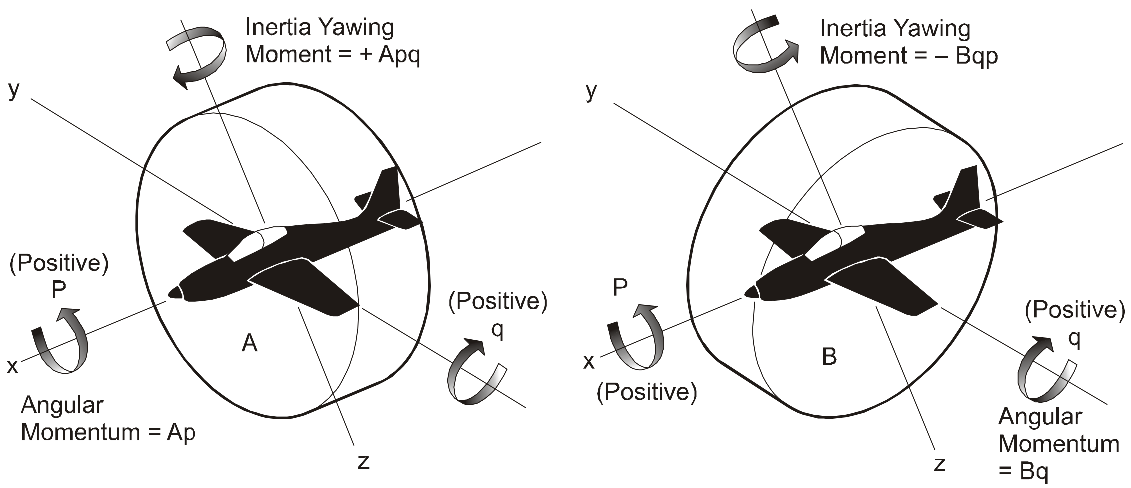
b. **Inertia Pitching Moments (Fig 2)**. The angular momentum in the rolling plane is Ap and imposing a yawing velocity of r on the rolling plane rotor causes it to precess in pitch in a nose-down sense due to inertia pitching moment (–Apr). Similarly, the angular momentum in the yawing plane is Cr, and imposing a roll velocity of p on the yawing plane rotor generates an inertia pitching moment (+Crp) in the nose-up sense. The total inertia moment is therefore (C – A)rp. In an erect spin, roll and yaw are always in the same direction and C is always greater than A. The inertia pitching moment is therefore positive (Nose-up) in an erect spin.

1-8 Fig 2 Total Inertia Pitching Moment



c. **Inertia Yawing Moments (Fig 3)**. Replacing the aircraft by a rotor having the same moment of inertia in the rolling plane, its angular momentum is the product of the moment of inertia and angular velocity (Ap). Imposing a pitching velocity (q) on the rotor will generate a torque tending to precess the rotor about the normal axis in the same direction as the spin. It can be shown that this inertia yawing moment is equal in value to +Apq where the positive sign indicates a pro-sign torque. Similarly, the angular momentum in the pitching plane is equal to Bq and imposing a roll velocity of p on the pitching plane rotor will generate an inertia yawing moment in an anti-spin sense equal to –Bpq. The total inertia yawing moment is therefore equal to (A – B)pq, or, if B > A, then – (B – A)pq.

1-8 Fig 3 Total Inertia Yawing Moment



CHAPTER 9 - WING PLANFORMS

Introduction

1. The preceding chapters discussed the basic considerations of lift, drag, stalling and spinning and explained the causes of these phenomena. However, it is now necessary to examine another important aspect of the design of wings, ie the planform. The planform is the geometrical shape of the wing when viewed from above; it largely determines the amount of lift and drag obtainable from a stated wing area and has a pronounced effect on the value of the stalling angle of attack.

2. This chapter is concerned mainly with the low-speed effects of variations in wing planforms. The high-speed effects are dealt with in the chapters on high-speed flight.

Aspect Ratio

3. The aspect ratio (A) of a wing is found by dividing the square of the wingspan by the area of the wing:

$$A = \frac{\text{Span}^2}{\text{Area}}$$

Thus, if a wing has an area of 250 square feet and a span of 30 feet, the aspect ratio is 3.6. Another wing with the same span but with an area of 150 square feet would have an aspect ratio of 6. Another method of determining the aspect ratio is by dividing the span by the mean chord of the wing. For example, a span of 50 ft with a mean chord of 5 ft gives an aspect ratio of 10.

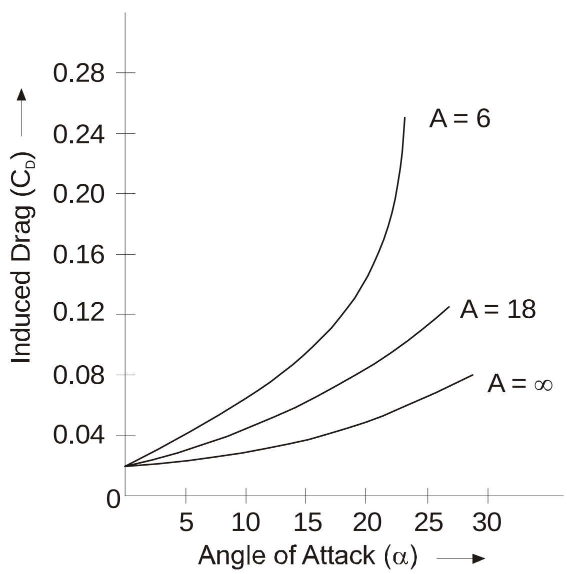
4. From the preceding examples it can be seen that the smaller the area or mean chord in relation to the span, the higher is the aspect ratio. A rough idea of the performance of a wing can be obtained from knowledge of the aspect ratio.

Aspect Ratio and Induced (Vortex) Drag

5. The origin and formation of trailing edge and wing tip vortices was explained in Volume 1, Chapter 5 where it was shown that the induced downwash was the cause of induced drag. The downwash imparted to the air is a measure of the lift provided by a wing.

6. Induced drag is inversely proportional to aspect ratio. A graph showing the curves of three different aspect ratio wings plotted against C_D and angle of attack can be seen in Fig 1.

1-9 Fig 1 Effect of Aspect Ratio on C_D



Aspect Ratio and Stalling Angle

7. A stall occurs when the effective angle of attack reaches the critical angle. As has been shown in Volume 1, Chapter 5 induced downwash reduces the effective angle of attack of a wing. Since induced drag is inversely proportional to aspect ratio it follows that a low aspect ratio wing will have high induced drag, high induced downwash and a reduced effective angle of attack. The low aspect ratio wing therefore has a higher stalling angle of attack than a wing of high aspect ratio.

8. The reduced effective angle of attack of very low aspect ratio wings can delay the stall considerably. Some delta wings have no measurable stalling angle up to 40º or more inclination to the flight path. At this sort of angle, the drag is so high that the flight path is usually inclined downwards at a steep angle to the horizontal. Apart from a rapid rate of descent, and possible loss of stability and control, such aircraft may have a shallow attitude to the horizon and this can be deceptive. The condition is called the 'super stall' or deep stall, although the wing may be far from a true stall and still be generating appreciable lift.

Use of High Aspect Ratio

9. Aircraft types such as sailplanes, transport, patrol and anti-submarine demand a high aspect ratio to minimize the induced drag (high performance sailplanes often have aspect ratios between 25 and 30).

10. While the high aspect ratio wing will minimize induced drag, long thin wings increase weight and have relatively poor stiffness characteristics. Also, the effects of vertical gusts on the airframe are aggravated by increasing the aspect ratio. Broadly it can be said that the lower the cruising speed of the aircraft, the higher the aspect ratios that can be usefully employed. Aircraft configurations which are developed for very high-speed flight (especially supersonic flight) operate at relatively low lift coefficients and demand great aerodynamic cleanness. This usually results in the development of low aspect ratio planforms.

The Effects of Taper

11. The aspect ratio of a wing is the primary factor in determining the three-dimensional characteristics of the ordinary wing and its drag due to lift. However, certain local effects take place throughout the span of the wing and these effects are due to the distribution of area throughout the span. The typical lift distribution is arranged in some elliptical fashion.

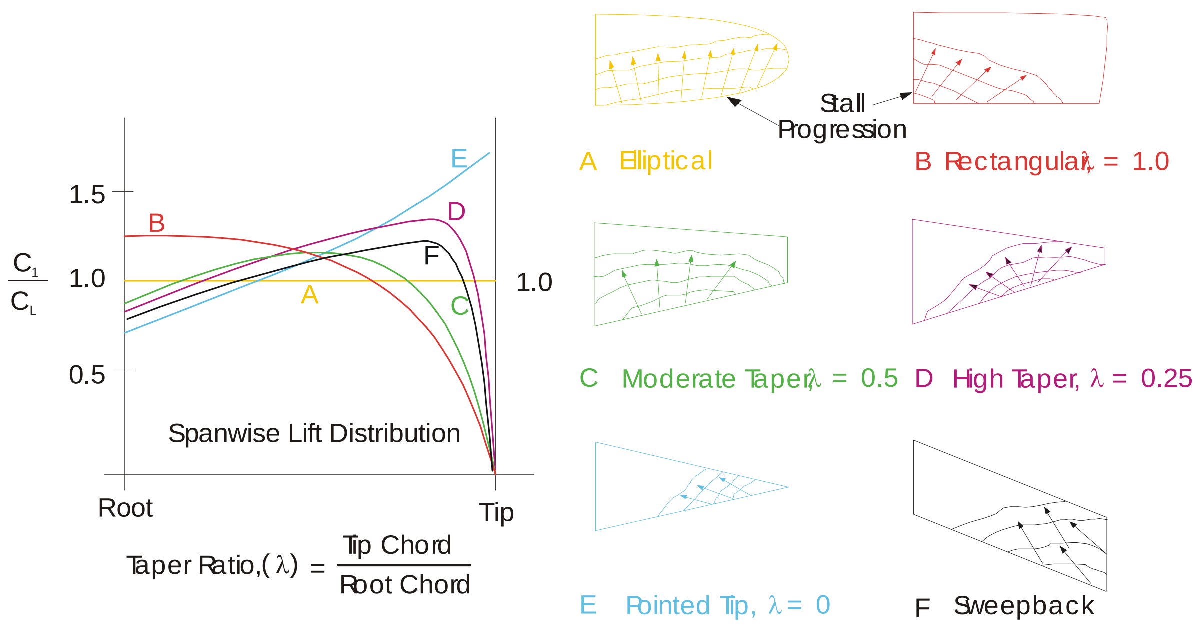
12. The natural distribution of lift along the span of the wing provides a basis for appreciating the effect of area distribution and taper along the span. If the elliptical lift distribution is matched with a planform whose chord is distributed in an elliptical fashion (the elliptical wing), each square foot of area along the span produces exactly the same lift pressure. The elliptical wing planform then has each section of the wing working at exactly the same local lift coefficient and the induced downflow at the wing is uniform throughout the span. In the aerodynamic sense, the elliptical wing is the most efficient planform because the uniformity of lift coefficient and downwash incurs the least induced drag for a given aspect ratio. The merit of any wing planform is then measured by the closeness with which the distribution of lift coefficient and downwash approach that of the elliptical planform.

13. The effect of the elliptical planform is illustrated in Fig 2 by the plot of local lift coefficient (C_1) to wing lift coefficient (C_L), $\frac{C_1}{C_L}$ against semi-span distance. The elliptical wing produces a constant

value of $\frac{C_1}{C_L}$ = 1.0 throughout the span from root to tip. Thus, the local section angle of attack (α_0), and local induced angle of attack (α_1), are constant throughout the span. If the planform area distribution is anything other than elliptical it may be expected that the local section and induced angles of attack will not be constant along the span.

14. A planform previously considered is the simple rectangular wing which has a taper ratio of 1.0. A characteristic of the rectangular wing is a strong vortex at the tip with local downwash behind the wing which is high at the tip and low at the root. This large non-uniformity in downwash causes similar variation in the local induced angles of attack along the span. At the tip, where high downwash exists, the local induced angle of attack is greater than the average for the wing. Since the wing angle of attack is composed of the sum of α_1 and α_0, a large, local α_1 reduces the local α_0 creating low local lift coefficients at the tip. The reverse is true at the root of the rectangular wing where low local downwash exists. This situation creates an induced angle of attack at the root which is less than the average for the wing and a local section angle of attack higher than the average for the wing. The result is shown by the graph in Fig 2 which depicts a local coefficient at the root almost 20% greater than the wing lift coefficient.

1-9 Fig 2 Lift Distribution and Stall Patterns



15. The effect of the rectangular planform may be appreciated by matching a near elliptical lift distribution with a planform with a constant chord. The chords near the tip develop less lift pressure than the root and consequently have lower section lift coefficients. The great non-uniformity of local lift coefficient along the span implies that some sections carry more than their share of the load while others carry less. Hence, for a given aspect ratio, the rectangular planform will be less efficient than the elliptical wing. For example, a rectangular wing of A = 6 would have 16% higher induced angle of attack for the wing and 5% higher induced drag than an elliptical wing of the same aspect ratio.

16. At the other extreme of taper is the pointed wing, which has a taper ratio of zero. The extremely small area at the pointed tip is not capable of holding the main tip vortex at the tip and a drastic change in downwash distribution results. The pointed wing has greatest downwash at the root and this

downwash decreases toward the tip. In the immediate vicinity of the pointed tip an upwash is encountered which indicates that negative induced angles of attack exist in this area. The resulting variation of local lift coefficient shows low C_1 at the root and very high C_1 at the tip. The effect may be appreciated by realizing that the wide chords at the root produce low lift pressures while the very narrow chords towards the tip are subject to very high lift pressures. The variation of $\frac{C_1}{C_L}$ throughout the span of the wing of taper ratio = 0 is shown on the graph of Fig 2. As with the rectangular wing, the non-uniformity of downwash and lift distribution result in inefficiency of this planform. For example, a pointed wing of A = 6 would have 17% higher induced angle of attack for the wing and 13% higher induced drag than an elliptical wing of the same aspect ratio.

17. Between the two extremes of taper will exist planforms of more tolerable efficiency. The variations of $\frac{C_1}{C_L}$ for a wing of taper ratio 0.5 are similar to the lift distribution of the elliptical wing and the drag due to lift characteristics is nearly identical. A wing of A = 6 and taper ratio = 0.5 has only 3% higher α_1 and 1% greater C_{D1} than an elliptical wing of the same aspect ratio.

18. The elliptical wing is the ideal of the subsonic aerodynamic planform since it provides a minimum of induced drag for a given aspect ratio. However, the major objection to the elliptical planform is the extreme difficulty of mechanical layout and construction. A highly tapered planform is desirable from the standpoint of structural weight and stiffness and the usual wing planform may have a taper ratio from 0.45 to 0.20. Since structural considerations are important in the development of an aeroplane, the tapered planform is a necessity for an efficient configuration. In order to preserve the aerodynamic efficiency, the resulting planform is tailored by wing twist and section variation to obtain as near as possible the elliptic lift distribution.

Stall Patterns

19. An additional effect of the planform area distribution is on the stall pattern of the wing. The desirable stall pattern of any wing is a stall which begins at the root sections first. The advantages of the root stall first are that ailerons remain effective at high angles of attack, favourable stall warning results from the buffet on the tailplane and aft portion of the fuselage, and the loss of downwash behind the root usually provides a stable nose-down moment to the aircraft. Such a stall pattern is favoured but may be difficult to obtain with certain wing configurations. The types of stall patterns inherent with various planforms are illustrated in Fig 2. The various planform effects are separated as follows:

a. The elliptical planform has constant lift coefficients throughout the span from root to tip. Such a lift distribution means that all sections will reach stall at essentially the same wing angle of attack and the stall will begin and progress uniformly throughout the span. While the elliptical wing would reach high lift coefficients before an incipient stall, there would be little advance warning of a complete stall. Also, the ailerons may lack effectiveness when the wing operates near the stall and lateral control may be difficult.

b. The lift distribution of the rectangular wing exhibits low local lift coefficients at the tip, and high local lift coefficients at the root. Since the wing will initiate the stall in the area of highest local lift coefficients, the rectangular wing is characterized by a strong root stall tendency. Of course, this stall pattern is favourable since there is adequate stall warning buffet, adequate aileron effectiveness, and usually strong stable moment changes on the aircraft. Because of the great aerodynamic and structural inefficiency of this planform, the rectangular wing finds limited application only to low cost, low speed, light planes.

c. The wing of moderate taper (taper ratio = 0.5) has a lift distribution which is similar to that of the elliptical wing. Hence the stall pattern is much the same as the elliptical wing.

d. The highly tapered wing of taper ratio = 0.25 shows the stalling tendency inherent with high taper. The lift distribution of such a wing has distinct peaks just inboard from the tip. Since the wing stall is started in the vicinity of the highest local lift coefficient, this planform has a strong 'tip stall' tendency. The initial stall is not started at the exact tip but at the station inboard from the tip where the highest local lift coefficients prevail.

e. The pointed tip wing of taper ratio equal to zero develops extremely high local lift coefficients at the tip. For all practical purposes the pointed tip will be stalled at any condition of lift unless extensive tailoring is applied to the wing. Such a planform has no practical application to a subsonic aircraft.

f. The effect of sweepback on the lift distribution of a wing is similar to the effect of reducing the taper ratio. The full significance of sweepback is discussed in the following paragraphs.

SWEEPBACK

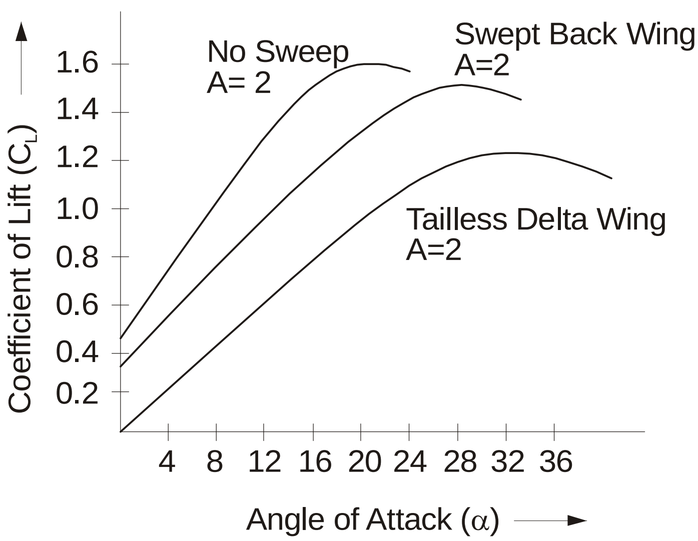
Swept-back Leading Edges

20. This type of planform is used on high-speed aircraft and may take the form of a sweptback wing, or a delta, with or without a tailplane. The reason for the use of these planforms is their low drag at the higher speeds. Volume 1, Chapter 22 deals fully with this aspect. However, the high speed/low drag advantages are gained at the cost of a poorer performance at the lower end of the speed range.

Effect of Sweepback on Lift

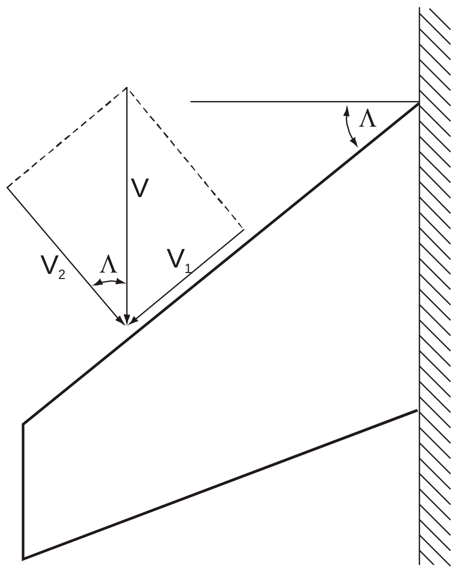
21. If a straight wing is changed to a swept planform, with similar parameters of area, aspect ratio, taper, section and washout, the C_{Lmax} is reduced. This is due to premature flow separation from the upper surface at the wing tips. For a sweep angle of 45º, the approximate reduction in C_{Lmax} is around 30%. Fig 3 shows typical C_L curves for a straight wing, a simple swept-back wing, and a tailless delta wing of the same low aspect ratio.

1-9 Fig 3 Effect of Planform on $C_{L\,max}$

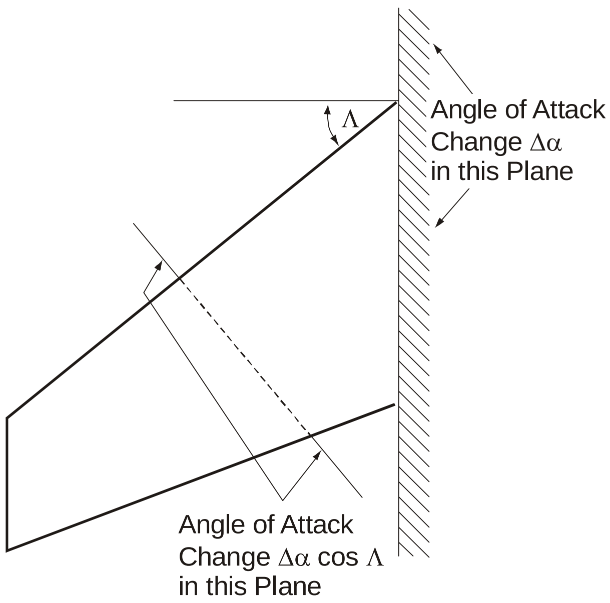


22. A swept wing presents less camber and a greater fineness ratio to the airflow. However, the reasons for the lowering of the C_L slope are more readily apparent from an examination of Figs 4 and 5. From Fig 4 it can be seen that the velocity V can be divided into two components, V_1 parallel to the leading edge which has no effect on the lift, and V_2 normal to the leading edge which does affect the lift and is equal to V cos Λ. Therefore, all other factors being equal, the C_L of a swept wing is reduced in the ratio of the cosine of the sweep angle.

1-9 Fig 4 Flow Velocities on a Swept Wing



1-9 Fig 5 Effect of Change in Angle of Attack



23. Fig 5 shows that an increase in fuselage angle of attack $\Delta\alpha$ will only produce an increase in the angle of attack $\Delta\alpha$ cos Λ in the plane perpendicular to the wing quarter chord line. Since we have already said that it is airflow in the latter plane which effects C_L, the full increment of lift expected from the $\Delta\alpha$ change is reduced to that of a $\Delta\alpha$ cos Λ change.

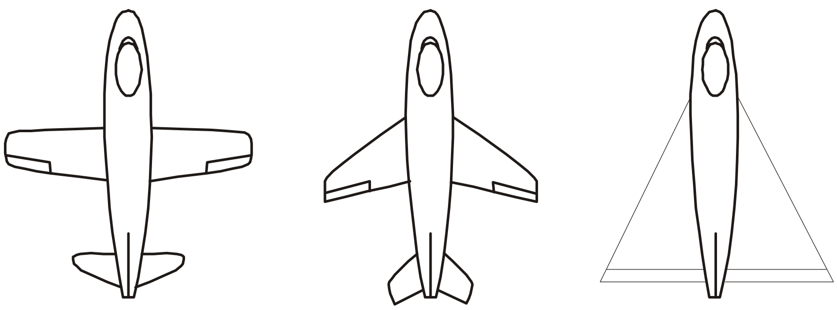
24. Considering Fig 3, the stall occurs on all three wings at angles of attack considerably greater than those of wings of medium and high aspect ratios. On all aircraft it is desirable that the landing speed

should be close to the lowest possible speed at which the aircraft can fly; to achieve this desirable minimum the wing must be at the angle of attack corresponding to the $C_{L\ max}$.

25. On all wings of very low aspect ratio, and particularly on those with a swept-back planform, the angles of attack giving the highest lift coefficients cannot be used for landing. This is because, as explained later, swept-back planforms have some undesirable characteristics near the stall and because the exaggerated nose-up attitude of the aircraft necessitates, among other things, excessively long and heavy undercarriages. The maximum angle at which an aircraft can touch down without recourse to such measures is about 15º, and the angle of attack at touchdown will therefore have to be something of this order. Fig 3 shows that the C_L corresponding to this angle of attack is lower than the C_{Lmax} for each wing.

26. Compared with the maximum usable lift coefficient available for landing aircraft with unswept wings, those of the swept and delta wings are much lower, necessitating higher landing speeds for a given wing loading. It is now apparent that, to obtain a common minimum landing speed at a stated weight, an unswept wing needs a smaller area than either of the swept planforms. The simple swept wing needs a greater area, and so a lower wing loading, in order that the reduced C_L can support the weight at the required speed. The tailless delta wing needs still more area, and so a still lower wing loading, to land at the required speed. Fig 6 shows typical planforms for the three types of wing under consideration, with the areas adjusted to give the same stalling speed. The much larger area of the delta wing is evident.

1-9 Fig 6 Planform Areas Giving a Common Stalling Speed



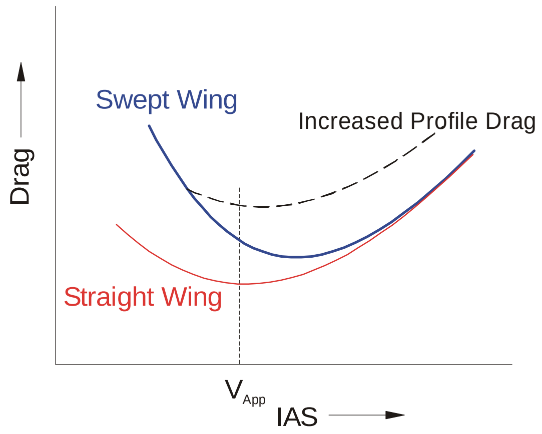
Effect of Sweepback on Drag

27. The main reason for employing sweepback as a wing planform is to improve the high-speed characteristics of the wing. Unfortunately, this has adverse effects on the amount of drag produced at the higher range of angles of attack. The induced drag increases approximately in proportion to $\frac{1}{\cos\Lambda}$. This is because, as already explained, by sweeping the wing, C_L is reduced, and therefore to maintain the same lift the angle of attack has to be increased. This increases the induced downwash and hence the induced drag.

28. The practical significance of this high increase in drag is the handling problems it imposes during an approach to landing. Because of the greater induced drag, the minimum drag speed is higher than for a comparable straight wing, and the approach speed is usually less than the minimum drag speed. Therefore, if a pilot makes a small adjustment to the aircraft's attitude by, for example, raising the nose slightly, the lift will be increased slightly, but there will be a large increase in drag which will result in a rapid fall off in speed, and a large increase in power to restore equilibrium. In fact, the stage may be reached where the use of full power is insufficient to prevent the aircraft from descending rapidly.

29. On some aircraft this problem is overcome by employing high drag devices such as airbrakes or drag-chutes to increase the zero lift drag. This results in a flatter drag curve with the minimum drag speed closer to the approach speed (see Fig 7). A further advantage is that more power is required on the approach which, on turbojet aircraft, means better engine response.

1-9 Fig 7 Improvement in Approach Speed Stability

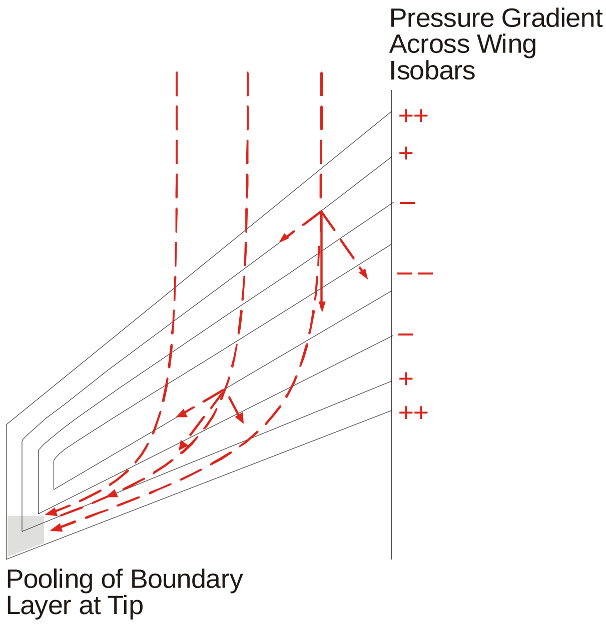


Effect of Sweepback on Stalling

30. When a wing is swept back, the boundary layer tends to change direction and flow towards the tips. This outward drift is caused by the boundary layer encountering an adverse pressure gradient and flowing obliquely to it over the rear of the wing.

31. The pressure distribution on a swept wing is shown by isobars in Fig 8. The velocity of the flow has been shown by two components, one at right angles and one parallel to the isobars. Initially, when the boundary layer flows rearwards from the leading edge it moves towards a favourable pressure gradient, ie towards an area of lower pressure. Once past the lowest pressure however, the component at right angles to the isobars encounters an adverse pressure gradient and is reduced. The component parallel to the isobars is unaffected, thus the result is that the actual velocity is reduced (as it is over an unswept wing) and also directed outwards towards the tips.

1-9 Fig 8 Outflow of Boundary Layer



32. The direction of the flow continues to be changed until the component at right angles to the isobars is reduced to zero, whilst the parallel component, because of friction, is also slightly reduced. This results in a 'pool' of slow moving air collecting at the tips.

33. The spanwise drift sets up a tendency towards tip stalling, since it thickens the boundary layer over the outer parts of the wing and makes it more susceptible to separation, bringing with it a sudden reduction in $C_{L\ max}$ over the wing tips.

34. At the same time as the boundary layer is flowing towards the tips, at high angles of attack, the airflow is separating along the leading edge. Over the inboard section it re-attaches behind a short 'separation bubble', but on the outboard section it re-attaches only at the trailing edge or fails to attach at all. The separated flow at the tips combines with the normal wing tip vortices to form a large vortex (the ram’s horn vortex). The factors which combine to form this vortex are:

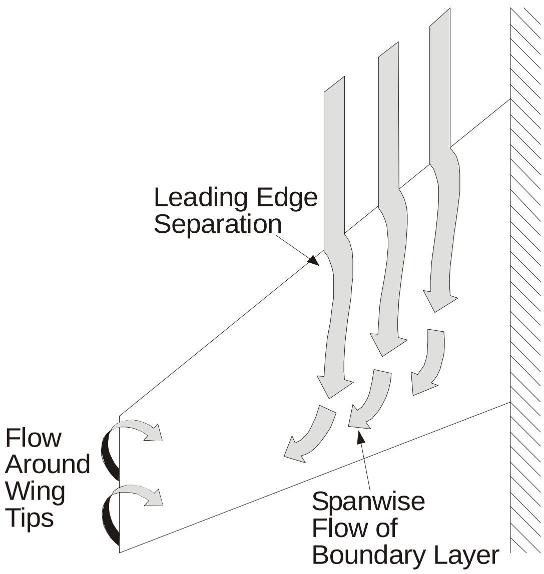
a. Leading edge separation.

b. The flow around the wing tips.

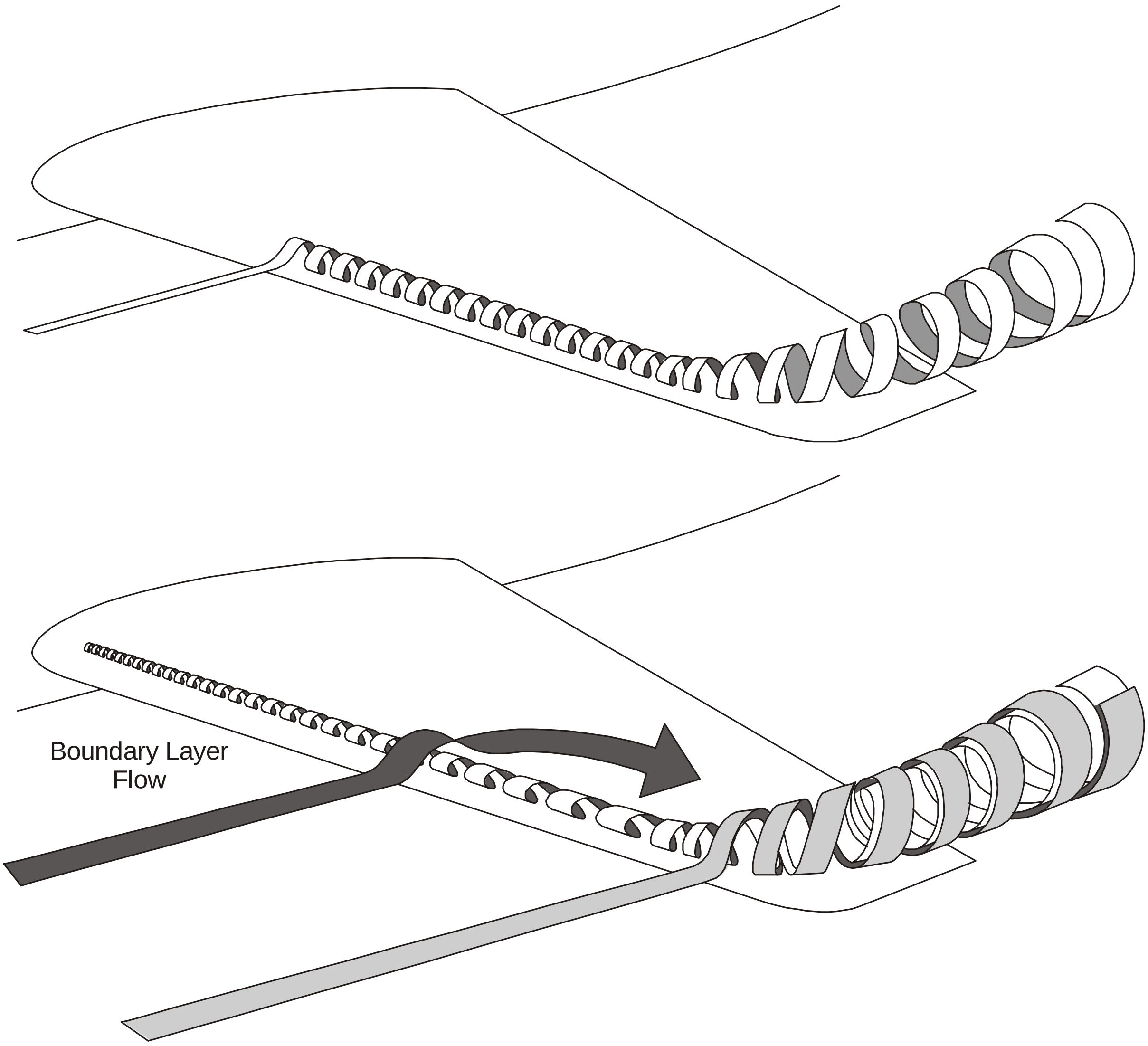
c. The spanwise flow of the boundary layer.

These factors are illustrated in Fig 9, and the sequence of the vortex development and its effect on the airflow over the wing is shown in Fig 10. From Fig 10 it can be seen that the ram’s horn vortex has its origin on the leading edge, possibly as far inboard as the wing root. The effect of the vortex on the air above it (the external flow) is to draw the latter down and behind the wing, deflecting it towards the fuselage (Fig 11).

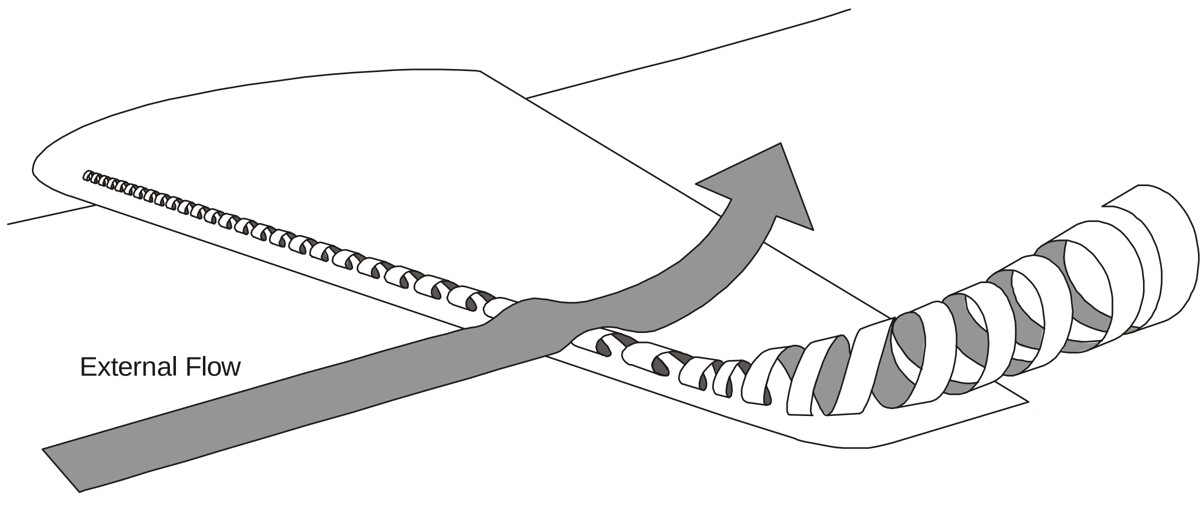
1-9 Fig 9 Vortex Development



1-9 Fig 10 Formation of Ram's Horn Vortex



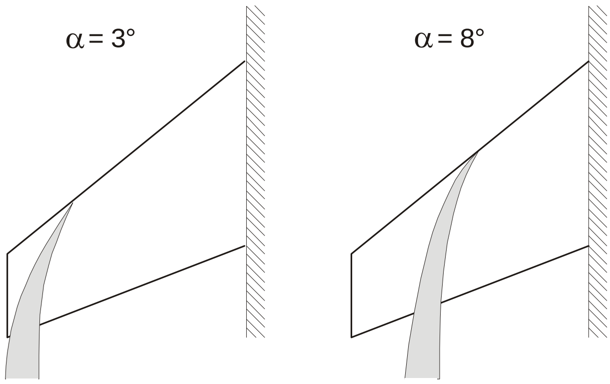
1-9 Fig 11 Influence on External Flow



35. The span wise flow of the boundary layer increases as angle of attack is increased. This causes the vortex to become detached from the leading edge closer inboard (see Fig 12). As a result, outboard ailerons suffer a marked decrease in response with increasing angle of attack. This, in turn,

means that comparatively large aileron movements are necessary to manoeuvre the aircraft at low speeds; the aircraft response may be correspondingly sluggish. This effect may be countered by limiting the inboard encroachment of the vortex as described below, or by moving the ailerons inboard. Another possible solution is the use of an all-moving wing tip.

1-9 Fig 12 Shift of Ram's Horn Vortex



Alleviating the Tip Stall

36. Most of the methods used to alleviate the tip stall aim either at maintaining a thin and therefore strong boundary layer, or re-energizing the weakened boundary layer:

a. **Boundary Layer Fences**. Used originally to restrict the boundary layer out-flow, fences also check the spanwise growth of the separation bubble along the leading edge (see para 34).

b. **Leading Edge Slots.** These have the effect of re-energizing the boundary layer (see also Volume 1, Chapter 10).

c. **Boundary Layer Suction.** Suitably placed suction points draw off the weakened layer; a new high-energy layer is then drawn down to take its place (see also Volume 1, Chapter 10).

d. **Boundary Layer Blowing**. High velocity air is injected into the boundary layer to increase its energy (see also Volume 1, Chapter 10).

e. **Vortex Generators**. These re-energize the boundary layer by making it more turbulent. The increased turbulence results in high-energy air in layers immediately above the retarded layer being mixed in and so re-energizing the layer as a whole. Vortex generators are most commonly fitted ahead of control surfaces to increase their effect by speeding up and strengthening the boundary layer. Vortex generators also markedly reduce shock-induced boundary layer separation and reduce the effects of the upper surface shockwave.

f. **Leading Edge Extension**. Also known as a 'sawtooth' leading edge, the extended leading edge is a common method used to avoid the worst effects of tip stalling. An illustration of a leading-edge extension can be seen at Volume 1, Chapter 22, Fig 29. The effect of the extension is to cut down the growth of the main vortex. A further smaller vortex, starting from the tip of the extension, affects a much smaller proportion of the tip area and in lying across the wing, behind the tip of the extension, it has the effect of restricting the outward flow of the boundary layer. In this way the severity of the tip stall is reduced and with it the pitch-up tendency. Further effects of the leading-edge extension are:

(1) The t/c ratio of the tip area is reduced, with consequent benefits to the critical Mach number.

(2) The Centre of Pressure (CP) of the extended portion of the wing lies ahead of what would be the CP position if no extension were fitted. The mean CP position for the whole wing is therefore further forward and, when the tip eventually stalls, the forward shift in CP is less marked, thus reducing the magnitude of the nose-up movement.

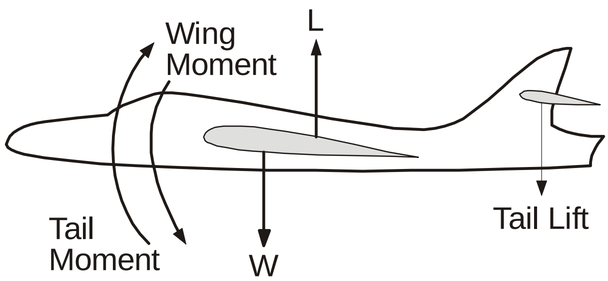
g. **Leading Edge Notch**. The notched leading edge has the same effect as the extended leading edge insofar as it causes a similar vortex formation thereby reducing the magnitude of the vortex over the tip area and with it the tip stall. Pitch-up tendencies are therefore reduced. The leading-edge notch can be used in conjunction with extended leading edge, the effect being to intensify the inboard vortex behind the devices to create a stronger restraining effect on boundary layer out-flow. The choice whether to use either or both of these devices lies with the designer and depends on the flight characteristics of the aircraft. An illustration can be seen at Volume 1, Chapter 22, Fig 30.

Pitch-up

37. **Longitudinal Instability.** Longitudinal instability results when the angle of attack of a swept wing increases to the point of tip stall. The instability takes the form of a nose-up pitching moment, called pitch-up, and is a self-stalling tendency in that the angle of attack continues to increase once the instability has set in. The aerodynamic causes of pitch-up are detailed in the following paragraphs.

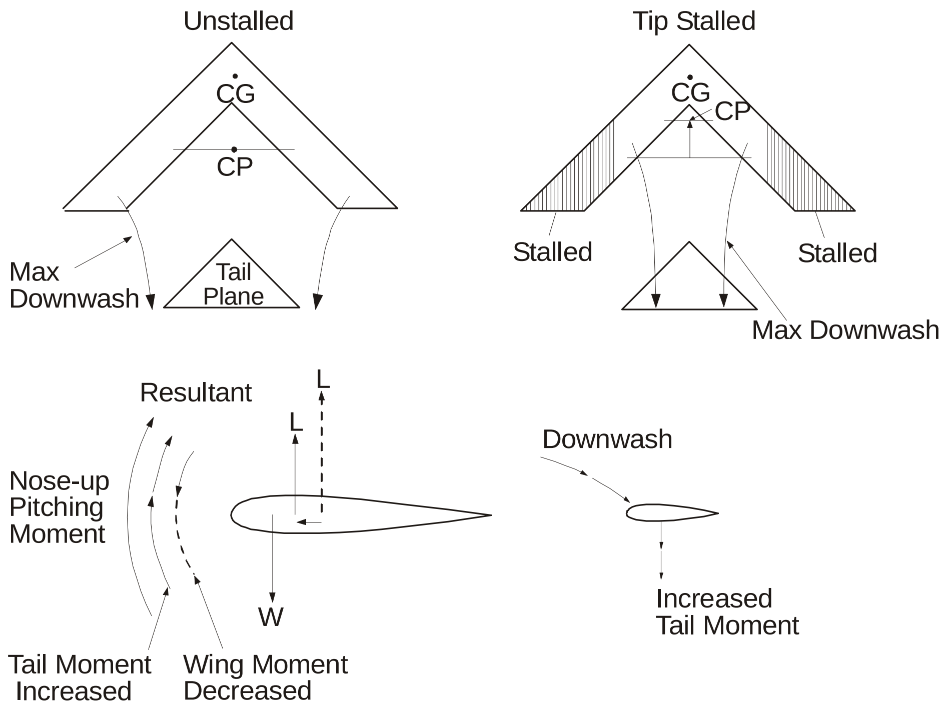
38. **Centre of Pressure (CP) Movement**. When the swept-back wing is unstalled, the CP lies in a certain position relative to the CG, the exact position being the mean of the centres of pressure for every portion of the wing from the root to the tip. When the tip stalls, lift is lost over the outboard sections and the mean CP moves rapidly forward; the wing moment (Fig 13) is reduced and a nose-up pitching moment results which aggravates the tendency.

1-9 Fig 13 Nose-up Pitching Moment Resulting from Tip Stalling



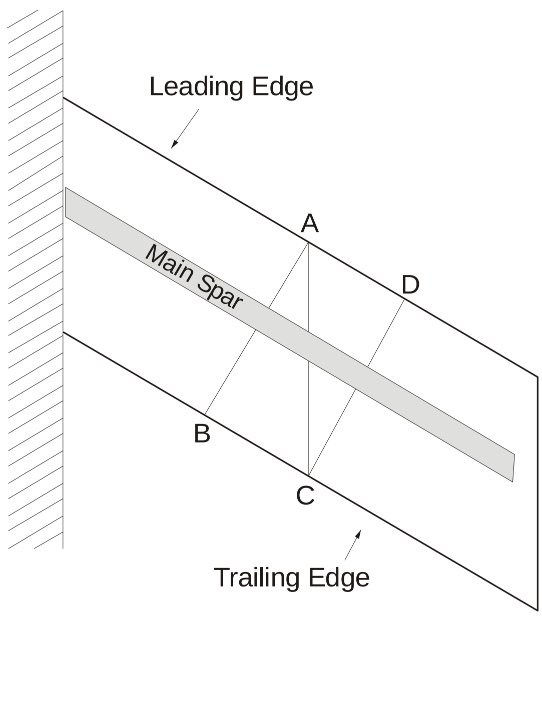
39. **Change of Downwash over the Tail plane.** Fig 14 shows that the maximum downwash from the swept-back wing in unstalled flight comes from the tip portions; this is to be expected since the C_L is highest over these parts of the wing. When the wing tips stall, effective lift production is concentrated inboard and the maximum downwash now operates over the tailplane and increases the tendency to pitch up. This effect can be reduced by placing the tailplane as low as possible in line with, or below, the wing chord line, so that it lies in a region in which downwash changes with angle of attack are less marked.

1-9 Fig 14 Variation of Downwash



40. **Washout Due to Flexure**. When a swept wing flexes under load, all chordwise points at right angles to the main spar are raised to the same degree, unless the wing is specially designed so that this is not so. Thus, in Fig 15, the points A and B rise through the same distance and the points C and D rise through the same distance but through a greater distance than A and B. Thus, C rises further than A and there is a consequent loss in incidence at this section. This aeroelastic effect is termed, 'washout due to flexure', and is obviously greatest at the wing tips. It is most noticeable during high g manoeuvres when the loss of lift at the tips, and the consequent forward movement of the centre of pressure, causes the aircraft to tighten up in the manoeuvre. A certain amount of washout due to flexure is acceptable provided the control in pitch is adequate to compensate for it, but it can be avoided by appropriate wing design.

1-9 Fig 15 Washout Due to Flexure



41. **Pitch-up on Aircraft with Straight Wings.** On aircraft with low aspect ratio, short-span wings, pitch-up can be caused by the effect of the wing tip vortices. As the angle of attack is increased the vortices grow larger until at or near the stall they may be large enough to affect the airflow over the tailplane. As each vortex rotates inwards towards the fuselage over its upper half, the tailplane angle of attack is decreased giving rise to a pitch-up tendency.

42. **Rate of Pitch-up**. From the pilot's point of view pitch-up is recognized when the pull force on the control column which is being applied to the aircraft near the stall has to be changed to a push force to prevent the nose from rising further; the more the speed decreases, the further forward must the control column be moved to restrain the nose-up pitch. Pitch-up in level flight or in any 1g stall is usually gentle, since the rate at which the stall is spreading is comparatively slow and is usually accompanied by the normal pre-stall buffeting. When the stall occurs in a manoeuvre, under g, the onset of pitch-up can be violent and sudden, corresponding to the rate of spread of the stall. This aspect is dealt with in more detail in Volume 1, Chapter 15.

The Crescent Wing

43. The crescent wing planform combines variable sweep with a changing thickness/chord ratio. At the root section where the wing is thickest, the angle of sweep is greatest. As the t/c ratio is reduced spanwise, so is the angle of sweep, so that the outboard sections are practically unswept. Hence there is little or no outflow of the boundary layer at the tips. The advantages of the crescent wing are:

a. The critical drag rise Mach number is raised.

b. The peak drag rise is reduced.

c. Because of the lack of outflow of the boundary layer at the tips, tip stalling is prevented.

FORWARD SWEEP

General

44. The benefits of wing sweep can be achieved by sweeping the wing backwards or forwards, yet only in recent years has the forward swept wing (FSW), become a serious alternative to sweepback. The reason for this lies in the behaviour of wing structures under load, see Para 40.

45. The main advantages lie in the sub/transonic regime. Taking the 70% chordline as the average position for a shockwave to form as the critical Mach number is approached, the sweep angle of this chordline influences wave drag.

46. The FSW can maintain the same chord-line sweep as the swept-back wing (SBW) but due to a geometric advantage, achieves this with less leading-edge sweep and enjoys the advantages accruing from this subsonically (Para 20, et seq).

47. The decision to employ FSW or SBW will depend inter alia, on the speed regime envisaged for the design. Due to better lift/drag ratio in the sub-sonic and near transonic speed range (typical of a combat air patrol) fuel consumption is improved over the SBW. For a high speed supersonic interception, the higher supersonic drag is a disadvantage.

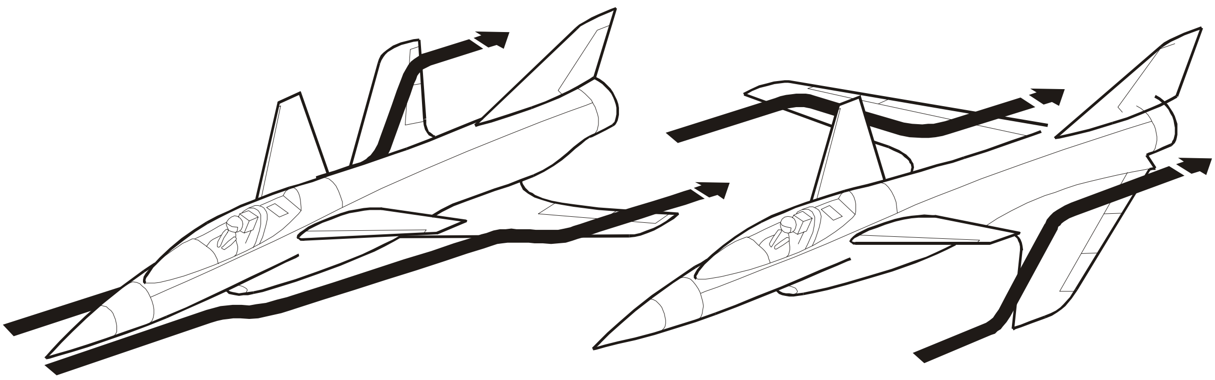
Wing Flexure

48. Under flexural load the airflow sees a steady increase in effective angle of attack from root to tip, the opposite effect to aft-sweep. Under g loading, lift will be increase at the tips, leading to pitch-up as the centre of pressure moves forwards. Additionally, the increased angle of attack at the tips now leads to increased wing flexure, which leads to increased effective angle of attack at the tips. The result of this aero-elastic divergence is likely to be structural failure of the wing, so it is not surprising that sweepback was considered to be a better option until comparatively recently. What changed the situation was the development of carbon fibre technology, which can produce controlled wing twist under load, such that the effect described is eliminated.

Vortex Generation

49. Fig 16 shows the difference in ram's horn vortex behaviour. In the swept forward design, the ram's horn vortex develops inwards towards the root, not outwards towards the tips.

1-9 Fig 16 Comparison of Ram's Horn Vortex Behaviour



50. There will, of course, still be vortices from the wing tips, but these no longer reinforce and aggravate the ram's horn vortex, which now lies along the fuselage, or slightly more outboard if a small section of the wing root is swept back.

Stalling

51. A swept forward wing will tend to stall at the root first. This stall can be controlled in a number of ways. Since a conventional tail plane would tend to lie in a vortex, a popular option is to combine sweep forward with a canard foreplane. Downwash from a carefully placed canard can delay root stall, (see Para 68), and even the vortices from the canard can be used to energize the airflow over inboard sections of the wing, maintaining lift to higher angles of attack.

52. The root-stall characteristics give better lateral control at the stall as aileron control is retained, but may incur a penalty in directional control as the fin and rudder are acting in the chaotic turbulence from the root separation.

DELTA WINGS

Tailless Delta

53. On aircraft using this type of wing the angle of attack is controlled by movement of the trailing edge of the wing; an upward movement produces a downward force on the trailing edge and so increases the angle of attack. When compared with an identical wing which uses a separate tailplane to control the angle of attack, the tailless delta reveals two main differences:

a. The $C_{L\ max}$ is reduced.

b. The stalling angle is increased.

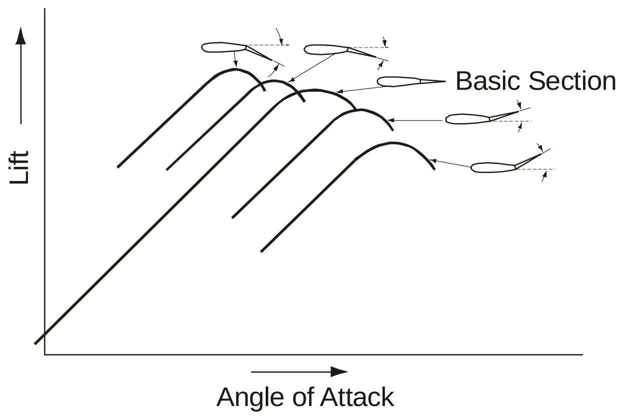
Reduction of $C_{L\ max}$

54. The chord line of the wing is defined as being a straight line joining the leading edge to the trailing edge. If a given wing/aerofoil combination has a hinged trailing edge for use as an elevator, then it can be said that when the trailing edge is in any given angular position the effective aerofoil section of the wing has been changed.

a. When such a wing reaches its stalling angle in level flight, the trailing edge elevator must be raised to impose a downward force on the trailing edge to maintain the wing at the required angle of attack. The raised trailing edge has two effects; it deflects upwards the airflow passing over it and so reduces the downwash, the amount of which is proportional to the lift, and it reduces the extent of both the low-pressure area over the upper surface of the wing and the high-pressure area below, thereby lowering the C_L.

b. The curves of Fig 17 show that a section with a raised trailing edge will suffer a decreased $C_{L\ max}$ compared to the basic section.

1-9 Fig 17 Effect of Hinged Trailing Edge on $C_{L\ max}$ and Stalling Angle

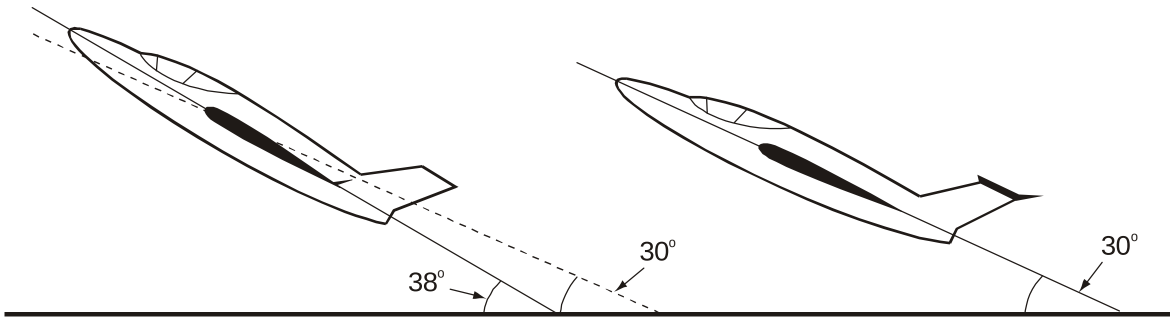


Increase in Stalling Angle

55. The planform of the delta wing gives it an inherently low aspect ratio and therefore a high stalling angle and a marked nose-up attitude at the stall in level flight. If a certain delta wing is used without a tail plane, ie the trailing edge is used as an elevator, then the stalling angle is higher than when the same wing is used in conjunction with a tail plane.

56. All else being equal (planform, aspect ratio, area, etc) changes in the amount of camber (by altering the angular setting of the trailing edge elevator) do not affect the stalling angle appreciably. That is, the angle between the chord line (drawn through the leading and trailing edges) and the direction of the airflow remains constant when at maximum C_L irrespective of the setting of the hinged trailing edge. Fig 18 illustrates this point and it can be seen that for both the 'tailed' and 'tailless', aircraft the stalling angle is the same when measured on the foregoing principles.

1-9 Fig 18 Comparison of Stalling Angle



57. However, it is normal practice and convention to measure the stalling angle with reference to the chord line obtained when the moveable trailing edge is in the neutral position, and not to assume a new chord line with each change in trailing edge movement. When the stalling angle is measured with reference to the conventional fixed chord line, it can be seen from Fig 18 that the angle is greater. Fig 18 also shows that, because the wing proper is set at a greater angle at the stall when a trailing edge elevator is used, the fuselage attitude is more nose-up, giving a more exaggerated attitude at the stall in level flight.

58. Since it is easier to refer to angle of attack against a fixed chord line, the basic chord line is always used as the reference datum. This convention is the reason for the apparently greater stalling angles of tailless delta wings; it is perhaps a more realistic method, as the pilot is invariably aware of the increased attitude of his aircraft relative to the horizontal but is not always aware of increases in the angle of attack.

The C_L Curve

59. Reference to Fig 3 shows that the peak of the curve for the lift coefficient is very flat and shows little variation of C_L over a comparatively wide range of angles. This very mild stalling behaviour enables the delta wing to be flown at an angle of attack considerably higher than that of the $C_{L\ max}$, possibly with no ill effects other than the very marked increase in the drag. The flat peak denotes a gradual stall, with a consequent gradual loss of lift as the stalling angle is exceeded.

The Slender Delta

60. The slender delta provides low drag at supersonic speeds because of its low aspect ratio. This, combined with a sharp leading edge, produces leading edge separation at low angles of attack. Paradoxically, this is encouraged. Up to now the vortex so produced has been an embarrassment as it is unstable, varies greatly with angle of attack, causes buffet, increases drag and decreases $C_{L\ max}$. By careful design, the vortex can be controlled and used to advantage.

61. **Vortex Lift.** The vortex on a slender delta is different in character from that on a wing of higher aspect ratio (greater than 3). On the slender delta the vortex will cover the whole leading edge from root to tip, rather than start at the tip and travel inwards at higher angles of attack. Its behaviour is therefore more predictable, and, as it is present during all aspects of flight, the following characteristics may be exploited:

a. Leading edge flow separation causes CP to be situated nearer mid-chord. Hence there will be less difference between CP subsonic and CP supersonic than before, and longitudinal stability is improved.

b. The vortex core is a region of low pressure, therefore an increase in C_L may be expected. On the conventional delta this cannot be utilized as the vortex seldom approaches anywhere near the wing root and most of its energy appears in the wake behind the wing, where it produces high induced drag. On the slender delta the low pressure in the vortex is situated above the wing and can result in an increase in C_L of as much as 30% under favourable conditions.

VARIABLE GEOMETRY WINGS

General

62. An aircraft which is designed to fly at supersonic speeds most of the time usually has poor low speed characteristics which have to be accepted, although various high lift devices are available for reducing take-off and landing speeds and improving the low speed handling qualities. In order to achieve the desired high-speed performance, the aircraft has thin symmetric wing sections and highly swept or delta wing planforms; these wings are very inefficient at low speeds where unswept wing planforms and cambered wing sections are required.

63. In the case of an aircraft which is required to be operated efficiently at both high and low speeds, variable wing sweep is a desirable feature to be incorporated in the design. The wings can then be swept back when the aircraft is being flown at high speeds and forward again when flying at low speeds.

Stability and Control Problems

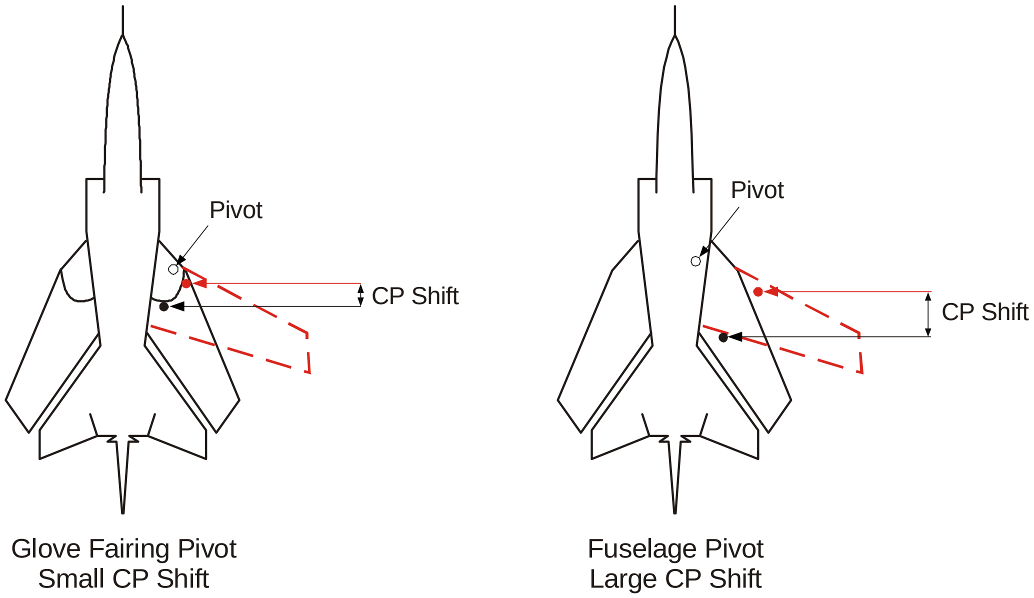
64. When the wing of an aircraft is moved backwards the aerodynamic centre moves rearwards. The CG of the aircraft also moves back at the same time, but, since most of the weight of an aircraft is concentrated in the fuselage, the movement is less than that of the aerodynamic centre. The rearwards movement of the aerodynamic centre produces a nose-down change of trim and an increase in the longitudinal static stability of the aircraft. Additional up-elevator is required to trim the aircraft, and this results in additional drag called 'trim drag'. This extra drag can be a relatively large part of the total drag of an aircraft at supersonic speeds and it is essential that it should be kept as small as possible. Various design methods are available for reducing or eliminating the trim changes produced by sweeping the wings.

65. **Wing Translation**. The aerodynamic centre can be moved forward again by translating the wing forwards as it is swept back. This method involves extra weight and structural complications.

66. **CG Movement**. The aircraft can be designed so that the CG moves rearwards in step with the aerodynamic centre by mounting some weight in the form of engines, etc at the wing tips. As engines would have to swivel to remain aligned with the airflow, additional weight and other complications result. Another possible method of moving the CG is by transferring fuel to suitable trim tanks in the rear fuselage.

67. **Leading Edge Fillet and Pivot Position**. Another solution can be obtained by positioning the pivot point outboard of the fuselage inside a fixed, leading edge fillet, called a 'glove'. The optimum pivot position for minimum movement of the CP depends on the wing planform, but it is usually about 20% out along the mid-span. However, the fixed glove-fairing presents a highly swept portion of the span at low-speed, forward-sweep settings. This incurs the undesirable penalties that variable geometry is designed to overcome. A compromise between sweeping the whole wing and a long glove giving the minimum CP shift is usually adopted (see Fig 19).

1-9 Fig 19 Movement of CP



CANARD WINGS

Use and Disadvantages

68. A canard-type configuration is one which has a fore-plane located forward of the wing instead of in the more conventional aft position. On an aircraft with a long slender fuselage with engines mounted in the tail and a CG position well aft, this layout has the obvious geometric advantage of a longer moment arm. This enables the stability and trim requirements to be satisfied by a fore-plane of smaller area. The trim drag problem will also be reduced because, at high speeds, an up-load will be required on the fore-plane to trim the aircraft. There are, however, certain disadvantages with this layout:

a. **Stalling Problems.** On a 'conventional' rear-tailplane configuration, the wing stalls before the tailplane, and longitudinal control and stability are maintained at the stall. On a canard layout, if the wing stalls first, stability is lost, but if the fore-plane stalls first then control is lost and the maximum value of C_L is reduced. One possible solution is to use a canard surface and a wing trailing edge flap in combination, with one surface acting as a trimming device and the other as a control. Alternatively, an auxiliary horizontal tailplane at the rear may be used for trim and control at low speed.

b. **Interference Problems**. In the same way as the airflow from the wing interferes with the tail unit on the conventional rear-tail layout, so the airflow from the fore-plane interferes with the flow around the main wing and vertical fin in a canard layout. This can cause a reduction in lift on the main wing and can also result in stability problems. The interference with the vertical fin can cause a marked reduction in directional static stability at high angles of attack. The stability may be improved by employing twin vertical fins in place of the single control vertical fin.

SUMMARY

Planform Considerations

69. Planform is the geometrical shape of the wing when viewed from above, and it largely determines the amount of lift and drag obtainable from a given area, it also has a pronounced effect on the stalling angle of attack.

70. Aspect ratio (A) is found by dividing the square of the wingspan by the area of the wing:

$$A = \frac{\text{Span}^2}{\text{Area}} \text{ or } \frac{\text{Span}}{\text{Mean Chord}}$$

71. The following wing characteristics are affected by aspect ratio:

a. Induced drag is inversely proportional to aspect ratio.

b. The reduced effective angle of attack of very low aspect ratio wings can delay the stall considerably. (Some delta wings have no measurable stalling angle up to 40º.)

72. In the aerodynamic sense, the elliptical wing is the most efficient planform because the uniformity of lift coefficient and downwash incurs the least induced drag for a given aspect ratio.

73. Any swept-back planform suffers a marked drop in $C_{L\ max}$, when compared with an unswept wing with the same significant parameters; also, the boundary layer tends to change direction and flow towards the tips.

74. The spanwise drift of the boundary layer sets up a tendency towards tip stalling on swept wing aircraft. This may be alleviated by the use of one or more of the following:

a. Boundary layer fences.

b. Leading edge slots

c. Boundary layer suction.

d. Boundary layer blowing.

e. Vortex generators.

f. Leading edge extension.

g. Leading edge notch.

75. The factors effecting pitch-up are:

a. Longitudinal instability.

b. Centre of pressure movement.

c. Change of downwash over the tail-plane.

d. Washout due to flexure.

76. The advantages of a crescent wing are:

a. The critical drag rise Mach number is raised.

b. The peak drag rise is reduced.

c. Because of the lack of outflow of the boundary layer at the tips, tip stalling is prevented.

77. A FSW stalls at the root first, prolonging aileron control. The configuration may offer an advantage in L/D ratio over sweepback in the appropriate speed range.

78. When compared with a delta which uses a separate tailplane to control angle of attack, the tailless delta reveals two main differences:

a. The $C_{L\,max}$ is reduced.

b. The stalling angle is increased.

79. Vortex lift has the following characteristics:

a. Leading edge flow separation causes the CP to be situated nearer to midchord.

b. The vortex core is a region of low pressure, therefore an increase in C_L may be expected.

80. The canard configuration has the following advantages and disadvantages:

a. Advantages:

(1) The control surface is ahead of any shocks which may form on the mainplane.

(2) On an aircraft with a long slender fuselage with engines mounted in the tail and the CG position well aft, the fore-plane has the advantage of a long moment arm.

(3) The stability and trim requirements can be satisfied with a smaller fore-plane area.

(4) Because up-loads will be required, the trim drag problem is reduced.

b. Disadvantages:

(1) If the wing stalls first stability is lost.

(2) If the fore-plane stalls first control is lost.

(3) In the same way as the airflow from the wing interferes with the tail unit on the conventional rear-tail layout, so the airflow from the fore-plane interferes with the flow around the main wing and vertical fin of the canard configuration.

CHAPTER 10 - LIFT AUGMENTATION

Introduction

1. As aircraft have developed over the years so their wing loading has increased from World War I figures of 12 lb to 15 lb per sq ft to figures in excess of 150 lb per sq ft. These figures are derived from the weight of the aircraft divided by the wing area, which of course, also represents the lift required in level flight per unit area of wing. Now, although aerofoil design has improved, the extra lift required has been produced by flying faster, with the result that where wing loading has gone up by a factor of 10, the stalling speed has also increased by a factor of $\sqrt{10}$. In other words, stalling speeds have increased from about 35 kt to 120 kt. This in turn has led to higher touchdown speeds on landing, and so longer runways and/or special retardation facilities such as tail chutes or arrester wires are needed. The aircraft also has to reach high speeds for unstick. Lift augmentation devices are used to increase the maximum lift coefficient, in order to reduce the air speed at unstick and touchdown.

2. The chief devices used to augment the C_{Lmax} are:

a. Slats - either automatic or controllable by the pilot. Slots are also in this classification.

b. Flaps - these include leading edge, trailing edge and jet flaps.

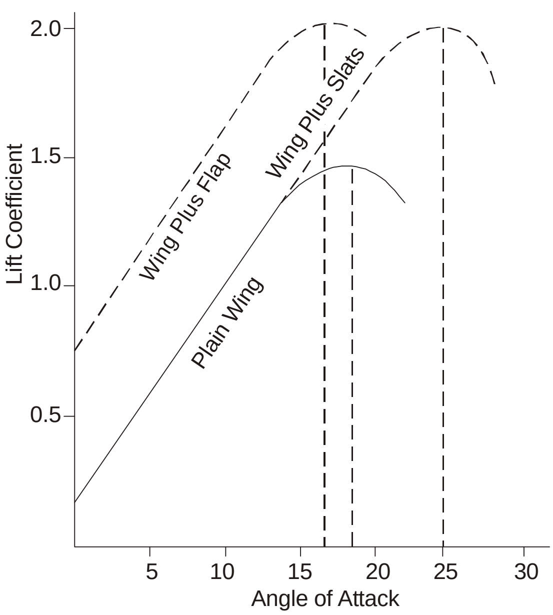
c. Boundary Layer Control (either sucking or blowing air over the wing to re-energize the boundary layer).

SLATS

Principle of Operation

3. When a small auxiliary aerofoil slat of highly cambered section is fixed to the leading edge of a wing along the complete span and adjusted so that a suitable slot is formed between the two, the C_{Lmax} may be increased by as much as 70%. At the same time, the stalling angle is increased by some 10°. The graph at Fig 1 shows the comparative figures for a slatted and unslatted wing of the same basic dimensions.

1-10 Fig 1 Effect of Flaps and Slats on Lift



4. The effect of the slat is to prolong the lift curve by delaying the stall until a higher angle of attack. When operating at high angles of attack the slat itself is generating a high lift coefficient because of its

marked camber. The action of the slat is to flatten the marked peak of the low-pressure envelope at high angles of attack and to change it to one with a more gradual pressure gradient. The flattening of the lift distribution envelope means that the boundary layer does not undergo the sudden thickening that occurred through having to negotiate the very steep gradient that existed immediately behind the former suction peak, and so it retains much of its energy, thus enabling it to penetrate almost the full chord of the wing before separating. Fig 2 shows the alleviating effect of the slat on the low-pressure peak and that, although flatter, the area of the low pressure region, which is proportional to its strength, is unchanged or even increased. The passage of the boundary layer over the wing is assisted by the fact that the air flowing through the slot is accelerated by the venturi effect, thus adding to the kinetic energy of the boundary layer and so helping it to penetrate further against the adverse gradient.

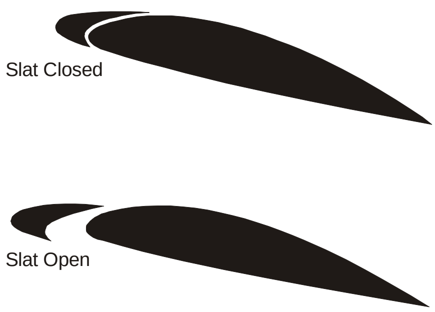
1-10 Fig 2 Effect of Slat on the Pressure Distribution

5. As shown in Fig 1, the slat delays separation until an angle of about 25° to 28° is reached, during which time the lift coefficient has risen steadily, finally reaching a peak considerably greater than that of an unslatted wing. Assuming that the C_{Lmax} of the wing is increased by, say 70%, it is evident that the stalling speed at a stated wing loading can be much reduced; for example, if an unslatted wing stalls at a speed of about 100 kt, its fully slatted counterpart would stall at about 80 kt. The exact amount of the reduction achieved depends on the length of the leading edge covered by the slat and the chord of the slat. In cases where the slats cover only the wing tips, the increase in C_L is proportionately smaller.

Automatic Slats

6. Since the slat is of use only at high angles of attack, at the normal angles its presence serves only to increase drag. This disadvantage can be overcome by making the slat moveable so that when not in use it lies flush against the leading edge of the wing as shown in Fig 3. In this case the slat is hinged on its supporting arms so that it can move to either the operating position or the closed position at which it gives least drag. This type of slat is fully automatic in that its action needs no separate control.

1-10 Fig 3 Automatic Slat



7. At high angles of attack the lift of the slat raises it clear of the wing leaving the required slot between the two surfaces which accelerates and re-energizes the airflow over the upper surface.

Uses of the Slat

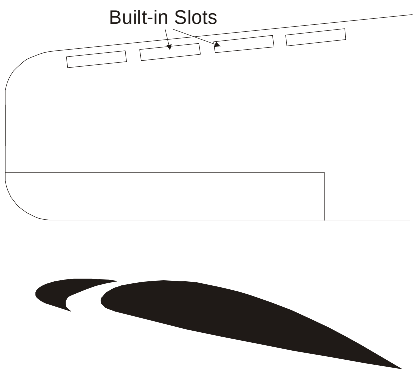
8. On some high performance aircraft the purpose of slats is not entirely that of augmenting the C_{Lmax} since the high stalling angle of the wing with a full-span slat necessitates an exaggerated and unacceptable landing attitude if the full benefit of the slat is to be obtained. When slats are used on these aircraft, their purpose may be as much to improve control at low speeds by curing any tendencies towards wing tip stalling as it is to augment the lift coefficient.

9. If the slats are small and the drag negligible they may be fixed, ie non-automatic. Large slats are invariably of the automatic type. Slats are often seen on the leading edges of sharply swept-back wings; on these aircraft the slats usually extend along most of the leading edge and, besides relieving the tip stalling characteristics, they do augment C_L considerably, even though the angle of attack may be well below the stalling angle.

10. Automatic slats are designed so that they open fully some time before the speed reaches that used for the approach and landing. During this period they still accomplish their purpose of making the passage of the boundary layer easier by flattening the pressure gradient over the front of the wing. Thus whenever the slat is open, at even moderate angles of attack, the boundary layer can penetrate further aft along the chord thus reducing the thickening effect and delaying separation and resulting in a stronger pressure distribution than that obtained from a wing without slats. As the angle of attack is increased so the effect becomes more pronounced.

11. **Built-in Slots*.*** Fig 4 shows a variation of the classic arrangement, in which suitably shaped slots are built into the wing tips just behind the leading edge. At higher angles of attack, air from below the wing is guided through the slots and discharged over the upper surfaces, tangential to the wing surface, thereby re-energizing the boundary layer to the consequent benefit of the lift coefficient.

1-10 Fig 4 Built-in Slot



12. **Stalling with Slats**. The effect of the slat at the highest angles of attack is to boost the extent of the low-pressure area over the wing. At angles of attack of about 25°, the low-pressure envelope has been considerably enlarged and a proportionately larger amount of lift is being developed. When the wing reaches a certain angle, the slat can no longer postpone events and the stall occurs. When the powerful low pressure envelope collapses, the sudden loss of lift may result in equally sudden changes in the attitude of the aircraft. This applies particularly if one wing stalls before the other; in this case a strong rolling moment or wing dropping motion would be set up.

FLAPS

Purpose

13. The operation of the flap is to vary the camber of the wing section. High lift aerofoils possess a curved mean camber line, (the line equidistant from the upper and lower surfaces); the greater the

mean camber the greater the lift capability of the wing. High speed aerofoils however, may have a mean camber line which is straight, and, if either or both the leading edge and trailing edge can be hinged downwards, the effect will be to produce a more highly cambered wing section, with a resultant increase in the lift coefficient.

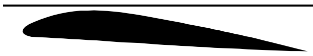
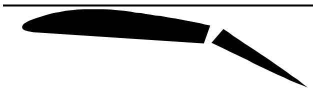
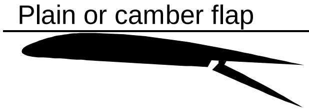
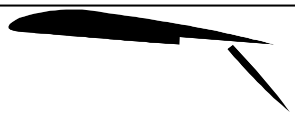
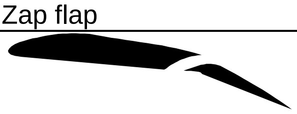
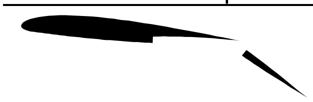
Action of the Flap

14. Increased camber can be obtained by bringing down the leading or trailing edge of the wing (or both). The use of leading edge flaps is becoming more prevalent on large swept-wing aircraft, and trailing edge flaps are used on practically all aircraft except for the tailless delta. The effect of this increased camber increases the lift, but since the change in camber is abrupt the total increase is not as much as would be obtained from a properly curved mean camber line. (Fig 1 shows the effect of flaps on the C_{Lmax} and stalling angle.)

Types of Flap

15. The trailing edge flap has many variations, all of which serve to increase the C_{Lmax}. Some, however, are more efficient than others. Fig 5 illustrates some representative types in use, the more efficient ones are usually more complicated mechanically.

1-10 Fig 5 Types of Flaps

High-lift devices	*Increase of Maximum lift*	*Angle of basic aerofoil at max lift*	*Remarks*
 Basic aerofoil	-	15°	Effects of all high-lift devices depend on shape of basic aerofoil
 Plain or camber flap	50%	12°	Increase camber. Much drag when fully lowered. Nose-down pitching moment.
 Split flap	60%	14°	Increase camber. Even more drag than plain flap. Nose-down pitching moment.
 Zap flap	90%	13°	Increase camber and wing area. Much drag. Nose-down pitching moment.
 Slotted flap	65%	16°	Control of boundary layer. Increase camber. Stalling delayed. Not so much drag.
 Double-slotted flap	70%	18°	Same as single-slotted flap only more so. Treble slots sometimes used.
 Fowler flap	90%	15°	Increase camber and wing area. Best flaps for lift. Complicated mechanism. Nose-down pitching moment.

16. The increase in C_{Lmax} obtained by the use of flaps varies from about 50% for the single flap, to 90% for the Fowler type flap. The effectiveness of a flap may be considerably increased if the air is constrained to follow the deflected surface and not to break away or stall. One method of achieving this is by the use of slotted flaps, in which, when the flap is lowered, a gap is made which operates as a slot to re-energize the air in a similar manner to leading edge slots. Some aircraft have double-slotted, or even triple-slotted flaps which may give a C_{Lmax} increase of up to 120%.

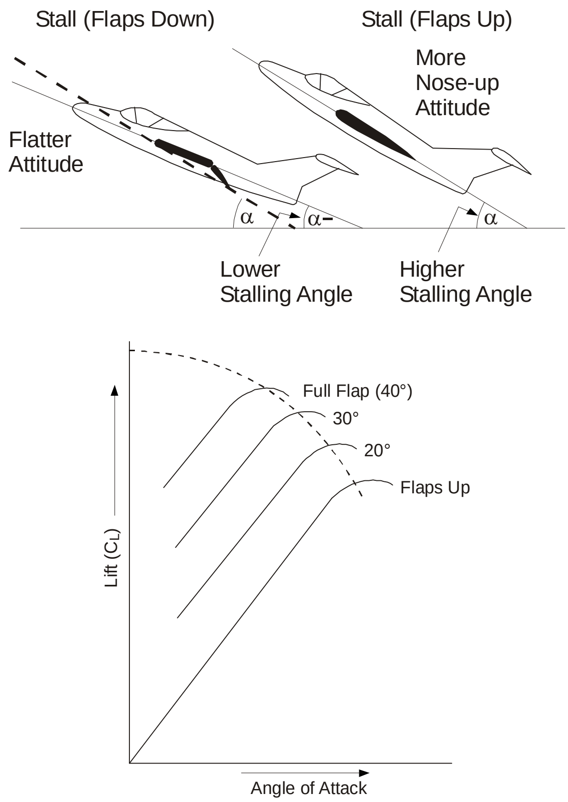
17. The angle of attack at which the maximum lift coefficient is obtained with the trailing edge flap is slightly less than with the basic aerofoil. Thus, the flap gives an increased lift coefficient without the attendant exaggerated angles made necessary with slats.

Effect of Flap on the Stalling Angle

18. When the trailing edge flap is lowered the angle of attack for level flight under the prevailing conditions is reduced. For each increasing flap angle there is a fixed and lower stalling angle. The lower stalling angle is caused by the change in the aerofoil section when the flap is lowered. Volume 1, Chapter 9, Paras 55 to 58 describe how the use of a trailing edge elevator on a delta wing increases the stalling angle and aircraft attitude at the stall; the same approach can be used to account for the effect of flap on the stalling angle.

19. The trailing edge flap is directly comparable to the trailing edge elevator insofar as the effect on stalling angle is concerned. The raised trailing edge elevator at the stall increases the stalling angle of attack and the aircraft attitude in level flight but the lowered trailing edge flap reduces the stalling angle and the aircraft attitude at the level flight stall. Fig 6 illustrates how the lowered flap affects the angle of attack and the aircraft attitude. Pilots should take care not to confuse attitude with angle of attack, for, as explained in Volume 1, Chapter 15, the attitude of the aircraft has no fixed relationship with the angle of attack while manoeuvring.

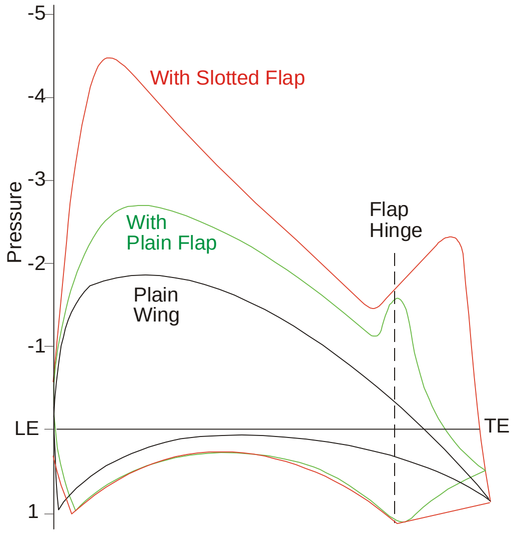
1-10 Fig 6 Effect of Flap on Stalling Angle and Level Flight Stalling Attitude



Change in Pressure Distribution with Flap

20. All types of flaps, when deployed, change the pressure distribution across the wing. Typical pressure distributions with flap deployed are shown in Fig 7 for plain and slotted flaps. The slotted flap has a much greater intensity of loading on the flap itself.

1-10 Fig 7 Change in Pressure Distribution with Flaps



Change in Pitching Moment with Flap

21. All trailing edge flaps produce an increased nose-down pitching moment due to the change in pressure distribution around the wing flap. Flaps, when lowered, may have the added effect of increasing the downwash at the tail. The amount by which the pitching moment is changed resulting from this increase in downwash depends on the size and position of the tail. These two aspects of change in pitching moment generally oppose each other and on whichever aspect is dominant will depend whether the trim change on lowering flap is nose-up or nose-down. Leading edge flaps tend to reduce the nose-down pitching moment and reduce the wing stability near the stall.

Lift/Drag Ratio

22. Although lift is increased by lowering flaps there is also an increase in drag and proportionately the drag increase is much greater when considered at angles of attack about those giving the best lift/drag ratio. This means that lowering flap almost invariably worsens the best lift/drag ratio.

23. For a typical split or trailing edge flap, as soon as the flap starts to lower, the lift and drag start increasing. Assuming that the flap has an angular movement of 90°, for about the first 30° there is a steady rise in the C_L; during the next 30° the C_L continues to increase at a reduced rate and during the final part of the movement a further very small increase occurs.

24. In conjunction with the lift the drag also increases, but the rate of increase during the first 30° is small compared with that which takes place during the remainder of the movement, the final 30° producing a very rapid increase in the rate at which the drag has been rising.

25. When flap is used for take-off or manoeuvring it should be set to the position recommended in the Aircrew Manual. At this setting the lift/drag ratio is such that the maximum advantage is obtained for the minimum drag penalty. For landings, however, the high drag of the fully lowered flap is useful since it permits a steeper approach without the speed becoming excessive (ie it has the effect of an airbrake).

The increased lift enables a lower approach speed to be used and the decreased stalling speed means that the touchdown is made at a lower speed. The high drag has another advantage in that it causes a rapid deceleration during the period of float after rounding out and before touching down.

Use of Flap for Take-Off

26. The increased lift coefficient when the flaps are lowered shortens the take-off run provided that the recommended amount of flap is used. The flap angle for take-off is that for the best lift/drag ratio that can be obtained with the flaps in any position other than fully up. If larger amounts of flap are used, although the lift is increased, the higher drag slows the rate of acceleration so that the take-off run, although perhaps shorter than with no flap, is not the shortest possible.

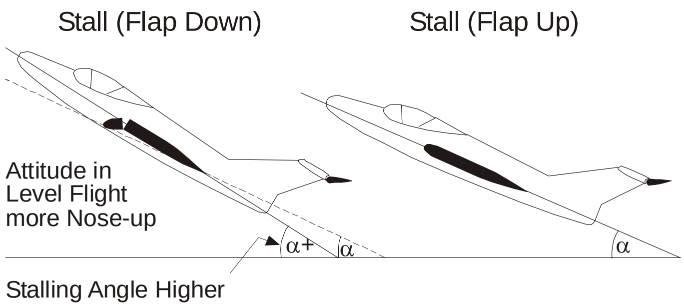
27. When the take-off is made at, or near, the maximum permissible weight, the flaps should invariably be set to the recommended take-off angle so that the maximum lifting effort can be obtained from the wing.

28. **Raising the Flaps in Flight**. Shortly after the take-off, while the aircraft is accelerating and climbing slightly, the action of raising the flaps causes an immediate reduction in the lift coefficient and the aircraft loses height or sinks unless this is countered by an increase in the angle of attack. If the angle of attack is not increased, ie if the pilot makes no correcting movement with the control column, the reduced lift coefficient results in a loss of lift which causes the aircraft to lose height until it has accelerated to a higher air speed that counterbalances the effect of the reduced C_L. When the flaps are raised and the sinking effect is countered by an increased angle of attack, the attitude of the aircraft becomes noticeably more nose-up as the angle of attack is increased. The more efficient the flaps the greater is the associated drop in lift coefficient and the larger the subsequent corrections that are needed to prevent loss of height. On some aircraft it is recommended that the flaps should be raised in stages so as to reduce the C_L gradually and so avoid any marked and possibly exaggerated corrections. This applies sometimes when aircraft are heavily loaded, particularly in the larger types of aircraft.

Leading Edge Flap

29. The effect of leading edge flaps is, as with other flaps, to increase the C_L and lower the stalling speed. However, the fact that it is the leading edge and not the trailing edge that is drooped results in an increase in the stalling angle, and the level flight stalling attitude. The difference is explained as before (Volume 1, Chapter 9, para 58, and para 19 of this chapter), by the fact that although the stalling angle measured with respect to the chord line joining the leading and trailing edges of the changed section may not be affected, the stalling angle is increased when measured, as is conventional, with respect to the chord line of the wing with the flap fully raised. Fig 8 illustrates this point. Leading edge flaps are invariably used in conjunction with trailing edge flaps. The operation of the leading edge flap can be controlled directly from the cockpit or it can be linked, for example, with the air speed measuring system so that the flaps droop when the speed falls below a certain minimum and vice versa.

1-10 Fig 8 Effect of Leading Edge Flap on Stalling Angle



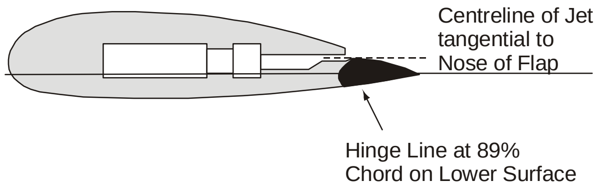
30. The effect of leading edge flaps is similar to that of slats except that the stalling angle is not increased as much. The amount of increase in the C_{Lmax} is about the same in both cases. The leading edge flap is sometimes referred to as a nose flap.

Jet Flap

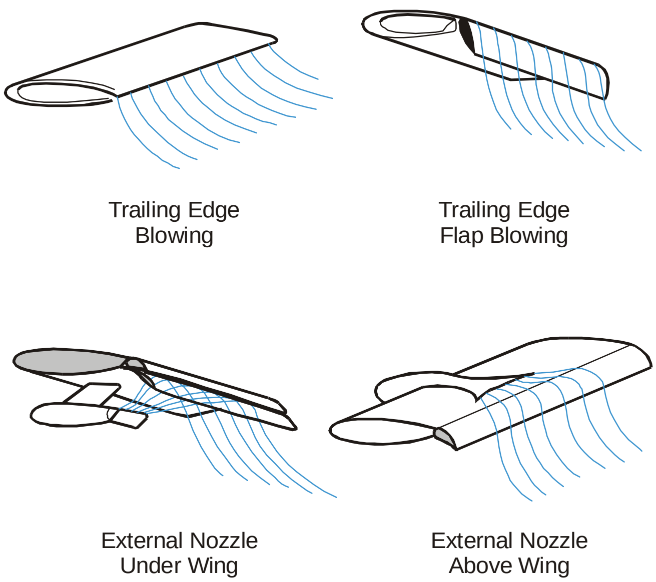
31. Jet flap is a natural extension of a slot blowing over trailing edge flaps for boundary layer control, using much higher quantities of air with a view to increasing the effective chord of the flap to produce so-called 'super circulations' about the wing.

32. The term jet flap implies that the gas efflux is directed to leave the wing trailing edge as a plane jet at an angle to the main stream so that an asymmetric flow pattern and circulation is generated about the aerofoil in a manner somewhat analogous to a large trailing edge flap. By this means, the lift from the vertical component of the jet momentum is magnified several times by 'pressure lift' generated on the wing surface, while the sectional thrust lies between the corresponding horizontal component and the full jet momentum. To facilitate variation of jet angle to the main stream direction, the air is usually ejected from a slot forward of the trailing edge, over a small flap whose angle can be simply varied (see Fig 9). Such basic jet flap schemes essentially require the gas to be ducted through the wing and so are often referred to as 'internal flow' systems. Fig 10 illustrates these, and the basics of their 'external flow' configurations.

1-10 Fig 9 Jet Flap



Some Jet Flap Systems



Effect of Sweepback on Flap

33. The use of sweepback will reduce the effectiveness of trailing edge control surfaces and high lift devices. A typical example of this effect is the application of a single slotted flap over the inboard 60% span to both a straight wing and a wing with 35° sweepback. The flap applied to the straight wing produces an increase in the C_{Lmax} of approximately 50%. The same type flap applied to the swept wing produces an increase in C_{Lmax} of approximately 20%. The reason for this is the decrease in frontal area of flap with sweepback.

Effect of Flap on Wing Tip Stalling

34. Lowering flaps may either increase or alleviate any tendency towards the tip stalling of swept wings. When the flaps are lowered the increased downwash over the flaps and behind them induces a balancing upwash over the outer portions of the wing at high angles of attack and this upwash may be sufficient to increase the angle of attack at the tip to the stalling angle. On the other hand, because of the lowered flaps, the higher suctions obtained over the inboard sections of the wing have the effect of restricting the outward flow of the boundary layer and thus a beneficial effect is obtained on wing tip stalling tendencies. The practical outcome of these opposing tendencies is dependent on the pressure configuration.

BOUNDARY LAYER CONTROL

General

35. Any attempt to increase the lifting effectiveness of a given wing directly and fundamentally concerns the boundary layer. If the boundary layer can be made to remain laminar and unseparated as it moves over the wing, then not only is the lift coefficient increased but both surface friction drag and form drag are reduced.

36. There are various methods of controlling the boundary layer so that it remains attached to the aerofoil surface as far as possible. They all depend on the principle of adding kinetic energy to the lower layers of the boundary layer.

Boundary Layer Control by Suction

37. If enough suction could be applied through a series of slots or a porous area on the upper surface of the wing, separation of the boundary layer at almost all angles of attack could be prevented. However, it has been found that the power required to draw off the entire boundary layer so that it is replaced by completely undisturbed air is so large that the entire output of a powerful engine would be required to accomplish this.

38. However, even moderate amounts of suction have a beneficial effect in that the tendency to separate at high angles of attack can be reduced. The effect of moderate suction is to increase the strength and stability of the boundary layer.

39. The effect of the suction is to draw off the lower layer (the sub-layer) of the boundary layer, so that the upper part of the layer moves on to the surface of the wing. The thickness of the boundary layer is thereby reduced and also its speed is increased, since the heavily retarded sub-layer has been replaced by faster moving air.

40. The suction is effected either through a slot or series of slots in the wing surface or by having a porous surface over the area in which suction is required. These devices are positioned at a point where the thickening effect of the adverse gradient is becoming marked and not at the beginning of the adverse gradient. Generally, suction distributed over a porous area has a better effect than the concentrated effect through a slot.

Boundary Layer Control by Blowing

41. When air is ejected at high speed in the same direction as the boundary layer at a suitable point close to the wing surface, the result is to speed up the retarded sub-layer and re-energize the complete boundary layer; again this enables it to penetrate further into the adverse gradient before separating.

42. Very high maximum lift coefficients can be obtained by combining boundary layer control with the use of flaps. In this case the suction, or blowing, of air takes place near the hinge line of the flap. An average C_{Lmax}, for a plain aerofoil is about 1.5, for the same aerofoil with a flap it may be increased to about 2.5; when boundary layer control is applied in the form of blowing or suction over the flap, the C_{Lmax} may rise to as high as 5. When this figure is put into the lift formula under a given set of conditions it can be seen that the amount of lift obtained is greatly increased when compared with that from the plain aerofoil under the same conditions.

1-10 Fig 10 Blown Flap

43. In addition to the use of boundary layer control over the flaps themselves, it can be used simultaneously at the leading edge. In this way even higher lift coefficients can be obtained; the practical limit is set, in the conventional fixed or rotating wing aircraft, by the large amount of power needed to obtain the suction or blowing which is necessary to achieve these high figures. By the use of the maximum amount of boundary layer control, wind tunnel experiments on a full size swept-wing fighter have realized an increase in C_{Lmax} such that the normal 100 kt landing speed was reduced to 60 kt. This is evidence of the importance of the part that the boundary layer plays in aerodynamics.

Vortex Generators

44. Vortex generators (VG) can either take the form of metal projections from the wing surface or of small jets of air issuing normal to the surface. Both types work on the same principle of creating vortices which entrain the faster moving air near the top of the boundary layer down into the more stagnant layer near the surface thus transferring momentum which keeps the boundary layer attached further back on the wing. An advantage of the air jet type is that they can be switched off for those stages of flight when they are not required and thus avoid the drag penalty. There are many shapes for the metal projections, such as plain rectangular plates or aerofoil sections, the exact selection and positioning of which depends on the detailed particular requirement of the designer. When properly arranged, VGs can improve the performance and controllability of the aircraft, particularly at low speeds, in the climb and at high angles of attack. In general, careful design selection can ensure that the increase in lift due to the generators can more than offset the extra drag they cause. Examples of how designers use VGs and other devices to modify the flight characteristics of aircraft can be seen in the case of the Hawk T2.

a. Hawk T2 VGs. The 'double density' vortex generators (VGs) carried on the Hawk T2 aircraft are attached at the 25% chord position and provide a means of mixing higher speed air into the boundary layer at transonic speeds. The term 'double-density' refers to the span-wise spacing which is half that seen on the T.Mk.1 aircraft. The VG array serves to delay the onset of buffet with increasing speed, or with increasing incidence at a fixed Mach number, and helps to fix the position of the shock above the critical Mach number. This ensures that in manoeuvring flight at any Mach number, the tailplane angle per 'g' characteristics remain nominally linear up to a manoeuvre limiting condition. On the Hawk T.Mk.1 aircraft the tailplane angle per 'g' generally decreases at higher 'g' levels. The VG array is sufficiently aft of the leading-edge so as to have no effect at low speed.

b. Hawk T2 Side-mounted Unit Root Fin (SMURF). When the Hawk T1 was developed, it was found that with the flap down at low AOAs a situation could arise where the tailplane lacked the authority to raise the nose and the resultant downhill rush was referred to as "tobogganing" or "phantom dive". To cure the problem on the T1, the flap vane was cut back to reduce the downwash at the tail. When the next generation of Hawk was developed, the loss of lift caused by the cut back flap vane could no longer be tolerated and so an alternate means had to be found to prevent the tobogganing. The problem of providing sufficient pitching moment control, flaps fully deflected and landing gear retracted, was solved by fitting fixed side-mounted strakes on the rear fuselage and parallel to aircraft datum. In this position the tailplane leading edge is adjacent to the trailing edge of the strake when full back stick is applied. When full, or almost full back stick is applied, the strakes create vortex flow over the tailplane lower surface, thus preserving the aerodynamic performance of the tailplane when a high down-load is required.

1-10 Fig 11 – Hawk T2 Vortex Generators and SMURF Locations

Deflected Slipstream

45. A simple and practical method of using engines to assist the wing in the creation of lift is to arrange the engine/wing layout so that the wing is in the fan or propeller slipstream. This, combined with the use of leading edge slats and slotted extending rear flaps, provides a reliable high lift coefficient solution for STOL aircraft. A more extreme example involves the linking together of engines and synchronization of propellers. The wings and flaps are in the slipstream of the engines and gain effectiveness by deflecting the slipstream downwards.

SUMMARY

Devices to Augment Lift

46. The chief devices used to augment C_{Lmax} are:

a. Slats.

b. Flaps.

c. Boundary layer control.

The effect of the slat is to prolong the lift curve by delaying the stall until a higher angle of attack; the effect of flap is to increase the lift by increasing the camber and the principle of boundary layer control is to add kinetic energy to the sub-layers of the boundary layer.

47. Since the slat is of use only at high angles of attack, at the normal angles its presence serves only to increase drag. This disadvantage is overcome by making the slat movable so that when it is not in use it lies flush against the leading edge of the wing.

48. Flaps may be on the leading or trailing edge and many aircraft have them on both. Besides increasing the camber of the wing, many types of flap also increase the wing area. Trailing edge flaps reduce the stalling angle; leading edge flaps increase the stalling angle.

49. The main methods of boundary layer control are:

a. Blowing.

b. Sucking.

c. Vortex generators.

CHAPTER 11 - FLIGHT CONTROLS

General Considerations

1. All aircraft have to be fitted with a control system that will enable the pilot to manoeuvre and trim the aircraft in flight about each of its three axes.

2. The aerodynamic moments required to rotate the aircraft about each of these axes are usually produced by means of flap-type control surfaces positioned at the extremities of the aircraft so that they have the longest possible moment arm about the CG.

3. There are usually three separate control systems and three sets of control surfaces, namely:

   a. Rudder for control in yaw.

   b. Elevator for control in pitch.

   c. Ailerons for control in roll (the use of spoilers for control in roll is also discussed in this chapter).

4. On some aircraft the effect of two of these controls is combined in a single set of control surfaces. Examples of such combinations include:

   a. **Elevons.** Elevons combine the effects of ailerons and elevators.

   b. **Ruddervator.** A ruddervator is a Vee or butterfly tail, combining the effects of rudder and elevators.

   c. **Tailerons**. Tailerons are slab tail surfaces that move either together, as pitch control, or independently for control in roll.

5. It is desirable that each set of control surfaces should produce a moment only about the corresponding axis. In practice, however, moments are often produced about the other axes as well, eg adverse yaw due to aileron deflection. Some of the design methods used to compensate for these cross-effects are discussed in later paragraphs.

Control Characteristics

6. **Control Power and Effectiveness.** The main function of a control is to allow the aircraft to fulfil its particular role. This aspect is decided mainly by:

   a. Size and shape of the control.

   b. Deflection angle.

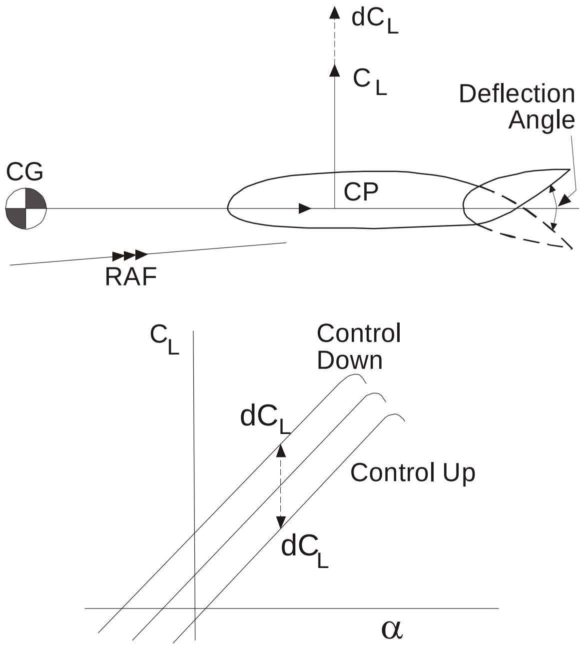
   c. Equivalent Air Speed ^{2} (EAS^2).

   d. Moment arm (distance from CG).

In practice, the size and shape of the control surface are fixed and, since the CG movement is small, the moment arm is virtually constant. The only variables in control effectiveness are speed and effective deflection angle.

7. **Control Moment.** Consider the effect of deflecting an elevator downwards, the angle of attack and the camber of the tailplane are both increased, thereby increasing the C_L value. The moment produced by the tail is the product of the change in lift (tail) $\times$ the moment arm to the CG (see Fig 1).

1-11 Fig 1 Effect of Elevator Deflection on the C_L Curve



8. **Effect of Speed**. The aerodynamic forces produced on an aerofoil vary as the square of the speed (see Volume 1, Chapter 2); it follows then, that for any given control deflection, the lift increment - and so the moment - will vary as the square of the speed, EAS^2. The deflection angle required to give an attitude change, or response, is inversely proportional to the EAS^2 . To the pilot this means that when the speed is reduced by half, the control deflection is increased fourfold to achieve the same result. This is one of the symptoms of the level flight stall, erroneously referred to as 'sloppy controls'.

9. **Control Forces**. When a control surface is deflected, eg down elevator, the aerodynamic force produced by the control itself opposes the downwards motion. A moment is thus produced about the control hinge line, and this must be overcome in order to maintain the position of the control. The stick force experienced by the pilot depends upon the hinge moment and the mechanical linkage between the stick and the control surface. The ratio of stick movement to control deflection is known as the stick-gearing, and is usually arranged so as to reduce the hinge moment.

Aerodynamic Balance

10. If control surfaces are hinged at their leading edge, and allowed to trail from this position in flight, the forces required to change the angle on all except light and slow aircraft would be prohibitive. To assist the pilot to move the controls in the absence of powered or power-assisted controls, some degree of aerodynamic balance is required.

11. In all its forms, aerodynamic balance is a means of reducing the hinge moment and thereby reducing the physical effort experienced in controlling an aircraft. The most common forms of aerodynamic balance are:

a. Horn balance.

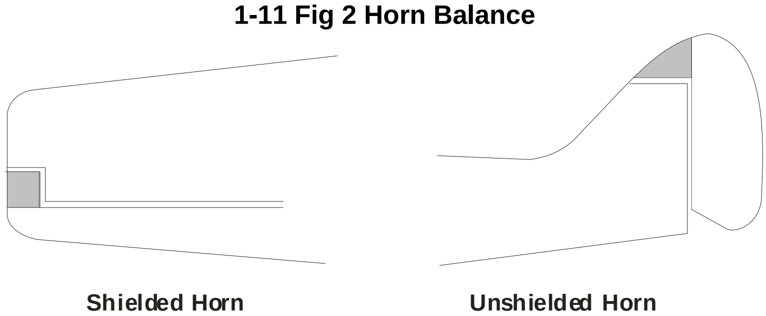
b. Inset hinge.

c. Internal balance.

d. Various types of tab balance.

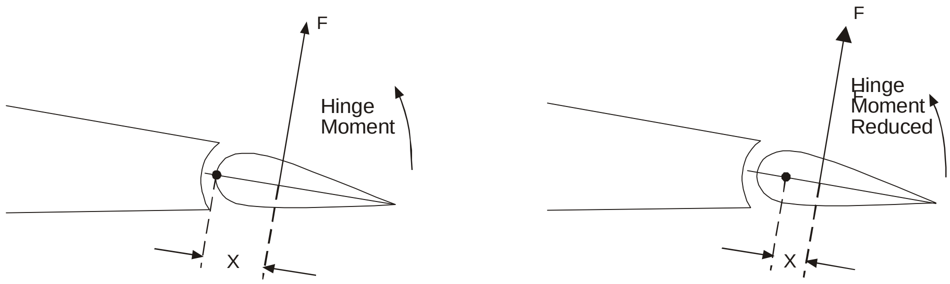
12. **Horn Balance.** On most control surfaces, especially rudders and elevators, the area ahead of the hinge is concentrated on one part of the surface in the form of a horn (see Fig 2). The horn thus produces a balancing moment ahead of the hinge-line. In its effect the horn balance is similar to the inset hinge.

1-11 Fig 2 Horn Balance



13. **Inset Hinge.** The most obvious way to reduce the hinge moment (see Fig 3a) is to set the hinge-line inside the control surface thus reducing the moment arm. The amount of inset is usually limited to 20% to 25% of the chord length, this ensures that the CP of the control will not move in front of the hinge at large angles of control deflection (see Fig 3b).

1-11 Fig 3 Hinge Moment and Inset Hinge



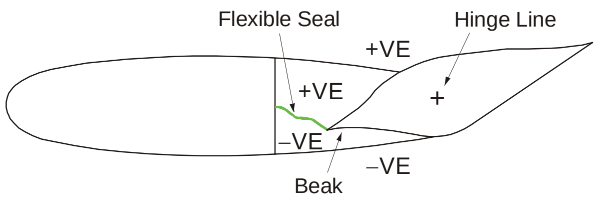
a Hinge Moment = FX

b Inset Hinge

14. **Over Balance**. Should the control CP move ahead of the hinge line, the hinge moment would assist the movement of the control and the control would then be over-balanced.

15. **Internal Balance.** Although fairly common in use, this form of aerodynamic balance is not very obvious because it is contained within the contour of the control. When the control is moved there will be a pressure difference between upper and lower surfaces. This difference will try to deflect the beak ahead of the hinge-line on the control producing a partial balancing moment. The effectiveness is controlled in some cases by venting air pressure above and below the beak (see Fig 4).

1-11 Fig 4 Internal Balance



16. **Tab Balance.** This subject is fully covered in Volume 1,Chapter 12, Trimming and Balance Tabs.

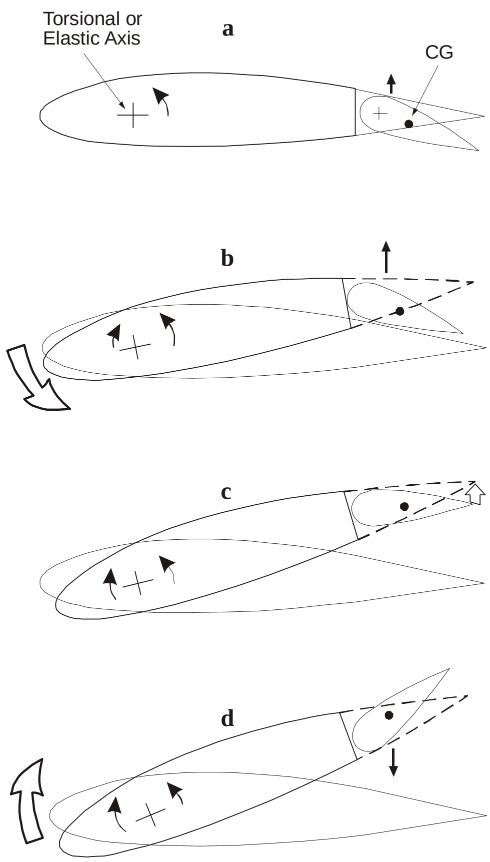
Mass Balance - Flutter

17. **Torsional Aileron Flutter**. This is caused by the wing twisting under loads imposed on it by the movement of the aileron. Fig 5 shows the sequence for a half cycle, which is described as follows:

a. The aileron is displaced slightly downwards, exerting an increased lifting force on the aileron hinge.

b. The wing twists about the torsional axis (the trailing edge rising, taking the aileron up with it). The CG of the aileron is behind the hinge line; its inertia tends to make it lag behind, increasing aileron lift, and so increasing the twisting moment.

1-11 Fig 5 Torsional Aileron Flutter



c. The torsional reaction of the wing has arrested the twisting motion but the air loads on the aileron, the stretch of its control circuit, and its upward momentum, cause it to overshoot the neutral position, placing a down load on the trailing edge of the wing.

d. The energy stored in the twisted wing and the reversed aerodynamic load of the aileron cause the wing to twist in the opposite direction. The cycle is then repeated. Torsional aileron flutter can be prevented either by mass-balancing the ailerons so that their CG is on, or slightly ahead of, the hinge line, or by making the controls irreversible. Both methods are employed in modern aircraft; those aircraft with fully powered controls and no manual reversion do not require mass-balancing; all other aircraft have their control surfaces mass-balanced.

18. **Flexural Aileron Flutter.** Flexural aileron flutter is generally similar to torsional aileron flutter, but is caused by the movement of the aileron lagging behind the rise and fall of the outer portion of the

wings as it flexes, thus tending to increase the oscillation. This type of flutter is prevented by mass-balancing the aileron. The positioning of the mass-balance weight is important; the nearer the wing tip, the smaller the weight required. On many aircraft, the weight is distributed along the whole length of the aileron in the form of a leading edge spar, thus increasing the stiffness of the aileron and preventing a concentrated weight starting torsional vibrations in the aileron itself.

19. **Other Control Surfaces**. So far only wing flutter has been discussed, but a few moments consideration will show that mass-balancing must be applied to elevators and rudders to prevent their inertia and the springiness of the fuselage starting similar troubles. Mass-balancing is extremely critical; hence to avoid upsetting it, the painting of aircraft markings etc is no longer allowed on any control surface. The danger of all forms of flutter is that the extent of each successive vibration is greater than its predecessor, so that in a second or two the structure may be bent beyond its elastic limit and fail.

Control Requirements

20. The main considerations are outlined briefly below:

a. **Control Forces.** If the stick forces are too light the pilot may overstress the aircraft, whereas if they are too heavy he will be unable to manoeuvre it. The effort required must be related to the role of the aircraft and the flight envelope.

b. **Control Movements.** If the control movements are too small the controls will be too sensitive, whereas if they are too large the designer will have difficulty in fitting them into the restricted space of the cockpit.

c. **Control Harmony.** An important factor in the pilot’s assessment of the overall handling characteristics of an aircraft is the 'harmony' of the controls with respect to each other. Since this factor is very subjective it is not possible to lay down precise quantitative requirements. One method often used is to arrange for the aileron, elevator and rudder forces to be in the ratio 1:2:4.

Control Response

21. **Ailerons.** Consider the effect of applying aileron. Deflection of the controls produces a rolling moment about the longitudinal axis and this moment is opposed by the aerodynamic damping in roll (the angle of attack of the down-going wing is increased while that of the up-going wing is decreased, see Volume 1, Chapter 17). The greater the rate of roll, the greater the damping. Eventually the rolling moment produced by the ailerons will be exactly balanced by the damping moment and the aircraft will attain a steady rate of roll. Usually the time taken to achieve the 'steady state' is very short, probably less than one second. Thus, over most of the time that the ailerons are being used, they are giving a steady rate of roll response and this is known as steady state response. The transient response is that which is experienced during the initiation period leading to a steady manoeuvre.

22. **Rate versus Acceleration Control**. A conventionally-operated aileron is therefore described as a rate control, that is the aircraft responds at a steady rate of movement for almost all of the time. The stick force required to initiate a manoeuvre may be less than, or greater than, the stick force required to sustain the manoeuvre, eg rolling. A favourable response is usually assumed to be when the initiation force is slightly greater than the steady force required, the difference being up to 10%. On some modern aircraft the transient response time to aileron deflection is increased. By the time the steady rate of roll is obtained the aileron is being taken off again. In these circumstances the aileron control is producing a rate of roll response which is always increasing and it is then described as an acceleration control.

23. **Elevators**. When the elevators are used, they produce a pitching moment about the lateral axis. The resulting pitching movement is opposed by the aerodynamic damping in pitch and by the longitudinal stability of the aircraft (see Volume 1, Chapter 17). The response to the elevator is a steady state change in angle of attack with no transient time, that is a steady change of attitude. Elevators are therefore described as a displacement control.

24. **Rudders.** The yawing moment produced by rudder deflection is opposed by aerodynamic damping in yaw and by the direction of the aircraft. The response to the rudder is a steady state of change of angle of attack on the keel surfaces of the aircraft, with no transient time. The rudder control has a similar response to the elevator and is therefore also described as a displacement control.

Steady State Response

25. In Volume 1, Chapter 17, it is shown that stability opposes manoeuvre; more precisely, the steady state response to control deflection is greatly affected by the static stability. This is easily seen when considering the response to rudder; the heavy rudder force required to sustain a steady yawing movement is a result of the strong directional static stability of most aircraft.

26. The steady state response to controls is of greater importance to the pilot and this subject is discussed in the following paragraphs.

PRIMARY CONTROL SURFACES

Elevators

27. The elevators are hinged to the rear spar of the tailplane and are connected to the control column so that forward movement of the column moves the elevator downwards and backward movement moves the elevator upwards. When the control column is moved back and the elevator rises, the effect is to change the overall tailplane/elevator section to an inverted aerofoil which supplies a downward force on the tail of the aircraft and, as seen by the pilot, raises the nose. The opposite occurs when a forward movement is made.

28. Elevators are normally free from undesirable characteristics, but large stick forces may be experienced on some aircraft if the aerodynamic balance of the elevators or the stability characteristics of the aircraft are at fault. The tail moment arm is determined by the position of CG, and to retain satisfactory handling characteristics throughout the speed range, the CG position must be kept within a certain limited range. If the CG moves too far forward, the aircraft becomes excessively stable and the pilot will run out of up-elevator before reaching the lowest speeds required.

29. Consider an aircraft in the round-out and landing where longitudinal static stability opposes the nose-up pitch. The downwash angle at the tail is much reduced by the ground effect thereby increasing the effective angle of attack at the tail. This results in a reduced tail-down moment in pitch, therefore a greater elevator deflection angle is required to achieve the landing attitude than would be required to achieve the same attitude at height. A forward movement of the CG would add to this problem by increasing the longitudinal stability.

30. Some tailless delta aircraft require a movement aft of the CG (fuel transfer) to ensure that the elevator movement is sufficient to achieve the landing attitude.

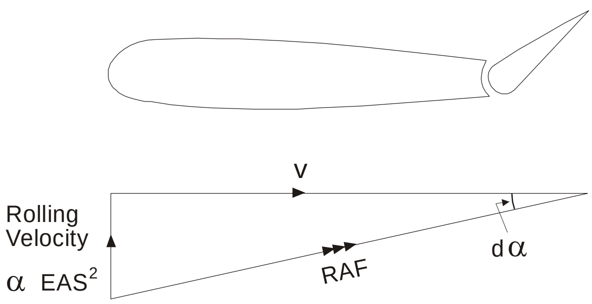
31. Some light aircraft, notably civilian flying club types with a tricycle undercarriage have undersized elevators in order to make the aircraft 'unstallable'.

Ailerons

32. It has been seen that the aileron is a rate control and the rate of roll builds up rapidly to a steady value dictated by the damping in roll effect. The value of the steady rate of roll produced by a given aileron deflection will depend upon the speed and altitude at which the aircraft is flying. Additional effects due to aero-elasticity and compressibility may be present and will modify the roll response.

33. **Effect of Altitude.** In a steady state of roll the rolling force and the damping in roll force are balanced. The rolling force is caused by the change in lift due to aileron deflection and is proportional to the amount of aileron deflection and to EAS. The damping force is due to the change in lift caused by the increase in angle of attack of the downgoing wing and the decrease in angle of attack of the upgoing wing. The value of the damping angle of attack can be found by the vector addition of the TAS and the rolling velocity, as illustrated in Fig 6. It can be seen that for a constant damping angle of attack the rolling force, and therefore rate of roll, will increase in direct proportion to TAS. When an aircraft is climbed at a constant EAS the rate of roll for a given aileron deflection therefore increases because TAS increases with altitude.

1-11 Fig 6 Damping in Roll Effect - Downgoing Wing

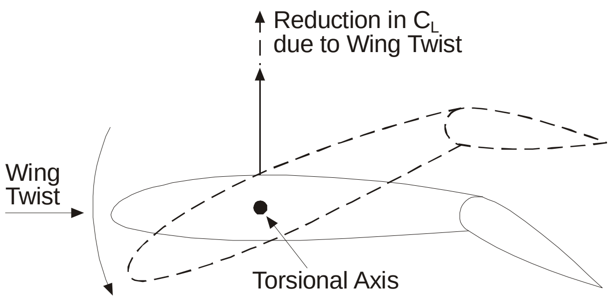


34. **Effect of Forward Speed.** It has been shown that the rolling force is proportional to the amount of aileron deflection and to EAS. It follows that rate of roll increases in proportion to EAS.

35. **Aero-elastic Distortion.** Ailerons are located towards the wing tips by the necessity, in most aircraft, to utilize trailing edge flaps for landing and take-off. This may cause the wing to twist when the ailerons are deflected and lead to an effect known as 'aileron reversal'. It must be recognized that aero-elastic distortion of the airframe may affect stability and control in pitch and yaw as well as in roll. Because the wings are usually the least rigid part of the airframe however, aileron reversal is important and reduces the ultimate rate of roll available at high forward speeds.

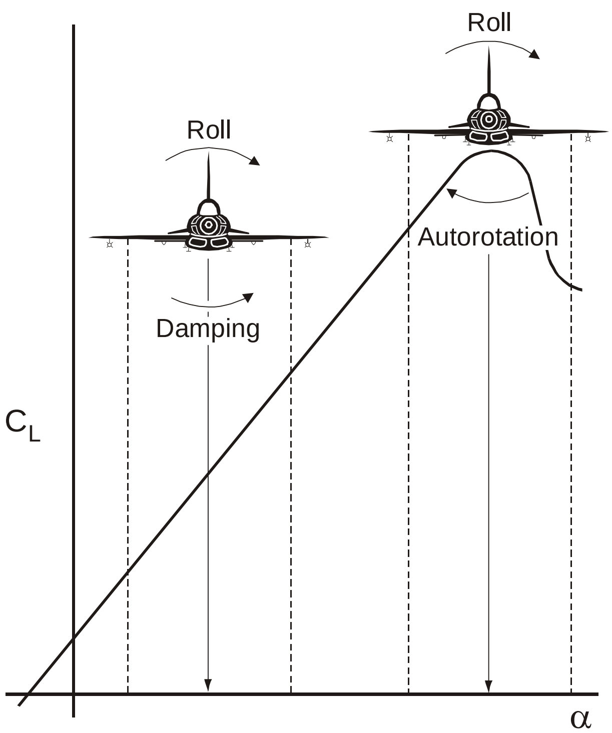
36. **Wing Twisting.** In Fig 7, it can be seen that deflection of the aileron down produces a twisting moment about the torsional axis of the wing. The torsional rigidity of the wing depends on the wing structure but will normally be strong enough to prevent any distortion at low speeds. Aileron power, however, increases as the square of the forward speed, whereas the torsional stiffness in the wing structure is constant with speed. At high speeds therefore, the twisting moment due to aileron deflection overcomes the torsional rigidity of the wing and produces a change in incidence which reduces the rate of roll. On the rising wing, illustrated in Fig 7, the incidence is reduced whereas on the downgoing wing the effect is to increase the incidence. A flight speed may be reached where the increment of CL produced by deflecting aileron is completely nullified by the wing twisting in the opposite sense. At this speed, called reversal speed, the lift from each wing is the same in spite of aileron deflection, and the rate of roll will be zero. At still higher speeds the direction of roll will be opposite to that applied by the pilot. Reversal speed is normally outside the flight envelope of the aircraft but the effects of aero-elastic distortion may be apparent as a reduction in roll rate at the higher forward speeds.

1-11 Fig 7 Aero-elastic Distortion



37. **Aileron Response at Low Speeds.** Deflecting an aileron down produces an effective increase in camber and a small reduction in the critical angle of attack of that part of the wing to which the aileron is attached (usually the wing tip). It is therefore important that the wings should stall progressively from root to tip in order to retain aileron effectiveness at the stall. Rectangular straight wings are not much of a problem because washout is usually incorporated to reduce vortex drag. Swept wings, however, are particularly prone to tip stall and design features may have to be incorporated to retain lateral control at low speeds. Some of these aspects were discussed in Volume 1, Chapter 9. Loss of effectiveness of the down-going aileron due to tip stall will not necessarily result in a reversal in the direction of roll (i.e. wing-drop). The up-going aileron will usually retain its effectiveness and produce a lesser but conventional response. The factor which has the most significant effect on aircraft response at high α is the damping in roll effect. It has been seen that when the aircraft is rolling, the angle of attack of the down-going wing is increased while that of the rising wing is decreased. If the aircraft is close to the stall the damping effect is reversed and the change in the rolling moments assists the rotation, see Fig 8. This leads to the phenomenon known as 'autorotation'.

1-11 Fig 8 Damping in Roll



38. **Adverse Aileron Yaw.** Unless they are carefully designed aerodynamically, the ailerons will materially alter the drag force on the wing in addition to the desired change in lift force. When an aileron is deflected downwards both the vortex drag and the boundary layer drag are increased. On

the aileron-up wing, however, the vortex drag is decreased, though the boundary layer drag may be still increased. The changes in the drag forces are such as to produce a yawing moment causing the aircraft to yaw in the opposite sense to the applied roll. Adverse yaw is produced whenever the ailerons are deflected but the effect is usually reduced by incorporating one or more of the following design features:

a. **Differential Ailerons.** For a given stick deflection the up-going aileron is deflected through a larger angle than the down-going aileron thus reducing the difference in drag and the adverse yaw.

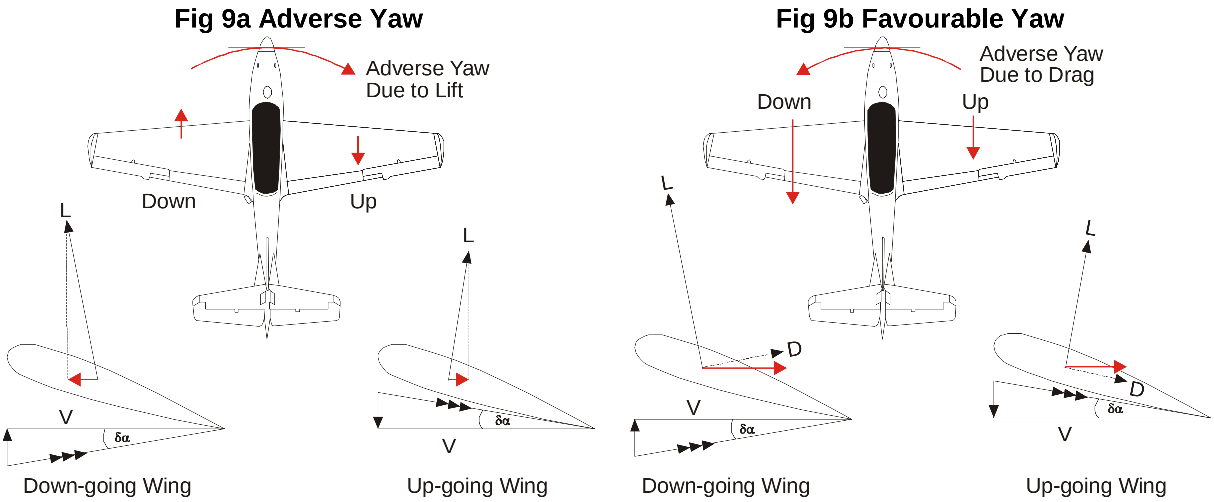
b. **Frise-type Ailerons**. The nose of the upgoing aileron protrudes into the airstream below the wing to increase the drag on the down-going wing. This arrangement has the additional advantage that it assists the aerodynamic balancing of the ailerons.

c. **Coupling of Controls.** A method used in some modern aircraft to overcome the adverse yaw is to gear the rudder to the ailerons so that when the ailerons are deflected the rudder moves to produce an appropriate yawing moment.

d. **Spoilers.** On some aircraft, spoilers in the form of flat plates at right angles to the airflow are used to increase the drag of the down-going wing. Spoilers are sometimes the only form of lateral control used at high speeds (eg the ailerons are used at low speeds, where spoilers are least effective, but spoilers alone at high speeds). Other uses of spoilers are as airbrakes, when the spoilers of both wings operate together, or as lift dumpers, when the aircraft has landed. They are usually hydraulically operated.

39. **Cross-coupling Response.** When an aircraft is rolling the increased angle of attack of the down-going wing increases both the lift and the drag, whereas on the up-going wing the lift and drag are reduced. The lift and drag forces are by definition, however, perpendicular and parallel respectively to the local relative airflow on each wing. The projections of the lift forces onto the yawing plane produce a yawing moment towards the rising wing (ie an adverse yaw), as shown in Fig 9a. This is partially offset by the projections of the drag forces onto the yawing plane which usually produce a yaw in the same direction as roll (Fig 9b). For a given rate of roll the change in angle of attack, $\delta\alpha$, will be greatest at low forward speeds. The adverse yaw due to roll is therefore greatest at low speed and may eventually become favourable at some high forward speed.

1-11 Fig 9 Cross-coupling Response



40. **Response to Sideslip.** The lateral response of the aircraft to sideslip is usually called the 'dihedral effect' and produces a rolling moment opposite to the direction of sideslip. In most conventional aircraft, this contribution is dominated by the yaw due to sideslip.

41. **Overbalance.** This may occur at any airspeed on some aircraft not fitted with power-operated controls, but usually only at the larger control angles. It is shown by a progressive decrease, instead of an increase, of the aileron stick force as the control column is moved, ie a tendency for the ailerons to move to their full travel of their own accord. In some cases this may happen fairly suddenly.

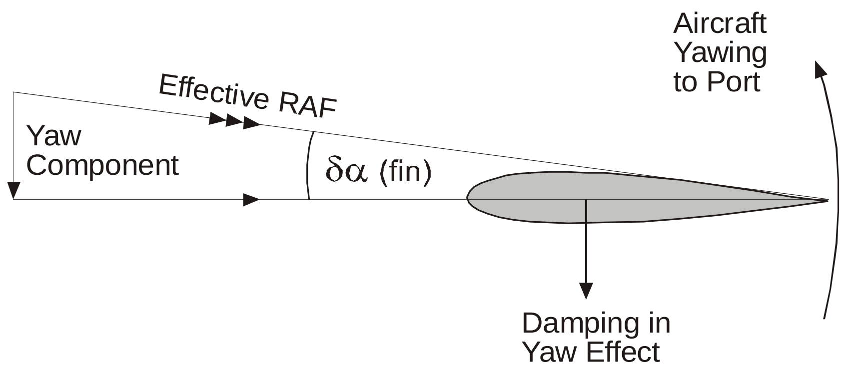
42. **Snatch.** Snatching usually occurs at or near the stall, or at high Mach numbers. It is caused by a continuous and rapid shifting of the centre of pressure of the aileron due to the disruption of the airflow over the surface, resulting in a snatching or jerking of the control, which may be violent.

The Rudder

43. The rudder, which is hinged to the rear of the fin, is connected to the rudder bar. Pushing the right pedal will cause the rudder to move to the right, and in so doing alter the aerofoil section of the fin/rudder combination. This provides an aerodynamic force on the rear of the aircraft which will move it to the left or, as the pilot sees it, the nose will yaw to starboard. Rudder effectiveness increases with speed; whereas a large deflection may be required at low speed to yaw a given amount, a much smaller deflection is needed at the highest speeds. The steady response of an aircraft to rudder deflection is complicated by the fact that yaw results in roll, and rudder deflection may also cause the aircraft to roll.

44. **Damping in Yaw.** Just as there is damping in pitch and roll, so also is the aircraft damped in yaw. The effect is similar in principle to damping in roll, in that the yawing velocity produces a change in the angle of attack of the keel surfaces. Keel surface moments fore and aft of the CG oppose the yawing movement, and these moments exist only while the aircraft is yawing. For simplicity, Fig 10 shows the damping effect on the fin and rudder. The wings also produce a small damping in yaw effect because the outer wing moves faster than the inner wing and therefore produces more drag. The damping in yaw effect decreases with altitude for the reasons stated in para 33.

1-11 Fig 10 Damping in Yaw



45. **Roll Due to Rudder Deflection**. The rudder control inevitably produces a rolling moment because the resultant control force acts above the longitudinal axis of the aircraft. Usually this effect is small but on aircraft with a tall fin and rudder it can be important and is sometimes eliminated by linking up the rudder and aileron circuits.

46. **Cross-coupling Response**. The response of the aircraft in roll to rudder deflection arises from the resulting yawing motion and/or sideslip. The roll due to sideslip, ie the dihedral effect, has a powerful cross-coupling effect, particularly on swept-wing aircraft. Rudder control is often used to pick

up a dropped wing at low speeds in preference to using large aileron deflection. If the starboard wing drops and port rudder is applied, the aircraft sideslips to starboard and the dihedral effect produces a rolling moment tending to raise the lower wing. The roll due to yaw (ie rate of yaw) arises because the outer wing moves faster than the inner wing and therefore produces more lift. The roll with yaw is greatest at high angles of attack/low speeds.

47. **Rudder Overbalance*.*** A fault which may be encountered is overbalance. This is indicated by a progressive lessening of the foot loads with increasing rudder displacement. If, owing to a weakness in design, the aerodynamic balance is too great, it will become increasingly effective as the rudder is moved and may eventually cause it to lock hard over when the centre of pressure moves in front of the hinge-line. At large angles of yaw (sideslip), the fin may stall causing a sudden deterioration in rudder control and directional stability and, at the same time, rudder overbalance. If this is encountered, the yaw must be reduced by banking in the direction in which the aircraft is yawing and not by stabilizing the yaw by instinctively applying opposite bank. The correct action reduces the sideslip by converting the motion into a turn from which recovery is possible once the fin has become unstalled. Sometimes slight apparent rudder overbalance may be noticed under asymmetric power when large amounts of rudder trim are used to decrease the foot load on the rudder bar. If this happens the amount of rudder trim should be reduced.

48. **Rudder Tramping*.*** On some aircraft the onset of rudder overbalance may be shown by 'tramping', or a fluctuation in the rudder foot loads. If the yaw is further increased overbalance may occur.

49. **Elevons and Tailerons*.*** Some aircraft with swept-back wings combine the function of the elevators and ailerons in control surfaces at the wing tips called elevons, or if tail mounted, tailerons. These are designed so that a backward movement of the control column will raise both surfaces and so, acting as elevators, they will raise the nose. If the control column is held back and also moved to one side, the surfaces will remain in a raised position and so continue acting as elevators, but the angular position of each surface changes so that the lift at each wing tip is adjusted to cause a rolling moment in the direction that the control column has been moved. If the control column is held central and then moved to one side the surfaces act as normal ailerons; a subsequent forward movement of the control column will lower both surfaces, maintaining the angular difference caused by the sideways movement of the control column, and the nose will drop while the aircraft is rolling.

The All-moving (Slab) or Flying Tail

50. At high Mach numbers, the elevator loses much of its effectiveness for reasons given in the chapters on high-speed flight. This loss of effectiveness is the cause of a serious decrease in the accuracy with which the flight path can be controlled and in the manoeuvrability. To overcome this deficiency, the tailplane can be made to serve as the primary control surface for control in the looping plane. When this is done some form of power assistance is usually employed to overcome the higher forces needed to move the tailplane during flight. With the flying tail, full and accurate control is retained at all Mach numbers and speeds. Forward movement of the control column increases the incidence of the tailplane to obtain the upward force necessary to lower the nose. On some aircraft the elevator is retained and is linked to the tailplane in such a way that movement of the tailplane causes the elevator, by virtue of its linkage, to move in the usual direction to assist the action of the tailplane. When no elevator is used the whole is known as a slab tailplane.

The Variable Incidence (VI) Tail

51. This is used on some aircraft as an alternative, and sometimes in addition, to trimming tabs. By suitably varying the incidence of the tailplane any out-of-trim forces can be balanced as necessary.

The VI tailplane is generally more effective than tabs at high Mach numbers. Its method of operation is usually electrical; the control being a switch which is spring-loaded to a central off position.

The Vee Tail

52. The Vee, or butterfly, tail is an arrangement whereby two surfaces, forming a high dihedral angle, perform the functions of the conventional horizontal and vertical tail surfaces. The effective horizontal tail area is the area of both surfaces projected on the horizontal plane. The effective vertical tail area is the area of both surfaces projected on the vertical plane. As the lift force from each surface acts normal to its span line, the vertical component acts to provide a pitching moment, and the horizontal component acts to produce a yawing moment. The two moveable portions, therefore, are capable of performing the functions of both the elevator and the rudder. They are sometimes called a ruddervator. It can be seen that if both surfaces are moved up or down an equal amount, the net result is a change in the vertical force component only, and a pitching moment only results. If the two surfaces are moved equal amounts in opposite directions, the result is a change in the net horizontal force component only, and a yawing moment only results. Any combination of the two movements results in combined pitching and yawing moments.

53. The elevator control in the cockpit is connected to give an equal deflection in the same direction. The rudder control is connected to give equal deflections in opposite directions to the two surfaces. The two cockpit controls are connected to the two surfaces through a differential linkage or gearing arrangement. Thus the Vee tail performs the horizontal and vertical tail control functions with normal cockpit controls. Some of the advantages claimed are:

a. Weight saving - less total tail surface required.

b. Performance gain - less total tail surface and lower interference drag, as only two surfaces intersect the fuselage.

c. Removal of the tail from the wing wake and downwash.

d. Better spin recovery, as the unblanketed portion of the tail acts both to pitch the aircraft down and to stop the rotation.

AIRBRAKES

General

54. Jet engine aircraft, having no propeller drag when the engine is throttled back, have comparatively low drag and lose speed only slowly. Further, having eventually reached the desired lower speed, any slight downward flight path causes an immediate and appreciable increase in speed.

55. An 'air brake' is an integral part of the airframe and can be extended to increase the drag of an aircraft at will, enabling the speed to be decreased more rapidly, or regulated during a descent. On some aircraft the undercarriage may be lowered partially, or completely, to obtain the same effect.

56. Although the area of the air brakes on a typical fighter is small, considerable drag is produced at high speeds. For example, an air brake with an assumed C_D of 1.2 and a total area of about 2.5 sq ft produces a drag of about 5,700 lb when opened at 500 kt at sea level. This figure is indicative of the large loads imposed on an aircraft when flying at high indicated speeds. The effectiveness of an air brake varies as the square of the speed and therefore at about 120 kt the same air brake gives a drag of about 330 lb only. The decelerating effect of air brakes can be seen from figures

obtained from an aircraft flying at 400 kt at low altitude. With the air brakes in, and power off, the aircraft takes 2 min 58 sec to slow to 150 kt; with air brakes out the time is reduced to 1 min 27 sec.

57. Ideally, air brakes should not produce any effect other than drag, although on some American aircraft the air brakes are designed to produce an automatic nose-up change of trim when opened. In practice, however, the opening of most air brakes is accompanied by some degree of buffet, with or without a change of trim; the strength of these adverse effects is usually greatest at high speeds, becoming less as the speed decreases.

Effect of Altitude on Effectiveness

58. Air brakes derive their usefulness from the fact that they are subjected to dynamic pressures (the $\frac{1}{2}\rho V^2$ effect) and so provide drag in proportion to their area. At high altitudes therefore, the effectiveness of all air brakes is much reduced since the drag, which is low in proportion to the TAS, takes longer to achieve a required loss in speed, ie the rate of deceleration is reduced.

59. An air brake which develops, say, 2,000 lb drag at a stated IAS/TAS at sea level will develop the same drag at the same IAS at high altitude; but whereas the TAS at sea level was equal to the IAS, the TAS at altitude may be as much as 2 or more times the IAS. Since the drag is required to decrease the kinetic energy, which is proportional to the TAS, it is apparent that the decelerating effect of the air brake (proportional to the IAS) is decreased, eg the time taken to decelerate over a given range of IAS will be doubled at 40,000 ft compared to that at sea level since the IAS at 40,000 ft is about half the TAS.

BRAKE PARACHUTES

General

60. Brake parachutes are used to supplement the aircraft's wheel brakes and so reduce the length of the landing run. In general, they produce enough drag to cause a steady rate of deceleration varying from about 0.25g to 0.35g, depending upon the particular installation. Below around 70 kt, the drag, varying as the square of the speed, falls to a much lower figure and the wheel brakes become the primary means of deceleration.

Parachute Diameter

61. The diameter of the parachute depends on the weight and size of the aircraft. For aircraft with a landing weight of around 5,000 kg, the 'flying' diameter of the parachute is from 2 m to 3 m. At a touchdown speed of 130 kt, this gives a drag of about 1,250 kg and a deceleration rate of about 0.25g.

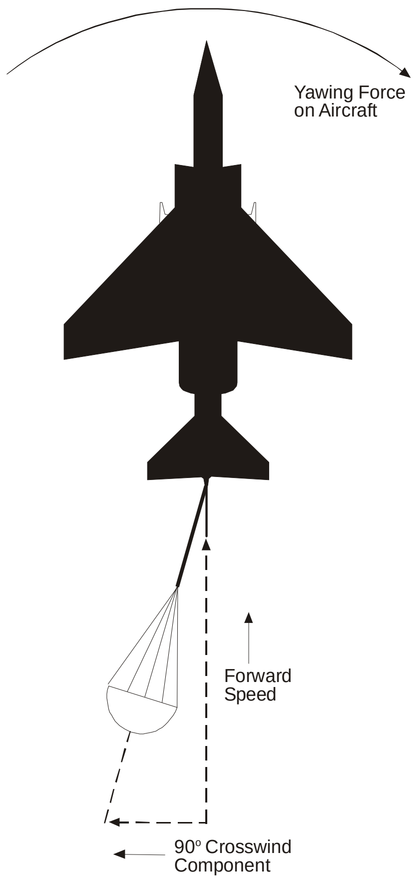
62. For large aircraft, with landing weights around 50,000 kg, the flying diameter of the brake parachute is about 11 m. This produces, at a touchdown speed of about 150 kt, a drag of some 25,000 kg and an initial rate of deceleration of about 0.35g.

Crosswind Landings

63. When used in a crosswind landing, the parachute aligns itself along the resultant between the vectors representing the forward speed of the aircraft and the 90° component of the crosswind vector (see Fig 11). This causes a yawing moment which increases the weather-cock characteristics of the aircraft. For 90°

crosswind components of up to 20 kt, the effect is small and can easily be countered by the pilot; at higher crosswind speeds it becomes progressively more difficult to keep the aircraft straight. As the aircraft decelerates, the flying angle between the parachute and the centre line of the aircraft increases, but the retarding force of the parachute will decrease with the reduction in V^2 . Thus the overall effect of decreasing speed is a reduction in the weathercocking tendency due to the brake parachute.

1-11 Fig 11 Effect of Crosswind When Using a Brake Parachute



64. Any difficulty in directional control tends to occur fairly early in the landing run, although not necessarily at the highest speed; this is mainly due to the very small flying angle and the relatively high rudder effectiveness.Tyre sideslip due to the side load imposed by the parachute may be another significant factor.

Jettisoning of Brake Parachute

65. At any time after the parachute has been streamed, it can be disconnected or jettisoned by the pilot in an emergency. It is also usual to jettison the parachute at the end of the landing run.

Inadvertent Streaming in Flight

66. If the parachute is opened inadvertently, at high speed, the opening load causes failure of a weak link and the parachute breaks away from the aircraft. If opened too early on the approach it can be disconnected by the pilot.

SUMMARY

Flight Controls

67. The main control surfaces are:

    a. Rudder.

    b. Elevator.

    c. Aileron.

On some aircraft, the effect of two of these controls is combined in a single set of control surfaces, e.g. elevons, ruddervator and tailerons (tail mounted 'elevons').

68. Control power and effectiveness is decided by:

    a. Size and shape of the control.

    b. Deflection angle.

    c. EAS^2.

    d. Moment arm.

69. Aerodynamic balance is a means of reducing the hinge moment and thereby reducing the physical effort experienced in controlling an aircraft. The most common forms of aerodynamic balance are:

    a. Horn balance.

    b. Inset hinge.

    c. Internal balance.

    d. Various types of tab balance.

70. Flutter can be prevented by:

    a. Mass balancing.

    b. Increasing structural rigidity.

    c. Irreversible controls.

71. The main considerations for control requirements are:

    a. Control force.

    b. Control movement.

    c. Control harmony.

72. Different control responses are given different terms. The ailerons are normally described as a rate control, and the rudder and elevator as displacement controls.

73. Methods of overcoming adverse aileron yaw are:

    a. Differential ailerons.

    b. Frise ailerons.

    c. Control coupling.

    d. Spoilers.

74. Some advantages claimed for the Vee tail are:

a. Weight saving.

b. Performance gain.

c. Removal of the tail from the wing wake and downwash.

d. Better spin recovery.

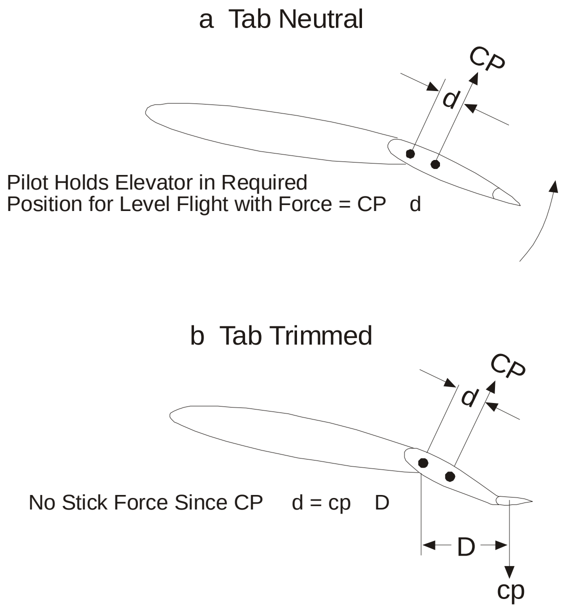
CHAPTER 12- TRIMMING AND BALANCE TABS

Introduction

1. The previous chapter covered flight controls and the methods by which these can be balanced. This chapter is really an extension of the last, but deals mainly with tabs, discussing the more important types and their method of operation.

2. A tab is the small, hinged, surface forming part of the trailing edge of a primary control surface. The principle of operation is described below and shown in Fig 1.

1-12 Fig 1 Principle of the Tab



3. Assume that the aircraft is slightly tail heavy and therefore requires a constant push force to maintain level flight (Fig 1a). If the position of the elevator trimming tab is then adjusted so that it is moved upwards, the result is a download on the elevator trailing edge, which moves the elevator down (Fig 1b). The downward movement continues until the downward moment of the tab is balanced by the upward moment of the elevator.

Fixed Tabs

4. Fixed tabs can only be adjusted on the ground and their setting determined by one or more test flights. Thus, under conditions of no stick force, the trailing position of the control surfaces is governed by the position of the tab (see Fig 2a). This type of tab is used on the ailerons of some aircraft and sometimes on the rudders of single engine jet aircraft.

5. An early form of fixed tab, sometimes still used on light aircraft, consists of small strips of cord doped above or below the trailing edge of the control surface. A strip of cord above the trailing edge deflects the surface downwards, the amount of deflection depending on the length and diameter of the cord used.

Trim Tabs

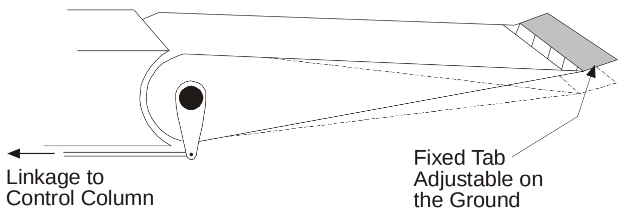
6. Trim tabs are used to trim out any holding forces encountered in flight, such as those occurring after a change of power or speed, or when the CG position changes owing to fuel consumption, dropping bombs or expending ammunition. Whenever the speed, power or CG position is altered, one at a time or in combination, the changed trim of the aircraft necessitates resetting of the tabs.

7. **Operation of Trim Tabs**. The tabs are pilot operated, either by handwheels in the cockpit or electrically (see Fig 2b). The handwheels always operate in the natural sense, ie the top of the elevator trim handwheel is moved forward to produce a nose-down trim change and vice versa. Tabs may also be electrically operated by small switches spring-loaded to the central (off) position; to eliminate a holding force the pilot moves the appropriate switch in the same direction as the holding force until the stick load is zero, then releases the switch.

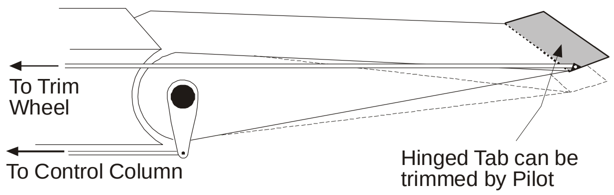
8. **Variation of Tab Effectiveness with Speed**. The sensitivity and power of a trim tab varies with speed in the same way as the control surfaces. At low speeds large tab deflections may be required to trim an aircraft, but at high speeds small movements have a marked and immediate effect on the trim.

1-12 Fig 2 Fixed and Trim Tabs

a Fixed Tab



b Trim Tab



Other Methods of Trimming

9. **Spring-Bias Trimmers**. Trimming is also done on some light aircraft by adjusting the tension in springs which are connected directly to the pilot's controls. Thus an adjustment in the spring tension will bias the control column or rudder bar in any desired position.

10. **Variable Incidence Tailplanes and All Flying (Slab) Tailplanes.** These are used to counter large changes in trim in the pitching plane. They are described in Volume 1, Chapter 11.

11. **Variable Incidence Wings**. Strictly speaking, the variable incidence wing is not a trimming device at all but a means of altering the attitude of the aircraft in flight. On swept-wing aircraft which utilize high lift devices on the leading edge, the net effect is to increase the stalling angle (see Volume 1, Chapter 10). A disadvantage of this is the nose high attitude of the aircraft in the take-off and landing configuration, with attendant poor cockpit vision and the requirement for long undercarriages. The variable incidence wing overcomes this disadvantage by the use of a variable rigging angle of incidence, controllable by the pilot, which allows a large angle of attack on the wing without a nose high attitude to the fuselage.

BALANCE TABS

General

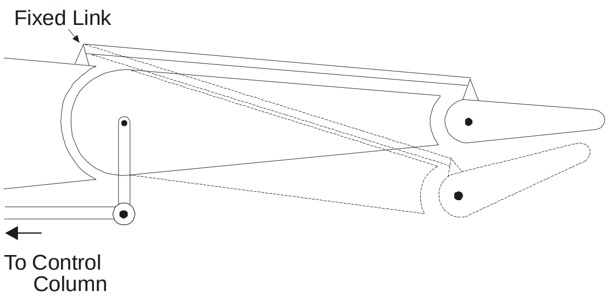
12. Balance tabs are used to balance or partially balance the aerodynamic load on the control surfaces, thus reducing stick loads. In its basic form, this type of tab is not under the control of the pilot, but the tab angle is changed automatically whenever the main control surface is moved. The cross-sectional shape of the control surface may have an important effect on aerodynamic balance. A convex shape can tend towards overbalance whilst a concave shape ('hollow-ground') can have the opposite effect. The latter shape is sometimes achieved by fitting narrow flat plates to the trailing edge of the tab to give the 'hollow-ground' effect.

Geared Tabs

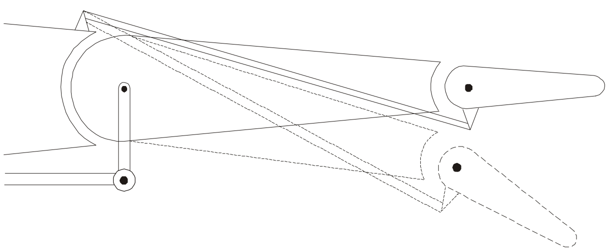
13. The geared tab is linked with the fixed surface ahead of the control surface and is geared to move at a set ratio, in the opposite direction to the movement of the control (Fig 3a). Since the tab moves in the opposite sense to the control, it produces a small loss of lift, but the reduction in control effectiveness is usually negligible.

1-12 Fig 3 Geared Tabs

a Balance Tab



b Anti-balance Tab



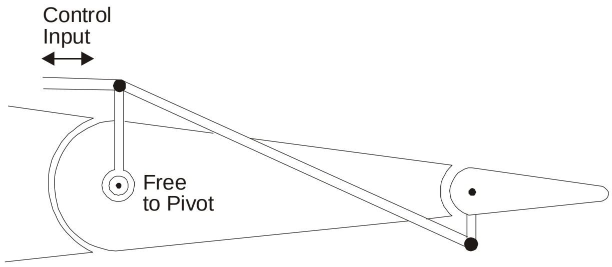
14. It is sometimes necessary to incorporate a large horn balance to improve the stick-free stability, and it may then be necessary to fit an anti-balance tab to increase the stick force (Fig 3b).

Servo Tabs

15. By connecting the tabs directly to the cockpit controls, the tab can be made to apply the hinge moment required to move the control surface. The pilot's control input deflects the tab and the moment produced about the hinge line of the control surface causes this surface to 'float' to its position of equilibrium. The floating control will then produce the required moment about the CG of the aircraft. The stick forces involved are only those arising from the hinge moments acting on the tab, which are much less than those on the main control surface. A schematic diagram of a servo tab is shown in Fig 4. One of the chief disadvantages of the simple servo tab is that it lacks effectiveness at low speeds, since any tab loses most of its effect when deflected through an angle of more than about 20°. The large control surface deflections required at low speeds need correspondingly large tab deflections and

therefore this system is not always satisfactory. The spring tab, described in para 16, is used to overcome this fault.

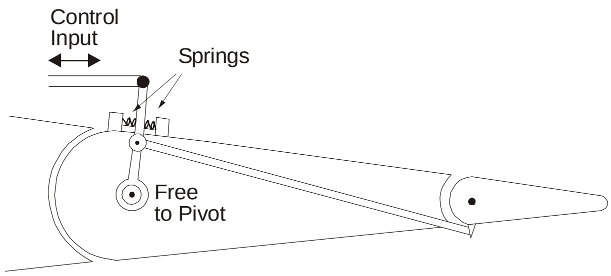
1-12 Fig 4 The Servo Tab System



Spring Tabs

16. The spring tab is a modification of the geared tab in which a spring is incorporated in the linkage permitting the tab gearing to be varied according to the applied stick forces. The actual arrangement of the mechanism may vary with different designs according to the space limitations where it is to be applied. A schematic arrangement of a spring tab is shown in Fig 5.

1-12 Fig 5 The Spring Tab System



17. Movement of the input rod deflects the tab against spring tension. The input force is transmitted through the spring to the control surface which moves as a result of the combined effect of input force and aerodynamic assistance provided by the tab.

18. The amount of servo-action depends on the rate (strength) of the spring employed. It can be seen that an infinitely strong spring produces no assistance from the tab whereas an infinitely weak spring causes the tab to behave as a servo-tab. The effect of the spring tab can be regarded as producing a change in the basic speed squared law.

19. Spring tabs may be pre-loaded to prevent them from coming into operation until the stick (or rudder) force exceeds a predetermined value. This is done to keep the spring tab out of action at low speeds, thus avoiding excessive lightening and lack of feel. A disadvantage of this arrangement is the sudden change in stick force gradient that occurs when the tab comes into operation.

20. The spring tab has been widely used in modern aircraft with much success. On very large surfaces, however, the spring tab or servo-tab may produce a 'spongy' feel in the controls to which the pilot may object. On the ground, when the control column is held in one position, the control surfaces can be moved by hand.

Dual Purpose Tabs

21. In some aircraft, the features of two or more of the foregoing types of tabs are combined. For example, servo, geared or spring tabs can be connected to the pilot-operated trim wheel so that the

basic position of the tab in relation to the control surface can be varied. In this way the tab functions both as a trim tab and as a pure servo, geared or spring tab.

Control Locks on Servo and Spring Tab Control Surfaces

22. On aircraft using servo or spring tabs, locking the control column or rudder bar will not prevent high winds from moving the control surfaces. In these cases, external control surface clamps are essential.

23. On certain spring tab installations, partial or full movement of the control column or rudder bar is possible when the external clamps are still in position. For this reason, it is vital that the clamps are removed before flight and a visual check of the correct movement of the control surfaces is made when the control column is moved. If the surface cannot be seen from the cockpit, movement should be checked with the assistance of the ground crew. Any restriction in the movement of the control column, other than the normal friction, must be investigated before flight.

CHAPTER 13 - LEVEL FLIGHT

Introduction

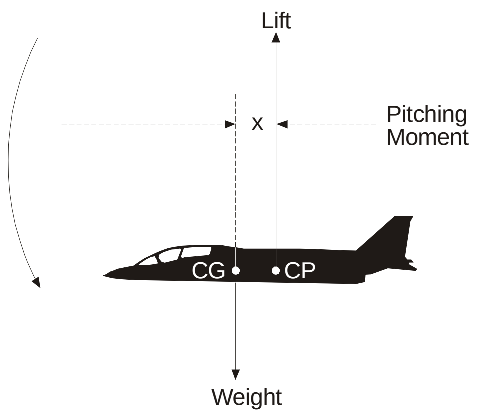
1. During flight four main forces act upon an aircraft: lift, weight, thrust and drag. When an aircraft is in steady level flight a condition of equilibrium must prevail. The unaccelerated condition of flight is achieved with the aircraft trimmed for lift to equal weight and the engine set for thrust to equal the aircraft drag.

2. The four forces mentioned in para 1 each have their own point of action: lift through the centre of pressure (CP), weight through the centre of gravity (CG), and for the purposes of this chapter (although not strictly valid), thrust and drag act in opposite directions parallel to the direction of flight through points varying with aircraft design and attitude. Although the opposing forces are equal there is a considerable difference between each pair of forces. The lift/drag ratio for example may be as low as ten to one depending on the speed and power used.

Pitching Moments

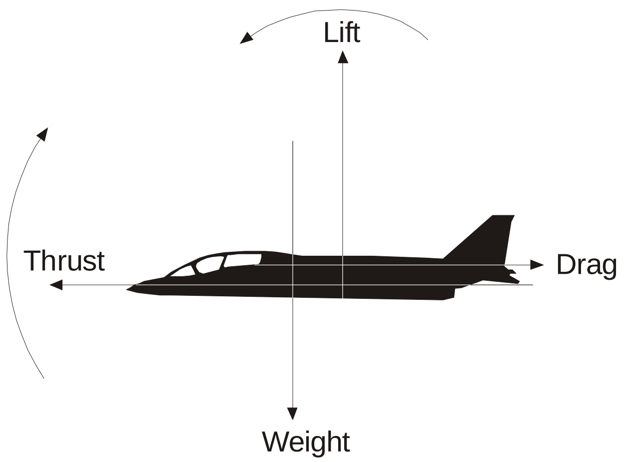
3. The positions of the CP and CG are variable and under most conditions of level flight they are not coincident. The CP changes its position with change in angle of attack and the CG with reduction in fuel or when stores are expended. The outcome is that the opposing forces of lift and weight set up a couple causing either a nose-up or a nose-down pitching moment depending on whether the lift is in front of, or behind, the CG, as illustrated in Fig 1. The same consideration applies to the position of the lines of action of the thrust and drag. Ideally, the pitching moments arising from these two couples should neutralize each other in level flight so that there is no residual moment tending to rotate the aircraft.

1-13 Fig 1 Pitching Moment



4. This ideal is not easy to attain by juggling with the lines of action of the forces alone but, as far as possible, the forces are arranged as shown in Fig 2. With this arrangement the thrust/drag couple produces a nose-up moment and the lift/weight couple a nose-down moment, the lines of action of each couple being positioned so that the strength of each couple is equal. If the engine is throttled back in level flight the thrust/drag couple is weakened and the lift/weight couple pitches the nose down so that the aircraft assumes a gliding attitude. If power is then re-applied the growing strength of the thrust/drag couple raises the nose towards the level flight attitude.

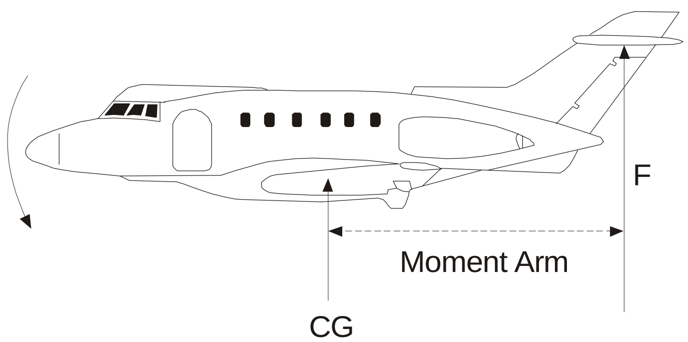
1-13 Fig 2 Disposition of the Four Forces



Tailplane and Elevator

5. The function of a tailplane is to supply any force necessary to counter residual pitching moments arising from inequalities in the two main couples, ie it has a stabilizing function. The force on the tailplane need only be small because the fact that the tailplane is positioned some distance from the CG means that it can apply a large moment to the aircraft (Fig 3). For this reason the area and lift of the tailplane is small compared with the mainplanes.

1-13 Fig 3 Action of Tailplane



6. In level flight, if any nose-up or nose-down tendency occurs, the elevator position can be altered to provide an upward or downward force respectively to trim the aircraft. If the tailplane, or trailing edge, has to produce a down-load balancing force this will add to the apparent weight of the aircraft and to maintain level flight at the same speed the angle of attack must be increased to increase the lift. The associated increase in drag is known as trim drag.

Variation of Speed in Level Flight

7. For level flight the lift must equal the weight. From the lift formula ($L = C_L \frac{1}{2} \rho V^2 S$), it can be seen that, for a given aircraft flying at a stated weight, if the speed factor is decreased, then the lift coefficient (angle of attack) must be increased to keep the lift at the same value as the weight.

8. **Aircraft Attitude in Level Flight.** At low speed the angle of attack must be high, while at high speed only a small angle of attack is needed to obtain the necessary amount of lift. Since level flight is being considered, these angles become evident to the pilot as an attitude which will be markedly nose-

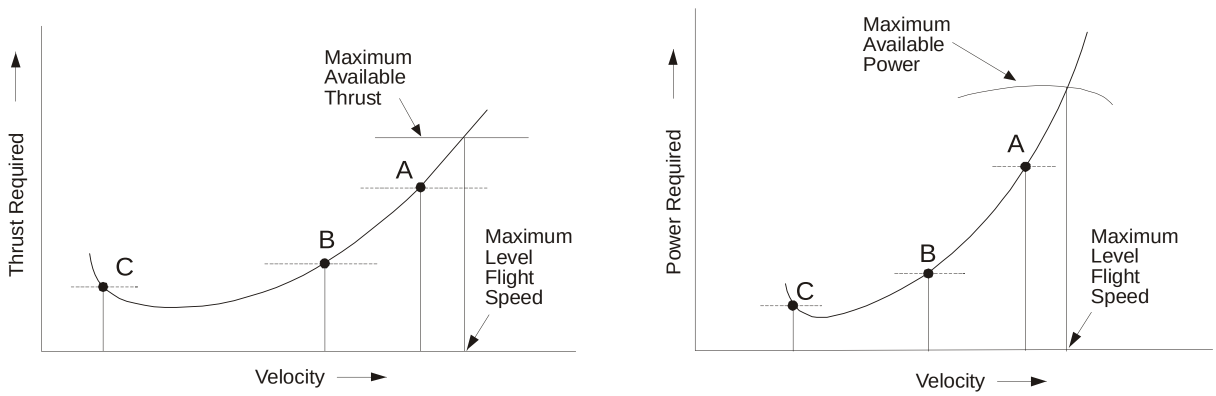
up at very low speeds and less so as speed increases. The difference between low and high speed attitudes is most marked on aircraft having swept-back wings or unswept wings of low aspect ratio, for the reasons given in Volume 1 Chapter 9.

9. **Aircraft Performance in Level Flight.** The variation of power required and thrust required with velocity is illustrated in Fig 4. Each specific curve of power or thrust required is valid for a particular aerodynamic configuration at a given weight and altitude. These curves define the power or thrust required to achieve equilibrium, ie lift equal to weight or constant altitude flight, at various air speeds. As shown by the curves of Fig 4, if it is desired to operate the aircraft at the air speed corresponding to point A, the power or thrust required curves define a particular value of thrust or power that must be made available from the engine to achieve equilibrium. Some different air speed, such as that corresponding to point B, changes the value of thrust or power required to achieve equilibrium. Of course, the change of air speed to point B would also require a change in angle of attack to maintain a constant lift equal to the aircraft weight. Similarly, to establish air speed and achieve equilibrium at point C will require a particular angle of attack and engine thrust or power. In this case, flight at point C would be in the vicinity of the minimum flying speed and a major portion of the thrust or power required would be due to induced drag. The maximum level flight speed for the aircraft will be obtained when the power or thrust required equals the maximum power or thrust available from the engine. The minimum flight air speed is not usually defined by thrust or power requirements since conditions of stall or stability and control problems generally predominate.

1-13 Fig 4 Level Flight Performance

a Thrust and Velocity

b Power and Velocity



Effect of Weight on Level Flight

10. If an aircraft is flying level at a given angle of attack, for example that for best L/D ratio (about 4°), and at a known weight and EAS, then if the weight is reduced by dropping stores the lift must also be reduced to balance the new weight. To maintain optimum conditions with the angle of attack at 4°, the speed must be decreased until the lift falls to the same value as the new weight.

11. The lower the weight, the lower is the EAS corresponding to a given angle of attack, the EAS at the required angle being proportional to the square root of the all-up weight.

Effect of Altitude on Level Flight

12. Since not only the speed but also the lift and drag vary with the $\frac{1}{2}\rho V^2$ factor, the relationship between EAS and angle of attack is unchanged at altitude, provided of course that the weight is

constant. However, compressibility effects tend to alter the relationship by reducing the C_L appropriate to a given angle of attack. This aspect is dealt with in Volume 1, Chapter 21.

CHAPTER 14 - CLIMBING AND GLIDING

Introduction

1. This chapter deals primarily with the principles of flight involved in both climbing and gliding, but in considering the factors affecting the climb, some performance details are discussed.

CLIMBING

General

2. During a climb an aircraft gains potential energy by virtue of elevation; this is achieved by one or, a combination of two means:

a. The expenditure of propulsive energy above that required to maintain level flight.

b. The expenditure of aircraft kinetic energy, ie loss of velocity by a zoom.

Zooming for altitude is a transient process of exchanging kinetic energy for potential energy and is of considerable importance for aircraft which can operate at very high levels of kinetic energy. However, the major portion of climb performance for most aircraft is a near steady process in which additional propulsive energy is converted into potential energy.

Forces in the Climb

3. To maintain a climb at a given EAS more power has to be provided than in level flight; this is first to overcome the drag as in level flight ($P_{REQ} = D \times V$), secondly to lift the weight at a vertical speed, which is known as the rate of climb, and thirdly to accelerate the aircraft slowly as the TAS steadily increases with increasing altitude.

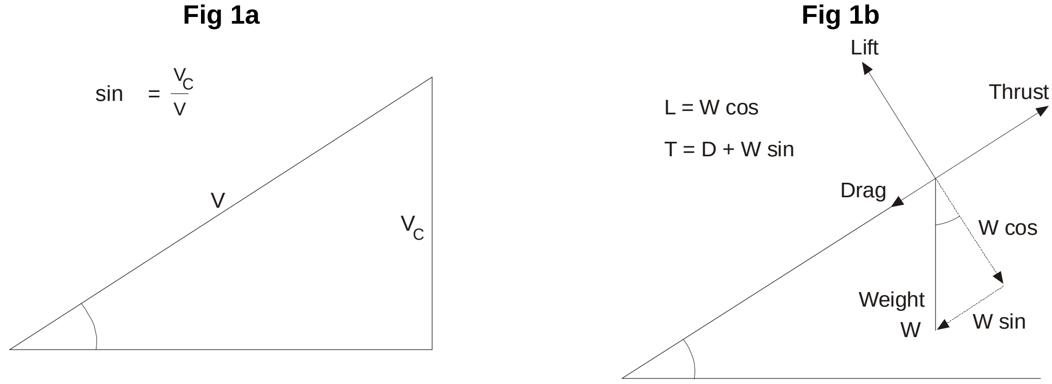
$$P_{REQ} = DV + WV_C + WV\,\frac{a}{g}$$

where V_C is the rate of climb and a is the acceleration.

The acceleration term can be ignored in low performance aircraft but has to be taken into account in jet aircraft with high rates of climb.

4. In Fig 1b it can be seen that, in the climb, lift is less than weight. If this is so then the lift dependent drag is less than at the same speed in level flight. It can be seen from the figure that L = W cos θ. However, it is still considered sufficiently correct to assume that lift equals weight up to about 15º climb angle (since cos 15º is still 0.9659, the error due to this assumption is less than 2%).

1-14 Fig 1 Forces in the Climb



5. Figs 1a and b can be used to show that rate of climb is determined by the amount of excess power and the angle of climb is determined by the amount of excess thrust left after opposing drag.

In Fig 1a: $\sin\theta = \dfrac{\text{rate of climb}}{V}$

In Fig 1b: $\sin\theta = \dfrac{\text{thrust} - \text{drag}}{\text{weight}}$

Since θ is the same in each case:

$$\frac{\text{rate of climb}}{V} = \frac{\text{thrust} - \text{drag}}{\text{weight}}$$

$$\text{Therefore, rate of climb} = \frac{V(\text{thrust} - \text{drag})}{\text{weight}}$$

$$= \frac{P_{AV} - P_{REQ}}{\text{weight}}$$

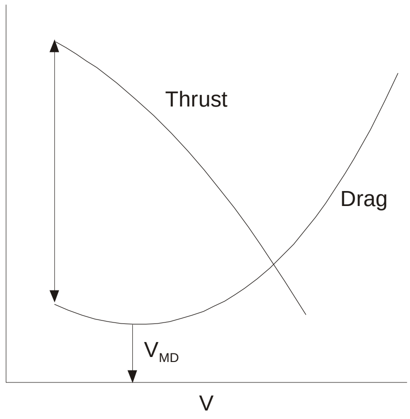
$$= \frac{\text{excess power}}{\text{weight}}$$

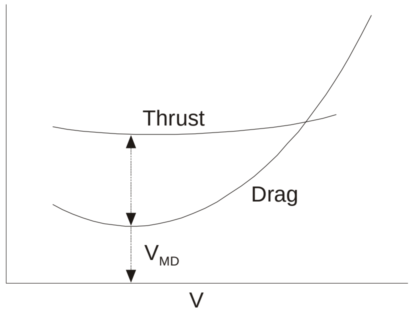
6. In practice aircraft do not, for varying reasons (e.g. engine cooling, avoidance of an exaggerated attitude), always use the exact speed for maximum rate of climb. In jet aircraft this speed is quite high and at low altitude is not very critical due to the shape of the power available curve (see para 8). In piston aircraft the speed is much lower and is normally found to be in the vicinity of the minimum drag speed.

7. When the maximum angle of climb is required it can be seen from Fig 1b, where $\sin\theta = \dfrac{\text{thrust} - \text{drag}}{\text{weight}}$, that the aircraft should be flown at the speed which gives the maximum difference between thrust and drag. For piston aircraft, where thrust is reducing as speed is increased beyond unstick, the best speed is usually as low as is safe above unstick speed. For a jet aircraft, since thrust varies little with speed, the best speed is at minimum drag speed (Fig 2).

1-14 Fig 2 Maximum Angle of Climb

Fig 2a Piston Aircraft **Fig 2b Jet Aircraft**

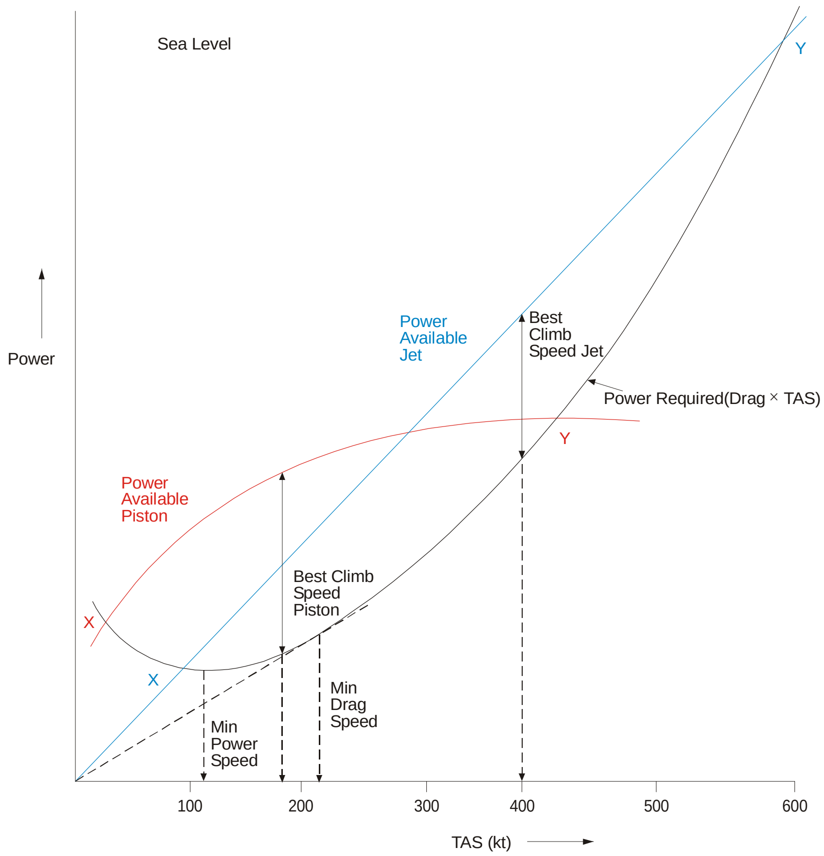




Power Available and Power Required

8. The thrust horsepower available curve is calculated by multiplying the thrust (in pounds (lb)) by the corresponding speed (in feet per second (fps)) and dividing by 550. The thrust power curve for a jet engine differs from that of a piston engine as shown in the upper curves of Fig 3. The main reason for this difference is that the thrust of a jet engine remains virtually constant at a given altitude, irrespective of the speed; therefore when this constant thrust is multiplied by the appropriate air speed to calculate thrust horsepower, the result is a straight line. The piston engine, on the other hand, under the same set of circumstances and for a given bhp, suffers a loss of thrust horsepower at both ends of its speed range because of reduced propeller efficiency. The thrust horsepower available curves are representative of a more powerful type of Second World War piston engine fighter and a typical jet fighter having a high subsonic performance.

1-14 Fig 3 Typical Power Available and Power Required Curves at Sea Level



9. The horsepower required to propel an aircraft in level flight can be found by multiplying the drag (lb) by the corresponding TAS (fps) and dividing by 550. The lower curve of Fig 3 is a typical example and can be assumed to apply to both a piston or jet-propelled airframe, ie the airframe drag is the same irrespective of the power unit used. Note that the speed for minimum drag, although low, is not the lowest possible nor is it that for minimum power. The increase in power required at the lowest speed is caused by the rapidly rising effects of induced drag.

Climbing Performance

10. The vertical distance between the power available and the power required curves represents the power available for climbing at the particular speed. The best climbing speed (highest rate of climb) is that at which the excess power is at a maximum; so that, after expending some power in overcoming the drag, the maximum amount of power remains available for climbing the aircraft. For the piston engine aircraft the best speed is seen to be about 180 kt, and for the jet about 400 kt. Notice that in the latter case a fairly wide band of speeds would still give the same amount of excess power for the climb but in practice the highest speed is used since better engine efficiency is obtained. At the intersection of the curves (points X and Y) all the available power is being used to overcome drag and none is available for climbing; these points therefore represent the minimum and maximum speeds possible for the particular power setting.

11. If power is reduced the power available curve is lowered. Consequently the maximum speed and maximum rate of climb are reduced, while the minimum speed is increased. When the power is reduced to the point when the power available curve is tangential to the power required curve, the points X and Y coincide and the aircraft cannot climb.

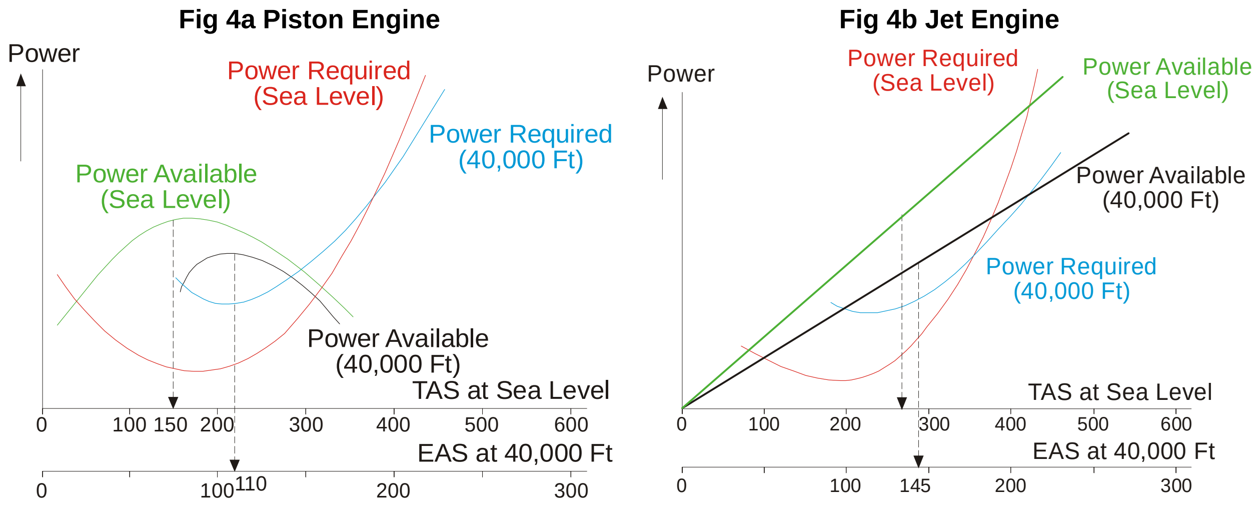
Effect of Altitude on Climbing

12. The thrust horsepower of both jet and piston engines decreases with altitude. Even if it is possible to prolong sea-level power to some greater altitude by supercharging or some other method of power boosting, the power will inevitably decline when the boosting method employed reaches a height at which it can no longer maintain the set power.

13. At the highest altitudes the power available curves of both types of engine are lowered as shown in Fig 4 and the power required curve is displaced upwards and to the right. Notice that the power required to fly at the minimum drag speed is increased; this effect is caused by the fact that although the minimum drag speed, in terms of EAS, remains the same at all heights, the speed used in the calculation of thp is the TAS, which increases with altitude for a given EAS. Therefore the power required to fly at any desired EAS increases with altitude. From Fig 4 it may be seen that the speed for the best rate of climb reduces with altitude and the range of speeds between maximum and minimum level flight speeds is also reduced.

Note: The height of 40,000 ft used in Fig 4 was chosen because at that height EAS = ½ × TAS; a piston engine would not normally operate at that height.

1-14 Fig 4 Effect of Altitude on Typical Power Available and Power Required Curves



14. **Ceilings**. The altitude at which the maximum power available curve only just touches the power required curve and a sustained rate of climb is no longer possible is known as the absolute ceiling. It is possible to exceed this altitude by the zoom climb technique which converts the aircraft's kinetic energy (speed) to potential energy (altitude). Another ceiling is the service ceiling which is defined as the altitude at which the maximum sustained rate of climb falls to 500 fpm (100 fpm for a piston aircraft).

Operating Data Manuals

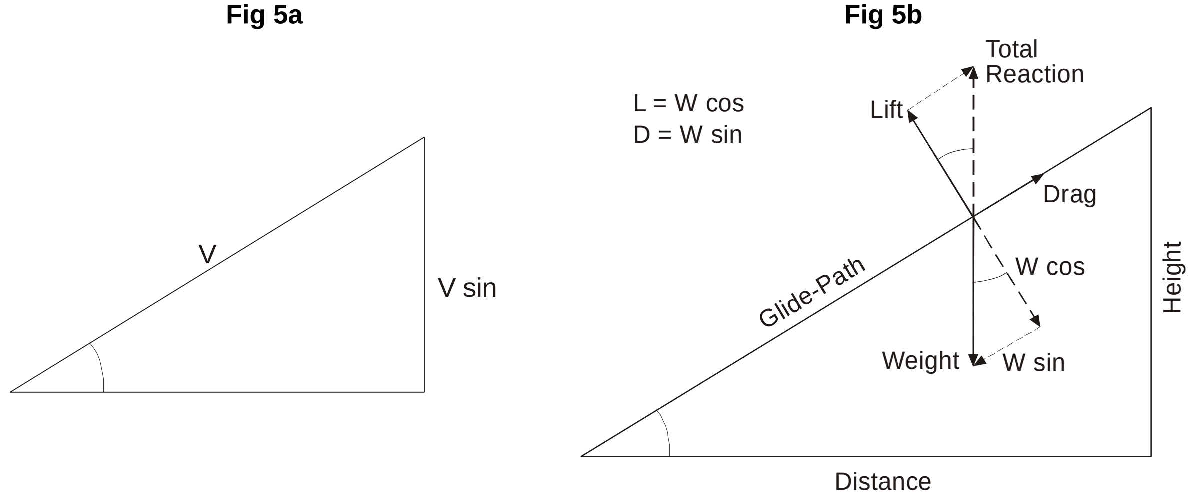
15. On older, simpler types of aircraft some basic take-off and climb data was given briefly in either narrative or tabular form in the Aircrew Manual. Now, with modern complex aircraft, take-off and climb data is much more comprehensive and is given in the aircraft Operating Data Manual (ODM) either in graphical or detailed tabular form. Further information on ODM is given in Volume 2, Chapter 16 and in Volume 8, Chapter 4.

GLIDING

Forces in the Glide

16. For a steady glide, with the engine giving no thrust, the lift, drag and weight forces must be in equilibrium (ignoring the deceleration term due to maintaining a constant IAS). Fig 5 shows that the weight is balanced by the resultant of the lift and drag; the lift vector, acting as it does at right angles to the path of flight, will now be tilted forward while the drag vector still acts parallel to the path of flight. To maintain air speed, energy must be expended to overcome this drag. When the engine is no longer working the source of energy is the potential energy of the aircraft (i.e. the altitude).

1-14 Fig 5 Forces in the Glide



17. The aircraft can either be made to have as low a rate of descent as possible, which will be obtained at the speed which requires least power, or it can be made to have as shallow an angle of glide as possible which will be the speed where least drag is produced (compare rate and angle of climb).

Gliding for Endurance

18. It is rare that aircraft, other than sailplanes, require to glide for minimum rate of sink. From Fig 5a it can be seen that for minimum rate of descent, V sin θ must be as small as possible. But D × V (ie power required) = WV sin θ (ignoring the deceleration term due to maintaining a constant IAS). Therefore, for a given weight, the rate of descent is least at the speed where the power required (DV) is least.

Gliding for Range

19. Note that the triangle formed by lift, drag and total reaction is geometrically similar to that formed by distance, height and glidepath. Now, if distance is to be maximum, gliding angle must be minimum. θ is minimum when cot θ is maximum and, from Fig 5b, it can be seen that:

$$\cot\theta = \frac{W\cos\theta}{W\sin\theta} = \frac{L}{D} = \frac{C_L}{C_D}$$

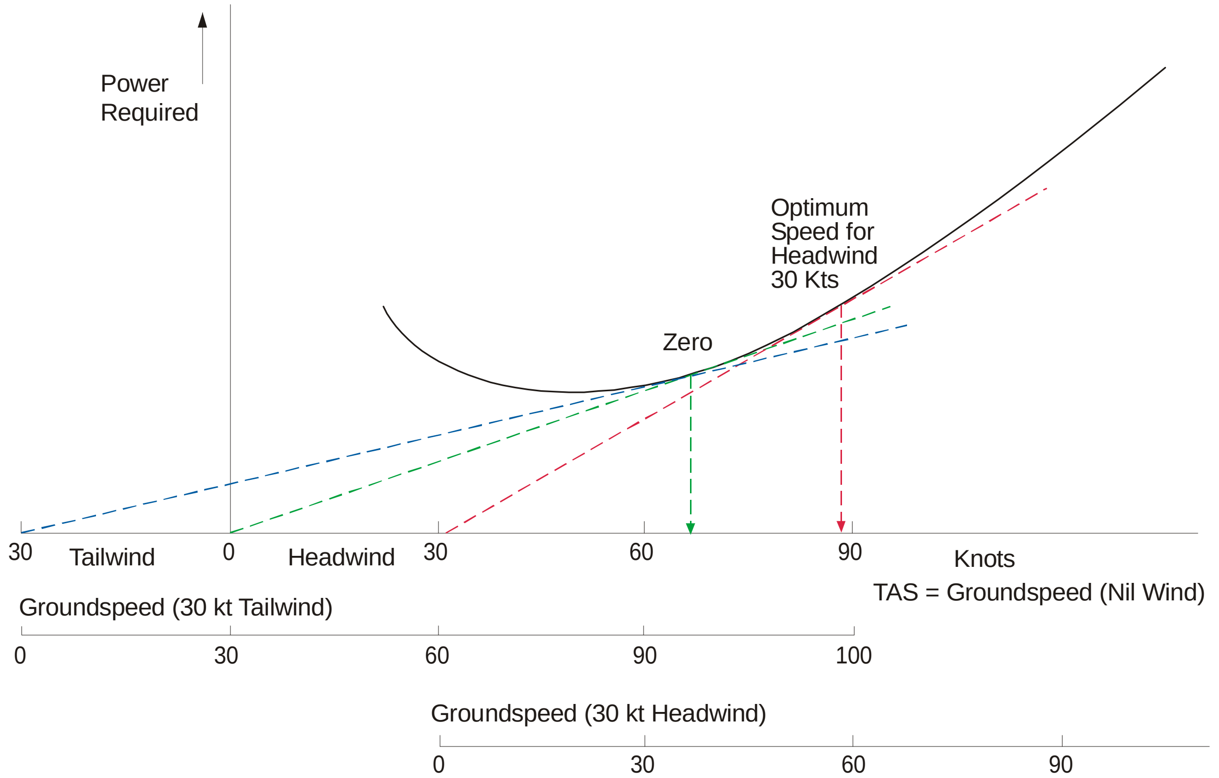
Therefore the best angle of glide depends on maintaining an angle of attack which gives the best lift/drag ratio. Therefore for maximum distance the aircraft should be flown for minimum drag. For angles of glide of less than 15º, cos θ approximates to 1 and so it is reasonable to use the level flight power curves. If the angle of descent exceeds about 15º then it is no longer sufficiently correct to assume that lift equals weight. The actual lift needed is less and so the gliding speed is reduced by a factor of $\sqrt{\cos\theta}$. Since, ignoring compressibility, drag depends on EAS, the best gliding speed at a given weight is at a constant EAS regardless of altitude. The rate of descent will, however, get less at lower altitudes as the TAS decreases.

Effect of Wind

20. Since, in a glide for minimum rate of descent the position of the end of the glide is not important, wind will not affect gliding for endurance. However, when gliding for range the all-important target is the point of arrival; the aim is maximum distance over the ground. In still air this is achieved by flying for minimum drag as explained in para 19. The effect of a headwind will be to decrease the ground distance travelled by approximately the ratio of the wind speed compared to the TAS. A calculated increase of airspeed reducing the time the wind effect could act could improve the ground distance

travelled. Similarly, if there were a tailwind the ground distance travelled would be increased. A reduction of speed towards V_{MP}, which gives the minimum rate of descent, could be beneficial because it would allow more time for the wind to act on the aircraft. The graphical method of finding the best speed to glide for maximum range in the presence of a head or tailwind is the same as that for finding the optimum range speed for a piston aircraft. From the wind value along the TAS axis the tangent should be drawn to the power required curve (see Fig 6).

1-14 Fig 6 The Graphical Method of Finding Maximum Range Gliding Speeds

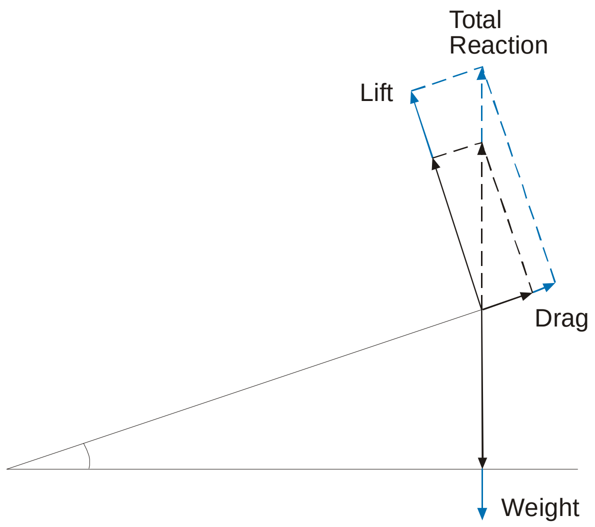


Effect of Weight

21. Variation in the weight does not affect the gliding angle provided that the speed is adjusted to fit the auw. The best EAS varies as the square root of the auw. A simple method of estimating changes in the EAS to compensate for changes in the auw up to about 20% is to decrease or increase the air speed by half the percentage change in the auw. For example, a weight reduction of 10% necessitates a drop in air speed of 5%; an increase of weight of the same amount would entail a 5% increase.

22. Fig 7 shows that an increase in the weight vector can be balanced by lengthening the other vectors until the geometry and balance of the diagram is restored. This is done without affecting the gliding angle. The higher speed corresponding to the increased weight is provided automatically by the larger component of the weight acting along the glide path; and this component grows or diminishes in proportion to the weight. Since the gliding angle is unaffected, the range is also unchanged in still air. Where weight does affect the range is when there is a tailwind or headwind component. The higher TAS for a heavier weight allows less time for the wind to affect the aircraft and so it is better to have a heavier aircraft if gliding for range into wind. If minimum rate of descent is required then the aircraft should be light. The lower drag requires a less rapid expenditure of power which is obtained from the aircraft's potential energy (height).

1-14 Fig 7 Effect of Weight on Glide



23. Although the range is not affected by changes in weight, the endurance decreases with increase of weight and vice versa. If two aircraft having the same L/D ratio but with different weights start a glide from the same height, then the heavier aircraft gliding at a higher EAS will cover the distance between the starting point and the touchdown in a shorter time; both will, however, cover the same distance in still air. Therefore the endurance of the heavier aircraft is less.

24. Usually the exact distance covered on the glide is not vitally important, therefore the gliding speed stated in the Aircrew Manual is a mean figure applying to the lower weights of a particular aircraft and giving the best all-round performance in the glide.

CHAPTER 15 - MANOEUVRES

Introduction

1. Changes in the attitude of an aircraft in flight can take place in any one, or combination of, the three major axes described below. During manoeuvres considerable forces are at work on the airframe and these may be large enough to cause damage or even structural failure if the aircraft is manoeuvred without consideration of the limits for which the airframe has been designed.

2. This chapter looks at the nature of the forces which affect limitations, the build-up of the manoeuvre envelope and the theory of level turns as a specific manoeuvre. Volume 1, Chapter 16 gives greater detail on turning performance.

Axes of Movement of an Aircraft

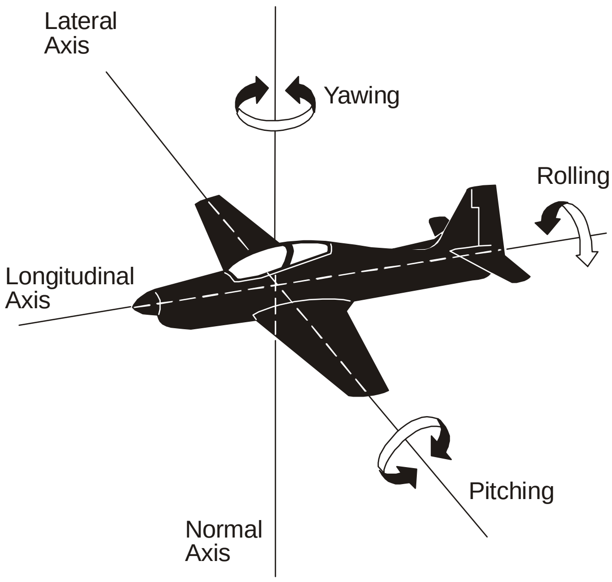
3. **Lateral Axis**. The lateral axis is a straight line through the CG normal to the plane of symmetry, rotation about which is termed pitching. This axis may also be known as the pitching or looping axis. If any component of the forward flight velocity acts parallel to this axis, the subsequent motion is called sideslip or skid.

4. **Longitudinal Axis**. The longitudinal axis is a straight line through the CG fore and aft in the plane of symmetry, movement about which is known as rolling. This axis is sometimes called the roll axis.

5. **Normal Axis**. The normal axis is a straight line through the CG at right angles to the longitudinal axis, in the plane of symmetry, movement about which is called yawing. This axis can be referred to as the yawing axis.

6. **Fixed Relationship**. The three axes are fixed relative to the aircraft irrespective of its attitude. Fig 1 shows the major axes and the possible movements about them.

1-15 Fig 1 The Three Major Axes



Acceleration

7. Any aircraft or body in motion is subject to three laws of motion (Newton's laws) which state that:

a. All bodies tend to remain at rest or in a state of uniform motion in a straight line unless acted upon by an external force, ie they have the property of inertia.

b. To change the state of rest, or motion in a straight line, a force is required. To obtain a given rate of change of motion and/or direction the force is proportional to the mass of the body. It follows that for a given mass, the greater the rate of change of speed and/or direction the greater is the force required.

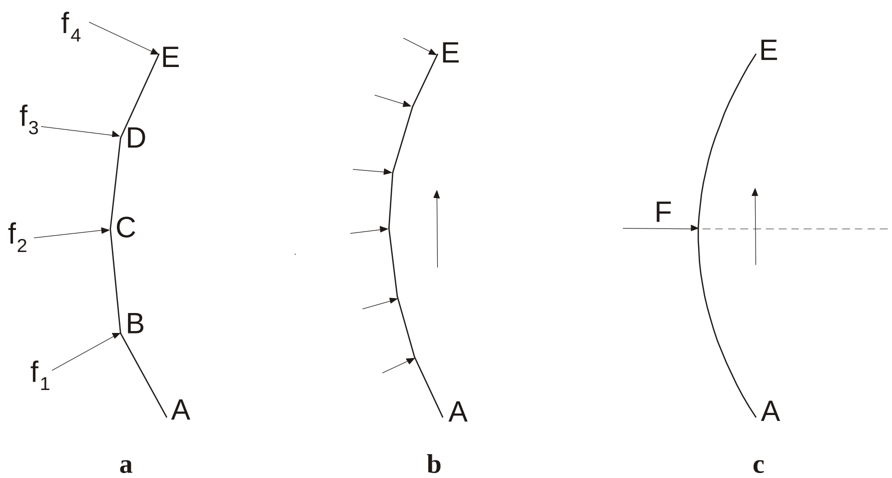
c. To every action there is an equal and opposite reaction.

Most manoeuvres involve changes in direction and speed, the degree of change depending on the manoeuvre involved. Any change of direction and/or speed necessarily involves an acceleration, which is often evident to the pilot as an apparent change in his weight. During an acceleration the aircraft is not in equilibrium since an out-of-balance force is required to deflect it continuously from a straight line.

8. While a body travels along a curved path it tries constantly to obey the first law and travel in a straight line. To keep it turning a force is necessary to deflect it towards the centre of the turn. This force is called centripetal force and its equal and opposite reaction is called centrifugal force. Centripetal force can be provided in a number of ways. A weight, on the end of a piece of string, that is swung in a circle is subjected to centripetal force by the action of the string. If the string is released the centripetal force is removed and the equal and opposite reaction (centrifugal force) disappears simultaneously. The weight, conforming to the first law, then flies off in a straight line at a tangent to the circle.

9. To clarify the difference between centripetal and centrifugal force, consider a body (Fig 2a) which moves along a series of straight lines AB, BC, CD and DE, each inclined to its neighbour at the same angle. When it reaches B it is subjected to an external force F1 acting at right angles to AB, which alters its path to BC. At C a force F2, acting at right angles to BC, alters the path to CD and so on. If the path from A to E consists of a greater number of shorter lines (Fig 2b) the moving body will be deflected at shorter intervals, and if the lines are infinitely short the intermittent forces blend into one continuous force F and the path becomes the arc of a circle (Fig 2c). F will act at right angles to the direction of motion, i.e. towards the centre of the arc; the reaction to it, the centrifugal force, is equal in size and opposite in direction.

1-15 Fig 2 Centripetal Force



Gravity

10. The symbol ‘g’ denotes the rate of acceleration of a body falling freely under the influence of its own weight, i.e. the force of gravity. The acceleration is about 9.8 m/sec/sec at the Earth’s surface when measured in a vacuum so that there is no drag acting upon the body. The force of gravity varies with the distance from the Earth’s centre and therefore differs slightly at different points on the Earth’s surface, since the Earth is not a perfect sphere. The weight of a body of a stated density and volume (mass) is proportional to the force of gravity and so varies slightly. For practical purposes g can be considered to be constant at sea level, irrespective of the geographical location. As altitude is increased, g decreases progressively, but in unaccelerated flight this effect is negligible at current operating heights.

11. If any object has, for example, a total force acting upon it equal to five times its own weight it will accelerate in the direction of the force at a rate five times greater than that due to gravity, namely 5 g. Although g is a unit of acceleration it is often used, inaccurately, to indicate the force required to produce a given acceleration. Among pilots g is often used in this way to express the force accompanying a manoeuvre in terms of a multiple of the static weight. For example, if a certain rate of turn (ie acceleration) necessitates a centripetal force of three times the aircraft weight then the turn is called a 3 g turn. Since the force is felt uniformly throughout the entire aircraft and its contents, the crew also experiences this force and acceleration and feel it as an apparent increase in weight which is proportional to the g. During straight and level flight an aircraft accelerometer shows 1 g, for this is the normal force of gravity that is acting at all times on all objects. During inverted level flight the instrument reads –1 g and the pilot’s weight, acting vertically downwards, is supported by his harness.

Calculation of Centripetal Force

12. The magnitude of the centripetal force during a given turn is directly proportional to the mass of the body (its static weight) and the square of the speed, and is inversely proportional to the radius of turn. It is calculated from the formula $\frac{WV^2}{gr}$ kg, where W is the weight (kg), V the TAS (m/s), r the radius of turn (m) and g is a constant 9. 8 m/s^2.

THE MANOEUVRE ENVELOPE

General

13. The manoeuvre envelope (or Vn diagram) is a graphic representation of the operating limits of an aircraft. The envelope is used:

a. To lay down design requirements for a new aircraft.

b. By manufacturers to illustrate the performance of their products.

c. As a means of comparing the capabilities of different types.

The axes of the envelope are:

d. On the vertical axis, load factor (n), positive and negative.

e. On the horizontal axis, EAS.

The Limits to the Basic Envelope (Theory)

14. One of the more obvious limits to the operation of an aircraft is the increase of the stalling speed as the load factor is increased. From basic theory (see Volume 1, Chapter 2):

$$\text{load factor (n)} = \frac{\text{lift}}{\text{weight}}$$

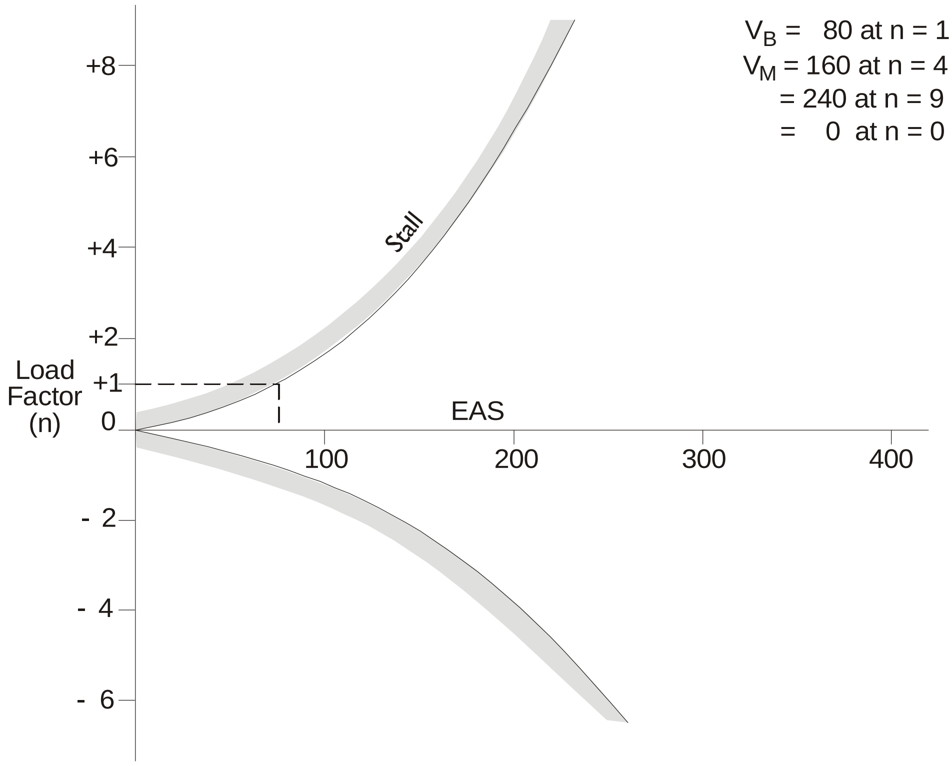
$$\therefore n = \frac{\text{Lift at manoeuvre stalling speed } (V_M)}{\text{Lift at basic stalling speed } (V_B)}$$

$$= \frac{C_{L\,max}\,\frac{1}{2}\rho V_M^2\,S}{C_{L\,max}\,\frac{1}{2}\rho V_B^2\,S}$$

$$= \frac{V_M^2}{V_B^2}, \text{so } V_M = V_B\sqrt{n}$$

It should be noted from Fig 3 that flight below the basic stalling speed is perfectly feasible at load factors below one.

1-15 Fig 3 Manoeuvre Stalling Speed

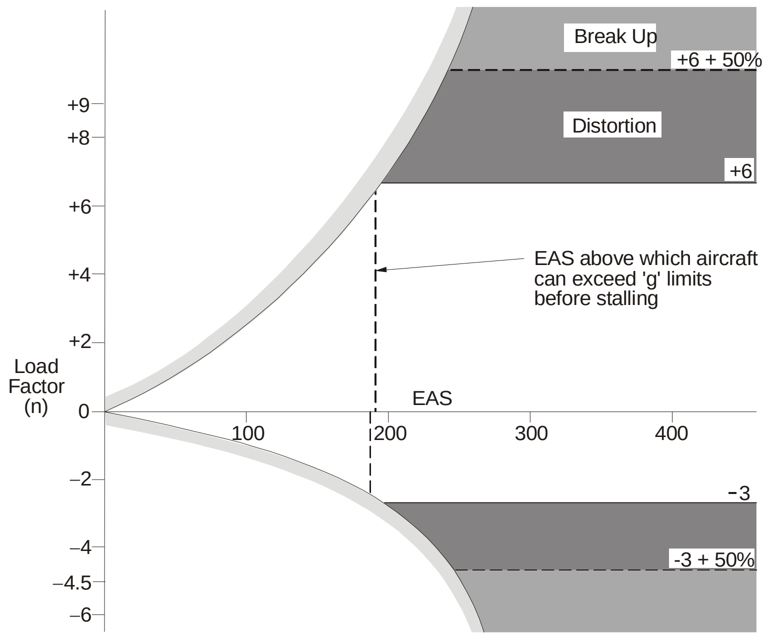


15. **g Limits**. Any aircraft is designed to certain strength requirements; fighter and training aircraft need to be stronger than transport aircraft. Extra strength normally means an increase in structural weight which in turn would need larger engines; therefore most aircraft are designed to a bare minimum strength which depends on their role. In this country the training and fighter aircraft are built to withstand about +7 g in use, which is about the maximum that an average human pilot can stand and still remain conscious; transport aircraft are built to about +2 g. Both classes usually have much lower negative g limits (e.g. about –3½ g and –½ g respectively). These values usually have a 50% safety margin above which permanent deformation and failure may occur. However this does not

mean that lesser damage will not occur; small excesses may cause rivets to 'pop' or panels to come off, and as the inroads into the safety margin increase so more severe damage may occur.

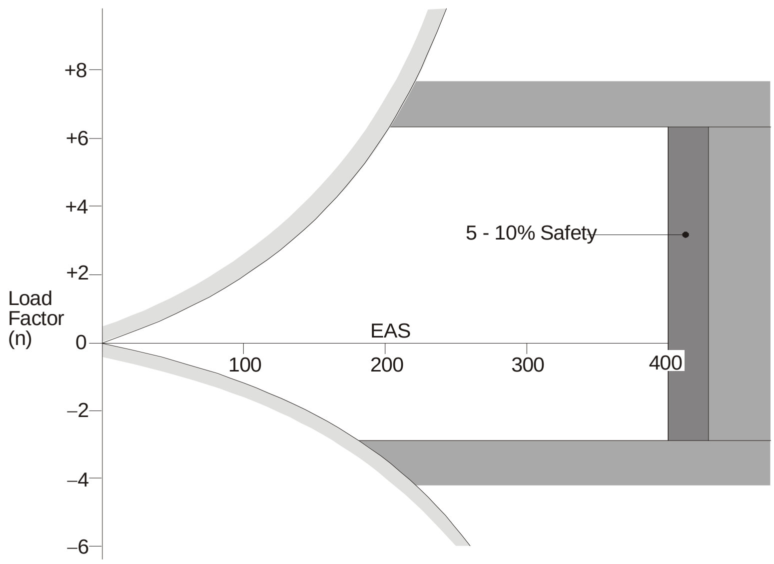
Since $g = \frac{\text{lift}}{\text{weight}}$, the g limits need to be reduced if the weight of the aircraft is increased in order to retain the same safety margins. (Fig 4).

1-15 Fig 4 g Boundary



16. **EAS Limitation**. An aircraft has a speed limit, (or maximum permissible diving speed) with a “small” (usually 5 or 10%) safety factor. Exceeding this may lead to loss of access panels, failure of the weakest structure (often the tailplane or canopy) or even control reversal. This limit completes the basic manoeuvre envelope, with operation outside the envelope either impossible or unsafe. (Fig 5). Variations in CL max

1-15 Fig 5 EAS Limit

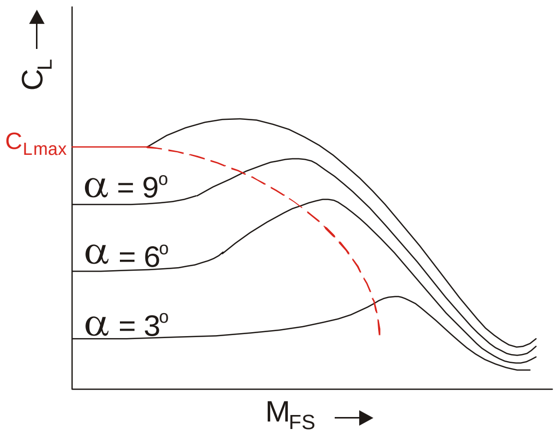


Variations in $C_{L\ max}$

17. The boundaries of the basic envelope constructed up to this point have been drawn on the assumption that $C_{L\ max}$ remains constant. However, when plotted against Mach number, $C_{L\ max}$ is seen to vary with changes of compressibility, Reynolds number and adverse pressure gradient.

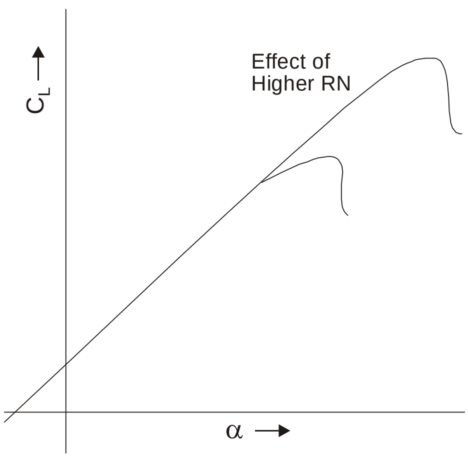
18. Compressibility and its effects are dealt with fully in Volume 1, Chapter 21, but for the purpose of this chapter it may be said that increasing the angle of attack increases the local acceleration over the wing so that the critical Mach number (M_{CRIT}) is reached at a progressively lower free stream Mach number (M_{FS}). The subsequent shockwaves are nearer the leading edge and as the flow behind the shock breaks away and is turbulent, a greater part of the wing is subject to separated flow and the loss of lift is greater. Fig 6 shows typical variations at different angles of attack with Mach number. With moderately thick wings this reduction could be at an extremely low M_{FS}.

1-15 Fig 6 Variation in C_L with Mach Number



19. The increase in Reynolds number due to speed only slightly affects the slope of C_L curve, ie when plotted against speed (ignoring compressibility), C_L is a straight line; its effect on $C_{L\ max}$ can, however, be quite marked because it delays separation to a higher angle of attack with the consequent increase in $C_{L\ max}$. When plotting $C_{L\ max}$ against speed, theory would indicate that $C_{L\ max}$ would increase until compressibility effects become predominant, when it would decline. (Fig 7). As speed increases, the adverse pressure gradient on the upper surface at high angles of attack becomes stronger; this causes the boundary layer to slow down and thicken; the separation point is moved forward and the $C_{L\ max}$ reduces.

1-15 Fig 7 Effect of Increased Reynolds Number



20. For a variety of reasons the whole aircraft may be incapable of using the highest angle of attack, so that the usable $C_{L\ max}$ is reduced. The reasons for this vary among types, but may include the following:

a. Reduction in effectiveness of the elevator, due to its operation behind a shockwave on either the wing or tailplane.

b. Buffet on the tailplane from the thick fluctuating wake of the wing, making it impossible accurately to hold angles of attack near the stall. As a result a lower angle has to be chosen.

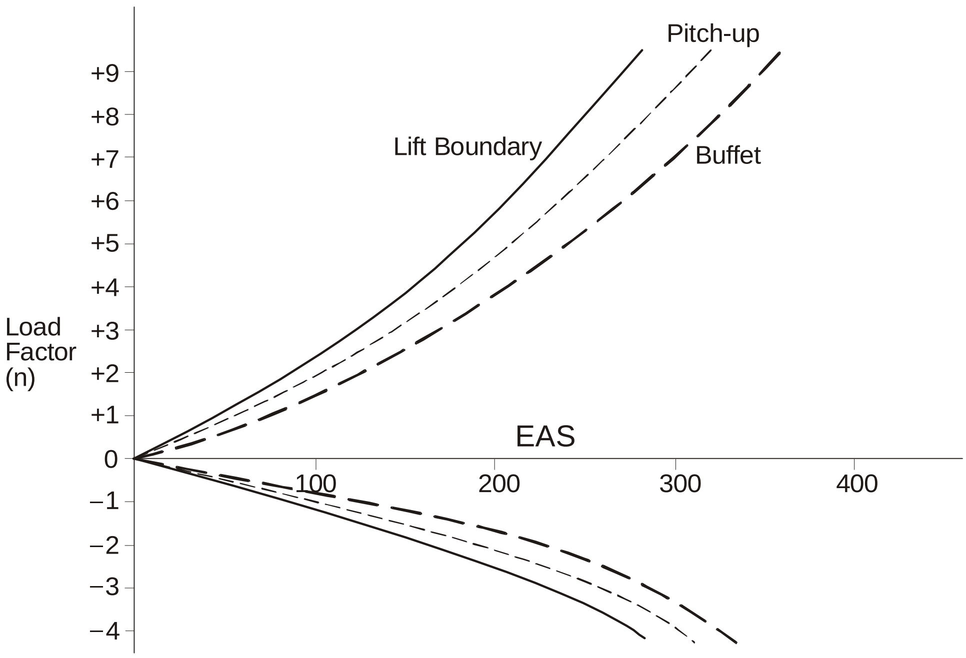
c. Wing tip stalling or the increase in downwash over the tailplane caused by the large vortices of a swept-wing aircraft may lead to pitch-up, again limiting the usable angles of attack.

21. What happens in practice depends very much on the wing in question but it is generally true that the theoretical C_{Lmax} cannot be used because control problems due to separated flow prevent the required angle of attack being selected. Generally, the obtainable maximum value of C_L for subsonic and most transonic aircraft decreases from a very low Mach number along the dashed line in Fig 6. For supersonic aircraft the C_{Lmax} may decline with increasing Mach number in the subsonic range, but usually recovers as the speed exceeds M1.0.

The Modified Envelope due to Reduction in CLmax

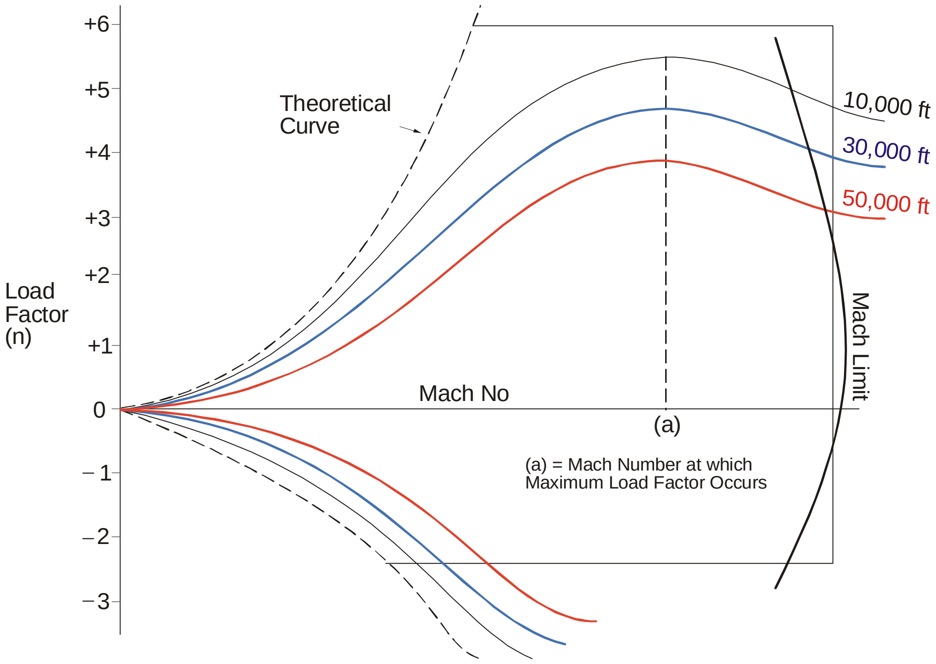
22. The reduction in $C_{L\ max}$ means that the wing gives less lift than was expected at any particular speed. Consequently the available load factor is reduced and a new lift boundary curve must be produced. (Fig 8).

1-15 Fig 8 Lift Boundary



23. As each EAS corresponds to a higher Mach number at altitude, so the $C_{L\ max}$ will decline further and new curves must be drawn for each altitude, although the Mach number at which the maximum load factor occurs will remain the same (see Fig 9). It should be noted that a lift boundary also exists for negative load factors and that a different lift boundary exists at different weights.

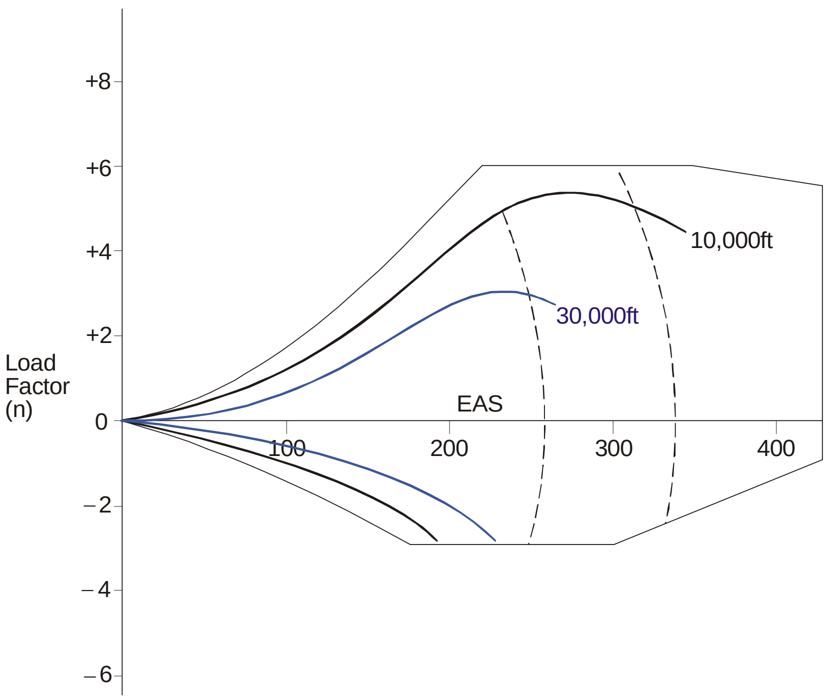
1-15 Fig 9 Variation of Lift Boundary with Altitude



Structural Limitations

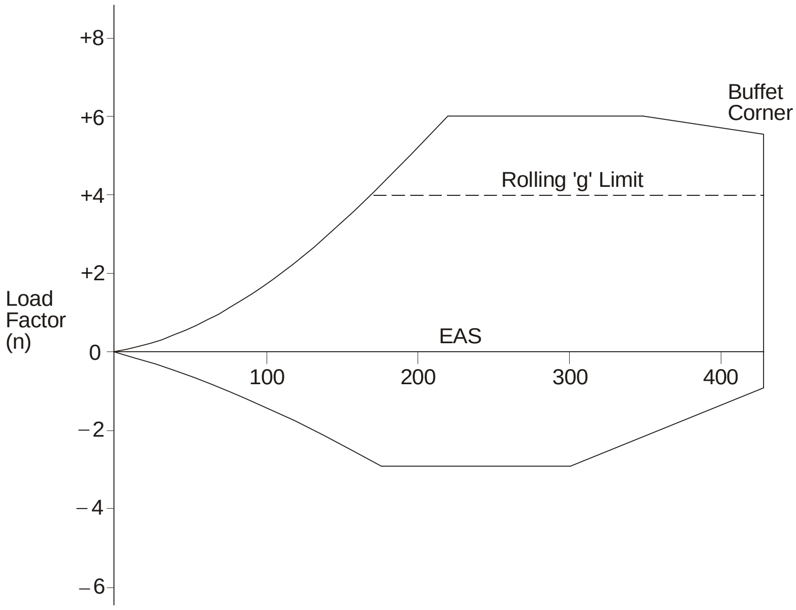
24. **Mach Limit**. Aircraft designed for subsonic or transonic flight usually have a compressibility Mach number. If this is exceeded the position and size of the shockwaves make control difficult or impossible and could even lead to structural failure. At high angles, of attack the airflow is accelerated more and the wave development will occur at a lower Mach number. As a result, this limit appears as a curved line (see Fig 9). In the complete envelope diagram, because the EAS/Mach number relationship changes with height, Mach number limits must be drawn for each altitude; these are shown in Fig 10.

1-15 Fig 10 The Complete Flight Envelope



25. **Rolling g Limit**. A rolling g limit is an additional limit imposed on the operation of an aircraft when the roll control is deflected. It is a lower figure than the ordinary g limit because the wing structure has to provide strength to withstand the twisting forces caused by the roll control deflection besides providing strength to withstand the normal g. It may also be imposed to avoid structural failure due to divergence caused by inertia cross-coupling, see Fig 11.

1-15 Fig 11 Rolling g Limit



26. **Buffet Corners**. If high air loads are combined with high loadings, the weakest part of the structure is more likely to fail. In the case of the tailplane this will be further aggravated if it is buffeted by the turbulent wake of the wings, possibly leading to fatigue failure.

27. **Other Limits**. Other limits which could be imposed include considerations for different weights, fuel states, stores, CG positions, etc.

Information Available from a Manoeuvre Envelope

28. When all the limits are drawn together they show the available range of manoeuvre for the aircraft at any height. The envelope shows:

a. Basic stalling speed.

b. Available load factor at any height and speed.

c. Maximum EAS at any height.

d. Stalling speed at any height and load factor.

These points are, of course, in addition to the Aircrew Manual limitations which can be read directly from the envelope:

e. Maximum permitted load factor.

f. Maximum EAS.

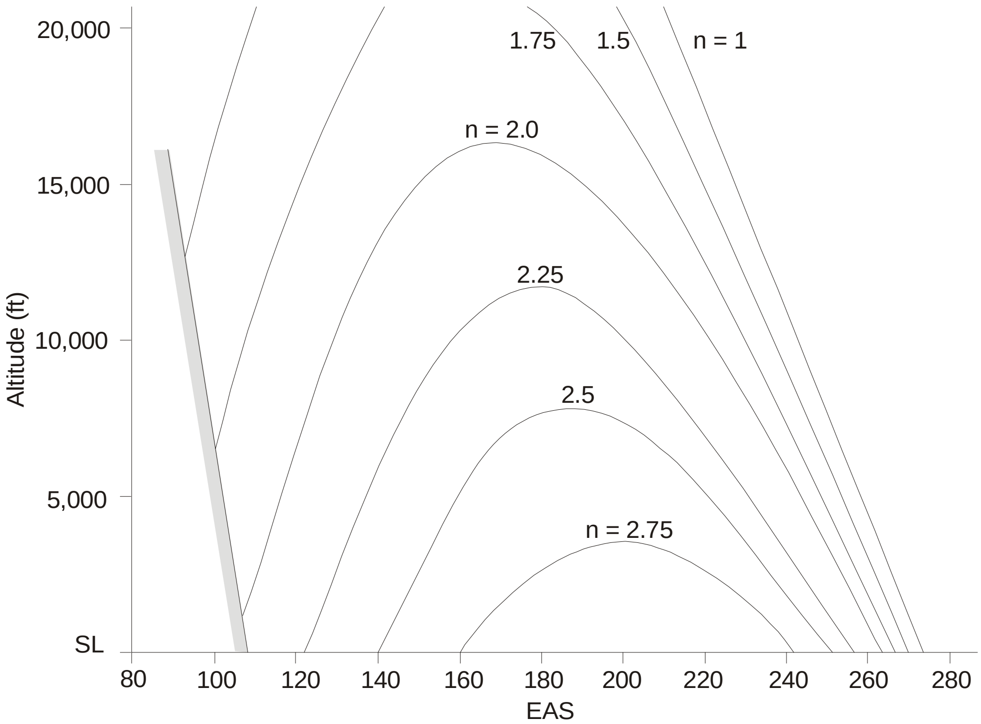
g. Compressibility Mach number.

h. Rolling g limits.

Limitations due to Power Unit

29. The thrust boundary shows the maximum altitude and/or speed range available at any given load factor. Just as the manoeuvre envelope shows the available range of manoeuvre for the aircraft at any height so the thrust boundary shows the limitations due to the amount of thrust available. A thrust boundary shows at what speeds, altitudes and load factors the thrust available from the engine is equal to the drag produced by the whole aircraft when in straight and level flight or in a level turn. Since the thrust varies with altitude and the drag varies with speed a typical boundary is as shown in Fig 12.

1-15 Fig 12 Thrust Boundary



TURNING

General

30. For an aircraft to turn, centripetal force is required to deflect it towards the centre of the turn. By banking the aircraft and using the horizontal component of the now inclined lift force, the necessary force is obtained to move the aircraft along a curved path.

31. If the aircraft is banked, keeping the angle of attack constant, then the vertical component of the lift force will be too small to balance the weight and the aircraft will start to descend. Therefore, as the angle of bank increases, the angle of attack must be increased progressively by a backward movement of the control column to bring about a greater total lift. The vertical component is then large enough to maintain level flight, while the horizontal component is large enough to produce the required centripetal force.

Effect of Weight

32. In a steady level turn, if thrust is ignored, then lift is providing a force to balance weight and a centripetal force to turn the aircraft. If the same TAS and angle of bank can be obtained, the radius of turn is basically independent of weight or aircraft type. However not all aircraft can reach the same angle of bank at the same TAS.

The Minimum Radius Turn

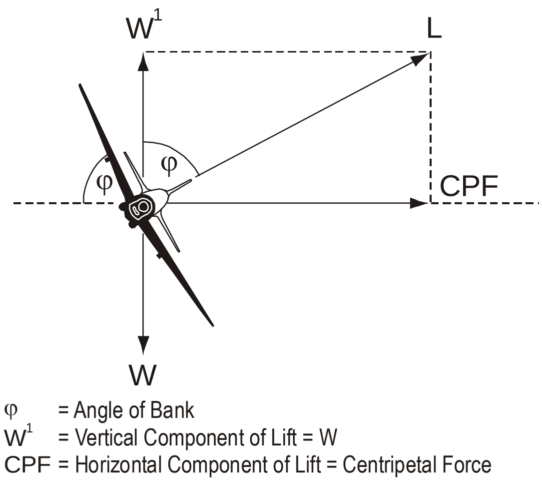
33. To achieve a minimum radius turn it can be shown (see Volume 1, Chapter 16) that the following must be satisfied:

a. Wing loading must be as low as possible.

b. The air must be as dense as possible, ie at Mean Sea Level (MSL) (achievable only theoretically).

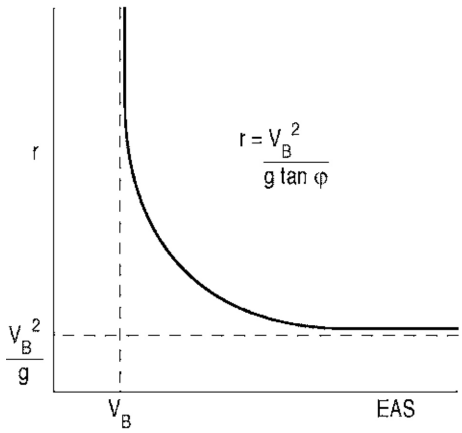
c. The maximum value of the product of C_L and angle of bank must be obtained. Note that this does not say maximum angle of bank, for the following reason. The angle of bank is increased to provide the increased lift force which is needed to give the centripetal force towards the centre of the turn. To do this, when already at the critical angle of attack, the speed must be increased. An increase in speed may cause a fall in the maximum value of C_L (see Fig 6). Thus bank and C_L may not both be at maximum value at the same time.

1-15 Fig 13 Forces in a Turn

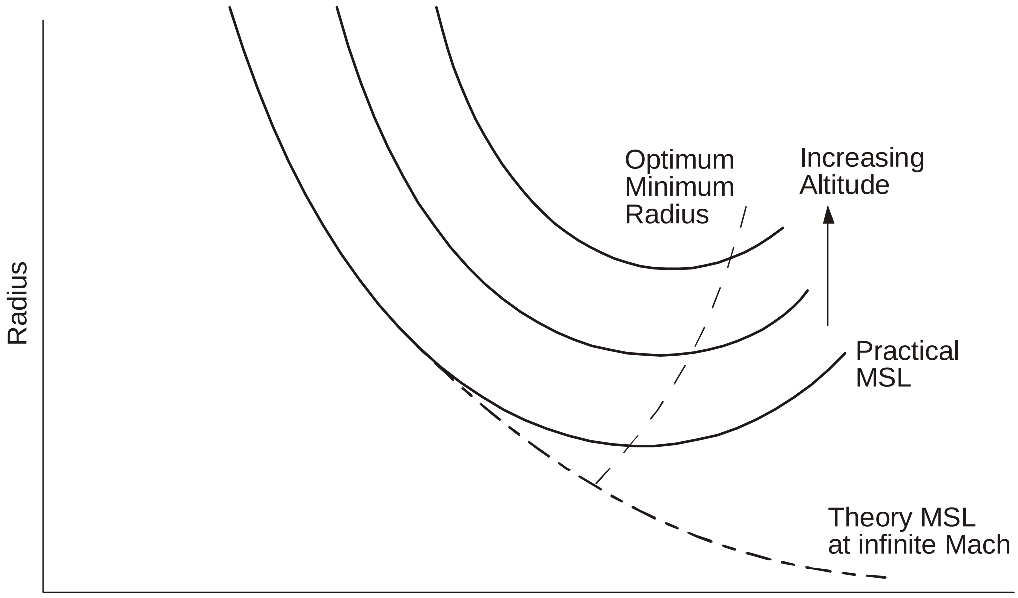


34. If the aircraft is kept on the verge of the stall to obtain C_{Lmax} and the speed is steadily increased from the basic stalling speed, the radius will decrease as bank is increased until, in theory, the minimum radius is obtained at infinite speed (see Fig 14). However, the C_{Lmax} changes with increasing Mach number. Therefore if, at a given speed, the C_L is less than expected, then less lift is generated and, as can be seen in Fig 13, the centripetal force is reduced; less bank can be applied and so the minimum radius starts to increase as speed is increased beyond a certain value. The overall result of this reduction in C_{Lmax} on the minimum radius is shown in Fig 15.

1-15 Fig 14 Theoretical Radius of Turn



1-15 Fig 15 Variation in Minimum Radius Turns with Speed and Altitude



35. **Altitude**. Fig 15 shows that there is an increase in the minimum radius with increase in altitude. This is commonly explained as being due to the EAS/TAS relationship. An additional increase in minimum radius is also caused by the greater reduction in C_{Lmax} because the Mach number is higher at altitude for a given TAS. In addition it should be remembered that thrust is also reduced at altitude, although so far thrust has been ignored. These factors combined give an optimum speed for minimum radius which is slightly higher than the ideal.

The Maximum Rate Turn

36. The rate of turn is $\frac{V}{r}$ radians per second. To achieve a maximum rate turn it can be shown that the following must be satisfied (see Volume 1, Chapter 16):

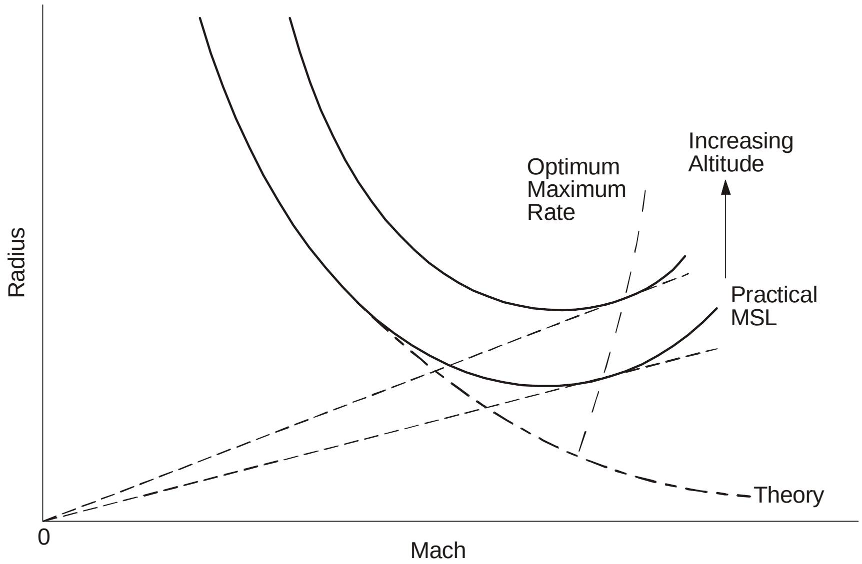
a. Wing loading must be as low as possible.

b. The air must be as dense as possible, ie at MSL.

c. The maximum value of the product of angle of bank, speed and C_L value must be obtained. Again note that this does not say maximum angle of bank for the following reason. The angle of bank is increased to provide the increased lift force which is needed to give the centripetal force towards the centre of the turn. To do this when already at the critical angle of attack the speed must be increased. An increase in speed may cause a fall in the maximum value of C_L (see Fig 6). Thus bank, speed and C_L may not all be at maximum value at the same time.

The speed for a maximum rate turn will be where a tangent from the origin just touches the minimum radius curve (see Fig 16). On the theoretical curve, with C_{Lmax} constant, the tangent is at infinite speed which is the same as the speed for minimum radius. However, when the practical reduction in C_{Lmax} with increasing Mach number is taken into account, the speed for a maximum rate turn is found to be somewhat higher than that for minimum radius. Increasing altitude will cause the rate of turn to decrease.

1-15 Fig 16 Variation in Maximum Rate Turns with Speed and Altitude

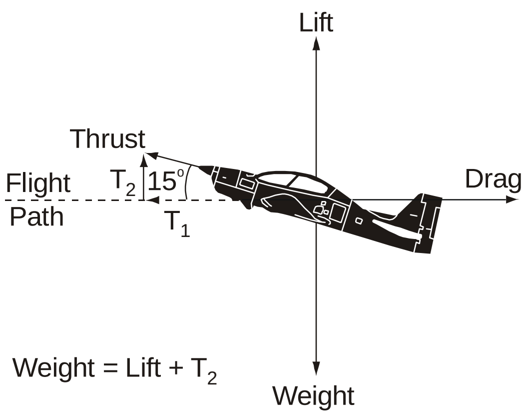


37. **Maximum g Loading**. The speed for maximum g loading is higher than the speed for a maximum rate turn (see Volume 1, Chapter 16).

Effect of Thrust

38. So far the effect of thrust on level turns has been ignored. However, the thrust, or lack of thrust, may be the determining factor as to whether the optimum speeds referred to in paras 34 to 37 can even be attained; this will depend on the thrust boundary (see para 29). Even in level flight it can be seen clearly with some aircraft that a component of thrust is acting in the same direction as lift due to the inclination of the thrust line from the horizontal. This effect becomes more pronounced as the critical angle of attack is approached (see Fig 17).

1-15 Fig 17 Thrust Vectors



In the minimum radius and maximum rate turns discussed, the aircraft is flown for $C_{L\ max}$ which is obtained at the critical angle. The thrust component assists lift so that either less lift is required from the wing (as simple turning theory requires for a particular turn) or the turn can be improved beyond that indicated by simple theory. Just as lift was split into two components in Fig 13, one to oppose weight and one to provide centripetal force, so the component of thrust that acts in the same direction as lift can also be split into two similar components (see Volume 1, Chapter 16). This is the reason for the remarkably small radius of turn of which some high performance aircraft are capable. However, the reduction of thrust with increasing altitude will cause a reduction in turning performance in addition to that caused by the TAS/EAS relationship and the greater C_{Lmax} reduction.

39. It should be realized that many aircraft do not have sufficient thrust to reach and sustain the optimum speed for a minimum radius turn at constant altitude. In this case the speeds for minimum radius, maximum rate and maximum g will all be the same at the maximum speed that can be sustained at constant altitude.

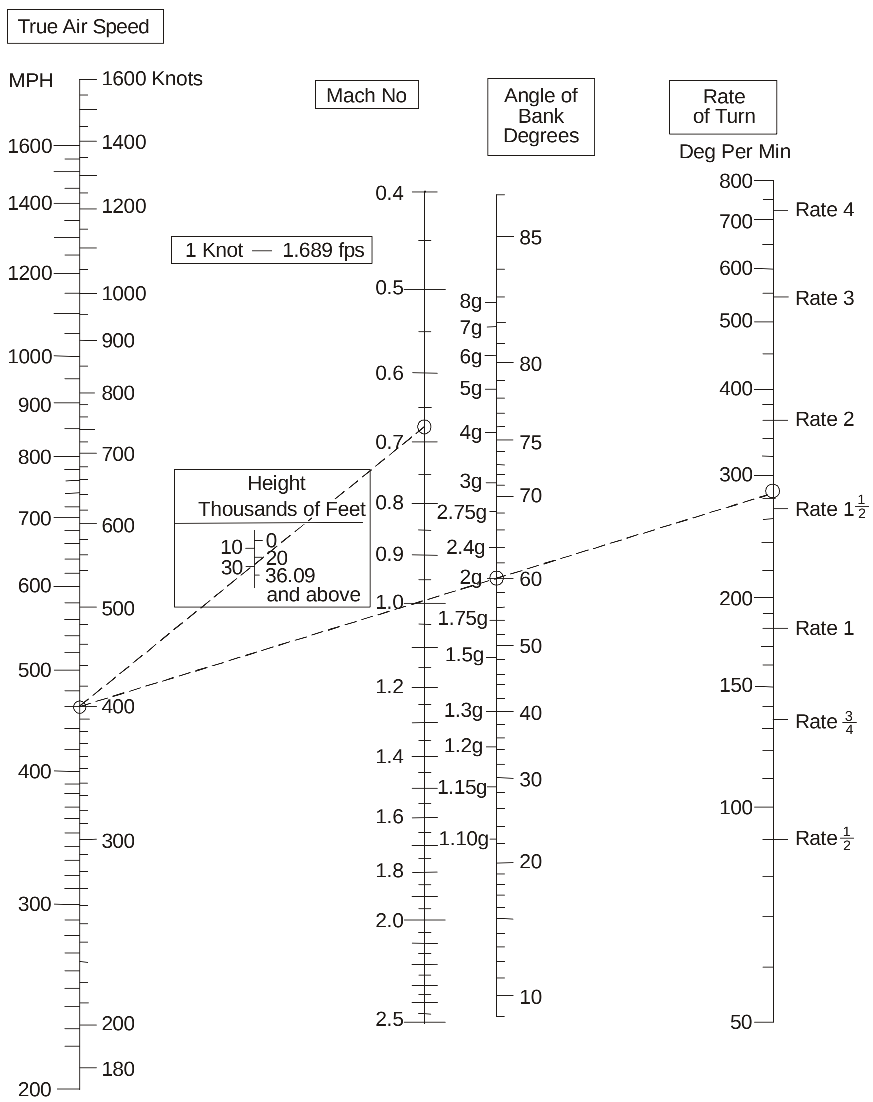
Effect of Flap

40. The lowering of flap produces more lift but also more drag at any given EAS. A smaller radius turn may be obtained when flap is lowered provided the flap limiting speed is not a critical factor and provided that there is sufficient thrust to overcome the extra drag.

Nomogram of Turning Performance

41. Fig 18 is a nomogram applicable to any aircraft, from which considerable information can be obtained on turning performance. Some examples of the use of the nomogram are given below.

1-15 Fig 18 Nomogram of Turning Performance



42. If the TAS and altitude are known, then the Mach number can be determined. Assume a TAS of 400 kt at 30,000 ft; the Mach number is found by placing a rule so that 400 kt (on the TAS scale) and 30,000 ft (on the "height in thousands of feet" scale) are in alignment and then reading off the Mach number from its scale. This example is shown by the upper dashed line on the nomogram and gives a Mach number of 0.68M. The example can equally well be worked in reverse, ie the altitude and Mach number can be used to determine the TAS.

43. If the TAS and bank angle are known then the rate of turn can be found. Assume a TAS of 400 kt and an angle of bank of 60º, place a rule on the nomogram to align these figures and then read off the rate of turn, in this case 284º per min, about rate 1½ . The bank scale also shows the g realized in a sustained turn at that angle of bank. The Mach scale has no significance in these examples. Any two known factors can be aligned to determine the unknown third factor; eg to find the bank angle corresponding to a rate 1 turn at 500 kt TAS the rule is aligned on these figures and the bank angle read off from its scale. The nomogram shows clearly that as speed increases the angle of bank must be increased to maintain a constant rate of turn (see examples in Table 1).

Table 1 Increase of Bank Angle

Rate of Turn	*Speed to*	*Bank angle*	*g*
1	200	28º	1.15
1	400	48º	1.5
1	600	58º	2

Angle of Bank Vs Rate of Turn

44. Volume 1, Chapter 16 gives a mathematical explanation of turning. Pilots may need to be able to calculate quickly and easily the angle of bank (AOB) required for a standard rate of turn.

45. An exact formula for converting rate of turn into AOB for a standard rate turn (3º/sec) is:

$$\text{AOB} = 57.296(\arctan(0.0027467 \times \text{TAS})) \qquad (1)$$

46. The above formula is not particularly user friendly to the pilot in the cockpit. The Federal Aviation Authority have published the following rule of thumb which states that:

To determine the approximate angle of bank required for a standard rate turn use 15% of the TAS. Thus:

$$\text{AOB} \sim 0.15\ \text{TAS} \qquad (2)$$

(Where ~ means approximately equal to).

e.g. At 100 kt approximately 15º angle of bank is required (15% of 100).

47. An alternative approximation is:

$$\text{AOB} \sim (0.1\ \text{TAS}) + 5 \qquad (3)$$

48. AOBs for a range of TAS values are shown in Table 2. The values have been calculated using each of the three formulas given above. Percentage differences from the exact value (formula 1) are given for the approximate formulas, (2) and (3). The data is presented graphically in Fig 19.

Table 2 Values of Angle of Bank Required for a Range of TAS Values Using Approximations

TAS	200	250	275	300	350	400	450	500	550	600	650	700
Formula												
(1)	28.7	34.5	37.1	39.5	44.0	48.0	51.0	54.0	56.5	59.0	61.0	62.5
(2)	30	37.5	41.2	45	52.5	60	67.5	75	82.5	90	97.5	105
% diff (2) from (1)	4.5	8.7	11.4	13.9	19.3	25.0	32.4	38.9	46.0	52.5	59.8	68.0
(3)	25	30	32.5	35	40	45	50	55	60	65	70	75
% diff (3) from (1)	12.9	13.0	12.2	11.4	9.1	6.3	2.0	1.9	6.2	10.2	14.8	20.0

Note: The values relate to a standard rate of turn of 3º/sec.

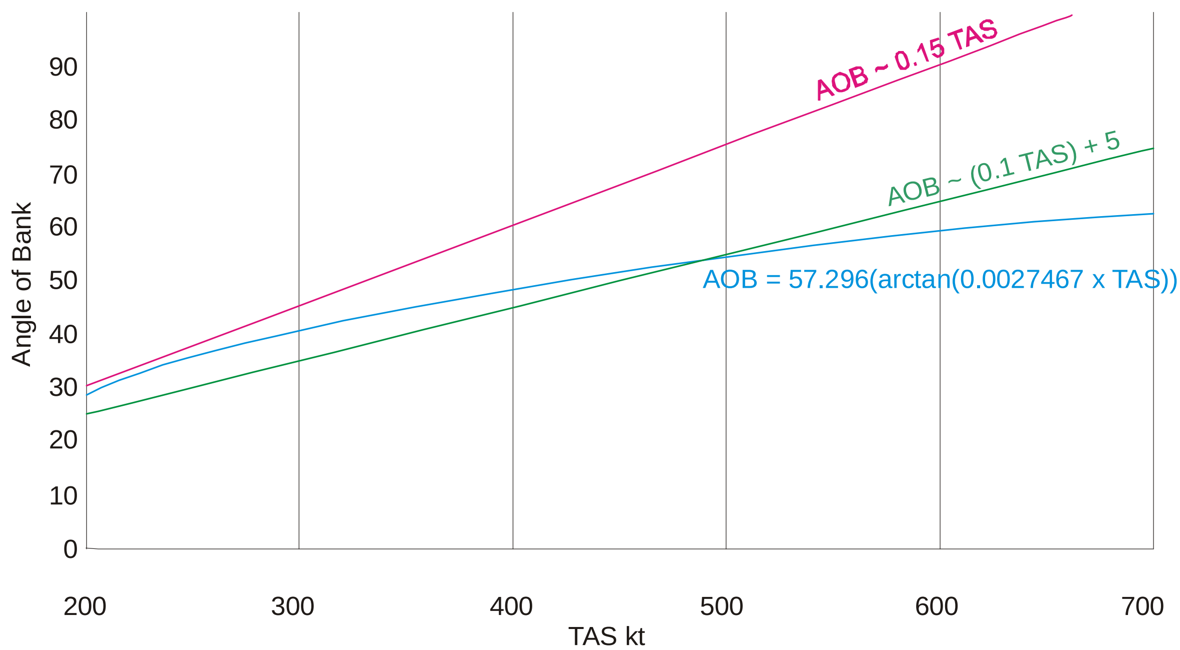
Formulas used:

Angle of Bank = 57.296(arctan(0.0027467 x TAS)) (1)

Angle of Bank ~ 0.15 TAS (2)

Angle of Bank ~ (0.1 TAS) + 5 (3)

1-15 Fig 19 Graph of AOB Against TAS



49. It can be seen from Table 2 and Fig 19 that up to 280 kt, formula 2 is the closest fit to formula 1. Above 280 kt, formula 3 is the closer fit, up to a speed approaching 600 kt where the divergence of the values becomes more significant. Above 600 kt neither of the approximations will give reasonable results.

SUMMARY

Manoeuvring Considerations

50. During a manoeuvre an aircraft is not in a state of equilibrium since an out-of-balance force is required to deflect it continuously from a straight line. This force is called centripetal force.

51. The manoeuvre envelope is used:

a. To specify design requirements for a new aircraft.

b. By manufacturers to illustrate the performance of their products.

c. As a means of comparing the capabilities of different types.

52. The axes of the manoeuvre envelope are load factor and velocity, and from it can be read:

a. Basic stalling speed.

b. Available load factor at any height and speed.

c. Maximum EAS at any height.

d. Stalling speed at any height and load factor.

e. Maximum permitted load factor.

f. Maximum EAS.

g. Compressibility Mach number.

h. Rolling g limits.

53. Variations in CLmax are caused by:

a. Compressibility effects:

(1) Separation behind shockwaves.

(2) Reduction of control effectiveness behind shockwaves.

(3) Buffet on the tailplane making it impossible to hold an accurate angle of attack.

b. Change in Reynolds number.

c. Increase in the adverse pressure gradient with speed.

54. For minimum radius turns the following must be satisfied:

a. Wing loading must be as low as possible.

b. The air must be as dense as possible.

c. The maximum value of the product of CL value and angle of bank must be obtained.

55. For maximum rate turns the following must be satisfied:

a. Wing loading must be as low as possible.

b. The maximum value of the product of angle of bank, speed and CL value must be obtained.

In both cases the use of thrust will improve turning performance.

CHAPTER 16 - MATHEMATICAL EXPLANATION OF TURNING

Minimum Radius Level Turns

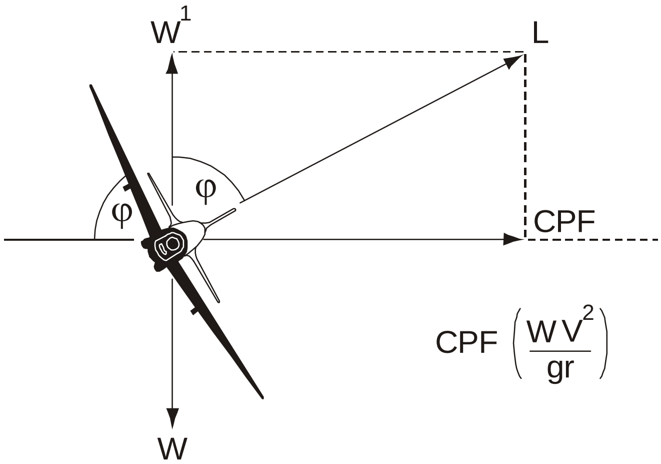
1. From Fig 1 it can be seen that:

$$\tan \varphi = \frac{WV^2}{Wgr} = \frac{V^2}{gr}$$

where φ is the angle of bank
V is the TAS in mps
r is in m
g is 9.8ms²

$$\therefore \quad r = \frac{V^2}{g \tan \varphi}$$

1-16 Fig 1 Forces in a Turn



φ = Angle of Bank
W^1 = Vertical Component of Lift = W
CPF = Horizontal Component of Lift = Centripetal Force

However, not all aircraft can achieve the same angle of bank at the same TAS and so the equation for radius may therefore usefully be broken down further:

The V in the formula is the manoeuvre stalling speed (V_M).

$$V_M = V_B\sqrt{\text{load factor}}$$

$$= V_B\sqrt{n}. \quad \text{But } n = \frac{L}{W} = \frac{1}{\cos\varphi} \quad \text{(from Fig 1)}$$

$$\therefore V_M{}^2 = V_B{}^2 \frac{1}{\cos\varphi}$$

$$\text{So } r = \frac{V_B{}^2}{g\tan\varphi}\frac{1}{\cos\varphi}$$

$$\therefore r = \frac{V_B{}^2}{g\sin\varphi}$$

At the basic stalling speed:

$$L = W = C_{L\,max}\frac{1}{2}\rho V_B{}^2 S$$

$$\therefore V_B{}^2 = \frac{2W}{C_{L\,max}\,\rho S}$$

$$\therefore r = \left(\frac{1}{C_{L\,max}\sin\varphi}\right)\left(\frac{W}{S}\right)\left(\frac{1}{\rho}\right)\left(\frac{2}{g}\right)$$

This means that for minimum radius:

a. $\left(\frac{1}{C_{L\,max}\sin\varphi}\right)$ must be a minimum. $\frac{1}{\sin\varphi}$ reduces towards unity as φ increases towards 90º. This requires an increase in speed to obtain the necessary increased lift. The speed should be increased up to that speed where $\left(\frac{1}{C_{L\,max}\sin\varphi}\right)$ stops decreasing because of a reducing C_{Lmax}.

b. Wing loading $\left(\frac{W}{S}\right)$ must be minimum.

c. ρ must be maximum; this occurs at MSL.

Maximum Rate Turns

2. From Fig 1:

$$\sin\ \varphi = \frac{CPF}{L} = \frac{W}{g}\frac{V^2}{r}\frac{1}{L}$$

$$\text{Rate of turn } (\omega) = \frac{V}{r}$$

$$\therefore \omega = \frac{L\ \sin\varphi\, g}{WV}.\ \text{But } L = C_L\frac{1}{2}\rho V^2 S$$

$$\therefore \omega = \frac{C_L\frac{1}{2}\rho V^2 S \sin\varphi\, g}{WV}$$

$$= (C_L V \sin\varphi)\rho\left(\frac{S}{W}\right)\left(\frac{g}{2}\right)$$

This means that for a maximum rate turn:

a. $(C_L V \sin\varphi)$ must be a maximum. Sin φ increases as φ increases towards 90º. This bank increase requires more speed, so speed should be increased as long as C_L does not decrease faster than (V sin φ) is increasing.

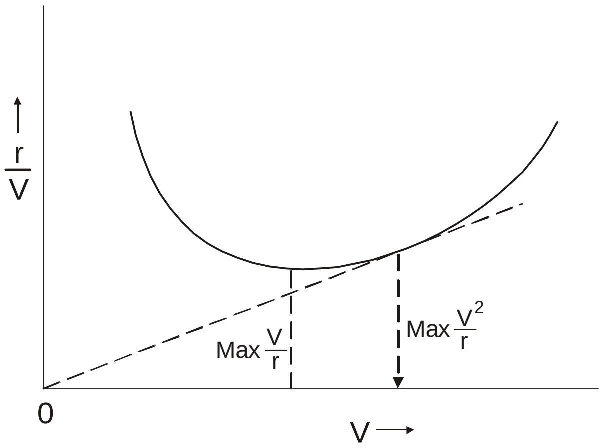
b. Wing loading $\left(\frac{W}{S}\right)$ must be minimum.

c. ρ must be maximum; this occurs at MSL.

Maximum g Loading

3. If C_{Lmax} were constant, the speed for minimum radius, maximum rate and maximum *g* loading would be the same. However C_{Lmax} decreases with increasing Mach number; maximum *g* loading (or n_{max}) depends on the lift produced; the acceleration towards the centre of the turn $\left(\frac{V^2}{r}\right)$ is proportional to this lift (see Fig 1). Therefore, if the speed for maximum $\frac{V^2}{r}$ is found, this will give nmax. In Fig 2, $\frac{r}{V}$ is plotted against V. The speed at which the tangent from the origin just touches this curve will be that for maximum $\frac{V^2}{r}$ and therefore maximum *g* loading. This is clearly at a higher speed than the minimum $\frac{r}{V}$ (or maximum $\frac{V}{r}$ which is maximum rate).

1-16 Fig 2 Variation of r/V with V



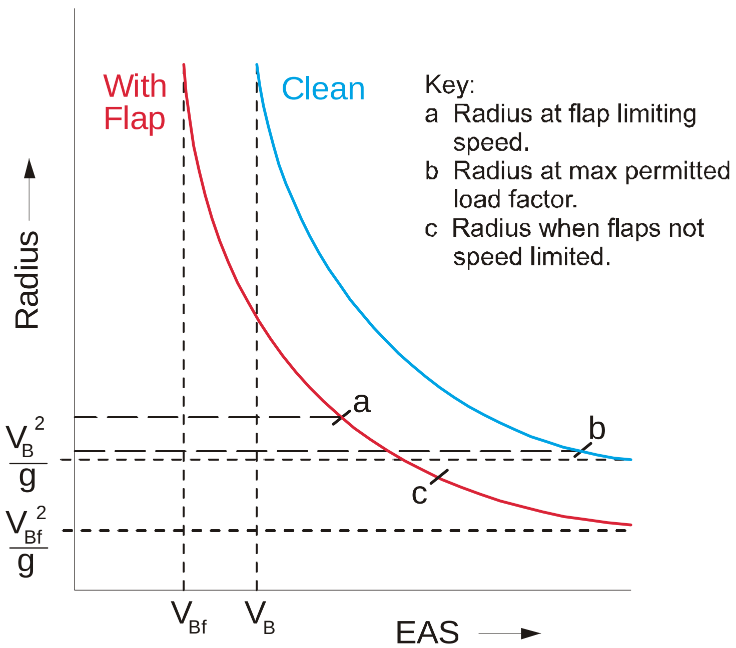
Effect of Thrust

4. The thrust line of an aircraft is vectored inwards in a turn so that thrust assists the lift in overcoming weight and centripetal force.

Use of Flap

5. The lowering of flap produces more lift but also more drag at any given EAS. A smaller radius turn may be obtained when flap is lowered provided that the flap limiting speed is not a critical factor and provided that there is sufficient thrust to overcome the extra drag. The theoretical absolute minimum radius will have been decreased since it depends on $\frac{V_{Bf}^{\,2}}{g}$ where V_{Bf} is the new stalling speed with flap (Volume 1, Chapter 15, para 34). Two possible cases, one where lowering flap is beneficial and one where it is not, are illustrated in Fig 3.

1-16 Fig 3 Comparison of Turning Radius with and without Flap



CHAPTER 17 - STABILITY

Introduction

1. The study of aircraft stability can be extremely complex and so for the purpose of this chapter the subject is greatly simplified: stability is first defined in general terms and it will then be seen how the aircraft designer incorporates stability into an aircraft. The subject is dealt with under two main headings: static stability and dynamic stability.

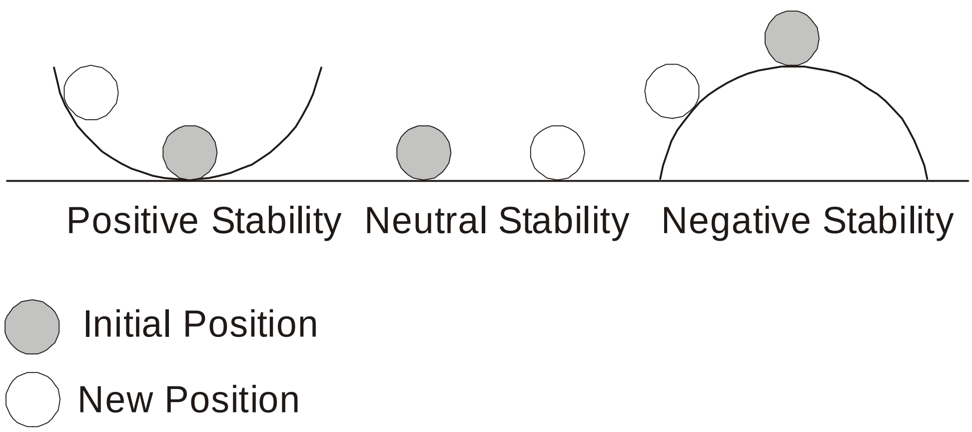
STATIC STABILITY

Definitions

2. To quote Newton’s first law again, "a body will tend to remain in a state of rest or uniform motion unless disturbed by an external force". Where such a body is so disturbed, stability is concerned with the motion of the body after the external force has been removed. Static stability describes the immediate reaction of the body after disturbance, while dynamic stability describes the subsequent reaction. The response is related to the original equilibrium state by use of the terms positive, neutral and negative stability. Positive stability indicates a return towards the position prior to disturbance, neutral stability the taking up of a new position of a constant relationship to the original, whereas negative stability indicates a continuous divergence from the original state. Note that, in colloquial usage, positively stable and negatively stable are usually 'stable' and 'unstable', respectively.

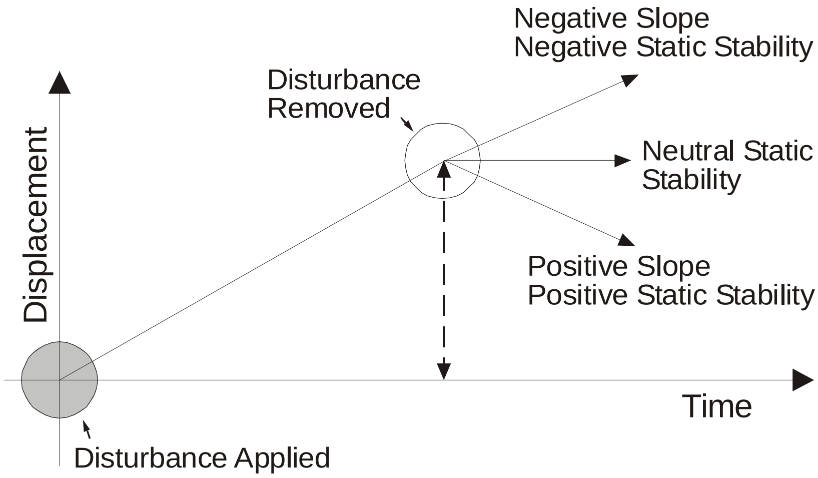
3. In order to relate the response of a body to its initial equilibrium state it is useful at this stage to use an analogy; the 'bowl and ball' can be used to illustrate this idea (see Fig 1). If the ball is displaced from its initial position to a new position, the reaction of the ball will describe its static stability. If it tends to roll back to its original position, it is said to have positive stability; if it tends to roll further away from its original position, it has negative stability and if the ball tends to remain in its new position, it has neutral stability.

1-17 Fig 1 Static Stability Analogies



4. The concept of stability 'degree' can be expressed more usefully in graphical form (Fig 2). Displacement, plotted on the vertical axis, may refer to any system, e.g. distance, moments, volts, etc. No scale is given to the horizontal axis which may vary from microseconds to hours, or even years.

1-17 Fig 2 Graph of the Degrees of Stability



5. Plotting the response in this form makes it possible to measure the actual degree of stability using the following two parameters:

a. The sign of the slope indicates whether the response is favourable or unfavourable.

b. The slope of the curve is a measure of the static stability.

6. Before considering the response of the aircraft to disturbance it is necessary to resolve the motion of the aircraft into components about the three body axes passing through the CG.

Axis	***Motion (About the Axis)***	***Stability***
Longitudinal (x)	Roll (p)	Lateral
Lateral (y)	Pitch (q)	Longitudinal
Normal (z)	Yaw (r)	Directional (Weathercock)

7. It is important to realize that the motion involved is angular velocity and the disturbance assumed is an angular displacement. In the first instance it is helpful to consider these components separately although, in other than straight and level flight, the motion of the aircraft is more complex, eg in a level turn the aircraft is pitching and yawing.

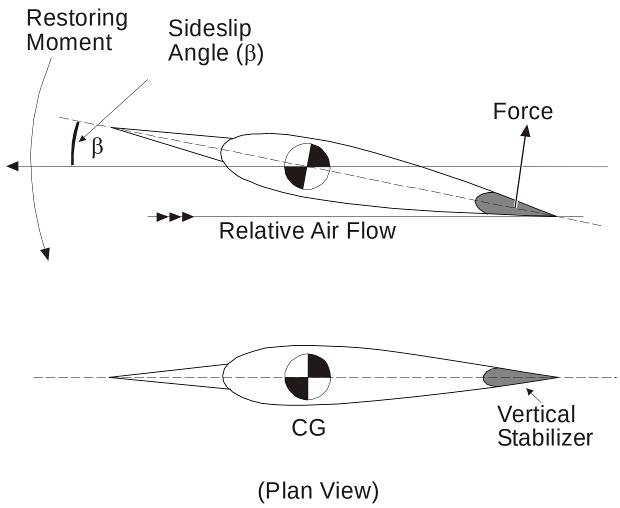
Directional Stability

8. A simple approach to both directional and longitudinal stability is to consider a simple dart. The flights or vanes of a dart ensure that the dart is aligned with the flight path. Consider first the pair of vanes which impart positive directional stability to the dart; these may be referred to as the vertical stabilizers. Fig 3 shows how a displacement in yaw through an angle β, resulting in sideslip, produces a restoring moment and therefore positive directional (static) stability. Two pointsare worth noting:

a. The dart rotates about the centre of gravity (CG).

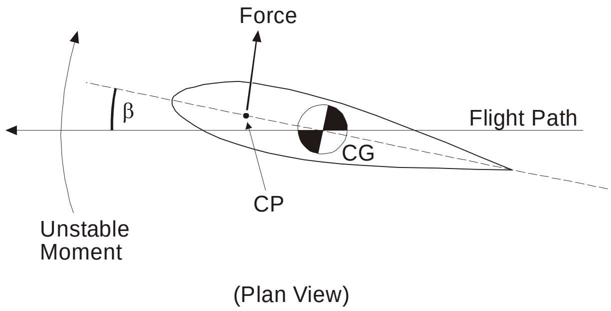
b. The momentum of the dart momentarily carries it along the original flight path, ie the Relative Air Flow (RAF) is equal and opposite to the velocity of the dart.

1-17 Fig 3 The Positive Stability of a Dart



9. An aerodynamic shape like a fuselage or drop-tank may be unstable. Reference to Fig 4 shows that this occurs when the centre of pressure (CP) is in front of the CG.

1-17 Fig 4 The Negative Static Stability of a Streamline Body when CP is ahead of CG



It is necessary therefore to add a vertical stabilizer or 'fin' to produce positive directional stability and this has the effect of moving the CP behind the CG (Fig 5). In general, it may be said that the keel surface of the fuselage (ahead of the CG) has an unstable influence, while the keel surface behind the CG has a stable influence. For simplicity, the rudder is considered to be 'locked'.

1-17 Fig 5 Positive Static Stability with the Addition of a Fin

Fig 5a Plan View

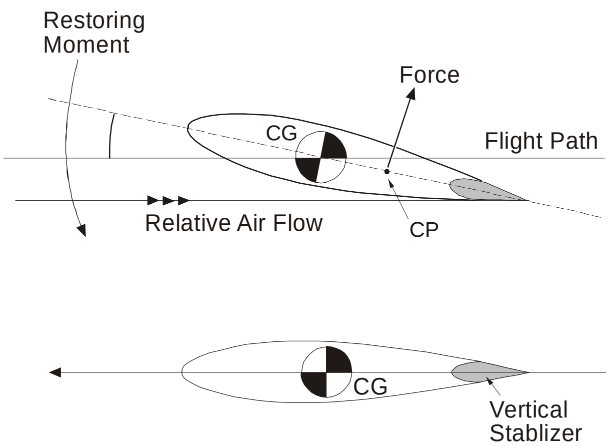
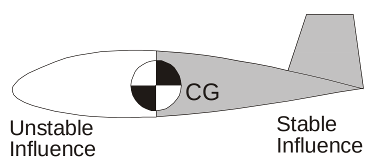


Fig 5b Side View



10. For a given displacement, and therefore sideslip angle, the degree of positive stability will depend upon the size of the restoring moment which is determined mainly by:

a. Design of the vertical stabilizer.

b. The moment arm.

11. **Design of the Fin and Rudder**. The vertical stabilizer is a symmetrical aerofoil and it will produce an aerodynamic force at positive angles of attack. In sideslip, therefore, the total side force on the fin and rudder will be proportional to the 'lift coefficient' and the area. The lift coefficient will vary, as on any aerofoil, with aspect ratio and sweepback. At high angles of sideslip it is possible for the fin to stall and to avoid this the designer can increase the stalling angle by increasing the sweepback, decreasing the aspect ratio, or by fitting multiple fins of low aspect ratio.

12. **Moment Arm**. The position of the centre of gravity, and therefore the distance between the CG and the centre of pressure of the vertical stabilizer, may be within the control of the pilot. Forward movement of the CG will lengthen the moment arm thereby increasing the directional stability; aft movement will decrease the directional stability.

Longitudinal Stability

13. The analogy used in para 8 can usefully be used to introduce the concept of static longitudinal stability. In this case the dart is viewed from the side and the horizontal stabilizers produce a pitching moment (M) tending to reduce the displacement in pitch.

14. On an aircraft, the tailplane and elevators perform the functions of a horizontal stabilizer and the conclusions reached in para 10 will be equally valid. For simplicity, the explanation is limited to stick fixed static stability, i.e. elevators locked.

15. Fig 6a shows a wing with the CP forward of the CG by the distance x. A nose-up displacement will increase the angle of attack, increase the lift (L) by the amount dL and increase the wing pitching moment by the amount dLx. The result is to worsen the nose-up displacement, an unstable effect. In Fig 6b, the CP is aft of the CG and the wing moment resulting from a displacement in pitch will be stabilizing in effect. The pitching moment is also affected by the movement of the CP with angle of attack and it follows therefore that the relative positions of the CP and CG determine whether the wings have a stable or unstable influence. Taking the worst case, therefore, the wing may have an unstable influence and the horizontal stabilizer must be designed to overcome this.

1-17 Fig 6 Variations in the Position of CP and CG

Fig 6a Unstable Contribution

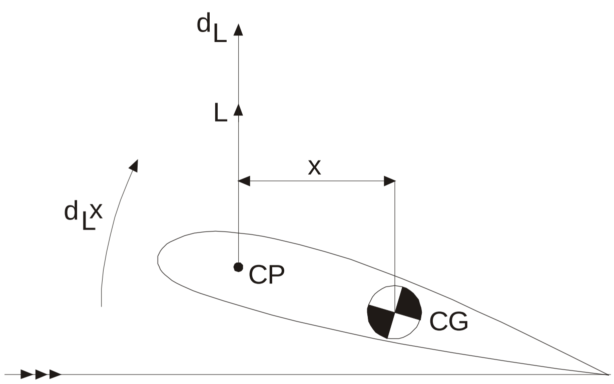
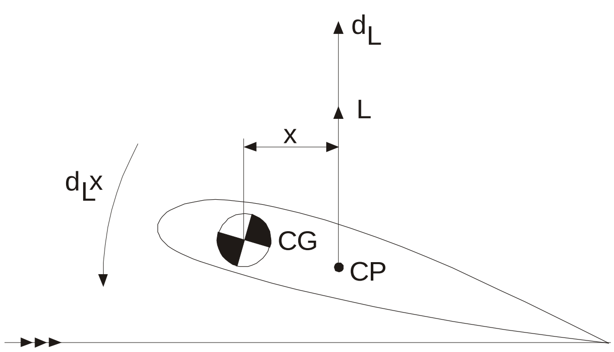
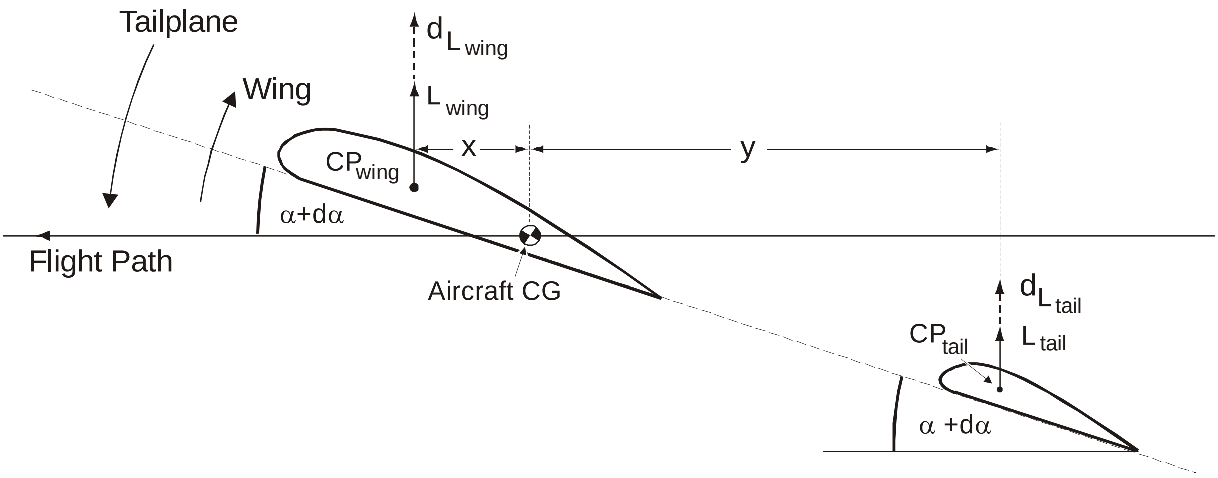


Fig 6b Stable Contribution



16. The simplified diagram at Fig 7 illustrates a system of forces due to displacement in pitch, in this case an increase in angle of attack. The tail contribution must overcome the unstable wing (and any other) contribution, for positive static longitudinal stability.

1-17 Fig 7 Changes in Forces and Moments due to a Small Nose-up Displacement(dα)



17. The degree of positive stability for a given change in angle of attack depends upon the difference between the wing moment and the tail moment. This difference is called the restoring moment,

i.e. (Total Lift $_{tail}$)y – (Total Lift $_{wing}$)x = Net pitching moment.

18. The main factors which affect longitudinal stability are:

a. Design of the tailplane.

b. Position of the CG.

19. **Design of the Tailplane**. The tailplane is an aerofoil and the lift force resulting from a change in angle of attack will be proportional to the C_{Ltail} and the area. The increment in lift from the tail will depend upon the slope of its C_L curve and will also be affected by the downwash angle behind the wing (if the downwash changes with angle of attack). The tail design features which may affect the restoring moment are therefore:

a. Distance from CP_{tail} to CG (moment arm).

b. **Tail Area**. The total lift provided by the wing = CL_{wing} qS and the total lift produced by the tail = CL_{tail} qS. For a given aerofoil of given planform, the C_L varies with angle of attack at a constant q (EAS). Therefore, in comparing tail moments with wing moments, it is necessary only to compare the respective area(s) and moment arms (CG position).

c. **Tail Volume**. The product of the tailplane area × moment arm is known as the tail volume. The ratio of the tail volume to the wing volume is the main parameter used by the designer in determining the longitudinal stability of the aircraft.

d. **Planform**. As was seen in Volume 1, Chapter 4, the slope of the C_L curve for a lifting surface is affected by aspect ratio, taper and sweepback. The planform of the tailplane therefore affects the change in C_L with change in angle of attack caused by a disturbance. For example, the C_L increments will be lower on a swept-back tail than on one of rectangular planform.

e. **Wing Downwash**. Where a disturbance in angle of attack results in a change in the downwash angle from the wings, the effective angle of attack at the tail is also changed. For example, if the aircraft is displaced nose-up and the downwash angle is increased, then the effective angle of attack on the tailplane is reduced. The total tail lift will not be as great as it would otherwise have been and so the restoring moment is reduced. This decrease in stability is compensated for by moving the CG further forward, thereby increasing the moment arm.

20. **Position of the CG**. The position of the CG may be marginally under the control of the pilot. From Fig 7 it can be seen that its position affects the ratio of the tail moment to the wing moment and therefore the degree of stability. In particular:

a. Aft movement of the CG decreases the positive stability.

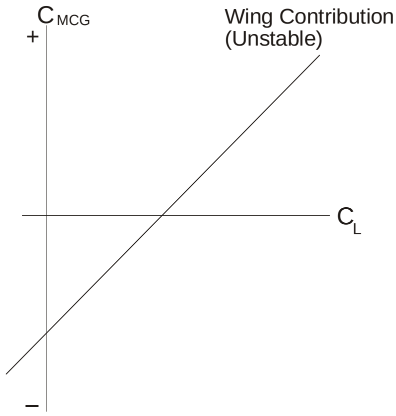
b. Forward movement of the CG increases the positive stability.

Because the position of the CG affects the positive longitudinal stability, it also affects the handling characteristics in pitch. The aerodynamic pitching moment produced by deflecting the elevators must override the restoring moment arising from the aircraft's positive stability, i.e. the stability opposes manoeuvre. For a given elevator deflection there will be a small response in an aircraft with a forward CG (stable condition) and a large response in an aircraft with an aft CG (less stable condition).

21. **Neutral Point**. Aircrew Manuals for every aircraft give the permitted range of movement of the CG. The forward position is determined mainly by the degree of manoeuvrability required in the particular aircraft type. Of greater importance to the pilot is the aft limit for the CG. If the CG is moved aft, outside the permitted limits, a position will eventually be reached where the wing moment (increasing) is equal to the tail moment (decreasing). In this situation the restoring moment is zero and the aircraft is therefore neutrally stable. This position of the CG is known as the neutral point. The aft limit for the CG, as quoted in the Aircrew Manual, is safely forward of the neutral point. If the loading limits for the aircraft are exceeded, it is possible to have the CG position on, or aft of, the neutral point. This unsafe situation is aggravated when the controls are allowed to 'trail', i.e. stick free.

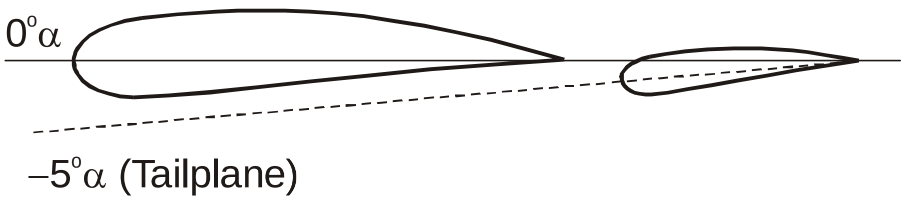
22. **CG Margin (Stick Fixed)**. The larger the tail area, the larger the tail moment, and so the further aft is the CG position at which the aircraft becomes neutrally stable. The distance through which the CG can be moved aft from the quoted datum, to reach the neutral point, is called the static or CG margin, and is an indication of he degree of longitudinal stability. The greater the CG margin, the greater the stability, e.g. a training, or fighter aircraft, may have a margin of a few inches but a large passenger aircraft may have a margin of a few feet. For stability in pitch it is necessary that when the angle of attack is temporarily increased a correcting force will lower the nose to reduce the angle of attack. As illustrated in Fig 6, if the CG is behind the centre of pressure an increase in the angle of attack results in a pitch-up or positive pitching moment coefficient. Thus the mainplane on its own has an unstable influence. If the CG is forward of the centre of pressure the pitching moment coefficient will be negative, a stable contribution. Fig 8 plots the coefficient of pitching moment (C_{MCG}) against the coefficient of lift, C_L for the mainplane only.

1-17 Fig 8 C_{MCG} Against C_L - Mainplane Only

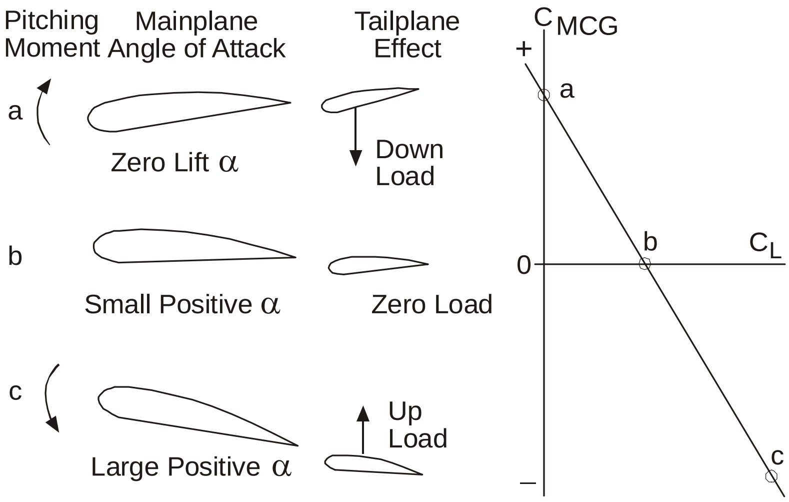


23. **Longitudinal Dihedral**. Longitudinal dihedral means that the tailplane is set at a lower angle of incidence than the mainplane, as illustrated in Fig 9. The effect of longitudinal dihedral is shown in Fig 10, where it can be seen that when the mainplane is at zero lift angle of attack, the tailplane is at a negative angle of attack at which it produces a downward force which tends to pitch the aircraft nose-up. When the mainplane is at a small positive angle of attack, such as is normally to be expected in straight and level flight, the tailplane produces no pitching moment. When the mainplane angle of attack increases the tailplane produces a nose-down pitching moment. Thus if the aircraft is disturbed in pitch, the tailplane will tend to restore it to level flight. The longitudinal dihedral therefore provides a stabilizing influence.

1-17 Fig 9 Longitudinal Dihedral

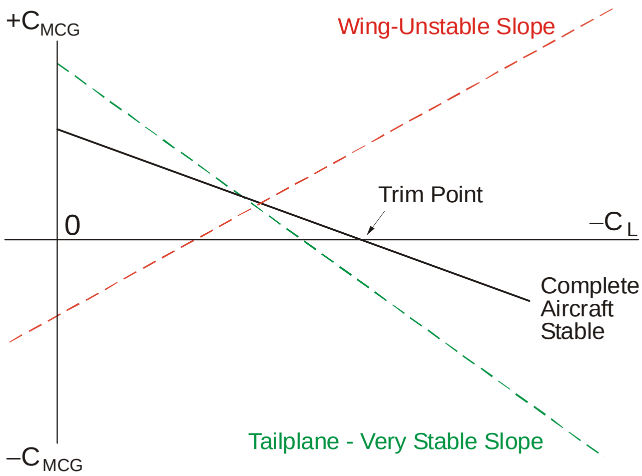


1-17 Fig 10 C_{MCG} Against C_L - Tailplane Contribution



24. The combined effects of wing and tail are shown in Fig 11. The negative slope of the graph shows positive longitudinal stability. The amount of longitudinal stability is measured by the slope of $\frac{C_{MCG}}{C_L}$.

1-17 Fig 11 Combined Effects of Wing and Tail



25. **Trim Point (Stick Fixed)**. Trim point is the C_L or angle of attack at which the overall moment about the CG is zero. This is where the wing moment is equal to the tail moment and the aircraft is then 'in trim'. The angle of attack or level flight speed at which this occurs is determined by the degree of longitudinal dihedral.

26. **Elevator Angle to Trim**. If the angle of attack is increased from the trim point, the aircraft's longitudinal stability will produce a stable, nose-down pitching moment. To maintain the new angle of attack, an equal and opposite moment, nose-up, will be required from the elevators. When this is achieved, by raising the elevators, a new trim point is established, ie at the higher angle of attack on the mainplane, the tail has been made to produce a greater nose-up moment by altering the effective camber on the tail. The reverse applies when the angle of attack on the mainplane is reduced. This does not usually affect the positive longitudinal stability.

Aerodynamic Centre

27. In text books on stability it is usual to find that the aerodynamicist writes of the 'aerodynamic centre' (AC), rather than the centre of pressure. The AC is a point within the aerofoil, and usually ahead of the CP, about which the pitching moment is independent of angle of attack; it is a convenient and calculated datum for the mathematical treatment of stability and control, and a full explanation may be found in Volume 1, Chapter 4.

Stick-free Longitudinal Stability

28. If the elevator is allowed to trail freely, the change in tail force due to a displacement will depend on the position taken up by the floating elevator. Usually the elevator will trail with the relative airflow and this will reduce the tail moment.

29. Under these conditions, with the tail moment reduced, the balance between the tail and wing moments is changed and so the position for the CG, about which the moments are equal, will be further forward, because the less effective tail requires a longer moment arm. That is, the neutral point is further forward, so reducing the stick-free CG margin. Since this margin is a measure of the longitudinal stability, it follows that when the elevators are allowed to float free the longitudinal stability is reduced.

Manoeuvre Stability (Steady Manoeuvres Only)

30. In the preceding paragraphs, the longitudinal static stability was discussed with respect to a disturbance in angle of attack from the condition of trimmed level flight. A pilot must also be able to hold an aircraft in a manoeuvre and the designer has to provide adequate elevator control appropriate to the role of the aircraft. The following paragraphs consider the effects on an aircraft of a disturbance in angle of attack and normal acceleration.

31. It should be carefully noted that the initial condition is, as before, steady level flight. The difference between static and manoeuvre stability is that manoeuvre stability deals with a disturbance in angle of attack (α) and load factor (n) occurring at constant speed, whereas static stability deals with a disturbance in angle of attack at constant load factor (n = 1).

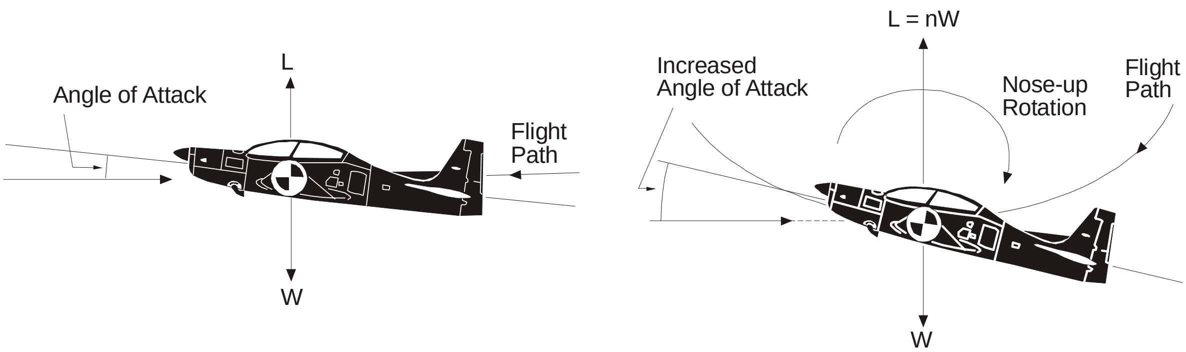
32. If an aircraft is trimmed to fly straight and level (the initial condition, Fig 12a), and is then climbed, dived and pulled out of the dive so that at the bottom of the pull-out it is at its original trimmed values of speed and height (Fig 12b), then the aircraft can be considered as having been 'disturbed' from its initial condition in two ways, both contributing to the overall manoeuvre stability:

a. It now has a greater angle of attack to produce the extra lift required to maintain a curved flight path (L = nW). This is the same as the static stability contribution discussed earlier.

b. It has a nose-up rotation about its CG equal to the rate of rotation about its centre of pull-out.

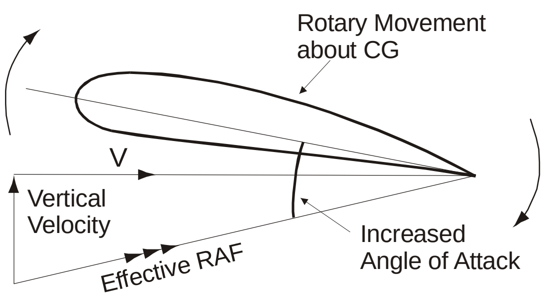
1-17 Fig 12 Forces Acting on an Aircraft in a Steady Manoeuvre

Fig 12a Level Flight **Fig 12b Pull-out**



33. Because the aircraft is rotating about its own CG, the tailplane can be considered to be moving downwards relative to the air or, alternatively, the air can be considered to be moving upwards relative to the tailplane. In either case the effective angle of attack of the tailplane will be increased (see Fig 13); thus the manoeuvre stability is greater than the static stability in level flight.

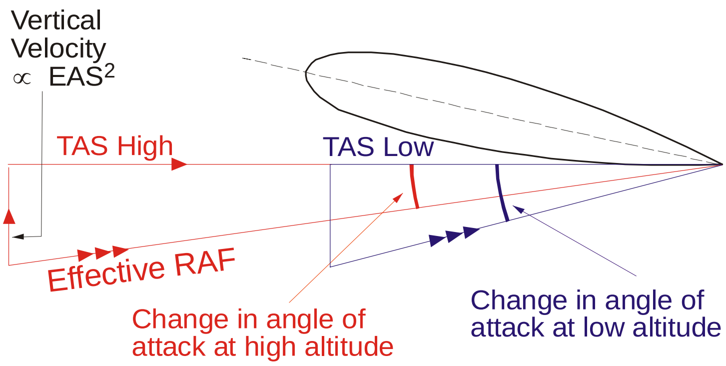
1-17 Fig 13 Increase in Tailplane Angle of Attack Due to its Vertical Velocity



34. If the aircraft's longitudinal stability is greater in manoeuvre, the position of the CG which achieves neutral stability will be further aft than for the straight and level case. This position of the CG is called the manoeuvre point (corresponding to the neutral point) and the distance between the CG and the manoeuvre point is called the manoeuvre margin. It will be seen that for a given position of the CG, the manoeuvre margin is greater than the CG margin.

35. **Effect of Altitude**. Consider an aircraft flying at two different heights at the same EAS and apply the same amount of up elevator in each case. The elevator produces a downward force on the tailplane to rotate the aircraft about its centre of gravity and increase the angle of attack of the wing. The rate of change of pitch attitude is dependent upon the magnitude of the force applied to the tailplane and also the TAS. As shown in Fig 14 the force is proportional to EAS^2 but its effect upon the change in angle of attack of the elevator is a function of the $\frac{EAS^2}{TAS}$ ratio. Fig 14 shows that the change in angle of attack at sea level is greater than at altitude and therefore the elevator is less effective at altitude and stability is decreased.

1-17 Fig 14 Effect of Altitude on Tailplane Contribution



Lateral Stability (Stick Fixed)

36. When an aircraft is disturbed in roll about its longitudinal axis the angle of attack of the down-going wing is increased and that on the up-going wing is decreased, (see Fig 15). As long as the aircraft is not near the stall the difference in angle of attack produces an increase of lift on the down-going wing and a decrease, on the up-going wing. The rolling moment produced opposes the initial disturbance and results in a 'damping in roll' effect.

1-17 Fig 15 The Damping in Roll Effect

Fig 15a Roll to Starboard

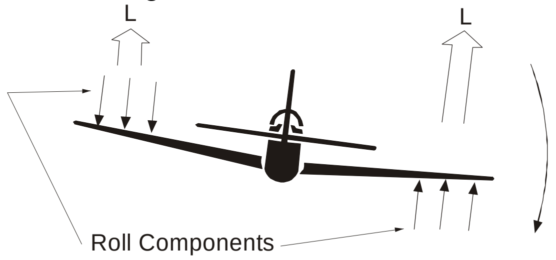


Fig 15b Up-going Wing

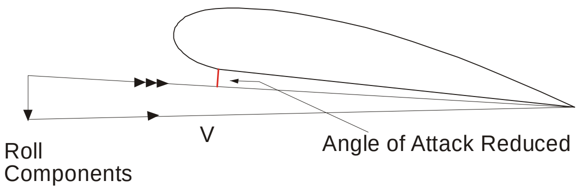
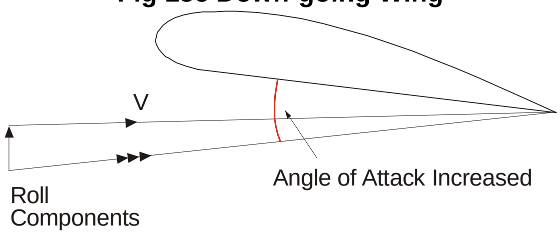


Fig 15c Down-going Wing



37. Since the damping in roll effect is proportional to the rate of roll of the aircraft, it cannot bring the aircraft back to the wings-level position; thus in the absence of any other levelling force, an aircraft disturbed in roll would remain with the wings banked. Therefore, by virtue of the damping in roll effect, an aircraft possesses neutral static stability with respect to an angle of bank disturbance. However, when an aircraft is disturbed laterally it experiences not only a rolling motion but also a sideslipping motion caused by the inclination of the lift vector (see Fig 16a).

38. The forces arising on the different parts of the aircraft as a result of the sideslip produce a rolling moment tending to restore the aircraft to its initial wings-level position. It is seen therefore that the lateral static stability of an aircraft reacts to the sideslip velocity (v) or a displacement in yaw, see Fig 16b. This effect has a considerable influence on the long-term response (lateral dynamic stability) of the aircraft.

1-17 Fig 16 Vector Action of Forward and Sideslip Velocities

Fig 16a

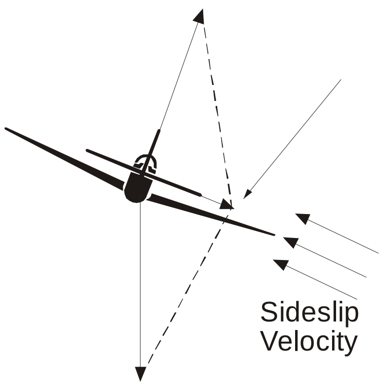
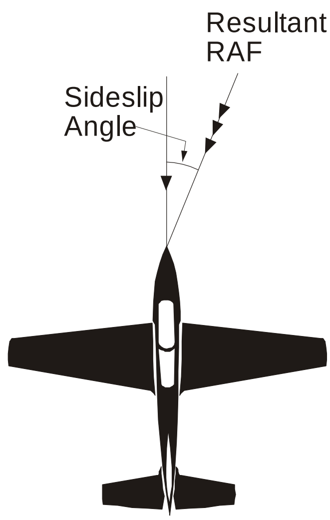


Fig 16b



39. Each different part of the aircraft will contribute towards the overall value of the lateral static stability and these contributions will be of different magnitude depending on the condition of flight and the particular configuration of the aircraft. The more important of these contributions are:

a. Wing contribution due to:

(1) Dihedral.

(2) Sweepback.

b. Wing/fuselage interference.

c. Fuselage and fin contribution.

d. Undercarriage, flap and power effects.

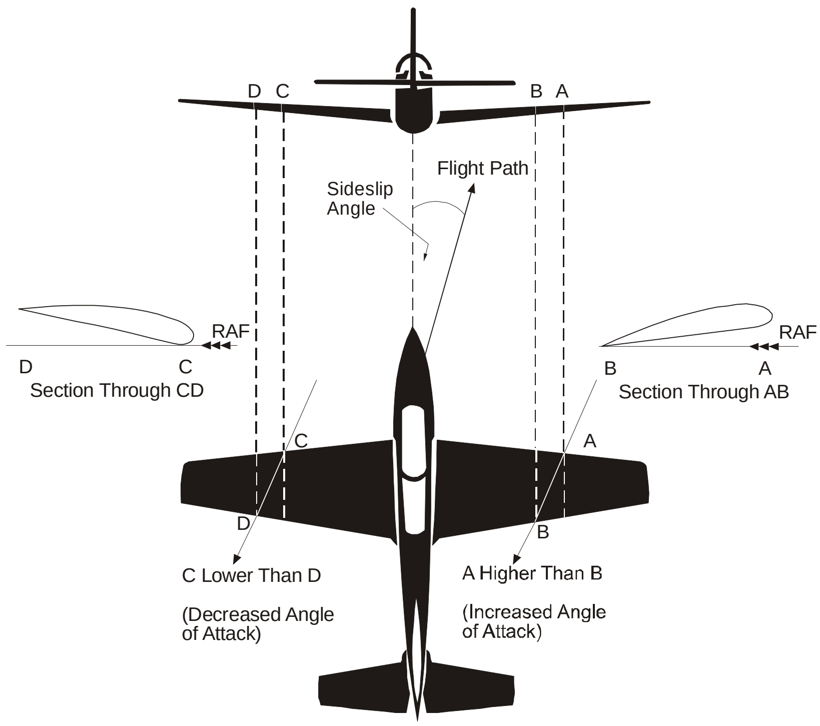
40. **Dihedral Effect**. Dihedral effect can be explained in a number of ways but the explanation illustrated at Fig 17 has the advantage of relating dihedral effect to sideslip angle. It will be seen that due to the geometric dihedral, a point nearer the wing tip (A or D) is higher than a point inboard (B or C). Therefore, a sideslip to starboard will produce the following effects:

a. **Starboard Wing**. The relative airflow will cross the wing (from A to B) at an angle equal to the sideslip angle. Since point A is higher than point B, this will produce the same effect as raising the leading edge and lowering the trailing edge, i.e. increasing the angle of attack. So long as the aircraft is not flying near the stalling speed, the lift will increase.

b. **Port Wing**. Similarly, the angle of attack on the port wing will reduce and its lift decrease.

A stable rolling moment is thus produced whenever sideslip is present (i.e. following a disturbance in yaw). This contribution depends on the dihedral angle and slope of the lift curve. It will therefore also depend on aspect ratio being increased with an increase in effective chord length (see Volume 1, Chapter 4, Fig 11). It is also affected by wing taper. This is one of the most important contributions to the overall stability and, for this reason, the lateral static stability is often referred to as the 'dihedral effect' although there are a number of other important contributions.

1-17 Fig 17 Dihedral Effect



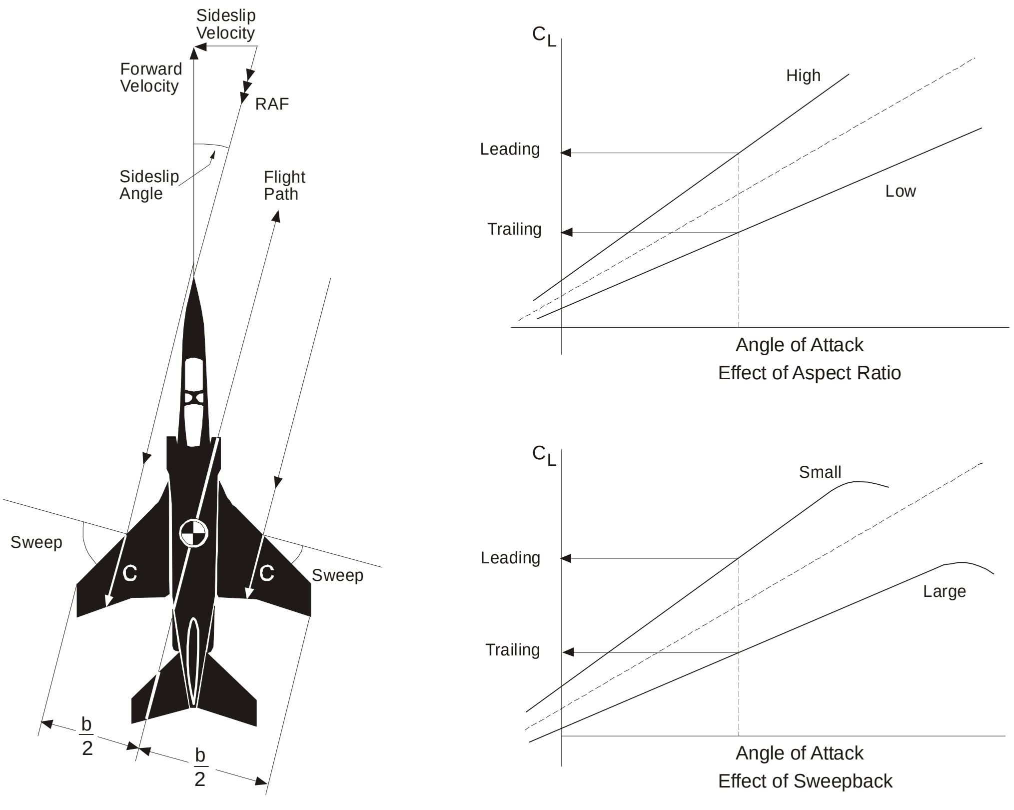
41. **Sweepback**. Wing sweepback has the effect of producing an additional stabilizing contribution thus increasing the 'effective' dihedral of the wing (10° of sweep has about the same effect as 1° of dihedral). Fig 18 illustrates the principal effects on the wing geometry of sideslip.

a. **Angle of Sweep**. The component of flow accelerated by the section camber is proportional to the cosine of the angle of sweep (Volume 1, Chapter 9). The angle of sweep of the leading (low) wing is decreased and that of the trailing wing is increased by the sideslip angle. A stable rolling moment is therefore induced by the sideslip.

b. **Aspect Ratio**. On the leading (low) wing the span is increased and the chord decreased which is an effective increase in aspect ratio. On the trailing (high) wing, the span is decreased and the chord is increased resulting in a reduction in aspect ratio. This again produces a stable rolling moment because the more efficient (low) wing produces more lift.

c. **Taper Ratio**. Another, smaller effect, arises from a tapered wing. An increase in taper ratio, defined as tip chord:root cord, affects the lift coefficient and also produces a small stable rolling moment in sideslip.

1-17 Fig 18 Effect of Sideslip on a Swept Planform



42. **Variation with Speed**. The changes in the slope of the lift curve associated with changes in aspect ratio and sweep result in variations in lift forces of the 'leading' and 'trailing' wings. It follows, therefore, that the contribution of sweep to the lateral (static) stability becomes more important at the higher values of C_L, ie at the lower forward speeds, because the C_L curves are divergent. This is very important because it means that the 'dihedral effect' varies considerably over the speed range of the aircraft. At high speeds a lower angle of attack is needed than for low speeds, therefore the stability at high speeds is much less than at low speeds. To reduce the stability to a more reasonable value at the higher angles of attack, it may be necessary to incorporate some negative dihedral (ie anhedral) on a swept-wing aircraft.

43. **Handling Considerations**. It has been shown that the 'dihedral effect' of sweepback in sideslip produces a strong rolling moment. This has been referred to somewhat imprecisely as roll with yaw. Two applications of this effect at low speeds, where it is strongest, are worth considering:

a. **Crosswind Landings**. After an approach with the aircraft heading into a crosswind from the right, the pilot must yaw the aircraft to port to align it with the runway prior to touchdown. This action will induce a sideslip to starboard and the pilot must anticipate the subsequent roll to port, in order to keep the wings level.

b. **Wing Drop**. The greater tendency of a swept-wing aircraft to drop a wing at a high angle of attack (aggravated by a steep curved approach) may be further increased by a large deflection of corrective aileron. In such cases, the dihedral effect of sweepback may be utilized by applying rudder to yaw the nose towards the high wing - sideslip to the left, roll to the right. It must be said, however, that modern design has reduced the tip-stalling tendency and improved the effectiveness of ailerons at high angles of attack and the problem is not as acute as it might have been in the 'transonic era'.

44. **Wing/Fuselage Interference**.

a. **Shielding Effect**. Most aircraft will be affected by the shielding effect of the fuselage. In a sideslip the section of the trailing wing near the root lies in the 'shadow' of the fuselage. The dynamic pressure over this part of the wing may be less than over the rest of the wing and therefore produces less lift. This effect will tend to increase the 'dihedral effect' and on some aircraft may be quite considerable.

b. **Vertical Location**. A stronger contribution towards lateral stability arises from the vertical location of the wings with respect to the fuselage. It is helpful to start by considering the fuselage to be cylindrical in cross-section. The sideslip velocity will flow round the fuselage, being deflected upwards across the top and downwards underneath. Superimposing a wing in this flow has the following effect, illustrated in Fig 19:

(1) **High Wing**. A high-mounted wing root will lie in a region of upwash on the up-stream side of the fuselage tending to increase its overall angle of attack. Conversely, on the down-stream side of the fuselage the wing root is influenced by the downwash tending to reduce its angle of attack. The difference in lift produced by each wing will cause a restoring moment to increase with sideslip. This effect has been demonstrated to be equivalent to 1°.to 3° of dihedral.

(2) **Low Wing**. The effect of locating the wing on the bottom of the fuselage is to bring it into a region of downwash on the up-stream side and into upwash on the down-stream side of the fuselage. The angle of attack of the leading (low) wing will be decreased and that of the trailing wing increased. This gives rise to an unstable moment equivalent to about 1° to 3° anhedral.

From these facts it can be seen that there is zero effect on lateral stability when the wing is mounted centrally on the fuselage. The effect is lessened as separation occurs at the wing/fuselage junction.

45. **Fuselage/Fin Contributions**. Since the aircraft is sideslipping, there will be a component of drag opposing the sideslip velocity. If the drag line of the aircraft is above the CG the result will be a restoring moment tending to raise the low wing. This configuration is therefore a contribution towards positive lateral stability. Conversely, a drag line below the CG will be an unstable contribution. The position of the drag line is determined by the geometry of the entire aircraft but the major contributions, illustrated in Fig 19, are:

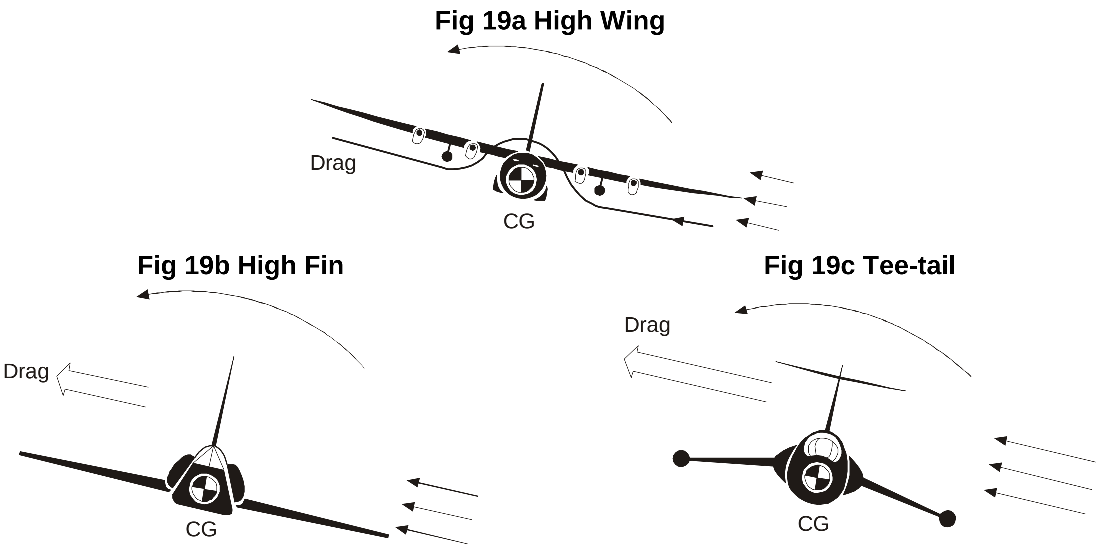
a. High wing.

b. High fin and rudder.

c. Tee-tail configuration.

The tee-tail configuration makes the fin more effective as well as contributing its own extra drag.

1-17 Fig 19 Wing/Fuselage Configurations



46. **Slipstream and Flap Contributions**. Two important effects which reduce the degree of positive lateral stability are illustrated in Fig 20:

a. **Slipstream**. Due to sideslip the slipstream behind the propeller or propellers is no longer symmetrical about the longitudinal axis. The dynamic pressure in the slipstream is higher than the free stream and covers more of the trailing wing in sideslip. The result is an unstable moment tending to increase the displacement. This unstable contribution is worse with flaps down.

b. **Flaps**. Partial-span flaps alter the spanwise distribution of pressure across a wing. The local increase in lift coefficient near the root has the effect of moving the 'half-span' centre of pressure towards the fuselage (in a spanwise sense). The moment arm of the wing lift is thus reduced and a given change in C_L due to the dihedral effect will produce a smaller moment.

The overall lateral stability is therefore reduced by lowering inboard flaps. The design geometry of the flap itself can be used to control this contribution. In particular, a swept-back flap hinge-line will decrease the dihedral effect, whereas a swept-forward hinge-line will increase it.

1-17 Fig 20 Destabilizing Effect of Flap and Slipstream

Fig 20a Destabilizing Effect of Slipstream

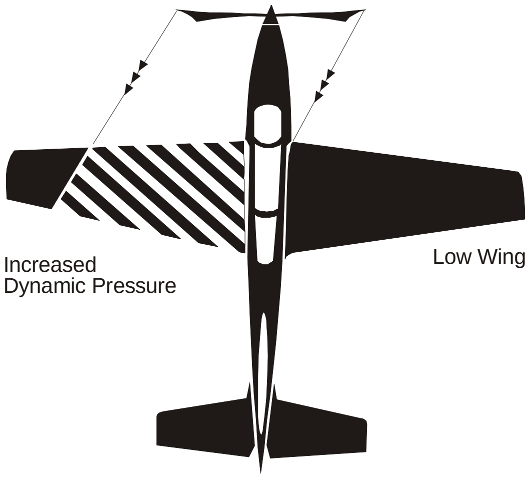
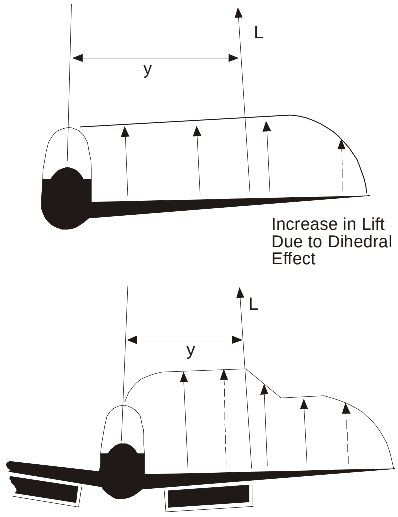


Fig 20b Destabilizing Effect of Flaps



47. **Design Problems**. It is desirable that an aircraft should have positive lateral static stability. If, however, the stability is too pronounced, it could lead to the dynamic problems listed below, some of which are discussed later:

a. Lateral oscillatory problems, i.e. Dutch roll.

b. Large aileron control deflections and forces under asymmetric conditions.

c. Large rolling response to rudder deflection requiring aileron movement to counteract the possibility of 'autorotation' under certain conditions of flight.

DYNAMIC STABILITY

General

48. When an aircraft is disturbed from the equilibrium state, the resulting motion and corresponding changes in the aerodynamic forces and moments acting on the aircraft may be quite complicated. This is especially true for displacement in yaw which affects the aircraft in both yawing and rolling planes.

49. Some of the factors affecting the long-term response of the aircraft are listed below:

a. Linear velocity and mass (momentum).

b. The static stabilities in roll, pitch and yaw.

c. Angular velocities about the three axes. }
} Angular Momentum
d. Moments of inertia about the three axes. }

e. Aerodynamic damping moments due to roll, pitch and yaw.

50. Consider a body which has been disturbed from its equilibrium state and the source of the disturbance then removed. If the subsequent system of forces and moments tends initially to decrease the displacement, then that body is said to have positive static stability. It may, however, overshoot the

equilibrium condition and then oscillate about it. The terms for possible forms of motion which describe the dynamic stability of the body are listed below:

a. Amplitude increased - negative stability.

b. Amplitude constant - neutral stability.

c. Amplitude 'damped' - positive stability.

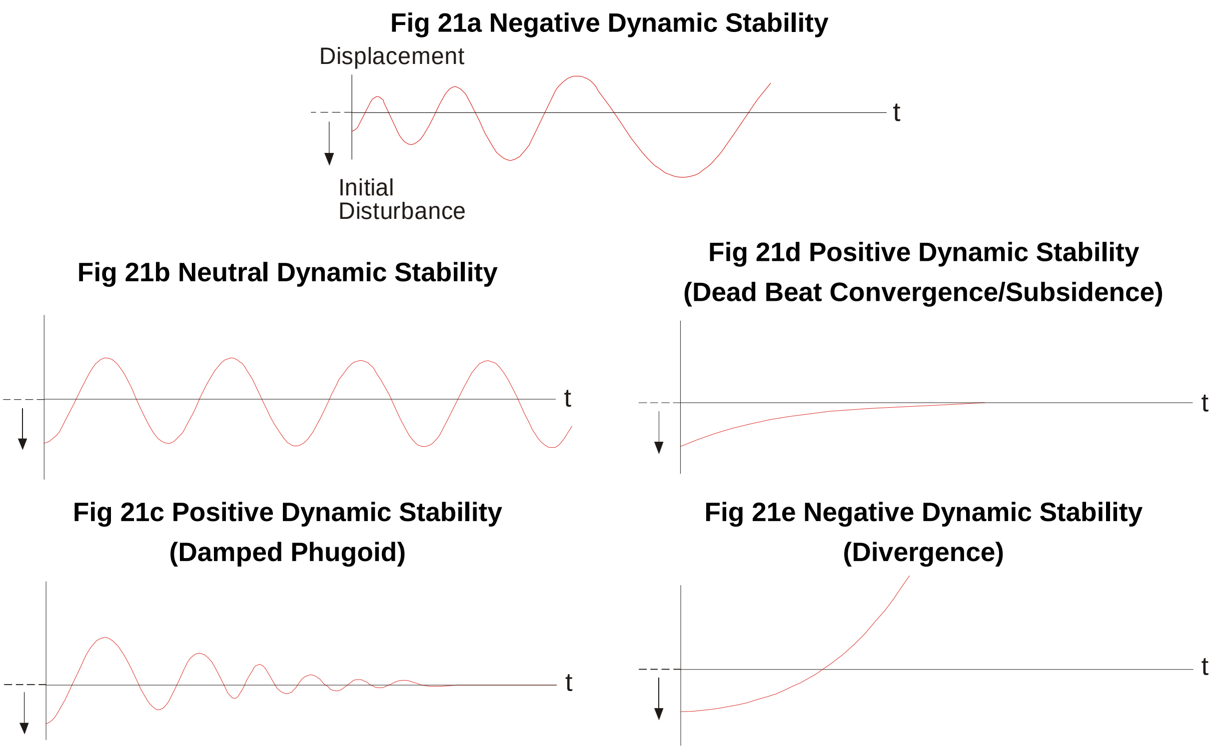
d. Motion heavily damped; oscillations cease and the motion becomes 'dead-beat' positive stability.

e. Motion diverges - negative dynamic stability.

Fig 21 illustrates these various forms of dynamic stability; in each case shown, the body has positive static stability.

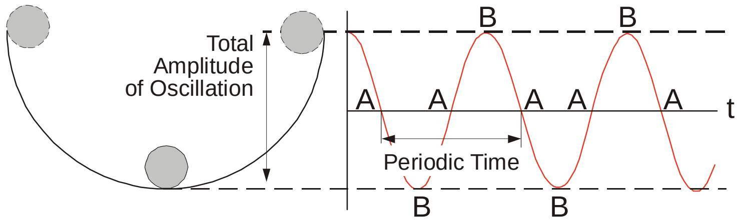
51. Dynamic stability is more readily understood by use of the analogy of the 'bowl and ball' described and illustrated at para 3. For example, when the disturbance is removed from the ball, it returns to the bottom of the bowl and is said to have static stability. However, the ball will oscillate about a neutral or equilibrium position and this motion is equivalent to dynamic stability in an aircraft.

1-17 Fig 21 Forms of Motion



52. If the oscillations are constant in amplitude and time then a graph of the motion would be as shown in Fig 22. The amplitude shows the extent of the motion, and the periodic time is the time taken for one complete oscillation. This type of motion is known as simple harmonic motion.

1-17 Fig 22 Simple Harmonic Motion



53. **Periodic Time**. The time taken for one complete oscillation will depend upon the degree of static stability, ie the stronger the static stability, the shorter the periodic time.

54. **Damping**. In this simple analogy it is assumed that there is no damping in the system; the oscillations will continue indefinitely and at a constant amplitude. In practice there will always be some damping, the viscosity of the fluid (air) is a damping factor which is proportional to the speed of the mass; damping can be expressed as the time required (or number of cycles) for the amplitude to decay to one half of its initial value (see Fig 21 - Damped Phugoid). An increase in the damping of the system (e.g. a more viscous fluid) will cause the oscillations to die away more rapidly and, eventually, a value of damping will be reached for which no oscillations will occur. In this case, after the disturbance has been removed, the mass returns slowly towards the equilibrium state but does not overshoot it, i.e. the motion is 'dead-beat' (see Fig 21d - Positive Dynamic Stability).

Dynamic Stability of Aircraft

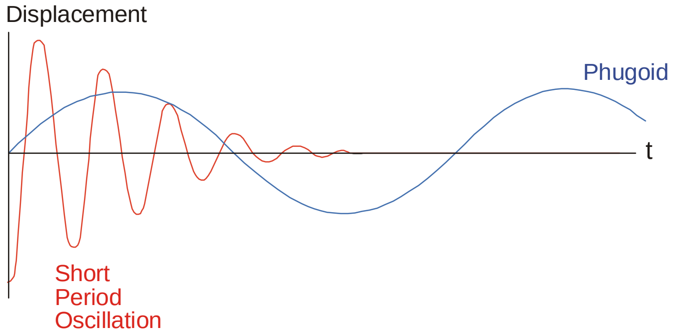
55. The dynamic stability of an aircraft depends on the particular design of the aircraft and the speed and height at which it is flying. It is usually assumed that for 'conventional' aircraft the coupling between the longitudinal (pitching) and lateral (including directional) motions of an aircraft can be neglected. This enables the longitudinal and lateral dynamic stability to be considered separately.

56. **Design Specification**. Oscillatory motions which have a long periodic time are not usually important; even if the motion is not naturally well damped, the pilot can control the aircraft fairly easily. To ensure satisfactory handling characteristics, however, it is essential that all oscillatory motions with a periodic time of the same order as the pilot's response time are heavily damped. This is because the pilot may get out of phase with the motion and pilot-induced oscillations (PIOs) may develop. The minimum damping required is that oscillation should decay to one half of their original amplitude in one complete cycle of the motion. Some modern aircraft however, do not satisfy this requirement and in many cases it has been necessary to incorporate autostabilization systems to improve the basic stability of the aircraft, eg pitch dampers or yaw dampers.

Longitudinal Dynamic Stability

57. When an aircraft is disturbed in pitch from trimmed level flight it usually oscillates about the original state with variations in the values of speed, height and indicated load factor. If the aircraft has positive dynamic stability, these oscillations will gradually die away and the aircraft returns to its initial trimmed flight condition. The oscillatory motion of the aircraft in pitch can be shown to consist of two separate oscillations of widely differing characteristics; the phugoid and the short-period oscillation, (Fig 23).

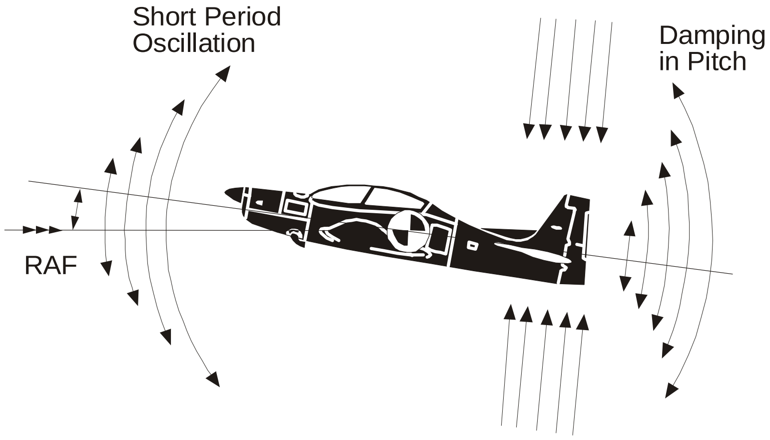
1-17 Fig 23 Basic Components of Longitudinal Dynamic Stability



58. **Phugoid**. This is usually a long period, poorly damped motion involving large variations in the speed and height of an aircraft but with negligible changes in load factor (n). It can be regarded as a constant energy motion in which potential energy and kinetic energy are continuously interchanged. The phugoid oscillation is usually damped and the degree of damping depends on the drag characteristics of the aircraft. The modern trend towards low drag design has resulted in the phugoid oscillation becoming more of a problem.

59. **Short-Period Oscillation**. This oscillatory motion is usually heavily damped and involves large changes of load factor with only small changes in speed and height. It can be regarded simply as a pitching oscillation with one degree of freedom. In para 53 it was stated that the time taken for one complete oscillation will depend upon the static stability, in this case it is the periodic time of the short-period oscillation (See Fig 24).

1-17 Fig 24 Short-Period Oscillation



60. **Stability Factors**. The longitudinal dynamic stability of an aircraft, i.e. the manner in which it returns to a condition of equilibrium, will depend upon:

a. Static longitudinal stability.

b. Aerodynamic pitch damping.

c. Moments of inertia in pitch.

d. Angle of pitch.

e. Rate of pitch.

Lateral Dynamic Stability

61. When an aircraft in trimmed level flight is disturbed laterally, the resulting motion can be shown to consist of the following components:

a. **Rolling Motion**. Initially the roll will only change the angle of bank and will be rapidly damped as explained in para 36.

b. **Spiral Motion**. A combination of bank and yaw will result in a gradually tightening spiral motion if the aircraft is unstable in this mode. The spiral motion is not usually very important because, even if it is divergent, the rate of divergence is fairly slow and the pilot can control the motion.

c. **Dutch Roll**. This is an oscillation involving roll, yaw and sideslip. The periodic time is usually fairly short and the motion may be weakly damped or even undamped. Because of these characteristics of the Dutch roll oscillation, lateral dynamic stability has always been more of a problem than longitudinal dynamic stability.

Spiral Stability

62. The lateral stability of an aircraft depends on the forces that tend to right the aircraft when a wing drops. However, at the same time the keel surface (including the fin) tends to yaw the aircraft into the airflow, in the direction of the lower wing. Once the yaw is started, the higher wing, being on the outside of the turn and travelling slightly faster than the lower, produces more lift. A rolling moment is then set up which opposes, and may be greater than, the correcting moment of the dihedral, since the roll due to yaw will tend to increase the angle of bank.

63. If the total rolling moment is strong enough to overcome the restoring force produced by dihedral and the damping in yaw effect, the angle of bank will increase and the aircraft will enter a diving turn of steadily increasing steepness. This is known as spiral instability. A reduction in fin area, reducing directional stability and the tendency to yaw into the sideslip results in a smaller gain in lift from the raised wing and therefore in greater spiral stability.

64. Many high performance aircraft when yawed, either by prolonged application of rudder or by asymmetric power, will develop a rapid rolling motion in the direction of the yaw which if uncorrected may cause it to enter a steep spiral dive due to the interaction of its directional and lateral stability.

Dutch Roll

65. Oscillatory instability is more serious than spiral instability and is commonly found to a varying degree in combinations of high wing loading, sweepback (particularly at low IAS) and high altitude. Oscillatory instability is characterized by a combined rolling and yawing movement or 'wallowing' motion. When an aircraft is disturbed laterally the subsequent motion may be either of the two extremes. The aerodynamic causes of oscillatory instability are complicated and a simplified explanation of one form of Dutch roll is given in the following paragraph.

66. Consider a swept-wing aircraft seen in planform. If the aircraft is yawed, say to starboard, the port wing generates more lift due to the larger expanse of wing presented to the airflow and the aircraft accordingly rolls in the direction of yaw. However, in this case the advancing port wing also has more drag because of the larger area exposed to the airflow. The higher drag on the port wing causes a yaw to port which results in the starboard wing obtaining more lift and reversing the direction of the roll.

The final result is an undulating motion in the directional and lateral planes which is known as Dutch roll. Since the motion is caused by an excessive restoring force one method of tempering its effect is to reduce the lateral stability by setting the wings at a slight anhedral angle.

67. The lateral dynamic stability of an aircraft is largely decided by the relative effects of:

a. Rolling moment due to sideslip (dihedral effect).

b. Yawing moment due to sideslip (weathercock stability).

Too much weathercock stability will lead to spiral instability whereas too much dihedral effect will lead to Dutch roll instability.

SUMMARY

Static and Dynamic Stability of Aircraft

68. Stability is concerned with the motion of a body after an external force has been removed. Static stability describes its immediate reaction while dynamic stability describes the subsequent reaction.

69. Stability may be of the following types:

a. Positive - the body returns to the position it held prior to the disturbance.

b. Neutral - the body takes up a new position of constant relationship to the original.

c. Negative - the body continues to diverge from the original position.

70. The factors affecting static directional stability are:

a. Design of the vertical stabilizer.

b. The moment arm.

71. The factors affecting static longitudinal stability are:

a. Design of the tailplane.

(1) Tail area.

(2) Tail volume.

(3) Planform.

(4) Wing downwash.

(5) Distance from CP_{tail} to CG.

b. Position of CG.

(1) Aft movement of the CG decreases the positive stability.

(2) Forward movement of the CG increases the positive stability.

72. Manoeuvre stability is greater than the static stability in level flight and a greater elevator deflection is necessary to hold the aircraft in a steady pull-out.

73. The factors affecting static lateral stability are:

a. Wing contributions due to:

(1) Dihedral.

(2) Sweepback.

b. Wing/fuselage interference.

c. Fuselage and fin contribution.

d. Undercarriage, flap and power effects.

74. Some of the factors affecting the long-term response of the aircraft are:

a. Linear velocity and mass.

b. The static stabilities in roll, pitch and yaw.

c. Angular velocities about the three axes. }
} Angular Momentum
d. Moments of inertia about the three axes. }

e. Aerodynamic damping moments due to roll, pitch and yaw.

75. The longitudinal dynamic stability of an aircraft depends upon:

a. Static longitudinal stability.

b. Aerodynamic pitch damping.

c. Moments of inertia in pitch.

d. Angle of pitch.

e. Rate of pitch.

76. The lateral dynamic stability of an aircraft is largely decided by the relative effect of:

a. Dihedral effect.

b. Weathercock stability.

CHAPTER 18 - DESIGN AND CONSTRUCTION

Introduction

1. The preceding contents of this manual explain the principles of flight without regard to the strength or method of construction of the aircraft. This chapter is intended to be a brief introduction to the study of airframes from the designer's point of view, including some of the general problems confronting him, and the procedure required to obtain a new type of aircraft for the Service.

2. To avoid misconceptions of the engineering terms used in this chapter the following list of definitions is included:

a. **Stress**. Stress is defined as the load per unit area of cross-section.

b. **Strain**. The deformation caused by stress. It is defined as the change of size over the original size.

c. **Elastic Limit**. When stress exceeds the elastic limit of a material, the material takes up a permanent 'set', and on release of the load it will not return completely to its original shape.

d. **Stiffness or Rigidity**. The ratio of stress over strain - Young's Modulus.

e. **Design Limit Load**. The maximum load that the designer would expect the airframe or component to experience in service.

THE DESIGN PROCESS

Military Aircraft Procurement

3. In the UK aircraft projects developed independently by aircraft manufacturers are known as private venture projects. Generally, military aircraft are developed in conjunction with MOD. Major aircraft projects are complicated by the necessity of obtaining agreement between the aircraft manufacturers, armed forces and governments of two or more countries. Simpler aircraft projects would normally follow the path shown below:

a. **The Idea**. The Air Staff identify a possible need for a new aircraft in the future and initiate an investigation.

b. **Preliminary Study**. A study is performed for the Air Staff, by, for instance, the Operational Requirements Branch, to identify more closely the best way of meeting the need.

c. **Staff Target (Air)**. Following on from the Preliminary Study the Air Staff publish a Staff Target (Air) (STA) which is a document that briefly states the performance requirements of the proposed aircraft. This is issued to aircraft companies who express a desire to tender for the project.

d. **Feasibility Study**. On the Government side, the management of an aircraft project is carried out by MOD Procurement Executive (MOD (PE)) and, after the issue of an STA, they may fund feasibility studies for a new project by one or more companies.

e. **Staff Requirement (Air)**. With more information available on, for instance, the cost implications of possible alternatives, a Staff Requirement (Air) (SRA) for the project aircraft is produced. The SRA is much more detailed a document than the STA and it leads to the Aircraft Specification, which forms the basis of contracts that are awarded to the aircraft manufacturer.

f. **Project Definition**. The project definition stage may include any research necessary to complete the detail design, and also includes production of the Aircraft Specification.

g. **Full Development**. The development stage includes the detail design and the setting-up of manufacturing facilities and may include the manufacture of prototype aircraft. This stage leads on to testing and trials.

h. **Acceptance and Approval**. Aircraft will not be accepted into service until they satisfy flight trials testing by MOD at DTEO Boscombe Down.

i. **Production**.

The Design

4. Def Stan 00-970 contains the design and airworthiness requirements for UK military aircraft. Contractual references to Def Stan 00-970 control and guide the translation of the Aircraft Specification into a finalized design (see also The Military Aviation Authority (MAA) 4000/5000 series of documents). The contents cover:

a. A series of design cases which define the extreme conditions that the aircraft is expected to encounter during its service life. For example, gust, manoeuvre and ground loads may be specified, together with the margins required to cater for structural variables and occasional excesses of normal operating limits.

b. The testing to be carried out to prove both static and fatigue strength, and the methods to be used to enable reliable estimates of fatigue lives to be made.

c. Provisions to guide the selection of materials and protective finishes which enhance long term durability.

5. Not all aircraft in service with the RAF were designed with reference to UK military requirements. Aircraft originating from other countries or from multi-national projects may be designed to other national standards such as the American range of military specifications (Mil Specs). Civil designs meet British-Civil Airworthiness Requirements (BCARs), Joint Airworthiness Requirements (JARS) or foreign equivalents.

OPERATING CONDITIONS FOR AIRFRAMES

Static Strength and Stiffness

6. Static strength requirements determine the design of a large proportion of aircraft structure. They are specified by applying specific safety factors to the design limit loads (DLLs) which result from each design case:

a. **Proof Load**. Proof load is normally 1.125 × DLL. When proof load is applied, the aircraft structure must not suffer any permanent deformation and flying controls and systems must function normally.

b. **Design Ultimate Load**. The design ultimate load (DUL) is 1.5 × DLL. The structure must withstand DUL without collapse.

7. Static strength is proved by loading a representative airframe to, first, proof load and then to DUL in the critical design cases. Such static strength testing is carried out before the type is released to service. Structural failure beyond DUL (as is usual) implies that the structural reserve factor is greater than 1.0.

8. Aeroelastic effects (in particular flutter) often set limits to the maximum speed of fixed wing aircraft in each configuration. Structural stiffness is normally checked during static and ground resonance testing. A flight flutter investigation is included in the development programme. Flutter is a violent, destructive vibration of the aerofoil surfaces, caused by interaction of their inertia loads, aerodynamic loads and structural stiffness.

9. When structures incorporate fibre-reinforced composite materials, design calculations and testing have to take account of additional environmental factors such as; temperature, moisture uptake, ultra-violet ageing and the potential effects of accidental damage.

10. **Temperature, Corrosion and Natural Hazards**. Airframes need to contend with elevated temperatures from two causes; local heating of the structure near to engines, heat exchangers, hot gas ducts etc and kinetic heating of the outside surface of the airframe at high Mach numbers. Corrosive conditions, caused by fluids such as water and general spillages, are a sizeable problem for airframe maintainers and corrosion is exacerbated by damage to paint and other protective finishes. Some materials, particularly certain aluminium alloys and steels, are susceptible to stress corrosion cracking where cracks grow in a stressed component from corrosion on the surface. New designs should, wherever possible, exclude the use of materials known to be highly susceptible to stress corrosion cracking or exfoliation corrosion and eliminate undesirable features such as water traps, dissimilar metal contact or ineffective protective coatings.

11. **Environmental Conditions**. Operating conditions also affect material choice and component design. In addition to the natural hazards of lightning and bird strikes, design attention must be paid to the effects of flight in saline environments and flight through erosive, sand laden atmospheres. Man-made hostile environments, such as atmospheric industrial pollution, also influence the range of materials which can be used in airframe construction.

Material Requirements

12. The ideal properties of an airframe material would include: low density, high strength, high stiffness, good corrosion resistance, high impact resistance, good fatigue performance, high operating temperature, ease of fabrication and low cost. This list is not exhaustive, and needless to say, there is no perfect material. Each material choice is a compromise which attempts to find the best balance of the most important requirements of the component. There is always a mix of materials on any particular aircraft type.

13. It is generally found that the stronger and stiffer an engineering metal is the more dense it is; therefore the strength/weight and stiffness/weight ratios for the commonly-used aircraft metals are similar. The increasing use of composite materials reflects their inherent advantages over metals. First, within their physical capabilities, they can be readily engineered to meet any specific requirement. Secondly, composite components can be designed and manufactured to complex shapes. For example, the weight and size of a gearbox casing can be minimized by the use of a skeletal composite structure in which each stress is reacted to by a specific feature of the structure, thus eliminating surplus material. This is almost impossible to achieve with conventional metal manufacture, although integral machining or chemical etching techniques can be used to manufacture minimalized metal components of such simple geometric shapes as wing and skin panels.

14. For each metal there may be dozens of different specifications, each with its own characteristics. The strength, hardness and ductility can vary greatly but the stiffness varies little. The 'specific' properties of a material, ie strength/weight and stiffness/weight ratios are important because the designer is always seeking to minimize mass. High stiffness of a material is important because we normally wish structures to deflect as little as possible. Even if two materials have similar specific properties, the less dense material may be the better choice as it will have a greater volume for a given load and the extra thickness is of advantage for components in compression or shear (doubling the thickness quadruples the load at which a sheet will buckle). However, the space available may, in some circumstances, restrict the use of the less dense materials.

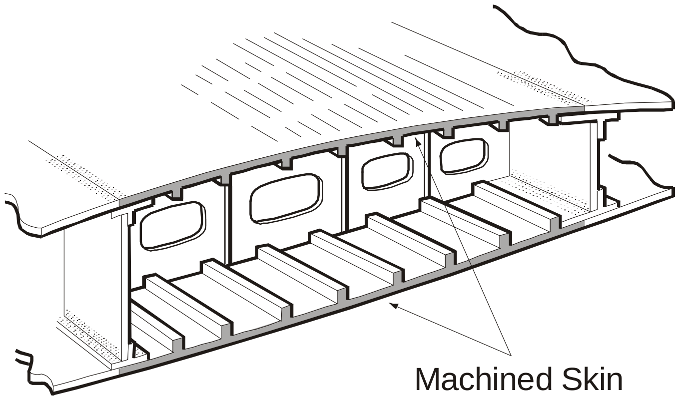
Use of Materials

15. Since the first days of aviation, the manufacture of safe and effective aircraft structures has demanded the use of the highest contemporary technology. Indeed, the majority of technological advances can be attributed to aviation. Early aircraft were constructed by the most skilled craftsmen using the best timber and other materials available. The need for more consistent and stronger materials led to developments in aluminium alloys during the 1930s, although wood and canvas were still widely used even during the Second World War. Arguably the first totally composite aircraft, the DH 98 Mosquito, owed much of its high performance and subsequent success to the ingenious use of a hard wood and balsa wood sandwich composite in its structure. Aircraft constructed since the War have relied almost entirely upon aluminium, magnesium, titanium and steel, although the structures of current aircraft, such as the EFA, make extensive use of advanced composite materials. Several all-composite aircraft are in production, and the trend towards their greater use is likely to continue.

16. The most commonly used material in the current generation of airframes is still aluminium alloy. Titanium alloy is used for structure adjacent to engines, heavily loaded fuselage frames and items like flap tracks, which are subject to wear and not easily protected against corrosion. Magnesium alloy is very rarely used except for small items like control-run brackets, and for helicopter transmission cases. Steel is sometimes used for heavily-loaded parts like wing attachments and undercarriage components, and of course for bolts and other hardware. Carbon Fibre Composites (CFC) are now common on large passenger transports for components such as control surfaces, fairings and even fins. Modern combat aircraft, such as the Typhoon and the Lightning II have a high proportion of CFC components.

17. Aluminium alloy is cheap, and is easy to produce, to machine and to fabricate. Pure aluminium has good natural corrosion resistance but is very weak. The stronger aluminium alloys have poorer corrosion resistance which is sometimes improved by cladding the alloy with a thin layer of pure aluminium. In any case, they are normally protected with a paint scheme. Components like wing skins, spars, ribs and fuselage frames are often machined by integral milling, where up to 90% of the material is removed (see Fig 1).

1-18 Fig 1 Machined Skin Wing Construction



Thinner skins may be chemically etched to remove the metal for weight saving. Many small parts are made by cutting from a sheet and bending to shape. Final assembly can be by adhesive bonding but is usually by riveting and bolting. The very highest strength aluminium alloys (with about 7% zinc as the main alloying ingredient) are often used for wing upper skins but they have a relatively poor fatigue performance. Slightly less strong, but more commonly used, are the Duralumin alloys with 4% copper. These alloys have to be heat-treated to produce the required properties and this means that their service temperature is limited to 120° C; if the material is accidentally heated above this, by fire for instance, it may have to be renewed. Under development is a series of aluminium alloys which contain about 3% lithium. These have about 10% increase in both specific strength and stiffness over established alloys, and may well compete with CFC.

18. Magnesium alloy is not as easy to work as aluminium, although it machines and casts quite readily. Its two major disadvantages are its reactivity (which means that it burns easily) and its extreme susceptibility to corrosion. Magnesium must be protected extremely well against corrosion and should be used only where it can be inspected easily.

19. Titanium is very expensive indeed owing to its natural scarcity and the difficulty of extraction from its ore. Titanium alloy is not easy to form or machine but it can be welded and cast with care. The advantages of titanium alloy are that it has a high specific strength, maintains its properties well up to 400° C, and has very good corrosion resistance (it needs no protective coating). It is used for fire-walls, helicopter rotor hubs and wing carry-through sections (e.g. on aircraft like Tornado, B-1 and F-14) and many other applications. Welding is generally done by the electron beam method.

20. **Superplastic Forming/Diffusion Bonding (SPF/DB)**. SPF/DB is a relatively new process which is being used at the moment to make production items, like a Tornado heat exchanger, out of titanium. It is also being developed for future use on aluminium. At about 900 °C, titanium will deform steadily under constant load to a strain of several hundred per cent; this is superplasticity. At the same temperature, if two clean titanium surfaces are pressed together they will weld by atomic diffusion across the joint line; this is diffusion bonding. SPF/DB combines the two processes in one operation. SPF/DB can save cost, weight and production time compared to alternative methods and can give a better finished product with less fasteners and excellent welds.

21. **Steel**. There are hundreds of different specifications of steel with widely varying properties; wide use is made of many different alloying constituents and various heat treatments. It is comparatively easy to machine, form and weld, except for the very high strength and stainless versions. Low alloy steels are used for fasteners and highly-loaded brackets. Steel has to be protected against corrosion by painting or

plating with cadmium or chromium. Stainless steels with upwards of 12% chromium are used infrequently in airframes.

22. **Non-metallic Composites**. The term 'composite' is usually taken to mean a matrix of a thermo-setting plastic material (normally an epoxy resin) reinforced with fibres of a much stronger material such as glass, Kevlar (a synthetic filament), boron or carbon. Glass provides a useful cheap material for the manufacture of tertiary components, whereas Kevlar, boron and carbon fibre composites (CFC) are used in the manufacture of primary structures such as control surfaces or helicopter rotor blades.

23. **Advantages of CFC**. The main advantages of CFC are its increased strength/weight and stiffness/weight compared to the normal airframe materials. This results in lighter structures and hence performance benefits; the usual weight saving is about 20%. CFC is very resistant to corrosion and does not necessarily have to be protected. Complex shapes can be made more easily than in metal. Large structures can be manufactured in one piece, thus saving machining and assembly time. CFC has a very good fatigue performance, especially at the stress levels used in the present generation of CFC structures.

24. **Disadvantages of Composite Materials**. Current composite materials have inherent drawbacks which limit their use, although continuing research and development will solve these problems in due course. Almost all such disadvantages result from limitations of the matrix materials used. The most relevant of these drawbacks are:

a. Most composite materials are relatively elastic, and unless adequately reinforced they tend to fail prematurely at such features as fastener holes.

b. Their use is limited to ambient temperatures below about 120° C.

c. Unless adequately protected, they tend to absorb atmospheric moisture.

d. They have poor impact resistance, the main effect of which is poor resistance to erosion caused by hail or sand. Leading edges of composite flying surfaces are frequently fitted with a titanium or stainless steel sheath to overcome this problem.

e. They are more difficult to repair than comparable metal structures. Curing the resins during manufacture and repair must be carried out within a narrow range of temperature and humidity conditions, and this requires the use of environmentally controlled facilities.

WING STRUCTURES

Design Considerations

25. A design to meet both subsonic and supersonic requirements must be less efficient in both modes of flight. The aircraft will be heavier and more expensive than one designed for a single role but nevertheless may be more useful overall. Even though a higher proportion of flying time is spent subsonic it will be supersonic considerations which control the design, since the price of inefficiency in that role, in terms of thrust and fuel required, is greater.

26. One obvious solution is to use variable sweep, if this is justified by combat endurance and/or radius of action requirements. Otherwise there will be penalties in weight and complexity. Alternatively, thin wings of moderate sweep may be provided with combat high lift devices, such as

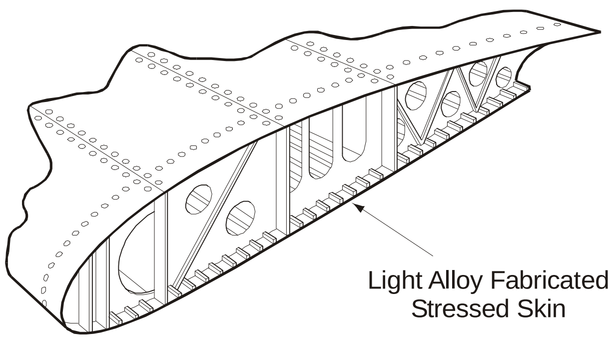
flaps and slats or leading edge droop, which reduce the drag at high lift. Additionally, leading edge strakes give more lift and reduced drag at high angles of attack (α) and at the same time delay the drag rise at low α. Practical limits to thinness are set by considerations of wing structure weight and fuel and undercarriage stowage volume.

Wing Loads

27. The loading on aircraft wings originates in airloads arising from the air pressure distribution and inertia loads due to the mass of the wing structure, fuel, stores and fuselage. This loading results in shear forces, bending moments and torques on the main structural box of the wing.

28. To a first approximation, the shear forces are reacted by the shear webs of the spars; the bending moments by the skin, stringers and spar booms, and the torque by the skin and shear webs. Composite wings, tailplanes, fins and fuselages have a similar basic structure to the metal versions described in the following paragraphs. Depending upon the required performance of the structure, a variety of composites or metals and composite hybrids are used to construct integrated skins, stringers, frames, ribs, spars and webs which are needed for the structures described.

1-18 Fig 2 Stressed-Skin Wing Construction



Structural Design

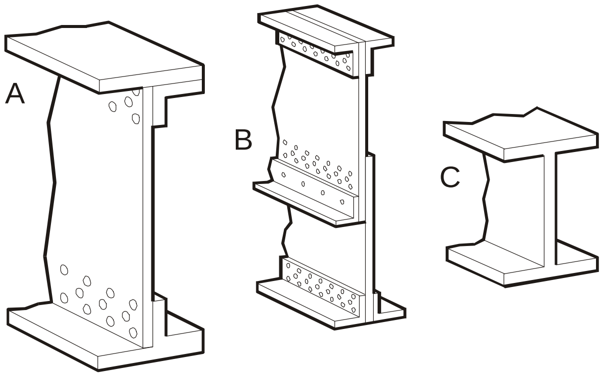
29. In more detail, the jobs of the various parts of the wing main structure are as follows:

a. **Skin**. The skin is subject to the pressure differences due to the air pressure and the pressure caused by the inertia forces of any fuel in the wing tanks. It reacts the bending moment by generating direct stresses (either tension or compression) in the spanwise direction. It reacts torsion by generating shear stresses.

b. **Stringers**. The stringers, provided that they are continuous in the spanwise direction, react bending moment in the same way as the skin. They also stiffen the skin in compression and shear by increasing the stress at which the skin buckles.

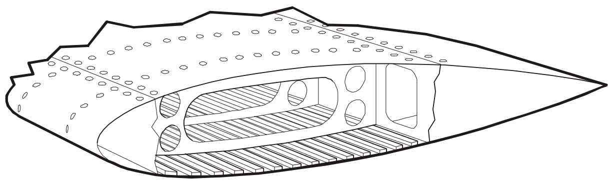
c. **Ribs**. The ribs maintain the aerodynamic shape of the wing, support the spars, stringers and skin against buckling, and pass concentrated loads (from stores, engines, undercarriages and control surfaces) into the skin and spars.

1-18 Fig 3 Typical Spar Sections



d. **Spars**. The webs of the spars react the applied shear forces by generating shear stresses. They also combine with the skin to form torsion boxes to react to torque by generating shear stresses. The spar booms (also known as flanges or caps, particularly if they are relatively small) react bending moment in a similar way to the skin. A conventional structure would consist of front and rear spars, the metal skin attached to the spar flanges forming a torsion box (Fig 4). There is a form of construction that uses a series of small spars to replace the two main spars. This results in a very rigid structure with good fail-safe characteristics.

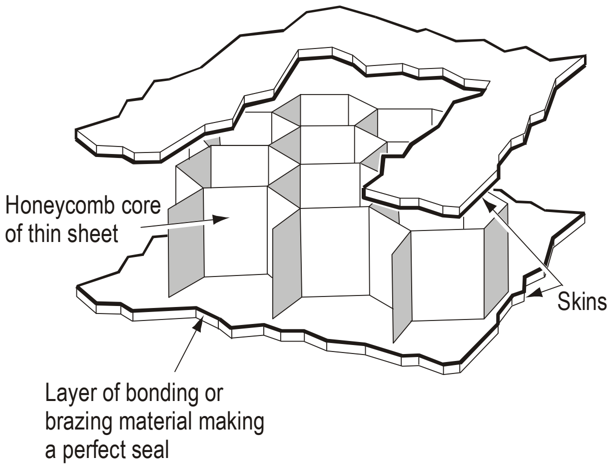
1-18 Fig 4 Torsion Box



e. **Intercostals**. Intercostals are short stringers which run between ribs. They do not go through the ribs and therefore do not transmit any end load. They act as local skin formers and stiffeners. They are not present on all aircraft.

30. **Honeycomb Construction**. Honeycomb construction (see Fig 5) is an efficient way of stabilizing (ie stiffening) relatively thin skins which are in close proximity. It is used on wing and tail trailing edges and access panels on many aircraft types. The honeycomb material is usually manufactured from strips of aluminium foil or resin impregnated Nomex cloth. The strips are bonded together alternately into a large block which can be machined to any desired shape. After machining, the block is stretched, and the strips deform into a matrix of hexagonal cells - similar to a honeycomb. These may then be bonded between two sheets of aluminium, or a composite material such as Kevlar, to form a light stiff sandwich. Sandwich panels are made using a variety of skin materials including stainless steel, titanium and CFC, and often with expanded foams instead of honeycomb. Such high strength materials are used extensively in current aircraft primary structures.

1-18 Fig 5 Honeycomb Sandwich Panel



TAILPLANE, CANARD AND FIN

Design Considerations

31. The purpose of the tailplane is to provide longitudinal stability and control. Considerable improvements in agility and reduction of overall size and weight are available in making a neutrally stable aircraft with computer-driven active controls to maintain the desired flight attitude. Large and rapid control movements are necessary with such aircraft, as they are with many current high performance aircraft when flying at the extremes of their flight envelopes. For this reason, most such aircraft have "all flying" tailplanes giving much greater control forces than are feasible with a separate tailplane and elevator.

32. With tailless aircraft, control power is provided by trailing edge elevators or elevons. Wing trailing edge flaps, however, cannot therefore be used.

33. Compared with a conventional aircraft, the tailless aircraft suffers from lift losses due to trimming and restricted supersonic manoeuvring resulting from the relative ineffectiveness of trailing edge controls at supersonic speeds. The trimmed lift of a tailless aeroplane can be improved by providing a canard, or foreplane.

34. The fin is sized to give the required level of directional stability and must be adequate to correct sideslip rapidly. This can be difficult to achieve at high angles of attack. An upper limit is set to the height of a tall fin by stiffness and strength in bending. Alternatives to a tall fin are the use of twin fins, the addition of aft strakes, or by forebody shaping to reduce the effect of body vortices.

35. The detailed structural components of tails and fins are similar to those of wings.

FUSELAGE DESIGN

Introduction

36. Unlike the wings, the shape of the fuselage is governed primarily by the load it has to carry. Aerodynamic features are secondary to the necessity to accommodate the payload.

37. Apart from payload there is the requirement to provide for pilot and crew, so visibility has also to be considered. Engine and jet pipe installation, particularly on combat aircraft, have an important effect on fuselage shaping.

38. Another aspect is the aircraft attitude in landing. For example if the landing attitude has a high angle of attack then either a long main undercarriage is required, or the tail must be swept up for ground clearance, or a combination of both.

39. The shape of the fuselage is very much a function of what it is asked to do, and in this respect there are some fundamental differences between combat and transport aircraft.

Combat Aircraft

40. **Introduction**. The main features influencing fuselage design for a combat aeroplane will be powerplant installation, fuel and undercarriage stowage and weapon carriage. Stealth technology (i.e. the avoidance of a recognizable infra-red and radar signature) will play an increasing role in future designs.

41. **Area Ruling**. In following the area rule to minimize drag, the design will aim at a low frontal area with a smooth build-up of cross-section area over the forebody, canopy and wings, followed by a gradual decrease over the afterbody and tail surfaces, the whole being spread over a length defined by fuel volume requirements.

42. **Powerplant Installation**. The choice of the number of engines has an important effect on fuselage shaping and directly influences the fuselage cross-section shape. The decision is largely a policy one, influenced by airworthiness, cost and maintainability factors.

43. **Intake Design**. The number of engines will also dictate the disposition of intake ducts and fuel tanks, again influencing fuselage shape. Intake design is governed by speed considerations and engine size, since the intake must deliver air to the engine with minimum pressure loss, evenly distributed over the engine face and free from turbulence. This is of particular importance at high supersonic speeds where the air must be slowed to subsonic speed.

44. **Weapon Carriage**. Internal weapon stowage has the advantage that area ruling is unchanged, whatever the aircraft role. Additionally, the external surfaces are aerodynamically clean. However, during the life of an aircraft, weapon requirements are likely to be diverse and to alter several times.

45. **Stowage Points**. External stowage is more adaptable to different roles and customer requirements, and may be under the fuselage or under or on top of the wings. The pros and cons of alternative methods of external stowage are shown in tabular form in Table 1.

Table 1 Features of External Store Positions

Position	Under fuselage	Under or overwing
Performance	Tandem carriage possible: reduces drag of rear stores.	Side-by-side carriage gives unfavourable interference drag, big stores sometimes affect induced drag adversely.
Stability and Control	Forward stores destabilizing directionally; stores remote from C of G can cause large pitch disturbances.	Destabilizing in pitch when in front of a low tailplane. Asymmetric carriage causes rolling moment - can limit g on pull out.
Loads and Flutter	Generally free from flutter and flexibility problems.	Generally causes flutter problem: requires big tanks to be compartmentalized to control C of G. Response to jettison of heavy stores can cause severe loads.
Release and Jettison	Usually good.	Release of light stores difficult; wing flow causes pitch disturbances: requires accurate store C of G control - can conflict with flutter considerations.
Accessibility	Generally satisfactory if pylon-mounted.	Usually good.

46. **Front Fuselage Structure**. For most types of interceptor/attack/strike aircraft the fuselage is a conventional stressed skin, stringer and longeron arrangement with frames and bulkheads. The front fuselage, forward of the wing/fuselage joint, contains the small pressurized area of the crew compartment. This is confined by a small front bulkhead, cockpit floor and a larger rear bulkhead. The forward pressure bulkhead carries the weapons system radar or weather radar, whilst the rear pressure bulkhead, which may slope, normally carries ejection seat attachment points, and may also accept nose undercarriage mounting loads. The fuselage upper longerons, taking end loads due to bending, may also carry the canopy rails and/or fixing points. This is the conventional front fuselage layout.

47. **Wing/Fuselage Transfer**. The manner in which the wing loads are transferred to the fuselage will depend to a large extent upon engine and wing position.

a. **Wing Configuration**. A low wing gives a shorter undercarriage and good crash protection whereas a high wing can give a better aerodynamic performance. A mid-wing position is attractive aerodynamically for supersonic performance.

b. **Light Loading**. If the wing structure is lightly loaded it is possible to transmit the loads to the fuselage through heavy root ribs and strong fuselage frames.

c. **Heavy Loading**. With a heavier loaded wing structure it is usual to make the whole centre section box continuous across the fuselage, thus avoiding transmitting primary wing-bending loads to the fuselage.

48. **Centre and Rear Fuselage**. The volume aft of the rear pressure bulkhead is available for the avionics bay, fuel stowage, engines and jetpipes. The strength and stiffness necessary for taking tailplane and fin bending and torsional loads will depend upon the particular design layout, for example one fin or two, or whether the aircraft is of tailless design. If deck landings are envisaged, then the rear structure must be capable of accepting loads from the arrester hook.

49. **Variable Wing Geometry**. The design of a variable wing geometry aircraft invariably leads to problems of cost and weight penalties which must be traded off against the better overall performance gained. The wing carry-through box and the wing attachment pivots must be adequately stiff, strong and crack resistant. Apart from the complexity of the wing sweep actuators there is the problem of providing the mechanism necessary to maintain parallel pylons. For good aerodynamic performance the seals between wing and glove must be effective over the full sweep range and allow for any structural flexibility. The pivot bearings themselves are in a highly loaded region, yet they must maintain low friction properties under dynamic loads in varying temperature conditions.

Transport Aircraft

50. **Pressurized Fuselage**. For a large pressurized fuselage the problem is complicated by the presence of cut-outs from windows, doors and undercarriage stowage and by the interaction effects at the wing/body junction. Adequate provision must be made against catastrophic failure by fatigue and corrosion and the task is further complicated by safeguarding against accidental damage. In practice the present trend is to design a fuselage in which structural safety is provided by the fail-safe concept. In detail, all fuselages are greatly different in design, but pressure cabins have a number of principles in common:

a. Low stress levels due to pressurization loads.

b. Cut-outs, such as doors and windows, are reinforced so that the fatigue life is at least equal to the basic fuselage.

c. Materials are chosen for good fatigue properties and the fuselage is provided with crack-arresting features.

51. **Fuselage Shape and Cross-section**. The use of a parallel fuselage eases seating and stowing arrangements. It also permits stretching of the fuselage, as development of the engine and airframe proceeds, to extend the life of the basic design. Usually the parallel part of the fuselage is continued as far aft as is consistent with tail clearance. There are four main factors which determine fuselage size:

a. Cabin length and width (i.e. usable floor area).

b. Freight volume.

c. Passenger and freight distribution to maintain the correct CG position (a serious problem with random seat loading).

52. **Wing Position**. A high-wing aircraft generally has a lower floor to ground height than a low-wing aircraft, which eases loading and may result in a reduction of weight. The overall height will be less, which eases ground maintenance. However, a low-wing aircraft provides a better cushion in case of a crash landing, whereas for a high-wing aircraft it is necessary to strengthen the main fuselage frames to prevent collapse of the wing. The problems of main undercarriage attachment and stowage are less for a low-wing aircraft.

Fuselage Structural Components

53. The function of the main structural components of the fuselage are:

a. **Skin**. The skin is the primary structural part of the fuselage and thicknesses are usually determined by hoop tension and shear forces. The skin is usually wrapped on the fuselage in sheets. This is convenient for a long parallel fuselage but becomes difficult in regions of double curvature where the sections of skin have to be preformed to shape.

b. **Stringers**. The function of the stringers is to stabilize the skin. Extruded stringers have good properties (due to extra material at the corners) and are more suitable for bonding since the round corners of rolled stringers are liable to cause peeling. Closed section stringers are not subject to instability and are even more suitable for bonding. The current trend for large aircraft is towards integral skin/stringer construction, particularly in areas of stress concentration such as the window band, in which case the necessary additional reinforcement is also machined into the panel.

c. **Longerons**. Longerons are heavy longitudinal members taking concentrated loads in direct tension and compression, usually replacing several stringers at cut-outs etc. They accept longitudinal loads due to fuselage bending.

d. **Frames and Bulkheads**. The function of the frames is to maintain the shape of the fuselage cross-section and to improve the stability of the stringers, particularly open section stringers, which are cleated to the frames. The frames also distribute local loading into the fuselage section, assist in carrying pressurization loads and act as crack stopping devices. Heavier frames and partial bulkheads will be used to transmit large concentrated loads into the fuselage section, eg wing, tailplane and fin connections, and at the undercarriage attachments. Pressure bulkheads can be either of the curved membrane type and react loads in tension, or flat and react loads in bending. The former are lighter but the difficulty of the circumferential join may preclude their use.

e. **Floors**. Floors give rise to both a local and an overall problem. Local loads are sufficiently high to require a robust construction, especially on freighter aircraft. These floors often tend to pick up loads in the longitudinal direction, and thus cause an overall problem when the fuselage is bending. As they are displaced from the neutral axis, the tendency is for the stress in the upper skin to increase. This is undesirable and hence the floor is often designed in short portions and supported laterally off the frames. Composite honeycombs are frequently used for the manufacture of floor panels. Their light weight and high stiffness are particularly suitable in this role, and flush repairs of the scuff and tear damage which is often inflicted on floors can be made more easily than with metals.

CHAPTER 19 - STRUCTURAL INTEGRITY AND FATIGUE

Introduction - Structural Integrity

1. Structural integrity is concerned with the structural airworthiness of aircraft. The objectives of structural integrity policy are:

a. **Flight Safety**. The principle aim is to prevent structural failure and to maintain the airworthiness of aircraft. Statistically, the probability of losing an aircraft through structural failure must not exceed 0.001.

b. **Operational Effectiveness**. Aircraft must remain operationally effective with minimal restrictions on the Release to Service. Front line availability should not be compromised by the need for structural maintenance.

c. **Life Cycle Costs**. Fatigue life consumption and structural maintenance workloads must be kept low to minimize life cycle costs, particularly as many aircraft are retained in-service beyond originally planned dates.

2. In high performance aircraft, fatigue is a primary threat to structural integrity. All aircraft structures incur insidious fatigue damage in normal service due to the fluctuations and reversals of loads which arise from flight manoeuvres, turbulence, pressurization cycles, landing, and taxiing. This chapter deals primarily with the ways in which the fatigue problem is dealt with in RAF aircraft, but other threats to structural integrity which should be noted include:

a. **Overstressing**. When loads in excess of design limitations are applied to an airframe structure, damage may result beyond a level that would contribute incrementally to fatigue. Overstressing can result in irreparable structural damage, necessitating withdrawal of the airframe from service, or even in catastrophic failure during flight.

b. **Operational Hazards.** Operational hazards include bird strikes, ricochet damage, and battle damage. Aircraft are designed to withstand specified levels of operational damage, based on the statistical probability of occurrence.

c. **Corrosion**. Corrosion is a chemical reaction occurring in metals exposed to water or other contaminants which not only weakens structures by causing metal decay, but also introduces stress concentrations that reduce fatigue life.

d. **Stress Corrosion Cracking**. Stress corrosion cracking occurs when certain metal alloys are subjected to permanent tensile stress in a corrosive environment. Such tensile stress is usually residual, as a result of forging, interference fit, or malalignment during assembly. The normal atmosphere is often sufficient to destroy the cohesion between individual metal grains, and stress corrosion cracking propagates around the grain boundaries.

Metal Fatigue

3. When a load is applied to a metal, a stress is produced, measured as the load divided by the cross-sectional area. The stress at which the metal fractures is known as its ultimate stress, which defines its ability to resist a single application of a static load. However, if a metal structure is loaded and unloaded many times, at levels below the ultimate stress, the induced stresses cause accumulative damage which will eventually cause it to fail. This type of damage constitutes metal

fatigue, and it is important to note that eventual failure can occur at a stress level well below the ultimate stress.

4. In metal, it is normally tensile loads that cause fatigue. Cracks almost always start on the surface of structures where there are discontinuities of shape, such as fastener holes, screw threads, machining marks, and sharp corners. Surface flaws caused by corrosion and fretting can also help to initiate fatigue cracks.

5. Composite materials too are subject to fatigue failure, but the failure mechanism is different, and is unlikely to pose comparable problems in service. Reinforcement materials have considerably higher inherent fatigue strengths than conventional metal alloys, and fatigue is not a consideration in composite materials at stress cycles below approximately 80% of ultimate stress. When fatigue does occur in composites, its mechanism is a fracture of individual filaments of the reinforcing material. Because composite materials are not homogeneous, such fractures do not precipitate cracking but more gradual weakening of the whole material mass. Thus, whilst metal structures suffering fatigue retain their design strength until cracking reaches a critical point when failure occurs very rapidly, composite structures only gradually loose their properties. Such 'soft failures' as they are termed provide ample evidence to be observed during routine inspection well before critical strength is lost.

Design Philosophy

6. There are basically two design philosophies used to counter the threat to structural integrity from fatigue:

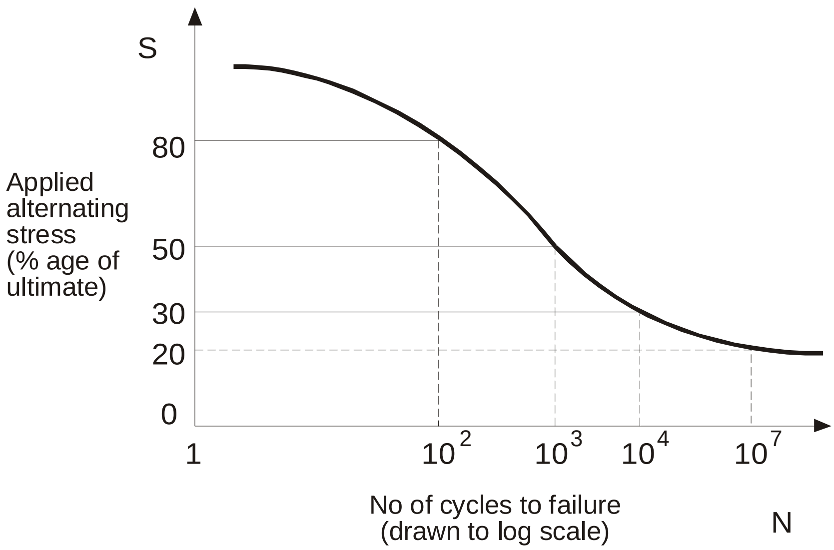
a. **Safe-life**. Safe-life design, which is used for most combat and training aircraft, aims to ensure that there will be no cracking of significant structure during the specified life of an aircraft. Crack arrest is not a feature of safe-life design, and should there be a significant risk that cracking has started, the structure is usually retired from service.

b. **Damage Tolerant**. The damage tolerant, or fail-safe approach, requires that structures maintain adequate strength in service until planned inspection reveals unacceptable defects. Fail-safe structures are characterized by crack-arrest features, redundant load paths, and relatively long critical crack lengths. Fracture mechanics analysis plays a large part in damage tolerant design, and the setting of inspection intervals. Most transport type aircraft are designed to damage tolerant principles.

Establishing Fatigue Life

7. The relationship between various alternating stress levels which can be applied to a material and its endurance to failure, can be established by testing a large number of typical structural elements. Fig 1 illustrates such a relationship in an 'S-N curve', which plots the applied percentage of ultimate stress against the number of cycles to failure. It can be seen in this example that if the applied stress was 80% of the ultimate stress, the specimen could be expected to fail after 100 applications, but if the applied stress was reduced to 20% of the ultimate stress, failure would not occur until there had been 10 million applications. The curve also shows that if the stress level in the structural element was to be increased by (say) 20% of the ultimate stress (e.g. from 30% to 50%), the life would be reduced by a factor of 10.

1-19 Fig 1 S-N Curve



8. Volume 1, Chapter 20 considers the basic applications of S-N curves to the calculation of fatigue life in structures. In order to apply fatigue formulae the design authority needs to know in considerable detail how an aircraft will be used in service. The range and frequency of 'g' forces which are likely to be applied during manoeuvres have the greatest influence on combat aircraft and trainers, but turbulence, cabin pressurization, flying control movements, landings, vibrations, and buffet are examples of other factors which must be taken into account.

9. To validate the theoretical life it is a requirement for Service aircraft that an early pre-production airframe undergoes a full-scale fatigue test (FSFT), during which the structure is subjected, in a test rig, to the load reversals expected to arise during typical operational sorties. Finally, a factor is applied to allow for S-N curve scatter, and a fatigue life is declared for that particular aircraft. The FSFT should ideally be completed early enough for any resulting modification to be embodied during production.

10. The fatigue life of an aircraft, or aircraft component, is established in terms of fatigue index (FI), which is a non-dimensional quantity related to the number of load cycles of varying intensity experienced during FSFT, and thus to the anticipated spectrum of fatigue loads in service. For safe-life aircraft, the fatigue life is normally declared as 100 FI.

11. A second FSFT will be started later in the aircraft production run, when it has been in operational service sufficiently long for the flight profiles and sortie patterns to be well established. This second test should make use of an operational loads measurement (OLM) programme, in which a small sample of the fleet is fitted with strain gauges to measure the loads in a dozen or more critical locations in the structure. These aircraft are flown in normal service, and continuous recordings of the strain gauge outputs, measuring the actual loads experienced, are available for analysis by the Design Authority. The safe-life may then be modified, and any extension would be established by declaring a life in excess of 100 FI.

Fatigue Life Monitoring

12. The object of monitoring is to measure the consumption of the fatigue life of individual aircraft in service. Each aircraft is equipped with a fatigue meter - a counting accelerometer mounted close to the centre of gravity - which senses the number of times each of several levels of acceleration along the vertical axis ('g') is exceeded in flight. The counting levels of the fatigue meter are selected for

each aircraft type after consideration of its expected operational usage pattern. Usually, five or six counters record 'g' levels above 1g (level flight), whilst two or three counters record 'g' levels below 1g. The counters record as 'g' increases or decreases about the 1g level; to avoid minor fluctuations the meter is so constructed that after a 'g' level has been counted, the same level cannot be counted again until a substantial return towards the 1g condition has been experienced.

13. The fatigue life (or number of FI) consumed during a flight is calculated by substituting evidence from the fatigue meter, and other post-flight MOD Form 725 data (eg the number of landings, AUW, sortie type, etc) into a formula which is prepared by the design authority and promulgated by MOD (PE). The formula is an approximation of Miner's Rule (see Annex A). A more sophisticated method of measuring fatigue accumulation involves a development of OLM, whereby each aircraft is fitted with strain gauges at critical locations and readings are collected and analysed by an on-board computer.

14. Because the initial calculations of fatigue life are based on the results of tests representing the projected use of aircraft in service, standard sortie profiles are established for each type. These sortie profiles are contained in the statement of operating intent (SOI) for each aircraft, which is subsequently published as topic 15S of the appropriate Air Publication. Programmes to analyse actual usage profiles are mounted at the appropriate stages of the in-service life of an aircraft, and the results are used to verify or to amend the SOI and the Aircrew Manual. The Manual contents allow the aircrew to select the correct sortie profile code (SPC) for post-flight entry in Form 725. It is essential that the SOI accurately reflects the flying that is actually being carried out in service, otherwise the assumptions made by the design authority in establishing fatigue life may be incorrect, and the structural integrity of the aircraft may be compromised. Any errors apparent in an SOI should be reported immediately through the appropriate air or engineering staffs who regularly assess and review the implications of changes in operational techniques or procedures.

Unmonitored Flying

15. Situations can arise in which information may not be available as a basis for calculating the fatigue life consumed during a flight - a fatigue meter may be unserviceable, or relevant Form 725 information may be indecipherable, clearly incorrect, or missing. The calculation of fatigue life consumption during unmonitored flying includes a safety factor which may overestimate the true figure by as much as 50%, and prolonged periods of flying with unserviceable meters are therefore extremely wasteful of fatigue life. Similar penalties may arise when corrupt or missing Form 725 data cannot be recovered by later research, eg the most severe sortie code will be applied when no record is available. If, because of inadequate flight data recording, neither fatigue meter readings nor a SPC are known, the resulting 'unmonitored/severe sortie' calculations could result in an estimated safe life consumption two or three times as high as that actually accumulated during the flight. Everyone concerned with fatigue recording should be aware of the importance of clear and accurate entries in Form 725, and of ensuring that unserviceable fatigue meters are changed promptly.

16. **Helicopters**. Helicopter flying represents an important exception to the practice of monitoring safe life consumption in service aircraft. In this case the airframe is not the main concern; rather it is the dynamic components of the rotor transmission system and the rotor assembly itself, which are susceptible. There is currently no equivalent of the fatigue meter for these components, and safe life consumption is determined solely on the basis of flying hours. Helicopters do however have an equivalent of OLM, called operational data recording (ODR), which provides useful information on their usage in service; and research is ongoing to develop the means of measuring stress loading in rotating assemblies.

Fatigue Life Conservation

17. One of the objectives of structural integrity policy given in para 1, was the minimizing of life cycle costs, which can be achieved in part by conserving fatigue life. Clearly, the way in which an aircraft is operated has a prime influence on the rate at which fatigue damage is accumulated, and hence on the consumption of its FI. Structures are designed to carry no more than the loads arising when aircraft are flown in accordance with the Release to Service, and pilots are responsible for ensuring that the flight limitations are not exceeded. A cockpit accelerometer provides a reasonably accurate means of monitoring the loads to which an aircraft is subjected during manoeuvres, although the positioning of the instrument away from the C of G may give rise to discrepancies with respect to the readings on the fatigue meter. The degree of under- or over-reading on a cockpit accelerometer will depend on the extent of this displacement, its damping characteristics, and the rate of entry into, or recovery from, a manoeuvre. Accurate cockpit readings and lower stress loadings on airframes are more likely when manoeuvres are entered and exited smoothly. Inadvertent overstressing must always be reported so that structural examination for damage is carried out.

18. Although high 'g' loading can be very wasteful of fatigue life, it is important to remember that because metal fatigue is a cumulative process, most fatigue life consumption occurs during routine sorties, at moderate 'g' levels, when aircraft are operated within the parameters laid down in the Release to Service, and in accordance with the SOI. Fatigue limiting practices should consequently be observed at all times. Weight, for example, is a critical factor; if an aircraft's weight is increased by 1%, the consumption of its fatigue life may go up by as much as 5%. Unnecessary weight should not be carried, and sorties should be so organized that the greatest degree of manoeuvring takes place towards the end of the flight, when fuel weight is down.

19. In summary therefore, there are a number of fatigue life conservation measures which should be the day-to-day concern of pilots, flight planners, and authorizing officers. These include:

a. Ensuring that aircraft are flown with serviceable fatigue meters.

b. Keeping tight turns and other high 'g' manoeuvres to a minimum consistent with the sortie operational requirements, and never exceeding Release to Service handling and clearance limitations.

c. Structure-friendly handling into and out of manoeuvres.

d. Carrying only the fuel actually required for sorties.

e. Avoiding turbulence whenever possible.

f. Minimizing close formation flying.

g. Flying only as low, and for as long as, the task requires.

h. Minimizing roller landings.

i. Keeping down taxiing speeds, particularly over rough surfaces.

CHAPTER 20 - THEORY OF METAL FATIGUE

Introduction

1. This chapter covers the basic theory of fatigue in metals. The concepts explained here are those that are applied to the design, testing and monitoring of aircraft structures by aircraft companies, the MOD and the RAF. As with any other aspect of aircraft design, the application of basic theory to practical problems involves much added complication, so a clear understanding of the essential characteristics of fatigue is important if one is to progress further.

2. **The Reasons for Fatigue in Airframes**. Fatigue is inevitable in any metal structure where alternating tensile stresses are repeated many times. In order to obtain a high performance from an aircraft, the mass of the airframe, as well as all other components, must be kept as low as possible and therefore the stresses in it will be high. This is indicated by the fact that the maximum service loads in structural components are often as high as 2/3 of their respective failure loads. The stresses in an airframe will alternate in a pattern determined predominantly by the aircraft flight manoeuvres, atmospheric turbulence, ground loads, cabin pressurization and thermal effects. It is mainly tensile stresses that cause fatigue hence the proneness of wing lower skins and lower spar booms, for example, to fatigue failure. The problem of high stresses is aggravated by the stress concentrations caused by, for instance, fastener holes. At these places the local stress is several times the average stress in the material and hence it is here that fatigue cracks start.

The Stages of Failure

3. The progression of fatigue cracks can be split into three stages: initiation, crack growth, and final fracture. The second and third stages are normally evident on a fatigue fracture surface. There are often marks which indicate the way in which the crack front has spread and these are called beach marks, striations, conchoidal marks or progression marks. The concept of S-N curves is used to cover the total process of fatigue whilst the recently-developed science of fracture mechanics can be used for the crack growth and final fracture stages.

4. **Crack Initiation**. Fatigue cracks almost always start at the surface of components, although exceptionally they will start at sub-surface flaws. These cracks tend to start at fastener holes, screw threads, machining marks, sharp corners and other geometrical features. For a round hole in a sheet of metal under tension, the local stress at the side of the hole is three times the average stress, (i.e. stress concentration factor = 3), and of course this area is where the cracks start.

5. **Crack Moderation**. Surface conditions created by corrosion and fretting will also help to initiate fatigue cracks, and residual tensile stresses from manufacturing processes speed up the fatigue process too. Crack initiation can be delayed and fatigue lives increased by measures such as cold-working of holes and bead-blasting of surfaces to leave residual compressive stresses.

6. **Crack Growth**. Once a fatigue crack has propagated to a length of, say, 1mm then the rate of growth depends less on the conditions that initiated the crack and will correlate more with the average stress in the surrounding material. With a large stress concentration factor the initiation period is shorter than with a small stress concentration factor, but once a crack is established it grows at about the same rate in both cases. If metal is being fatigued in a very corrosive environment then not only will the crack initiation period be shorter than otherwise, but the subsequent rate of crack growth will be

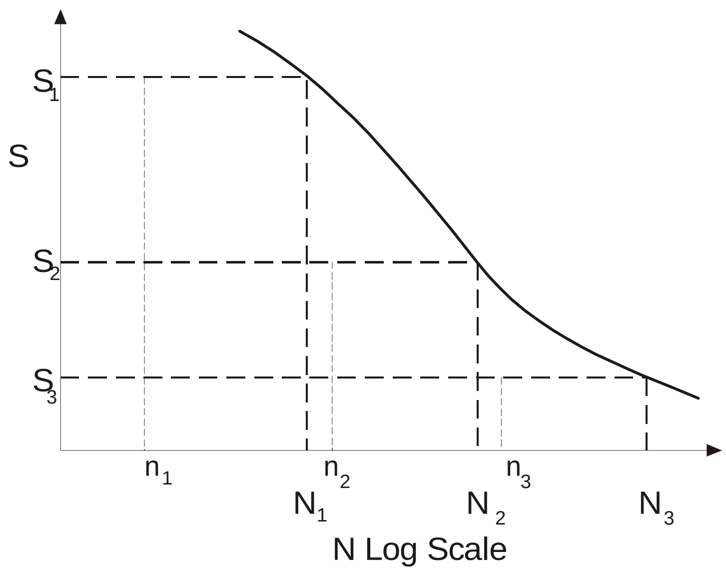
faster because corrosion speeds up crack growth. Also, corrosion may enable fatigue cracks to grow at low stress levels which would normally not cause fatigue at all.

7. **Final Fracture**. Final fracture will normally occur in a brittle manner (i.e. with no plastic deformation) as the result of a fatigue crack. It will happen when the crack reaches the critical crack length for the applied stress. The failure does not depend on the average stress in the cracked section reaching the ultimate tensile stress for the material, but on the conditions at the crack tip.

Miner's Rule

8. The cumulative fatigue damage caused by varying amplitude loading cycles is normally estimated using miner's rule. The appropriate S-N curve, discussed in the previous chapter, is used for the estimation.

1-20 Fig 1 Miner's Rule - Use of S-N Curve



9. Plotted on the S-N curve are three different alternating stress levels, S_1, S_2, S_3 which are applied n_1, n_2, n_3 times respectively. In a real calculation there may be several more stress levels, depending on the overall loading. Miner's rule states that the structure will fail when:

$$\frac{n_1}{N_1} + \frac{n_2}{N_2} + \frac{n_3}{N_3} + ... = 1$$

where N_1, N_2, N_3, etc are the number of cycles to failure at the applied stresses. In a simple example, taking just three stresses if we get for each 1,000 flying hours:

S_1 occurring 10 times - hence $n_1 = 10$

S_2 occurring 2,000 times - hence $n_2 = 2,000$

S_3 occurring 200,000 times - hence $n_3 = 200,000$.

And from the S-N curve:

$N_1 = 10^3$ cycles

$N_2 = 10^5$ cycles

$N_3 = 10^7$ cycles

Then proportion of fatigue life used is given by:

$$\frac{n_1}{N_1} + \frac{n_2}{N_2} + \frac{n_3}{N_3} = \frac{10}{1{,}000} + \frac{2{,}000}{100{,}000} + \frac{200{,}000}{10{,}000{,}000}$$

$$= 0.05$$

Since this damage occurred in 1,000 hours flying, the total safe life in flying hours (assuming the S-N curve has been factored to allow for 'scatter') is equal to:

1,000 divided by 0.05, which is 20,000 hours.

Critical Crack Length (CCL)

10. The critical crack length is the length at which a crack propagates catastrophically to failure. As has already been implied, it depends on the material and the average applied tensile stress. The overall size of a component has only a secondary effect and so, for example, if the CCL was 100mm for a 1m wide skin panel with a certain applied stress, then it would also be about 100mm for a 3m wide panel of the same material and applied stress.

11. All other things being equal, the higher the average applied stress on a component, the shorter will be its critical crack length. In general, aluminium alloys tend to have relatively long CCLs, titanium alloys shorter CCLs and high strength steels the shortest of all.

CHAPTER 21 - TRANSONIC AND SUPERSONIC AERODYNAMICS

Introduction

1. Aircraft flying at speeds well below the speed of sound send out pressure disturbances, or waves, in all directions. This enables an approaching aircraft to be heard and, more importantly for the aircraft, the air is warned of its approach. This warning gives the air time to divide and allows the passage of the aircraft with minimum disturbance.

2. If air was incompressible, the speed at which pressure disturbances travelled would be infinite, therefore the disturbance created by an aircraft would be felt everywhere instantaneously, regardless of the aircraft's speed; also, the flow pattern around the aircraft would be independent of its speed. However, air is compressible, and a change of density and temperature accompanies a change of pressure. Because of this, the speed of propagation of the pressure wave has a finite value; the speed of sound.

3. As the speed of an aircraft is increased, there is a decrease in the distance ahead of the influence of the advancing pressure waves; there is also a change in the flow and pressure patterns around the aircraft and this will ultimately change its manoeuvrability, stability and control characteristics. To understand the changes in these patterns it is helpful to examine the changes that take place when a source of small pressure waves is moved through the air. The distinguishing feature of small pressure waves is that they travel at the same speed (i.e. sonic speed) as they radiate from their source. Sound waves are audible pressure waves and, of course, also travel at the speed of sound.

Definitions

4. **Definitions of Terms**. Definitions of terms which will be used throughout this chapter are:

a. **Speed of Sound (a)**. The speed at which a very small pressure disturbance is propagated in a fluid under specified conditions.

b. **Mach Number (M)**.

$$M = \frac{\text{the speed of an object or flow}}{\text{the speed of sound in the same part of the atmosphere}} = \frac{V}{a}$$

Notes:

1. Both V and a may vary.

2. M is a ratio and has no units.

c. **Free Stream Mach Number (MFS)**. This is the Mach number of the flow sufficiently remote from an aircraft to be unaffected by it.

Notes:

1. $$M = \frac{V}{a}$$

$$\text{therefore } M = \frac{\text{TAS}}{\text{local speed of sound}}$$

2 M_{FS} is sometimes called flight Mach number.

3. Ignoring small instrument errors, M_{FS} is the true Mach number of an aircraft as shown on the Machmeter.

d. **Local Mach Number (M_L)**. When an aircraft flies at a certain M_{FS}, the flow is accelerated in some places and slowed down in others. The speed of sound also changes because the temperature around the aircraft changes. Hence,

$$M_L = \frac{\text{speed of flow at a point}}{\text{speed of sound at the same point}}$$

M_L may be higher than, the same as, or lower than M_{FS}.

e. **Critical Mach Number (M_{CRIT})**. As M_{FS} increases, so do some of the local Mach numbers. That M_{FS} at which any M_L has reached unity is called the critical Mach number. As will be seen later, M_{CRIT} for a wing varies with angle of attack; it also marks the lower limit of a speed band wherein M_L may be either subsonic or supersonic. This band is known as the transonic range.

f. **Detachment Mach Number (M_{DET})**. If an aerofoil has no leading-edge radius, there will be a M_{FS} at which the bow shockwave of an accelerating aircraft attaches to the sharp leading edge (or detaches if the aircraft is decelerating from supersonic speed). This value of M_{FS} is M_{DET}. At M_{DET} all values of M_L are supersonic except in the lower part of the boundary layer. In practice this rarely happens since wings have a significant radius to the leading edge, and behind the shock immediately in front of the leading edge there will be a small area of subsonic flow. Nevertheless, M_{DET} can still usefully be defined as that M_{FS} above which there is only a small movement of the bow shockwave with an increase in speed. M_{DET} can therefore be used to indicate the upper limit of the transonic range.

g. **Critical Drag Rise Mach Number (M_{CDR})**. (M_{CDR}) is that M_{FS} at which, because of shockwaves, the C_D for a given angle of attack has increased significantly. Different criteria are often used, i.e. 0.002 rise in C_D, 20% rise in C_D

5. **Definitions of Flow**. Fig 1 illustrates the flow speed ranges. Changes in airflow occur at $M_L = 1.0$ and the boundary between each region of flow is that M_{FS} which produces an M_L appropriate to that region.

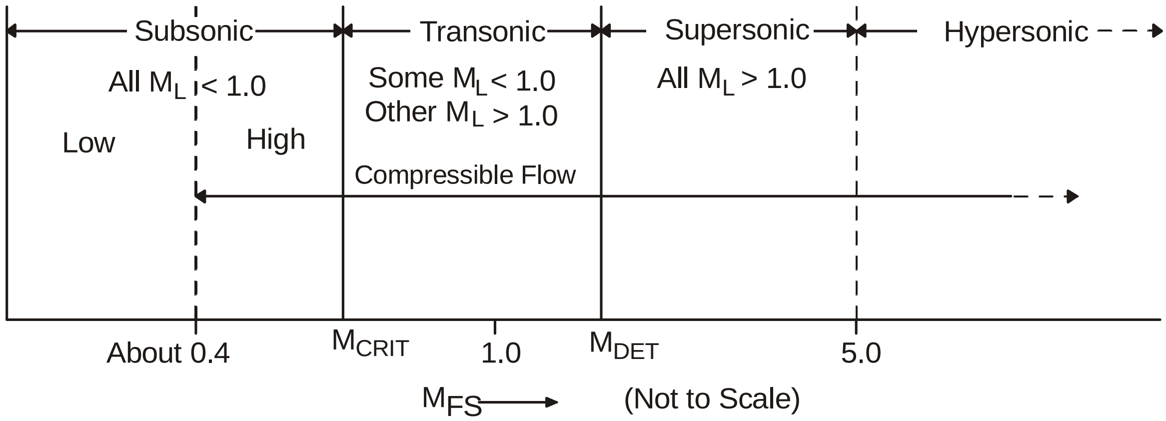
Notes:

1. The subsonic region has been subdivided at M = 0.4 since below this Mach number errors in dynamic pressure, assuming incompressibility, are small. However, compressibility effects can be present even at M = 0.4; whether they are or not depends on wing section and angle of attack.

2. The actual values of M_{CRIT} and M_{DET} depend on individual aircraft and angle of attack.

3. A third definition of the transonic range is: that range M_{FS} during which shockwaves form and move significantly.

1-21 Fig 1 Flow Speed Ranges



Wave Propagation

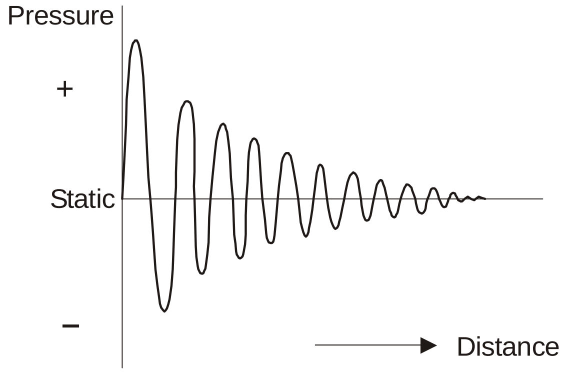
6. The two properties of air which determine the speed of sound within it are resilience and density. Resilience describes how readily the air regains its original state having been disturbed. If one leg of a vibrating tuning fork is considered as the leg moves sideways, the air adjacent to it is compressed and the molecules are crowded together. The air, recovering from its compression, expands and compresses the air adjacent to it and so a pressure wave is propagated.

7. As each particle of air is compressed, the individual speed of the crowded molecules is increased and this affects the expansion and compression of the particles surrounding it. The more the air that can be compressed, the higher will be the increased speed of the molecules, the more rapid will be the expansion and so the faster the wave will travel. Temperature is a measure of molecular speed, and therefore increasing molecular speed suggests a rise in temperature in the air. However, this is not so because the situation is adiabatic, ie there is no overall rise in temperature and the temperature change in the production of small pressure waves is infinitesimal. But if the temperature of the air is increased as a whole, then the added increase in molecular speed as the air is compressed is sufficient to cause an increase in the rate of expansion, and therefore an increase in the speed of propagation of the pressure wave, i.e. the speed of sound has been increased by an increase in temperature.

8. In terms of pressure and density, $a^2 = \frac{\gamma P}{\rho}$ where γ is the ratio of the specific heats (SH) $\left(\frac{\text{SH constant pressure}}{\text{SH constant volume}}\right) = 1.4$ for air. This can be rewritten as $a^2 = \gamma RT$ where R is the gas constant and T is in K; therefore $a = \sqrt{\gamma RT}$. Allowing for the two constants, it can be seen that $a \propto \sqrt{T}$.

9. It should be noted that, in the transmission of pressure waves, air is not physically displaced, it merely vibrates about a mean position as does the prong of a tuning fork. The wave motion can be seen from a graph showing pressure variations against distance from a source at an instance of time (Fig 2).

1-21 Fig 2 Pressure Waves from a Stationary Source

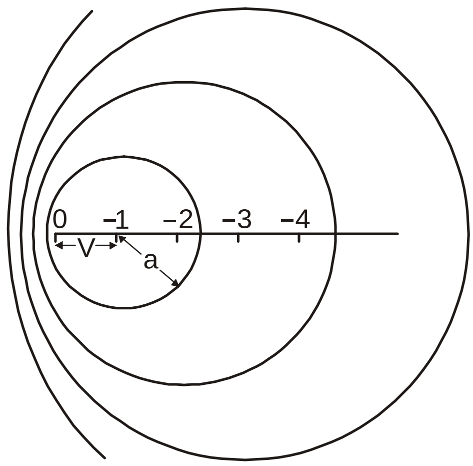


10. The size of the wave is measured by its amplitude, ie the maximum change of pressure from static. The amplitude of a sound-wave is approximately one millionth of an atmosphere. In free air the waves weaken with distance from source because the energy in the wave is spread over an ever-increasing surface. The amplitude decreases rapidly at first, and then more gradually until the wave becomes too weak to measure.

Pressure Waves from a Moving Source

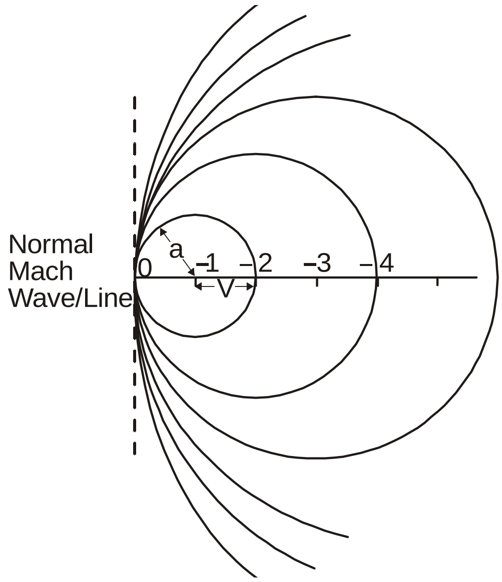
11. **Source Moving at Subsonic Speed**. Fig 3 shows the pattern produced at a given instant in time. The sound-waves emanate in all directions relative to the source, although they are closer together ahead of the source than behind it. The waves maintain their separation and there is no tendency for them to bunch.

1-21 Fig 3 Pattern from a Subsonic Source



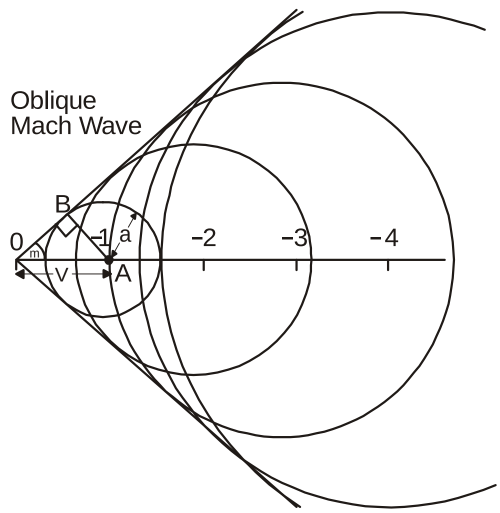
12. **Pattern from a Subsonic Source**. A different pattern is produced when the source is moving at the speed of sound (Fig 4). The waves cannot move ahead of the source; they bunch up and form a Mach wave ahead of which the air is quite unaware of the existence of the sound source. The Mach wave is not a shockwave, it is merely a line dividing area where the source can be heard and areas where it cannot. As the Mach wave is at right angles to the direction of movement of the source the wave is called a normal Mach wave.

1-21 Fig 4 Pattern from a Sonic Source



13. **Source Moving at Supersonic Speed**. At supersonic speeds yet another pattern is produced (Fig 5), again with a boundary beyond which no wave can pass, i.e. the limit of influence of the source. This boundary is called an oblique Mach wave and the angle it makes with the flight path is called the Mach angle (m). From Fig 5 it can be seen in triangle AOB that sin m = $\frac{a}{V}$, but as M = $\frac{V}{a}$ (para 4b), sin m = $\frac{1}{M}$. In the air, this pattern is three-dimensional, so the boundary becomes a surface called the Mach cone. To distinguish between the regions where the air is affected by the source and where it is not, the terms Mach after-cone and Mach fore-cone respectively are used.

1-21 Fig 5 Pattern from a Supersonic Source



14. **Mach Waves**. It should be noted that a flow passing through a Mach wave is affected infinitesimally since the pressure change across the wave is infinitely small. A Mach wave, or line, is a line along which a pressure disturbance is felt, and the significance of Mach lines will be appreciated when expansions in supersonic flow are considered later.

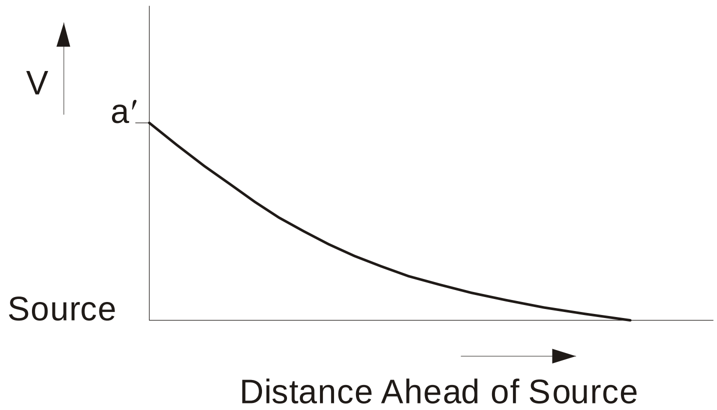
15. **Large Pressure Waves**. In para 3 it was explained that small pressure waves travel at sonic speed. When large pressure waves are produced (say, by a wing) there is a significant temperature increase at the source. This increase reduces with distance from the source and changes the speed of wave propagation as the waves radiate from the source. Due to this temperature rise, there is an increase in the local speed of sound and the initial speed of propagation of the pressure wave is at that higher sonic speed.

SHOCKWAVES

Formation of a Bow Shockwave

16. The pressure pattern from an aerofoil section at some subsonic speeds is well known. In particular, the region close to the leading edge will be experiencing pressure higher than atmospheric. Since the pressure and temperature are increased, then the speed of sound in that region will be higher than in the undisturbed flow some distance ahead of the wing. Using the same reasoning, by beginning at the leading edge where the pressure is highest, then moving up-stream ahead of the wing, pressure and temperature will reduce until finally a point is reached where they have reduced to the free stream static values. Between this point and the leading edge, the speed of sound will be higher than that of the free stream because of the compression and consequent temperature rise. The symbol conventionally given to this locally increased speed of sound is a′. If a typical curve of pressurization is plotted from its static value ahead of the leading edge, the result would be as shown in Fig 6.

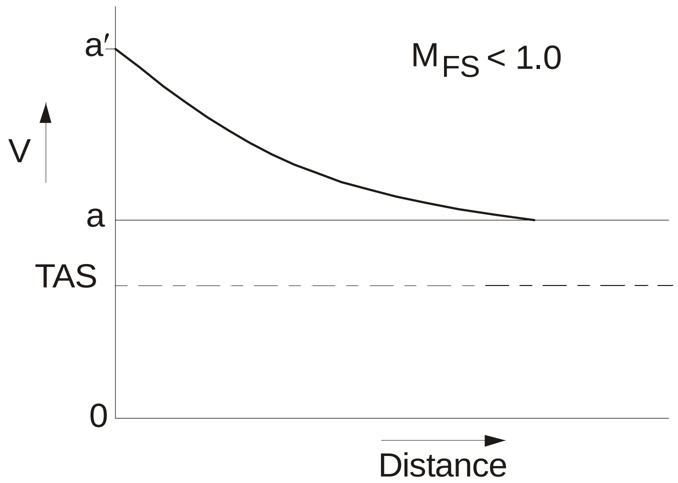
1-21 Fig 6 Pattern from a Supersonic Source



17. When a wing or any object is placed in a flow, the flow around the object is deflected. If the deflection is such that the air is compressed, then the deflection and speed of the flow will determine the compression and temperature rise and, consequently, the local increase in the speed of sound. Consider an infinitely thin flat plate; if it is at 0º angle of attack, there will be no pressure changes round it. If it is set at a positive angle of attack, the pressure rise underneath will increase the temperature such that the increase in speed of propagation of the pressure waves ahead of the plate is a′ (where a is the speed of sound in the undisturbed flow). The initial value of a′ depends on the shape and speed of the source; the blunter the source, the greater the increase in the local speed of sound.

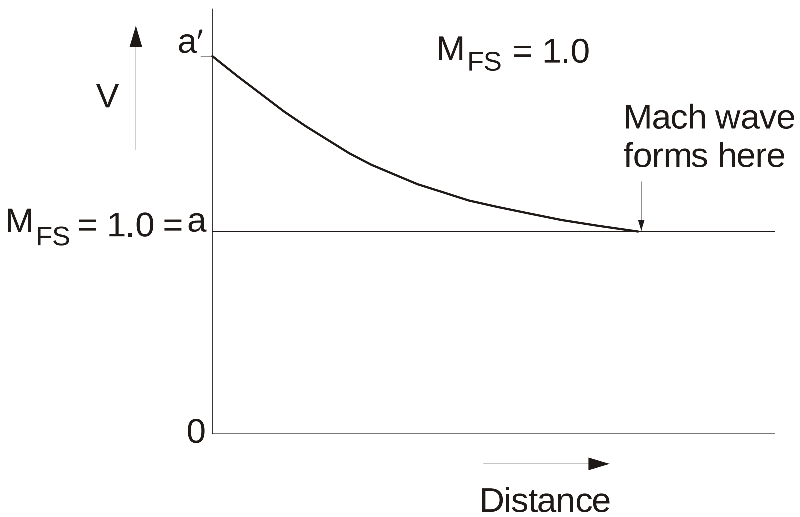
18. Considering now a wing at an M_{FS} < 1.0, the pressure disturbances from the wing will propagate forward initially at some value of a′, which will decrease with distance until its value is equal to a. Since the aircraft is subsonic, i.e. its speed is less than a, the disturbances will continue to progress up-stream until they finally die out; they will not bunch, and no Mach wave will form. This is shown in Fig 7.

1-21 Fig 7 Pressure Variation from a Subsonic Source



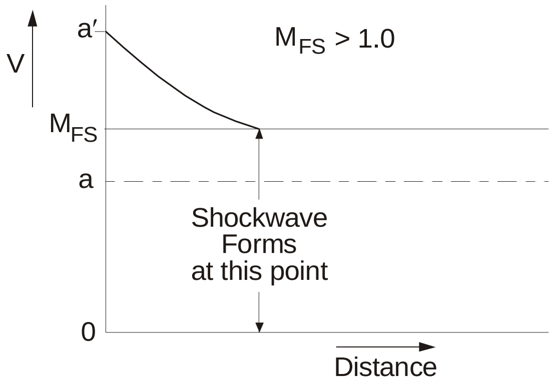
19. When M_{FS} = 1.0, the pressure waves propagate initially at a value of a′ greater than M_{FS} resulting from the heated high-pressure region ahead of the wing. They can, therefore, escape forward away from the advancing wing for as long as the temperature and pressure remain higher than ambient and P_0 (the free stream static pressure) respectively at which stage the disturbance cannot make any further progress up-stream. Thus, a Mach wave will form at this point well ahead of the aircraft and the blunter the wing leading edge (more disturbance) the further away it will be. The location of the Mach wave is shown on Fig 8. Note that it is still a Mach wave - not a shock wave - because pressure is P_0 at this point.

1-21 Fig 8 Pressure Variation from a Source at the Speed of Sound



20. An increase in M_{FS} above M 1.0 results in an increase in the initial a′, but at some distance ahead of the aircraft the reducing value of a′ will equal the forward speed of the aircraft, ie a′ will equal M_{FS}. The pressure waves will therefore be unable to penetrate further up-stream, as shown in Fig 9. The pressure waves will bunch and form a shockwave called the bow shockwave; this wave is the forward limit of influence of the aeroplane; air ahead of it receives no warning of the aircraft's approach.

1-21 Fig 9 Pressure Variation from a Supersonic Source



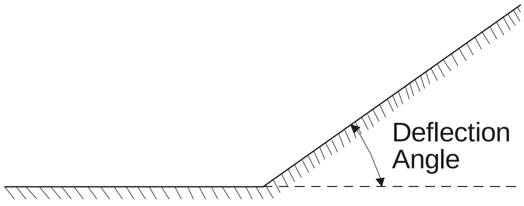
21. As far as an observer on the ground is concerned, there is no aircraft noise ahead of the bow shockwave, but as the shockwave passes the observer there is an intense increase in noise (the sonic 'bang') and then the normal aircraft sound is heard. A second 'bang' comes from the shockwave formed around the wings and fuselage of the aircraft and will be explained in a later paragraph.

22. If the aircraft continues to accelerate, the pressure waves reach their forward limit of travel even closer to the source. Once again the bow shockwave will be positioned where the decreasing a′ is equal to the M_{FS} and will be more intense. Eventually, when the M_{FS} reaches the same value as the initial a′ of the disturbance, the bow shockwave attaches to the leading edge; this is at the detachment Mach number (M_{FS}) and any further increase in speed results in the shockwave becoming more oblique. This stage will only be reached if the leading edge is very sharp, since the value of a′ is a function of speed and shape and increases with speed. Where there is a rounded leading edge, an increase in speed will give rise to increased stagnation temperatures and the shock will be 'held off'.

23. The disturbances which cause an aerofoil to generate shockwaves originate at the points where the airflow suddenly changes direction - notably at the leading and trailing edges. The angle through which the airflow changes direction at those points is known as the deflection angle. Below is a table of detachment Mach numbers for particular deflection angles with regard to the bow wave.

Deflection Angle	M_{DET}
10°	1.41
20°	1.83
30°	2.00

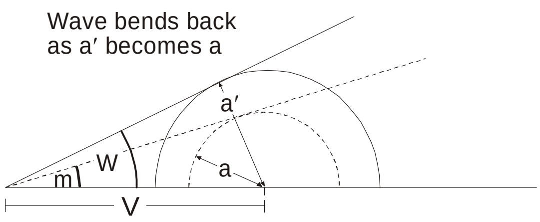
1-21 Fig 10 Deflection Angle



24. Comparing the case V > a (small source) and V > a′ (large source), it is found that because a′ is greater than a, the wave angle (W) is always greater than the Mach angle (m), as shown in Fig 11.

Also, from Fig 11, sin $W = \frac{a'}{V}$. Multiplying by $\frac{a}{a}$, gives sin $W = \frac{a'}{a} \times \frac{a}{V} = \frac{a'}{a} \times \frac{1}{M}$ from which W can be calculated.

1-21 Fig 11 Wave Angle Greater than Mach Angle



25. To summarize:

a. M_{FS} < 1.0 - no bow shockwave.

b. M_{FS} > 1.0 < a′ - detached normal shockwave.

c. M_{FS} > a′ - oblique attached shockwave.

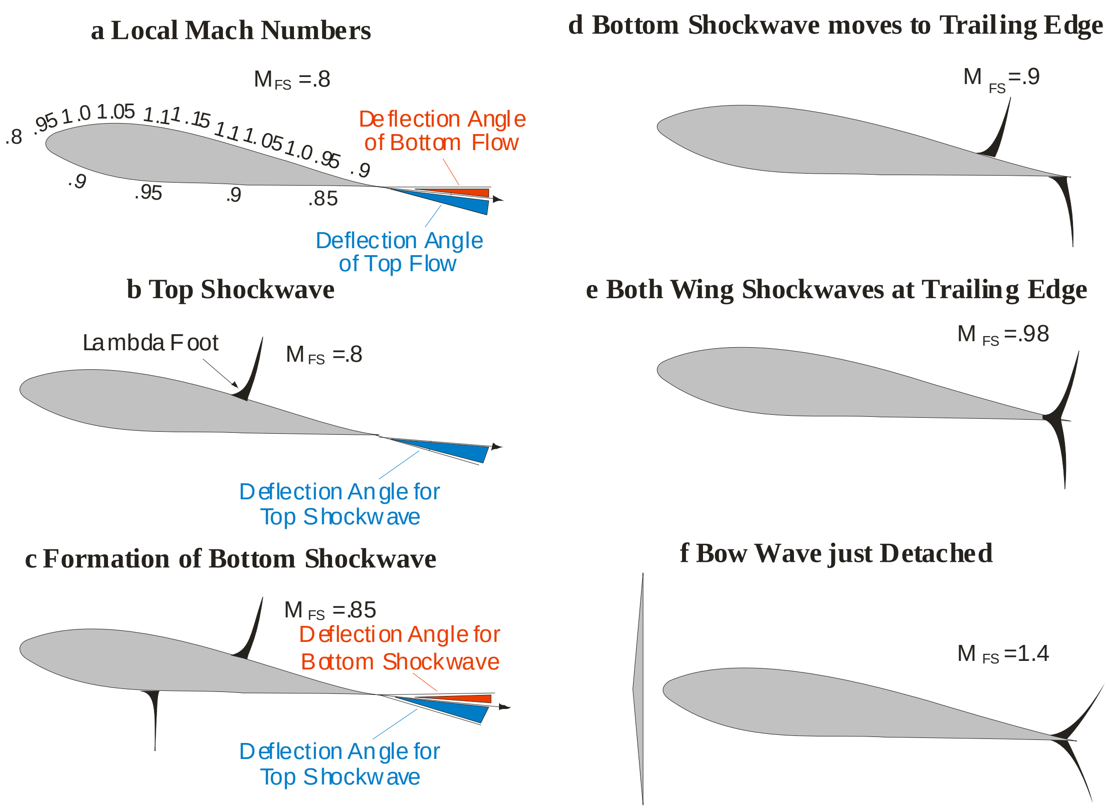
d. The blunter the source, the higher M_{DET} will be.

Wing Surface Shockwaves

26. The bow shockwave is not the first shockwave to form. Long before the aircraft as a whole has reached Mach 1, some portions of the flow around the aircraft have exceeded the speed of sound. When the first local Mach number (M_L) equals 1.0 somewhere around the aerofoil, the aircraft has reached its critical Mach number (M_{CRIT}) and shockwaves will begin to form.

a. **Conditions for Shockwave Generation**. Fig 12a shows the local Mach numbers around a wing at a speed in excess of M_{CRIT} (M_{FS} = 0.8 in the example) at the instant in time before any shockwave has had time to develop. Where the top and bottom airflows meet at the trailing edge, they are deflected violently (each through a different deflection angle) to follow a common path. At the point where they meet there will be another region of high pressure similar to the region of high pressure at the leading edge. Beneath the wing, since the highest M_L = 0.95, the pressure waves travelling at sonic speed are able to escape forwards. Similarly, above the wing nearer the trailing edge, in the flow where M_L = 0.9 and 0.95, pressure waves can progress forwards but they are soon brought to rest at the point where M_L = 1.0.

1-21 Fig 12 The Formation of Wing Shockwaves



b. **Top Shockwave**. Fig 12b shows the situation an instant later when only a top shockwave forms. Shockwave generation is complicated somewhat by the presence of the boundary layer, part of which must always be subsonic. This enables the pressure rise to be communicated forward of the shock, thickening the boundary layer and giving rise to a further compressive corner ahead of the main shockwave known as the lambda foot. This interaction between the shockwave and the boundary layer is dealt with in more detail in Design for High Speed Flight (Volume 1, Chapter 22).

c. **Bottom Shockwave**. As M_{FS} is increased the top shockwave will grow steadily stronger and start to move rearwards, eventually attaching to the trailing edge. At some stage during this movement, some local Mach numbers below the wing will reach unity and a bottom shockwave will form (Fig 12 c). This will be weaker than the top shockwave because a wing at a positive angle of attack will feature a smaller deflection angle for the bottom shockwave. This smaller deflection angle also means that the bottom shockwave will move rearwards faster than the top shockwave and will attach to the trailing edge before it (Fig 12d).

d. **Progression to M_{DET}**. As M_{FS} is increased further, both surface shockwaves become attached to the trailing edge (Fig 12e) and, as speed is increased even further, become more and more oblique. The bow wave forms once M_{FS} exceeds 1.0 and moves towards attachment but, for a blunt leading edge, does not quite achieve this stage as explained in para 22 and illustrated in Fig 12f. Conversely, the sharper the leading edge the sooner will the weaker shockwave become attached (at M_{DET}) and any further increase in Mach number will cause the normal shockwave to become oblique.

Acceleration of a Supersonic Flow

27. To examine the changes that take place when supersonic flow is accelerated, consider the flow through a venturi tube, as in Fig 13a. The up-stream end is connected to a reservoir containing air, the total pressure P_0 and temperature T_0 of which are maintained constant. The down-stream end is connected to another tank in which the pressure may be varied. If no pressure difference exists along the duct there is no flow. If the pressure P is decreased slightly to P_1, then a low speed flow results, producing a pressure distribution as shown in Fig 13b. Gradual lowering of the pressure P will lead to larger flow speeds until at pressure P_4 there is a throat pressure of $0.528P_0$ and a flow speed of M = 1.0. If the pressure P is decreased even further to P_5, the change in pressure will be transmitted by a pressure wave which will travel at Mach 1.0 up-stream from the downstream end of the venturi. This can reach the throat but there it will meet the flow travelling downstream at the same speed and so it will be unable to pass through the throat. The flow up-stream of the throat will therefore be unaware of any changes in pressure downstream of the throat, and will remain unchanged no matter how large or small are further reductions in P. This produces several important results:

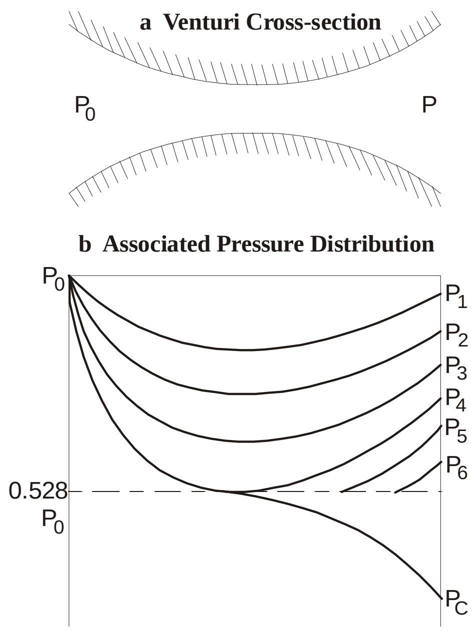
a. M = 1.0 is the highest Mach number that can be obtained at the throat.

b. At M = 1.0, the pressure will always be 52.8% of the total head pressure. (In this case, THP is P_0 because the situation started with no flow.)

c. Maximum mass flow is obtained with M = 1.0 at the throat. The flow at the throat is sonic and so mass flow is independent of the down-stream pressure.

d. The streamlines are parallel at the throat.

1-21 Fig 13 Supersonic Flow Acceleration in a Duct

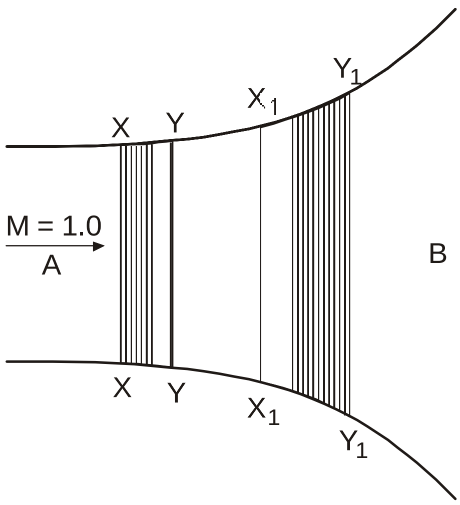


28. The reduction in pressure is, however, felt up to the throat and the adverse pressure gradient is removed in a small section of the duct. Now a sonic flow is encountering diverging conditions in a cross-sectional area of **no pressure gradient**. What happens to the flow can be best explained by considering the small section of the duct which has just started to diverge from the throat.

29. Fig 14 shows a very small section of a venturi at, and just beyond, the throat through which there is a steady flow. A small, very thin disc of air lying between XX and YY travelling at sonic speed will

be considered as it moves to the position X_1X_1 and Y_1Y_1. There is no pressure difference at the moment between A and B.

1-21 Fig 14 Supersonic Flow in a Divergent Duct



30. In seeing how velocity and Mach number will change, three separate stages are considered:

a. On moving out of the throat, because of inertia, the disc faces will want to maintain their velocity. The duct has, however, diverged and the disc will expand to fill the available space. In so doing the density, and with it the temperature, will decrease; this increases the Mach number.

b. In comparison with X_1X_1, the greater expansion, and therefore the greater density change, is felt at position Y_1Y_1. Thus, there will be a pressure difference across the disc and the disc will accelerate. If the velocity increases, Mach number increases.

c. In a steady flow the mass flow is constant, and the same number of discs will pass a point in unit time. However, the velocity has increased so there has been an apparent increase in mass flow. To keep the mass flow constant the thickness of the disc must increase, and this means a further drop in density and temperature. As a result, Mach number again increases.

Notes:

1. Because of the density changes, a large increase in Mach number can be achieved for a relatively small increase in velocity.

2. The shape of the duct controls the expansion and thus the acceleration.

3. This effect could be produced in subsonic flow if conditions of no pressure gradient could be achieved from A to B.

Thus, we have sonic flow expanding and accelerating in conditions of divergency. Streamlines of the flow will also diverge.

31. Referring back to the whole venturi tube in Fig 13, a reduction of pressure to P_6 means that the area of no pressure gradient will be increased, and the flow will be free to expand and accelerate along the curve to P_C. This curve is the plot of flow if the speed is supersonic for the whole of the venturi's diverging length, and its shape is entirely dependent on the shape of the diverging part of the duct (see para 30, note 2).

32. At some distance down the duct the now supersonic flow will encounter an adverse pressure gradient. The decreasing pressure of the expanding supersonic flow is equal to the pressure in the duct and the flow will be forced to decelerate. This deceleration takes place through a shockwave.

33. Considering a curved wing surface as part of a duct the important fact to appreciate is that the flow will be parallel to the surface and the change in surface ahead of the flow will be sensed as an area of divergence. When an $M_L = 1.0$ is reached the flow will be free to accelerate within the limits of the shape of the camber of wing and until it encounters no more divergence or a strong adverse pressure gradient.

34. The flow through a duct can be summarized as:

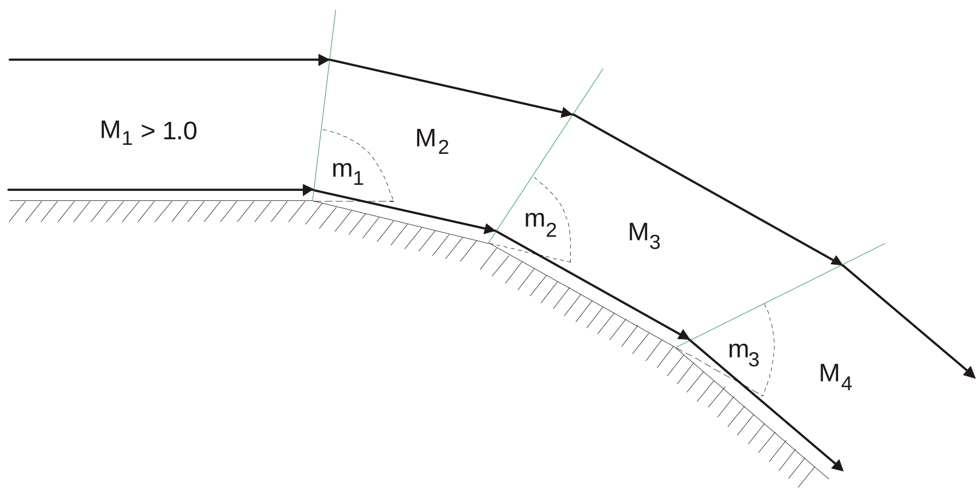
Flow	**Duct or Stream lines**	
	Divergent	Convergent
Subsonic	P↑ ρ↑ V↓	P↓ ρ↓ V↑
Supersonic	P↓ ρ↓ V↑	P↑ ρ↑ V↓

35. Flow through a duct has been considered but, to understand the reason behind the design of a fully supersonic wing section, it is necessary to examine in rather more detail how a supersonic flow negotiates sharp corners.

Expansive Corners

36. As its name implies, an expansive corner is a convex corner which allows the flow to expand and thus accelerate. Any corner can be considered to consist of a series of infinitely small changes in surface angle and Fig 15 shows a single corner broken down into a series of greatly exaggerated steps.

1-21 Fig 15 Supersonic Flow Round a Convex Corner

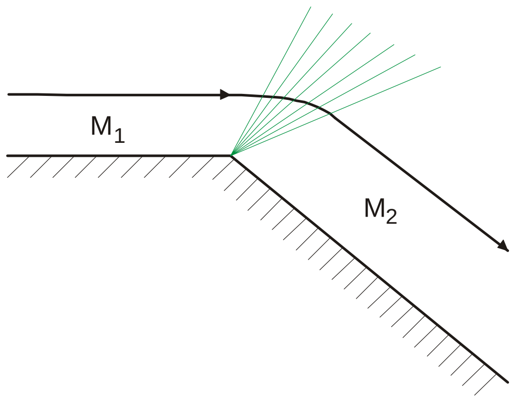


37. Consider the two streamlines in the supersonic flow M_1 as they reach the corner. The one adjacent to the surface will sense the change in surface as soon as it gets to the corner, the flow will expand, accelerate and, associated with the decrease in density, the pressure will decrease. This pressure disturbance will be felt along a Mach line appropriate to the flow M_1. The streamline further away from the surface will be 'unaware' of the corner until it reaches the Mach line originating at the corner. This Mach line is a boundary between relatively high and low pressure, therefore a pressure gradient is felt across it such that the

streamline is accelerated slightly and turned through an angle equal to the change in surface angle. It now continues parallel to the new surface at a higher Mach number (V increased, a decreased because of lower temperature). The streamlines are now farther apart. The same process will be repeated at the next change in surface angle with one important difference: the Mach angle of the flow M_2 will be smaller.

38. The process outlined above will take place at a corner through an infinite number of Mach lines and, because the lines 'fan' out, the expansion and acceleration is smooth. The region within which the expansion takes place is limited by the Mach lines appropriate to the speed of the flow ahead and behind the corner; this is called a Prandtl-Meyer Expansion (Fig 16).

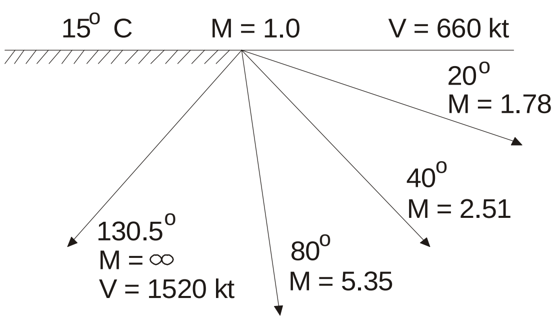
1-21 Fig 16 Prandtl-Meyer Expansion



39. The decrease in pressure, ie a favourable pressure gradient, round an expansive corner in supersonic flow allows an attached boundary layer to be maintained; this is exactly the opposite to what happens in a subsonic flow where an adverse pressure gradient would exist causing the boundary layer to thicken and break away. Consequently, there is no objection to such corners on essentially supersonic aircraft and design features are permitted which would be quite unacceptable in subsonic aircraft.

40. An example of expansive corners and associated velocity and Mach number increases is given in Fig 17. The initial flow is M = 1.0 in standard conditions. The theoretical maximum angle of expansion for sonic flow is 130.5°. The vectors indicate true velocity of the flow.

1-21 Fig 17 Mach Number Increased Round Expansive Corners

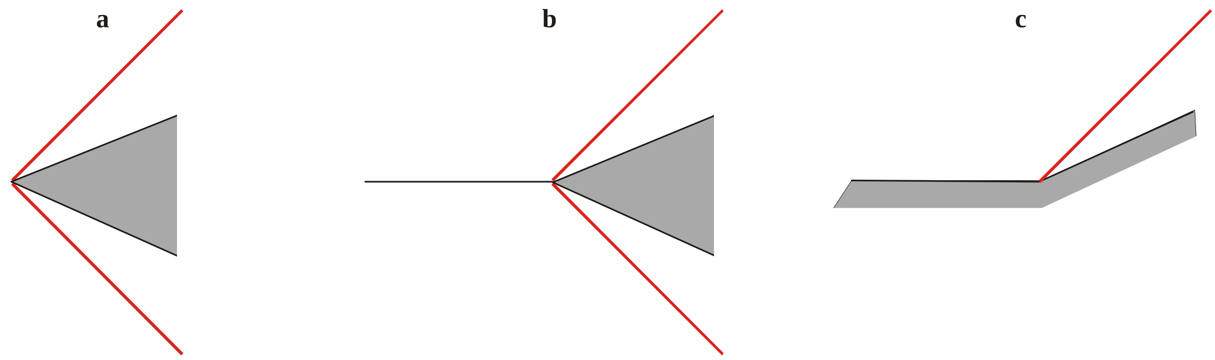


It should be noted that to increase the final speed, the temperature of the initial flow would have to be increased.

Compressive Corners

41. A compressive corner causes supersonic streamlines to converge, creating a shockwave whose formation can be visualized by treating the corner as a source of pressure waves creating a bow wave. The shockwave exhibits all the characteristics of a bow wave; it first forms as a detached normal shockwave, moves back to the source with an increase in speed, attaches and becomes oblique. Fig 18 illustrates a compressive corner obstructing supersonic flow above M_{DET}.

1-21 Fig 18 Subsonic Flow at a Compressive Corner



A sharp wedge creating an oblique bow shockwave

A thin plate, parallel to the initial flow, has been added. This does not influence the flow or shockwave

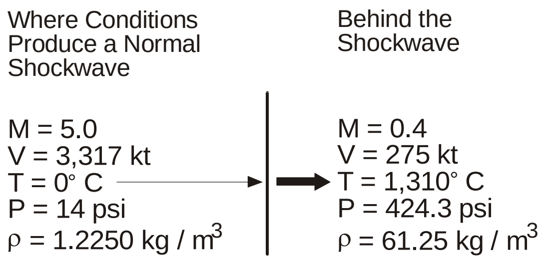
The wedge has been divided. A single oblique shockwave is originating from the change in surface direction, the compressive corner.

Nature of a Shockwave

42. **Physical Effects of a Shockwave**. A shockwave is a very narrow region, 1/400 mm thick, within which the air is in a high state of compression. The molecules within the shock are continually being replaced by those from the free flow ahead of the shock. Mach number can be considered as being the ratio of directed energy (DE) to random energy (RE) within the flow. The velocity is a measure of DE whilst RE is related to temperature. On encountering a shockwave, a flow is violently compressed such that some DE is converted to RE (ie velocity is decreased, temperature is increased), the Mach number is reduced, and pressure and density are increased. The total energy of the flow remains constant, but there has been an increase in entropy because of the temperature increase. This process is irreversible. This is significant because this energy is lost to the aircraft and represents drag. It is also the reason why designers try to ensure correct engine intake design for a particular Mach number, the intention being to achieve the deceleration smoothly and with minimum loss of heat energy through as many weak shocks as possible. The stronger the shock, the greater the conversion to heat energy. $M = \frac{V}{a}$, therefore $M^2 = \frac{V^2}{a^2}$ which is proportional to $\frac{\text{kinetic energy}}{(T°K)}$ which, in turn, is proportional to $\frac{\text{directed energy}}{\text{random energy}}$. Therefore, if DE decreases and RE increases, M^2 decreases.

43. **Normal Shockwave**. A normal shockwave is one perpendicular to the direction of flow. It is always detached from its source, behind it the direction of flow is unchanged, and the Mach number is always subsonic (see Fig 19).

1-21 Fig 19 Flow Through a Normal Shockwave



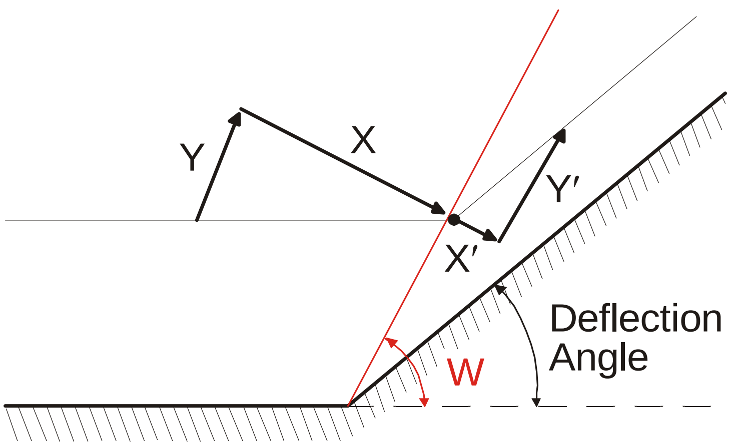
44. **Oblique Shockwave**. The flow behind an oblique shockwave has its Mach number reduced, but if the wave angle (W) is less than about 70° it will still be supersonic. The direction of flow behind an oblique shockwave is turned through an angle equal to the deflection angle of the obstruction giving rise to the shockwave.

45. **Reduction in Flow Speed Behind Oblique Shockwave**. Fig 20 shows why the reduction in flow speed behind an oblique shockwave is less than that behind a normal shockwave. The actual approach flow has been resolved into two vectors thus:

a. X is the component normal to the shockwave and is therefore affected by it.

b. Y is the component parallel to the shockwave and is therefore unaffected by it.

1-21 Fig 20 Flow Through an Oblique Shockwave



Once the flow has passed through the shockwave:

c. Component X has been reduced to a subsonic value, X′.

d. Component Y is unaffected.

e. The flow Mach number has been reduced and travels parallel to the new surface.

With the foregoing in mind two very important statements can be made about the pressures experienced round an aircraft above M_{DET}:

f. All forward-facing surfaces experience greater than atmospheric pressure.

g. All rearward-facing surfaces experience less than atmospheric pressure.

THE EFFECTS OF COMPRESSIBILITY ON LIFT

Introduction

46. In considering the effects of compressibility on lift, it is necessary to start at a speed where they become significant and see how they vary with an increasing Mach number. The lower limit will be determined by section and angle of attack; this naturally leads to the consideration of three quite distinct speed ranges:

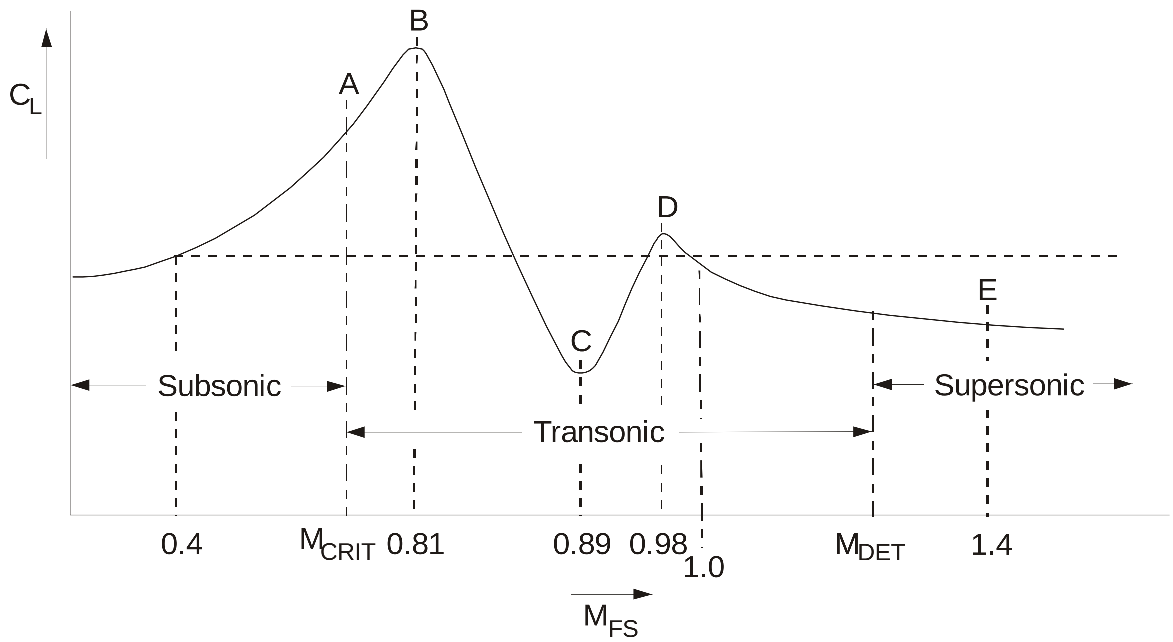
a. Fully subsonic flow.

b. Fully supersonic flow.

c. The transonic region within which there is a mixture of both.

In the case of a and b, the flow behaviour is basically simple and relatively easy to predict. In the transonic range, however, the change from stable subsonic flow to stable supersonic flow is governed by the formation and movement of shockwaves. Unless the wing is of a suitable design this will lead to undesirable lift and drag characteristics with consequent problems in stability and control. The C_L curve shown in Fig 21 is for a small symmetrical wing of 12% thickness chord ratio set at a constant angle of attack of +2°.

1-21 Fig 21 Variation of C_L with Mach Number at a Constant Angle of Attack

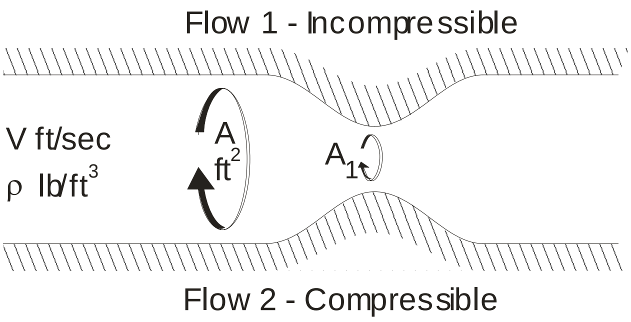


47. It is apparent from the graph that a section producing such a curve is totally unsuitable for use in the transonic range. Even so, it is instructive to examine the reasons for its shape and see how a thin wing will eliminate the valley at point C, which causes most of the undesirable handling characteristics.

Subsonic Rise in C_L

48. Consider two identical ducts each as shown in Fig 22 with identical flows in the parallel section before the throat. The flows have the same velocity and density but flow 1 is considered to be incompressible and flow 2 compressible.

1-21 Fig 22 Comparison of Flows in a Venturi



49. Since the initial flows are the same and the ducts have the same cross-sectional area, then the mass flow of both will be exactly the same and, since we are considering stream tubes, they must remain constant, i.e. $\rho \times V \times A = K$. For the purpose of this discussion give these symbols values: let $\rho = 4$ units and V = 10 units, and then compare the flows in the parallel (6 units) and throat (3 units) sections. In the parallel section the mass flow of both flows will be $4 \times 10 \times 6 = 240$ units.

50. Now compare the flows at the throat:

a. In the incompressible flow, with density unchanged

$$4 \times V \times 3 = 240$$
$$\therefore V = 20$$

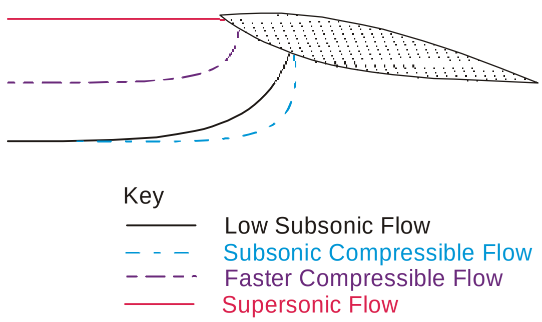
b. In the compressible flow a density change will occur at the throat associated with the pressure decrease. Assuming the density reduces to 2 units, then

$$2 \times V \times 3 = 240$$
$$\therefore V = 40$$

51. An increase in velocity is always accompanied by a decrease in pressure and since the velocity increase in the compressible flow is greater than in the incompressible flow for the same duct (and the same wing), the pressure will be lower and thus the lift will be greater for a wing in a compressible flow. At low speed, where air can be considered incompressible, lift is proportional to V^2, and C_L can be assumed to remain constant for the same angle of attack. At moderately high speeds where density changes are becoming significant, lift increases at a rate higher than indicated by V^2. In other words C_L increases for the same angle of attack.

52. Another factor which, together with the density change outlined above, affects C_L, is the amount of warning the air gets of the wing's approach, or how far ahead of the wing the streamlines start to adjust. In Fig 23 the low subsonic streamline is flowing to the stagnation point.

1-21 Fig 23 Streamline Variation with Speed



As the speed increases, compressibility effects increase, and the reduced warning causes the flow displacement to start closer to the wing; this is shown as the subsonic compressible flow. This effectively increases the angle of attack of the wing and so increases C_L. As the speed increases the air cannot be displaced fast enough and so the stagnation point moves forward as shown in the faster compressible flow. This forward movement causes a reduction in up-wash before the wing which results in a loss of lift. When the speed increases to M_{DET} the stagnation point is theoretically at the leading edge and there is no up-wash at all: this is the supersonic flow.

53. A mathematical expression has been derived known as the Prandtl-Glauert Factor, representing the increase in C_L due to compressibility. A simplified expression is:

$$\text{Prandtl-Glauert Factor} = \frac{1}{\sqrt{1-M^2}}$$

ie, the C_L may be calculated by this factor such that

$$C_L \text{ (compressible)} = \frac{C_L \text{(incompressible)}}{\sqrt{1-M^2}}$$

This would indicate that C_L would be infinite at M = 1.0 and so it needs qualification; in fact it is limited to relatively thin wings at small angles of attack with no shockwaves present.

Transonic Variations in C_L

54. In considering the C_L curve, the shockwave development and movement in the transonic range, five significant speeds are selected: The points A, B, C, D and E in Fig 21.

55. In analysing the flow, a percentage total head pressure, or stagnation pressure (P_S), diagram is used as a convenient method of showing instantaneous values of speed and pressure at any point on the wing for any given M_{FS}. The method of interpreting a P_S diagram is as follows.

56. In Figs 24 to 28, the pressure at the forward stagnation point, where the flow is brought to rest, is given the value of unity. This pressure is the sum of the dynamic and static pressures, and is known as the total head, or stagnation pressure; it is always greater than atmospheric. On either side of the stagnation point the flow accelerates causing a rise in M_L and an associated fall in pressure. When the flow has accelerated to a value equal to M_{FS} the pressure will again be atmospheric and a further acceleration above M_{FS} will cause the local pressure to fall below atmospheric.

Notes:

1. Pressures are plotted as percentages of the P_S value on the left-hand axis. The value decreases from the origin.

2. M_L is plotted on the right-hand axis, increasing from 0 at the origin.

3. For every M_L there is a corresponding pressure for the flow. For M_L = 1.0, percentage P_S is 52.8% (see para 27b).

4. The horizontal axis represents the wing chord.

5. The position of the centre of pressure and variations in C_L at different M_{FS} may also be judged from the diagrams.

6. Where the pressure in the flow has been reduced to static pressure, the Mach number of the flow is the free stream Mach number.

7. The percentage total head pressures are shown as follows:

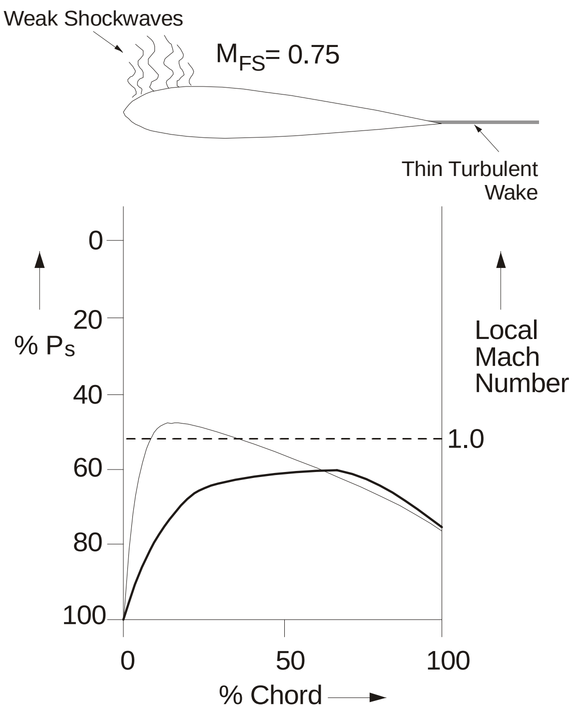
Upper surface - thin black line

Lower surface - solid black line

8. The area between the lines is a measure of the amount of lift being delivered. If the upper surface pressure is lower than the lower surface pressure this will be positive lift, and vice versa.

57. At point A in Fig 21 (see Fig 24), the M_{FS} is 0.75 and the flow accelerates rapidly away from the stagnation point along both upper and lower surfaces, giving a sharp drop in pressure through and below the atmospheric figure. Values of M_L greater than 1.0 are found between 4% and 25% chord; thus, the wing is above its M_{CRIT}.

1-21 Fig 24 Pressure Pattern at Fig 21 Point A



58. When M_L reaches 1.0, the streamlines, which previously were converging during acceleration, are not parallel. Any further movement across a convex surface will cause the streamlines to diverge, thus giving a supersonic acceleration which would theoretically continue right to the trailing edge. However, this does not happen in practice because of the 'whiskering effect'.

59. The 'whiskers' are, in effect, weak shock waves through each of which the flow is retarded slightly, the cumulative effect being to prevent further supersonic acceleration and eventually to decelerate the flow smoothly back to a subsonic value. Their origin can best be imagined by considering the compressive corner formed at the trailing edge. The separated flow leaving the trailing edge is sensed by the streamline flow as a deflection angle and, since the wake is turbulent, the value of the deflection angle will continually be changing, as will the amplitude of the pressure waves originating there. The longer ones will progress further into the supersonic flow before coming to rest than will the smaller ones. As M_{FS} is increased the Mach waves are swept together and they then form a sudden pressure discontinuity, i.e. a shockwave.

60. Summarizing the situation at M_{FS} 0.75, the wing is travelling at just above M_{CRIT}, giving the following conditions over the top surface of the wing:

a. A supersonic bubble exists; as yet there is no shockwave.

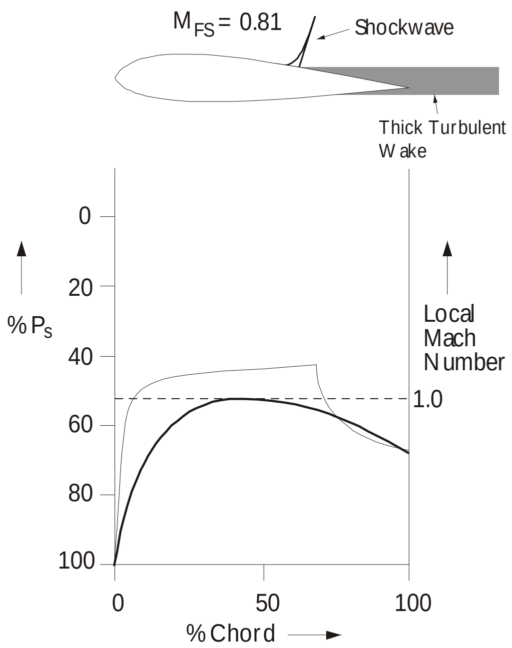
b. C_L has risen by 60% of its low speed value for the same angle of attack.

c. The centre of pressure is about 20% chord.

Over the bottom surface, the flow speed is still subsonic.

61. At Point B in Fig 21 (see Fig 25), the M_{FS} is 0.81. The top shockwave has strengthened and is located at approximately 70% chord; there is still no shockwave on the under surface. In comparing this P_S diagram with the one for M_{FS} = 0.75 it is significant to note that whereas behind the shockwave on the rear part of the wing there is no real change in the pressure differential between upper and lower surfaces, ahead of it and behind the 40% chord position the pressure differential has increased considerably due to the supersonic acceleration up to the shockwave. This effectively increases the value of C_L to approximately double its incompressible value and causes the centre of pressure to move rearwards to approximately 30% chord.

1-21 Fig 25 Pressure Pattern at Fig 21 Point B



62. The shape of the shockwave near the surface (Lambda foot) is caused by the pressure rise behind the shockwave being communicated forward via the subsonic part of the boundary layer. This effectively thickens the boundary layer, presenting a compressive corner to the flow. A series of oblique shockwaves appear which merge into the main normal shockwave. The boundary layer separates from the base of the shockwave causing a thick turbulent wake.

63. Summarizing the situation at M_{FS} = 0.81, the following conditions exist:

a. There is a shockwave at about 70% chord.

b. C_L has risen to 100% of low subsonic value.

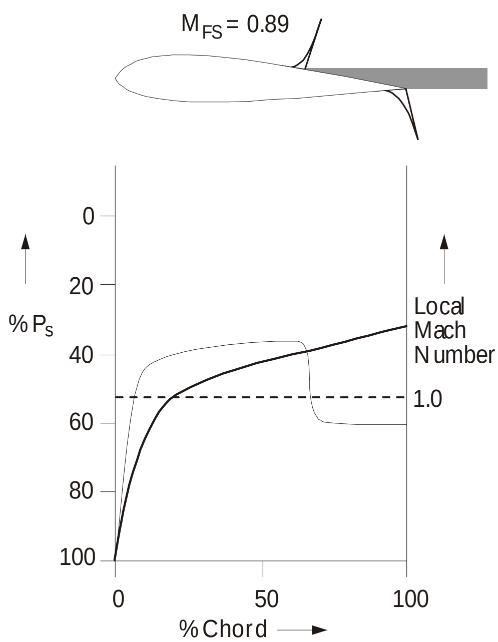
c. The centre of pressure is about 30% chord.

d. The flow under the bottom surface is just sonic.

64. Between M_{FS} = 0.81 and M_{FS} = 0.89, a shockwave has formed on the under surface and moved to the trailing edge while the upper surface shockwave has remained virtually stationary (see Fig 26). The reason for the differing behaviour of the shockwaves is the effect that each has on the boundary layer. On the relatively thick wing under consideration the top shockwave has a large pressure rise across it, producing a very strong adverse pressure gradient in this region. Since part of the boundary layer is subsonic, the increased pressure is felt ahead of the shockwave and the adverse pressure gradient is strong enough to cause a marked thickening and separation of the boundary layer. The thickening of the boundary layer gives rise to a compressive corner which tends to anchor the shockwave to this point on the wing. On the under surface the acceleration is much less because of the angle of attack of its section; this results in a weaker shockwave. The pressure rise across it is insufficient to cause significant boundary layer separation and the shockwave moves quickly to the trailing edge. Such an arrangement of shockwaves leads to the pressure distribution as shown, where it can be seen that the wing behind the upper shockwave is producing negative lift. In computing lift, and thus C_L, this negative area has to be subtracted from the positive lift-producing area. C_L has reduced to a value 30% below its incompressible value and the centre of pressure has moved forward to approximately 15% chord.

65. Referring back to Fig 21, the slope of the curve between points B and C is governed by the relative movement between the upper and lower shockwaves. The lower shockwave moves faster and, from the time it moves aft of the upper one, a small area of negative lift develops and the C_L starts to decrease. This situation progresses until the lower shockwave reaches the trailing edge, by which time the negative lift area is at its maximum and the C_L at its minimum. With any further increase in M_{FS} the upper shockwave moves aft and thereby reduces the negative lift area.

1-21 Fig 26 Pressure Pattern at Fig 21 Point C



66. Summarizing the situation at M_{FS} = 0.89, the following conditions exist:

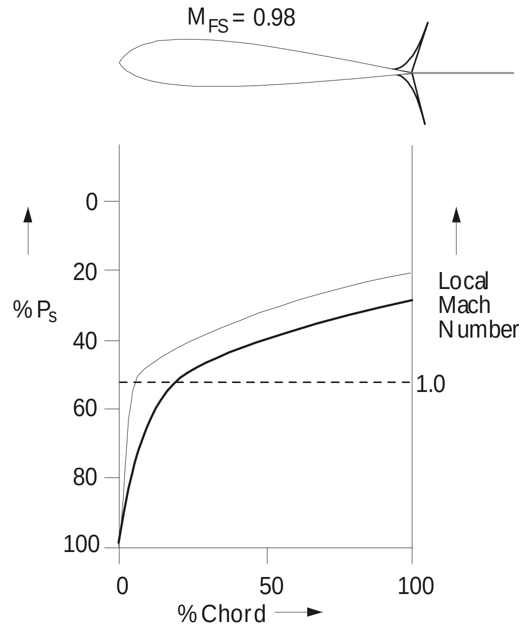
a. The bottom shockwave is at the trailing edge.

b. C_L has reduced to about 70% of its low subsonic value.

c. The centre of pressure is about 15% chord.

67. At point D in Fig 21 (see Fig 27) the M_{FS} is 0.98. Eventually the increase in M_L will overcome the resistance to rearward travel and the top surface shockwave will be forced back to the trailing edge (shown as M_{FS} = 0.98 in Fig 27). The area of negative lift has been replaced by an orthodox pressure differential over the entire chord of the wing and the turbulent wake has been reduced to its low subsonic thickness.

1-21 Fig 27 Pressure Pattern at Fig 21 Point D



68. Summarizing the situation at M_{FS} = 0.98, the following conditions exist:

a. Both shockwaves are at the trailing edge.

b. The turbulent wake is at its low speed thickness.

c. C_L is about 10% above its basic value.

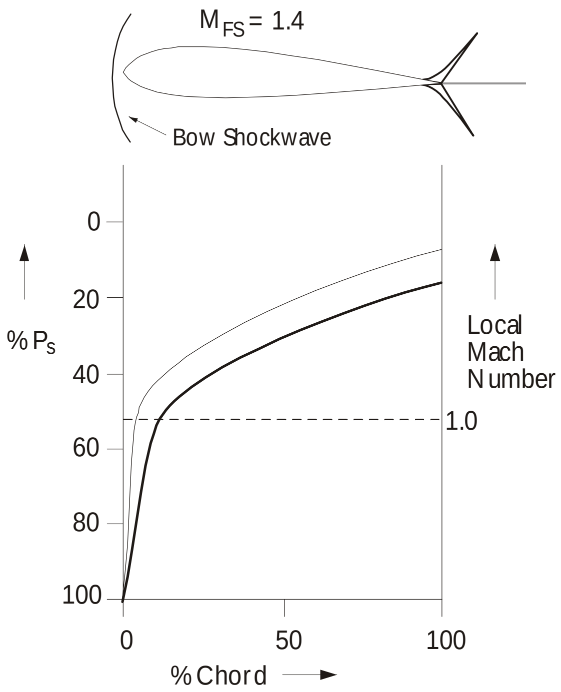
d. The centre of pressure has moved back to about 45% chord. This is the rearward movement of the centre of pressure experienced by all aircraft going through the transonic range.

69. At Fig 21 point E (see Fig 28), the M_{FS} is 1.4. Above M = 1.0 the bow shockwave forms and at M = 1.4 is almost attached to the leading edge (had the leading edge been very sharp, the bow wave would have been attached at M_{DET}). The whole of the wing is producing lift and the centre of pressure is in the approximate mid-chord position. C_L has reduced to a value of 30% less than its incompressible value. The reasons for this are:

a. The loss of circulation due to the stagnation point moving to the most forward point on the leading edge (see para 52).

b. There has been a loss of pressure energy through the bow shockwave.

1-21 Fig 28 Pressure Pattern at Fig 21 Point E



70. Summarizing the situation at M_{FS} = 1.4, the following conditions exist:

a. C_L is 70% of its low subsonic value.

b. The centre of pressure is about 50% chord.

71. The effect of changing angle of attack on shockwaves at constant M_{FS} is now considered. An increase in angle of attack will cause the minimum pressure point to move forward along the top surface; lower pressures and a higher M_L will occur for a given M_{FS}. The effects are as follows:

a. The top surface shockwave forms at a lower M_{FS} and moves forward as angle of attack is increased.

b. Separation affects a greater length of chord.

c. The influence of the shockwave on the boundary layer is more severe. Lower pressures are reached as a result of local acceleration and give higher adverse pressure gradients with the shockwave. This pressure gradient affects the thickness of the boundary layer under the shockwave.

d. The position and intensity of the two shockwaves alter considerably as angle of attack changes. This erratic movement has a large effect on the total lift and drag of the aerofoil and on the centre of pressure position. These changes give rise to transonic stability and control problems in manoeuvre.

The consequences of varying the angle of attack are confined to the transonic range. While the bow shockwave remains attached, there will be little movement of the centre of pressure. The limiting factor when manoeuvring supersonically is usually determined by the amount of tailplane movement available.

Supersonic Fall in C_L

72. In fully supersonic flow (above M_{DET}) for thin wings, regardless of camber, Ackeret's Theory states that $C_L = \frac{4\alpha}{\sqrt{M^2 - 1}}$. Substituting this in the lift formula:

$$\text{lift} = \frac{4\alpha}{\sqrt{M^2 - 1}} \tfrac{1}{2}\rho V^2 S$$

But, we saw in para 8 that $a^2 = \frac{\gamma P}{\rho}$. Therefore, by transposition, $\rho = \frac{\gamma P}{a^2}$. If we now substitute this expression for ρ in the lift formula we get:

$$\text{lift} = \frac{4\alpha}{\sqrt{M^2 - 1}} \tfrac{1}{2}\gamma P \frac{V^2}{a^2} S$$

From para 42, $M^2 = \frac{V^2}{a^2}$ which gives:

$$\text{lift} = \frac{4\alpha}{\sqrt{M^2 - 1}} \tfrac{1}{2}\gamma P M^2 S$$

$$\text{re-arranging, lift} = (2\alpha P \gamma S)\left(\frac{M^2}{\sqrt{M^2 - 1}}\right)$$

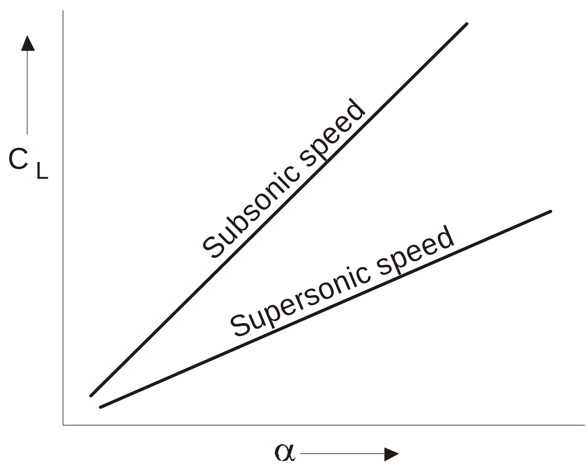
If α is a constant, then

$$\text{lift} = \text{some number} \times \frac{M^2}{\sqrt{M^2 - 1}}$$

Lift is now purely a function of $\frac{M^2}{\sqrt{M^2 - 1}}$ and theoretically lift becomes infinite at M = 1.0, decreases to a minimum when M = 1.4 and then slowly increases as M_{FS} is increased. In practice, any decrease in lift between M_{DET} and M_{FS} = 1.4 would be masked by trim changes produced in passing through the transonic range.

73. The practical result of this is that, supersonically, the lift curve slope becomes progressively gentler with an increase in M_{FS} (see Fig 29).

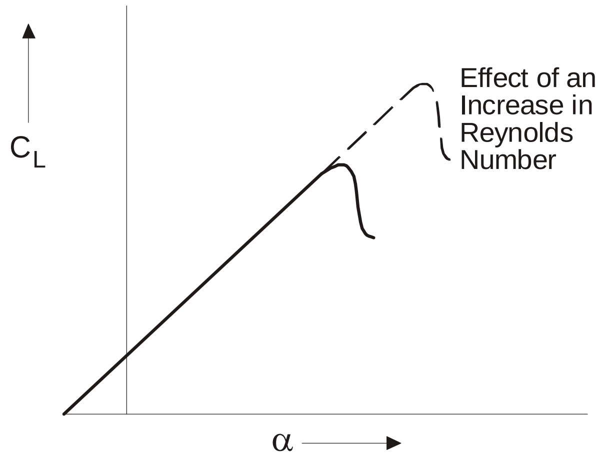
1-21 Fig 29 Slope of C_L Curve



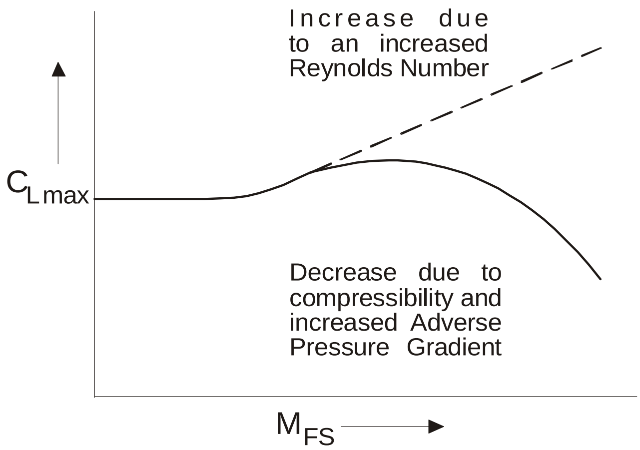
Variations in Maximum Value of C_L

74. In the derivation of the lift formula in Volume 1, Chapter 3, C_L was shown to be dependent on Mach number and Reynolds number (RN): as the aircraft speed increased, so did RN. Therefore with an increase in speed it would appear that the maximum value of C_L would increase (Figs 30 and 31). An increase in flow speed across a wing causes an increase in the adverse pressure gradient, therefore the transition point moves forward. The turbulent boundary layer has more energy in its flow and will therefore remain attached to the wing longer, moving the separation point aft. The wing can, therefore, be taken to a higher angle of attack and a higher value of C_L before it reaches its critical angle of attack. However, with a further increase in speed the Mach number increases and so do compressibility problems.

1-21 Fig 30 Effect of an Increase in Reynolds Number



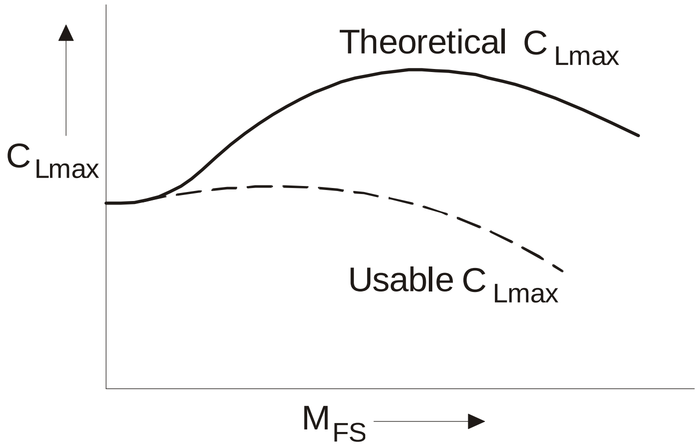
1-21 Fig 31 Theoretical Variation of $C_{L\,max}$



75. Another factor to be considered is that with a further increase in speed the adverse pressure gradient will be even greater, causing the separation point to move forward. This, therefore, lowers the maximum value of C_L.

76. The theoretical result is that there is an increase in $C_{L\,max}$ for an increase in speed until the effects of compressibility and adverse pressure gradient are felt. In practice the value of $C_{L\,max}$ depends on the effectiveness of the aircraft's elevators as to whether the pilot is able to use the increased value of $C_{L\,max}$. Generally, the usable value of $C_{L\,max}$ decays with a speed increase, as shown in Fig 32.

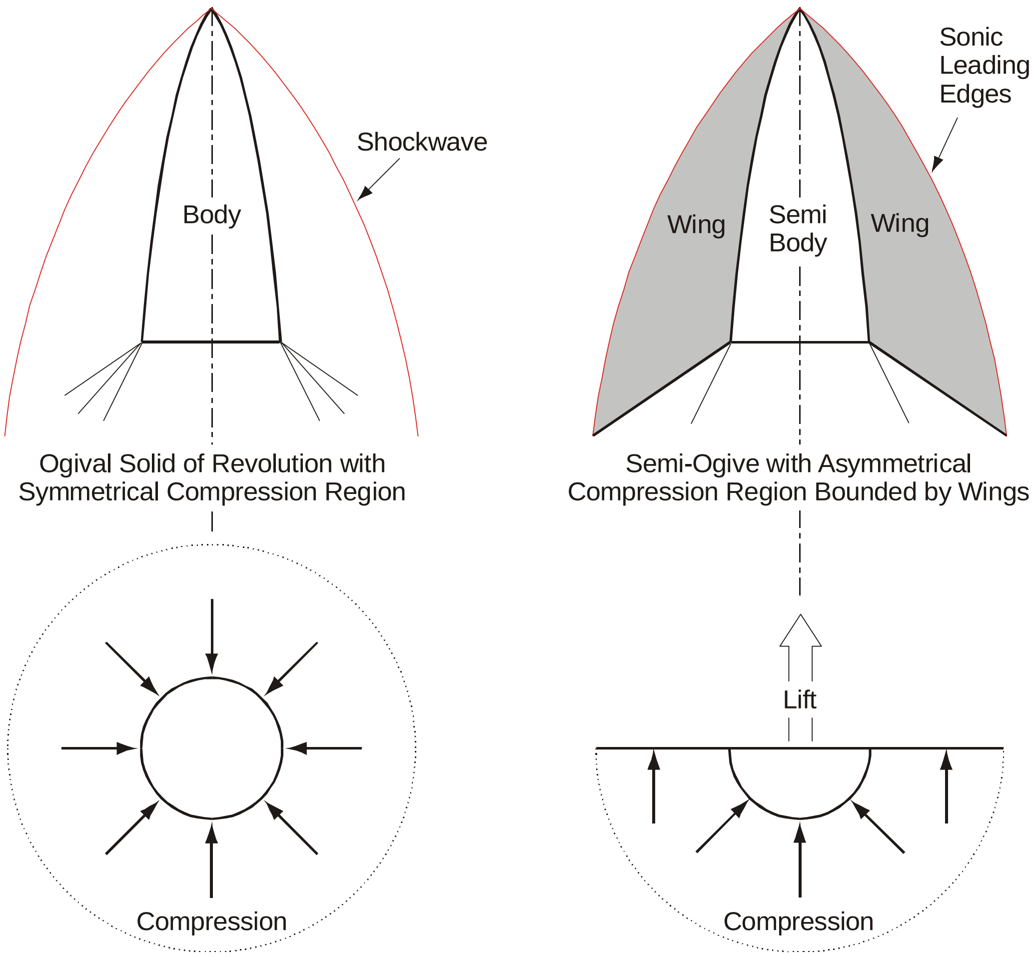
1-21 Fig 32 Practical Variation of $C_{L\ max}$



Compressive Lift

77. Beyond M = 3.0, shockwaves are the predominant feature of the relative airflow and since they cannot be avoided, they may be deliberately cultivated to provide lift. A vehicle designed to utilize compressive lift is called a wave rider and, in its simplest form, is a semi-ogive fitted with wings. The large under-body creates a semi-conical shock and at design M_{FS} the shape of the wings is such that the shock lies along the leading edge. Fig 33 shows that the under-surfaces experience a high pressure, thus providing lift.

1-21 Fig 33 Compressive Lift

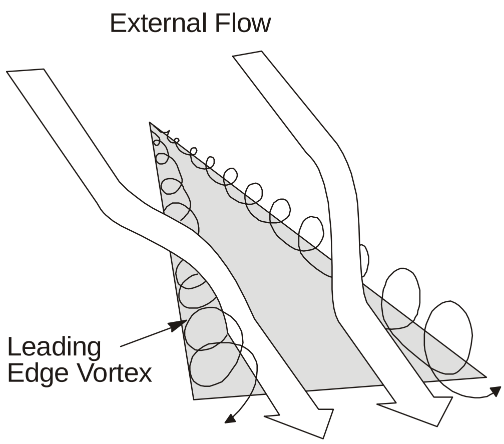


Vortex Lift

78. Due to the spillage of air around the wing tip caused by the mixing of the relatively high pressure air below the wing and the relatively low pressure air above the wing, a vortex is formed. The strength of this vortex varies as the angle of attack. Normally this vortex is left behind but on certain aircraft it is used to produce lift.

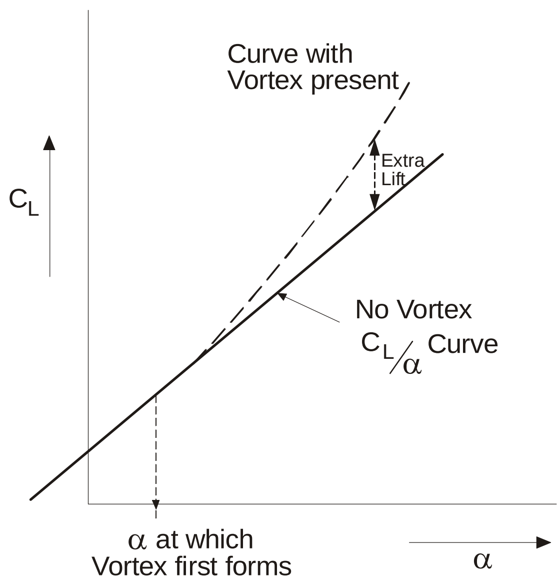
79. On slender delta aircraft at subsonic speeds the vortex is arranged so that it lies on top of the wing and has its origin at or near the wing root. As the core of a vortex is a region of low pressure, there is a vortex core lying along the top of the slender delta wing (Fig 34) creating a greater pressure change between the top and bottom surface of the wing than there would be if the vortices were attached to the wing tips. The vortex system also delays separation from the rear of the wing.

1-21 Fig 34 Slender Delta Aerofoil



80. The stalling angle for this type of aerofoil can be very large (40° or more) but the lift is accompanied by very high drag and a low lift/drag ratio. Also, any irregularity in the vortex formation will cause adverse stability problems. However, in moderate angles of attack the value of C is increased as shown in Fig 35.

1-21 Fig 35 Effect of Vortices on C_L



THE EFFECTS OF COMPRESSIBILITY ON DRAG

Introduction

81. As the speed of an aircraft increases, some factors affecting drag assume greater significance and some arise due to shockwaves and compressibility. Those that assume greater significance are:

a. Interference drag.

b. Trim drag.

These can be reduced to a minimum or virtually eliminated by aircraft design. However, the drag that arises due to the formation of shockwaves is called wave drag and although this cannot be eliminated it can be minimized.

Wave Drag

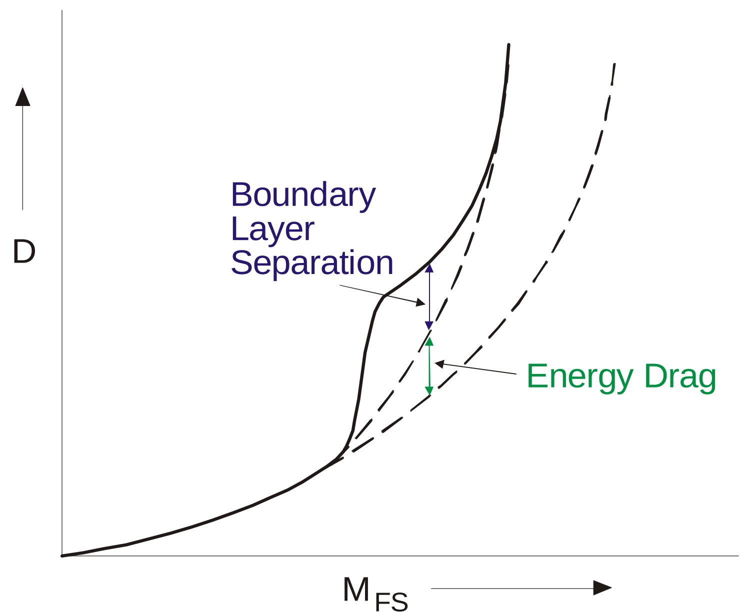
82. **Sources of Wave Drag**. Wave drag arises from two sources: energy drag and boundary layer separation.

83. **Energy Drag**. Energy drag stems from the irreversible nature of the changes which occur as a flow crosses a shockwave. Energy has to be used to provide the temperature rise across the shockwave and this energy loss is drag on the aircraft. The more oblique the shockwaves, the less energy they absorb, but because they become more extensive laterally and affect more air, the energy drag rises progressively as M_{FS} increases.

84. **Boundary Layer Separation**. In certain stages of shockwave movement there is considerable flow separation, as shown in Figs 25 and 26. This turbulence represents energy lost to the flow and contributes to the drag. As M_{FS} increases through the transonic range the shockwaves move to the trailing edge and the separation decreases; hence the drag decreases.

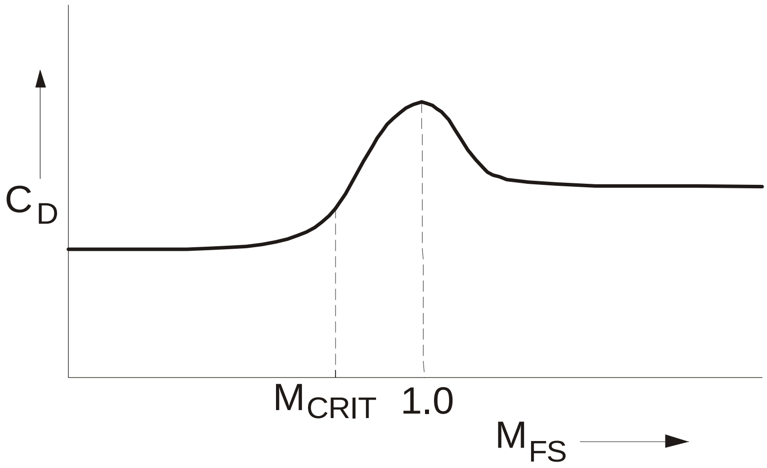
85. **Effect of Wave Drag**. Taken together, these low sources of drag modify the drag curve as shown in Fig 36.

1-21 Fig 36 Effect of Wave Drag



The change in drag characteristics is also shown by the C_D curve for a section at a constant angle of attack (Fig 37).

1-21 Fig 37 Variation of C_D with Mach Number at a Constant Angle of Attack



The hump in the curve in Fig 37 is caused by:

a. The drag directly associated with the trailing edge shockwaves arising from the non-isentropic compression process (energy loss) (see para 42).

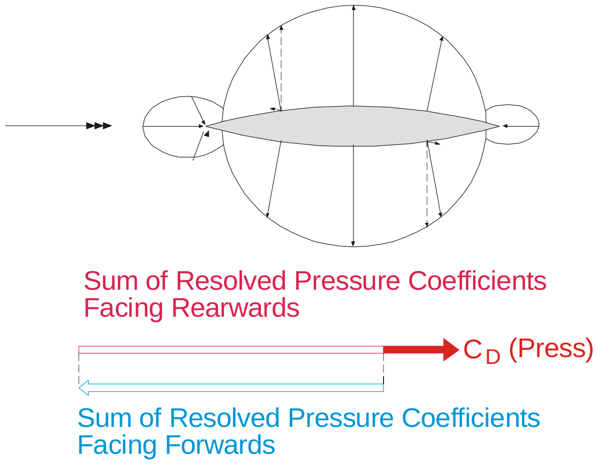
b. Separation of the boundary layer.

c. The formation of the bow shockwave above M = 1.0.

86. **Analysis of Drag Due to Shockwaves**. The mechanics of how a shockwave creates a drag force on a wing can be seen from Figs 38 to 40, where a symmetrical bi-convex is shown at three speeds. (A pressure coefficient is a non-dimensional method of showing pressure acting at a point. Its vectorial quantity is $\frac{P - P_0}{q}$ and it is drawn at right angles to the surface. A positive coefficient indicates a pressure greater than static and points towards the surface.)

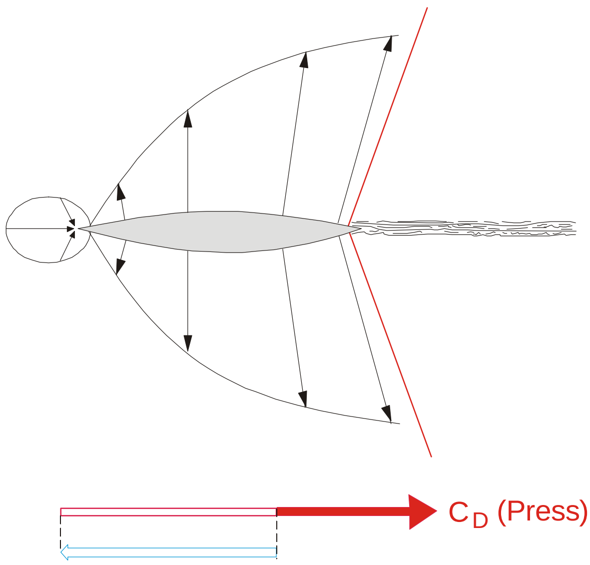
a. **Below M_{CRIT}.** Below M_{CRIT} the pressure coefficients can be split into two components relative to the flight path. The sum of all rearward-acting components minus all the forward-acting components gives a measure of the pressure drag. At low subsonic speeds a rough rule of thumb gives the percentage of the zero-lift drag, which is pressure drag, as being equal to the thickness chord ratio expressed as a percentage, e.g. for 10% t/c wing the pressure drag would be 10% of the zero lift drag, the remaining 90% being surface friction drag. Compressibility only affects pressure drag and, as with C_L, the pressure drag coefficient can be found by applying Glauert's factor, but the overall effect on C_D up to M_{CRIT} is very small. The important point to note from Fig 38 is the pressure recovery which is taking place between the point of maximum thickness and the trailing edge, i.e. flow velocity is decreasing and pressure increasing. Fig 24 also shows this occurring.

1-21 Fig 38 Subsonic Pressure Coefficients



b. **Just below M_{FS} = 1.0**. Fig 39 shows the pressure distribution at M_{FS} just below M = 1.0.

1-21 Fig 39 Transonic Pressure Coefficients



Again, summing the horizontal components of the pressure coefficients will give a measure of the pressure drag. A comparison of Figs 38 and 39 shows an obvious increase in the drag coefficient. The points to note are:

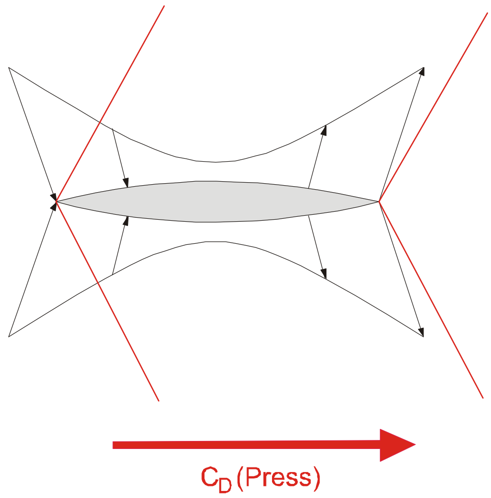
(1) Instead of decelerating after the point of maximum thickness, the flow continues to accelerate supersonically until it reaches the shockwaves, thus producing lower pressures on the rearward-facing surfaces.

(2) These lower pressures will increase the drag force on the wing and it is most important to realise that there is a direct relationship between the pressures ahead of the shockwave and the strength of the shockwave itself. The stronger the shockwave, the greater the energy loss, but also the lower the pressure ahead of the shockwave. This is how a shockwave exerts a drag force on a body.

c. **Above M_{DET}.** Through the leading edge shockwave, the flow is decelerated and deflected such that behind it the flow is tangential to the surface and still supersonic (Fig 40). The flow now accelerates all the way to the trailing edge and the pressure change is such that up to mid-chord the wing surface experiences pressures greater than atmospheric and from 50%-100% chord the pressures are less than atmospheric. Therefore, all the wing surface is experiencing a drag force (which is not so subsonically). This force is dependent on the strength of the shockwave which in turn determines the energy loss.

Above M_{DET} the C_D settles at approximately 1.5 times its low subsonic value and decreases slowly as M_{FS} is increased.

1-21 Fig 40 Supersonic Pressure Coefficients



87. **Reduction of Wave Drag**. To reduce wave drag the shockwaves must be as weak as possible; therefore, wings must have a sharp leading edge as well as a thin section to keep the deflection angle to a minimum and so produce a weak bow shockwave. The thin wing will have a reduced camber thus lowering the pressure drop over the wing. The adverse pressure gradient across the wing shockwaves will be smaller and the strength of the shockwaves will be reduced. Fuselages can be treated in a similar manner. For a given minimum cross-section, an increase in length, within reason, will reduce the wave drag.

Interference Drag

88. Where the flows over two or more parts of an aircraft meet, and these flows are at dissimilar speeds, there is a certain amount of mixing. The faster flows slow down, the slower flows speed up and this all takes place in an area of turbulence; this is interference drag.

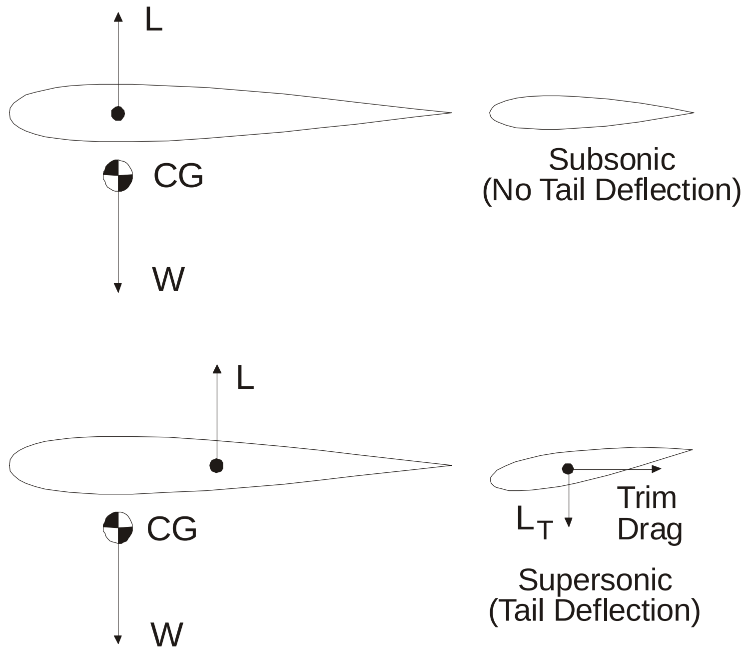
89. High speed aircraft tend to have low aspect ratio wings, and to produce lift the flow over these wings is usually much faster than the flow over other parts of the aircraft when compared with the flow speed of slower aircraft. When these two flows meet - usually at the wing root/fuselage junction - then due to the speed gradient of the two flows, there can be a lot of interference drag produced.

90. If the aircraft is flying at a speed where shockwaves are forming with their associated increase in drag, the combination of wave drag, unstable wave formation at the wing/fuselage junction and the increased interference drag could have a marked effect on the performance of the aircraft. On a poorly-designed aircraft interference drag is by far the largest component of total drag around $M_{FS} = 1.0$.

Trim Drag

91. Trim drag is that increment in drag resulting from the aerodynamic trimming of an aircraft. Unless the thrust, drag, lift and weight forces are so arranged that there is no resultant pitching moment, an aircraft will require a trimming force arranged about the centre of gravity so as to provide the necessary balancing moment to maintain equilibrium. In producing this force an aircraft experiences an increase in drag. This extra drag, trim drag, is not new but has only recently achieved prominence because of the large change in the position of the centre of pressure which takes place through the transonic range, necessitating extremely large trimming forces (Fig 41).

1-21 Fig 41 Balancing the Shift of the Centre of Pressure



92. If equilibrium is to be achieved, all forces must balance, ie the horizontal forces (thrust and drag) must be equal, and so must the vertical forces. Since L_T is acting downwards, in order to maintain level flight, lift must be made equal to W + L_T. To produce this extra lift a higher angle of attack is required which, in turn, will mean more drag.

93. Trim drag then has two sources, and it will readily be seen that it can be present to some degree at any speed. Subsonically, because the movement of the centre of pressure with speed is relatively small, trim drag can be minimized by suitable design. Consider a supersonic aircraft of conventional design with static longitudinal stability subsonically. As it accelerates through the transonic range there will be a steady rearwards movement of the centre of pressure. This gives a strong nose-down pitching moment requiring very large trimming forces and resulting in a large increase in drag. It is because of this rearward shift of the centre of pressure associated with supersonic flight that trim drag has become such a problem.

94. With careful design this rearward shift of the centre of pressure can be minimized but not, as far as is known at the moment, eliminated. For example, the Concord design restricts the movement of the CP to something like 8% of the chord. In terms of feet, this is still a considerable distance, and so the CG is also moved rearwards as the aircraft accelerates, by pumping fuel aft. However, this is only one method; the subject is discussed more fully in Volume 1, Chapter 22.

THE EFFECTS OF INCREASING MACH NUMBER ON STABILITY

Introduction

95. Certain aspects of stability have already been discussed; the purpose of the following paragraphs is to cover, very briefly, the main effects of compressibility on an aircraft's stability.

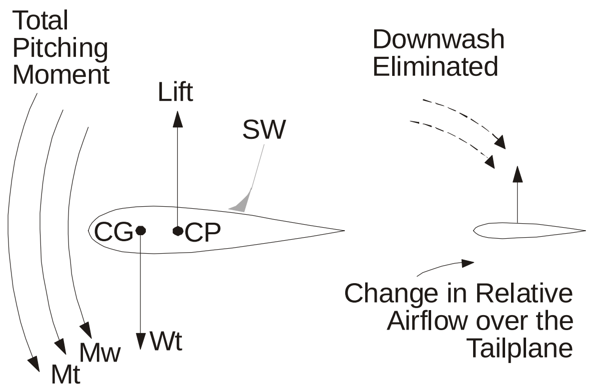
Transonic Longitudinal Stability

96. Most aircraft which operate in the transonic speed range experience a nose-down pitch with an increase in speed. Although there are two causes of this, the relative importance of each cannot be stated without reference to specific aircraft. The two causes are:

a. Rearward movement of the centre of pressure which increases the longitudinal stability. Because of the supersonic acceleration at the higher speed, pressure continues to decrease past the 50% chord point, thus increasing the amount of lift produced by the mid-chord part of the wing.

b. Modification to the airflow over the tailplane (Fig 42). Most tailplanes work in a region of downwash from the mainplane; in the case of swept-wing aircraft the downwash can be considerable. The formation of shockwaves on the mainplane modifies the flow such that the downwash is reduced; this will pitch the aircraft nose-down.

1-21 Fig 42 Elimination of Downwash on the Tailplane



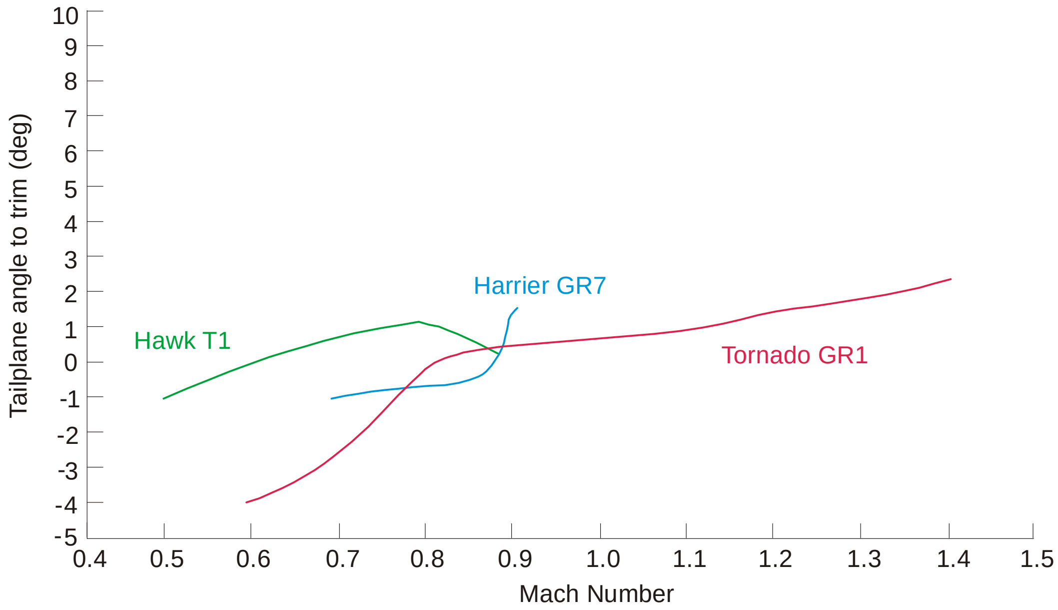
97. The effects on an aircraft's handling characteristics of nose-down pitch with an increase in speed are twofold:

a. At some Mach number, an aircraft will become unstable with respect to speed, ie an increase in Mach number will necessitate a rearward movement of the control column. This is potentially dangerous since, if the pilot's attention is diverted, a small increase in Mach number will give a nose-down pitch which will give a further increase in Mach number. This, in turn, leads to a further increase in the nose-down pitching moment. The designer's answer to this problem is to ensure positive stability by use of a Mach trimmer. This is a device, sensitive to Mach number, which simply deflects the elevator/tailplane by an amount greater than that just to compensate for the trim change. This ensures that the aircraft has positive stability and that the pilot will have to trim nose-down as speed increases.

b. The requirement for a large up-deflection of the elevator/tailplane just to maintain straight and level flight reduces the amount of control deflection available for manoeuvring. This is especially significant on tailless deltas and in one instance is the deciding factor in limiting the maximum speed.

98. The graph in Fig 43 illustrates the magnitude of trim changes experienced in the transonic speed range. With reference to the curve for the Harrier GR 7, a marked tailplane trim change occurs when accelerating through M = 0.9. There is a smooth reversal of trim direction above M = 0.8 in the Hawk. Note that, in the Tornado, transition problems have been almost designed out throughout the range - the largest trim change occurring in the region M = 0.8.

1-21 Fig 43 Trim Changes with Mach Number



Supersonic Longitudinal Stability

99. The rearwards movement of the centre of pressure in the transonic region continues as the aircraft accelerates into fully supersonic flight. Thus, all aircraft experience a marked increase in longitudinal stability with a consequent increase in the size of trimming forces and reduction in their manoeuvring capability. The adverse effect of having to produce large trimming forces has already been covered.

100. There are several possible methods of reducing either the rearward movement, or the effects of rearward movement, of the centre of pressure; these are discussed in Volume 1, Chapter 22.

Transonic Lateral Stability

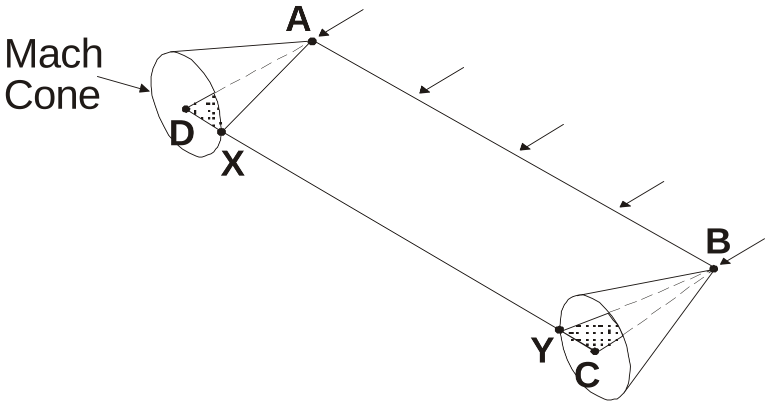
101. Disturbances in the rolling plane are often experienced in transonic flight; on certain aircraft one wing starts to drop when M_{CRIT} is exceeded. This tendency occurs due to the difference in lift from the two wings because in most aircraft the shockwaves do not form at identical Mach numbers and positions on each wing.

102. Design features which normally provide lateral stability may possibly have the reverse effect and aggravate the wing-drop. In a sideslip the down-going wing gains lift from a greater effective aspect ratio (sweepback) or a greater angle of attack (dihedral); each method produces a greater local acceleration of the flow. This higher local Mach number may promote or intensify an already existing shockwave on the lower wing, possibly causing the wing to drop further.

Supersonic Lateral Stability

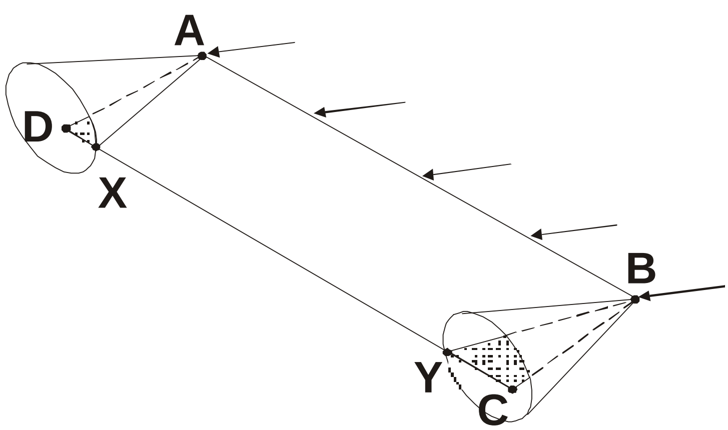
103. Lateral stability depends, after sideslip, on the lower wing developing lift. CL decreases in supersonic flight and thus the correcting force is reduced; dihedral and sweepback are consequently less effective. An additional effect is also worth considering; Fig 44 illustrates a rectangular wing at supersonic speed. There will be three-dimensional flow only within the Mach cones originating at A and B, ie the Mach cones are bounded by the Mach lines and air flowing through these Mach lines will be aware of, and respond to, the pressure difference from lower to upper surface. In the triangles ADX and BCY the average pressure is approximately half that of the two-dimensional flow in area ABYX and so the lifting efficiency is reduced. So is the drag, but surface friction drag is not reduced and so the lift/drag ratio is adversely affected. This is the reason why the 'tip triangles' on some wings are cut off, e.g. in some missiles.

1-21 Fig 44 Mach Cones on a Supersonic Wing



104. Consider now the flow pattern with the wing sideslipping. Fig 45 shows a wing-drop to starboard. Because of the changed relative airflow, the Mach cones affect different areas of the wing, the lower wing is producing much less lift and, all other things being equal, a strong destabilizing rolling moment is set up. Again, there is a requirement for features giving lateral stability, usually in the form of an autostabilization system.

1-21 Fig 45 Mach Cones Displaced by Sideslip

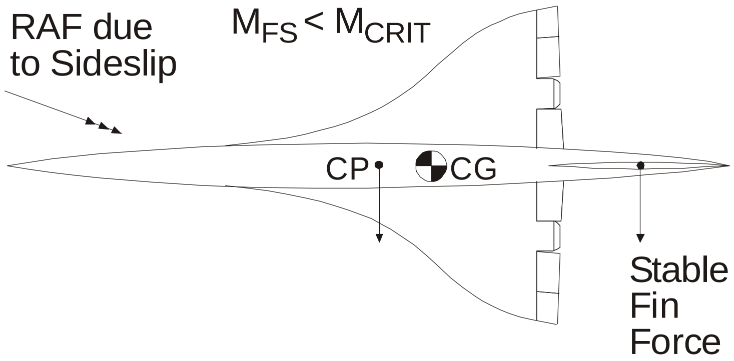


Directional Stability

105. The trend towards rear-mounted engines, and consequently an aft CG, has meant a decreased arm about which the fin can act. Also, the supersonic decrease in C_L for a given angle of attack caused by sideslip means a reduction in fin effectiveness.

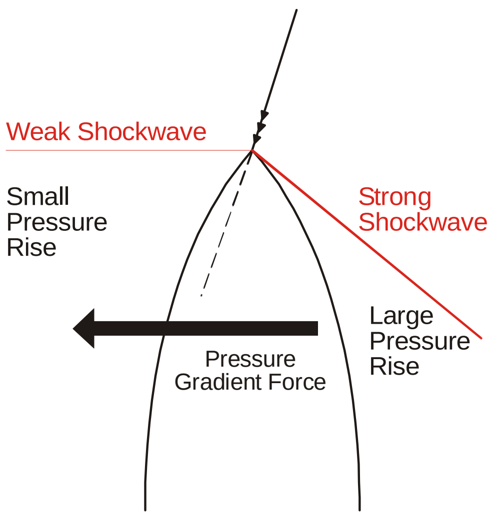
106. Subsonically, the fuselage side-force in a sideslip acts in front of the CG and the vertical tail surfaces are able to overcome this destabilizing situation, as shown in Fig 46. But, in supersonic flight, the fuselage side-force moves forward. As long as the aircraft is in balanced flight there is no problem, but if the relative airflow is off the longitudinal axis a destabilizing force at the nose results. This is caused by asymmetry in the strength of the two shockwaves which produces a pressure gradient across the nose. A strong shockwave has a greater pressure rise across it compared to a weak shockwave.

1-21 Fig 46 Subsonic Directional Stability

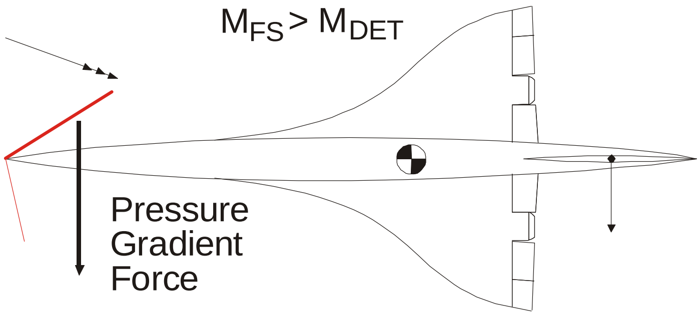


107. The nose force illustrated in Fig 47 is tending to prevent the nose being turned into the relative airflow and is therefore destabilizing. This force increases with speed and has a longer moment arm than the fin. Thus, above about M = 2.0, the moment becomes greater than that of the fin and rudder and the aircraft becomes directionally unstable (Fig 48). The point of application of this force is difficult to define but it is located at that part of the fuselage where the cross-sectional area is increasing.

1-21 Fig 47 Formation of a Sideways Nose Force



1-21 Fig 48 Instability Above M = 2.0



108. One answer to this problem is to fit longer fins and to increase their number, but obviously there is a limit if only for wave drag considerations. A better method is to fit yaw dampers that sense any out-of-balance condition before the pilot notices it and will correct the situation by automatically applying the correct amount of rudder.

Inertia Cross-coupling

109. One of the problems associated with the shape and distribution of weight in modern aircraft is that of inertial cross-coupling which imposes limitations on manoeuvrability both in combat and in weapon delivery. There are three particular cases which should be clearly understood.

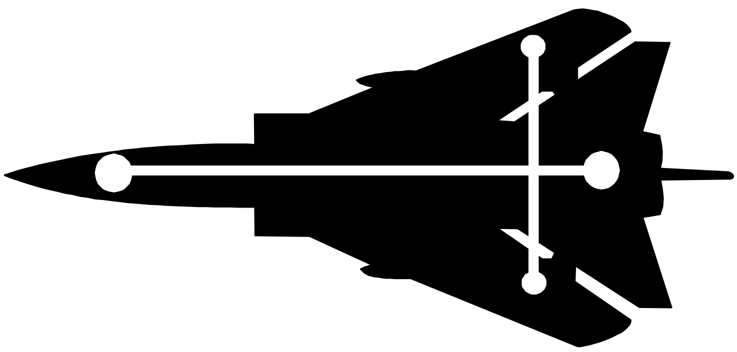
a. Yaw divergence.

b. Pitch divergence.

c. Autorotation.

110. Up to the end of the Second World War aircraft behaviour in manoeuvre was determined almost exclusively by aerodynamic characteristics. In more recent times, however, fuselages have grown longer and heavier in relation to the wings. This distribution of weight can be shown as in Fig 49. The dumb-bells represent the moments of inertia which are a product of the weight and the moment arm. The moment of inertia of the fuselage is usually referred to as B, whilst that of the wings is A. As an example, ignoring the effect of tip/drop tank fuel, the B:A ratio of the Spitfire was 4:3, that of the F-104, 10:1 and the Jaguar 7:1. The Tornado has a ratio of 4:1 when unswept with full weapons load, increasing to 7:1 when fully swept.

1-21 Fig 49 Distribution of Weight in a Supersonic Aircraft

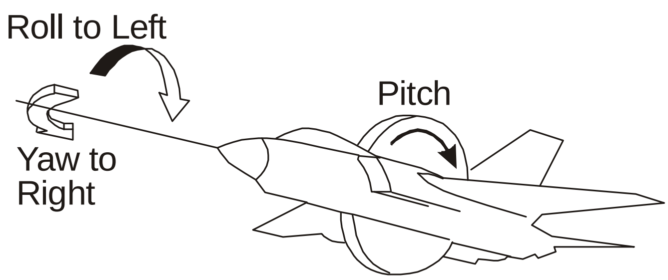


Yaw Divergence

111. Yaw divergence is the most common form of inertial cross-coupling encountered in practice. It results when high rates of roll are applied during pull-up manoeuvres in aircraft with high B:A ratios, good longitudinal static stability and poor directional stability. If the rate of roll is high, then a surprisingly small rate of pitch can be sufficient to start the sequence.

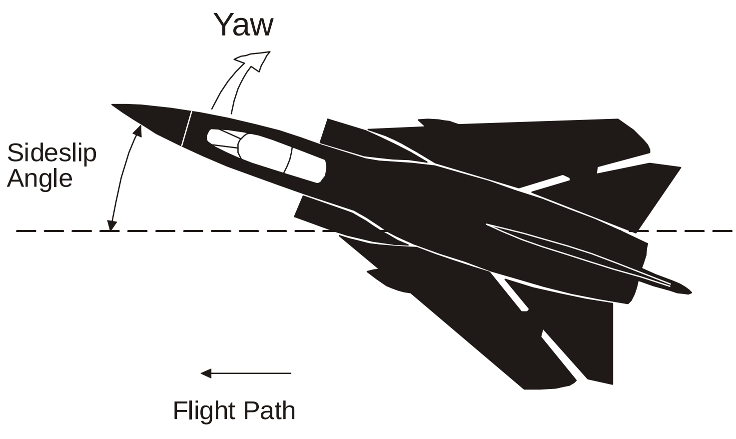
112. A simple and practical explanation of the cycle of events starts with the aircraft being rolled in a pull-up manoeuvre from level flight. Because of its longitudinal moment of inertia, the fuselage could be likened to the wheel of a gyroscope (see Fig 50).

1-21 Fig 50 Explanation of Yaw Divergence



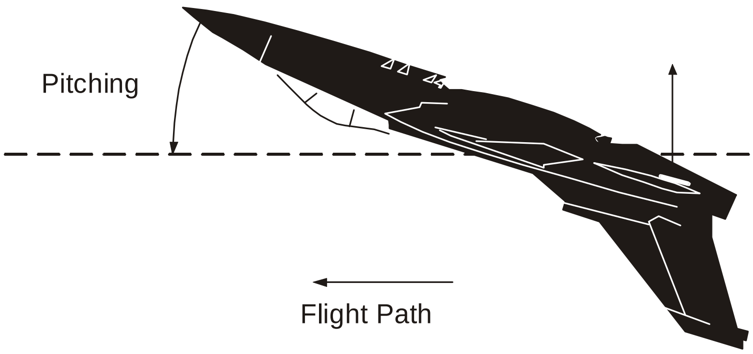
As the aircraft is pitching nose-up, the wheel is rotating clockwise viewed from the port side. If the aircraft is now rolled to the left it has the effect of applying a torque to the top of the gyro which, when translated through 90° by the normal laws of precession, causes the aircraft to yaw to the right. As the basic laws of precession state that the rate of precession is inversely proportional to the rate of rotation of the gyro, if the rate of roll (torque) is high enough only a small amount of pull-up (rotation) will induce a yaw (precession). In fact a badly designed aircraft might diverge at this point, especially if it lacks sufficient directional (or weathercock) stability, or if the situation is aggravated by adverse aileron drag. More often the aircraft will continue rolling but it may have quite a large sideslip angle as it passes through the 90° bank position since the yawing action will dominate the directional stability (Fig 51).

1-21 Fig 51 Yaw Divergence – Roll and Pull-up



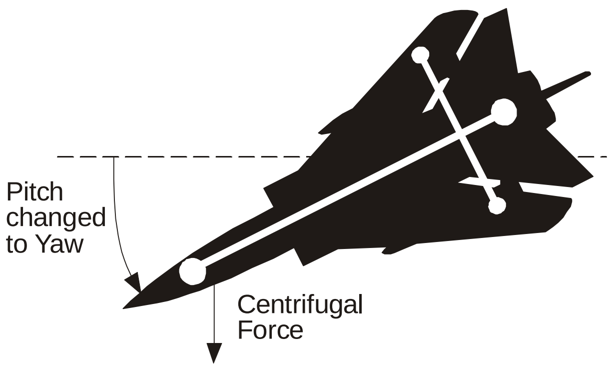
113. As the aircraft reaches the inverted position (Fig 52) the longitudinal stability of the tailplane becomes dominant and produces a pitching moment.

1-21 Fig 52 Yaw Divergence - Pitching



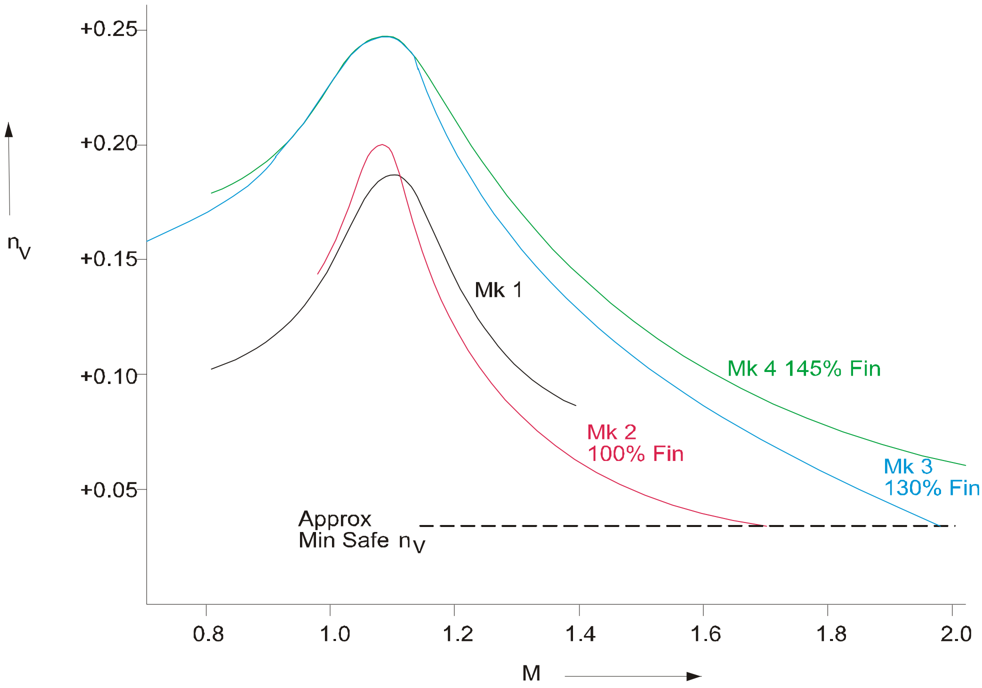
114. Having rolled through 270° the aircraft again reaches the 90° bank position (Fig 53). The pitching moment is translated into a yawing moment, the gyroscopic couple adds to this and the only stabilizing force is the weathercock stability due to the fin. If the aircraft lacks directional stability at this point yaw divergence and structural failure, or a spin, may follow.

1-21 Fig 53 Yaw Divergence – Pitch change to Yaw



115. The most common cause of yaw divergence is a rolling pull-out at high Mach numbers. The pull-out establishes the large angle of attack which translates quickly into sideslip as the aircraft rolls. The high Mach number reduces the effectiveness of the fin. Fig 54 shows four examples of fin stability. The aerodynamic derivative of yaw due to sideslip (ie weathercock stability) is n_V. Increasing values of n_V signify increasing fin stability. These curves are for a typical modern aircraft. The degradation of n_V at both ends of the speed scale shows that divergence in yaw is possible at high supersonic speeds, and also at low speeds. In describing the sequence of events, a full roll was used as an example, however a complete roll is not always necessary by any means. Aircraft are being developed with features designed to reduce the hazard by improving directional stability; the Lightning was given a larger fin; the Phantom had its wing tips cranked upwards; the twin fins of the Foxbat indicated another possible solution. Restricted aileron movement at high speed is another solution.

1-21 Fig 54 Yawing Moment due to Sideslip at 40,000 ft



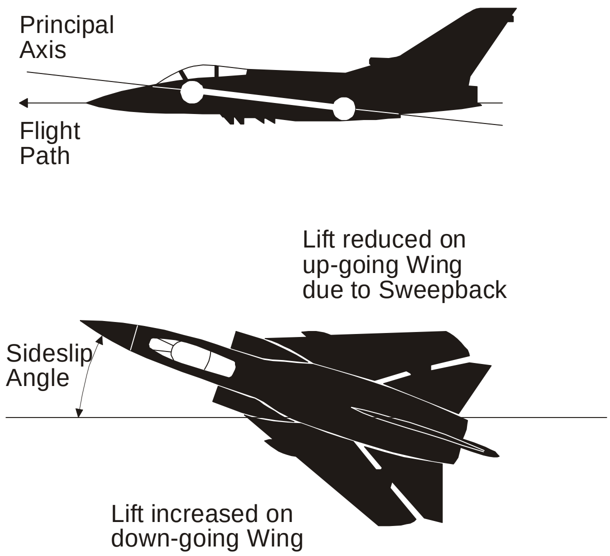
Pitch Divergence

116. The cycle of events which leads to a pitch divergence is exactly the same as yaw divergence except that the aircraft will have good directional stability and poor longitudinal static stability. It is not very common and is normally only encountered at very high altitudes.

Autorotation

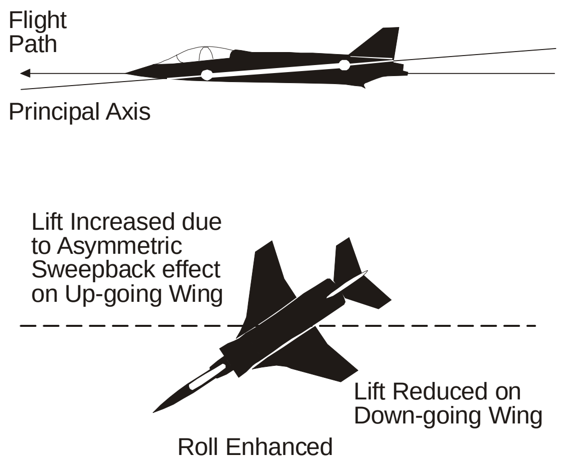
117. The third type of inertial cross-coupling is autorotation. If a normal aircraft flying at moderate air speed rolls through 90° it will tend to roll about its inertial axis, or principal axis, and sideslip along its flight path, as shown in Fig 55.

1-21 Fig 55 Roll Through 90º (Nose-up Principal Axis)



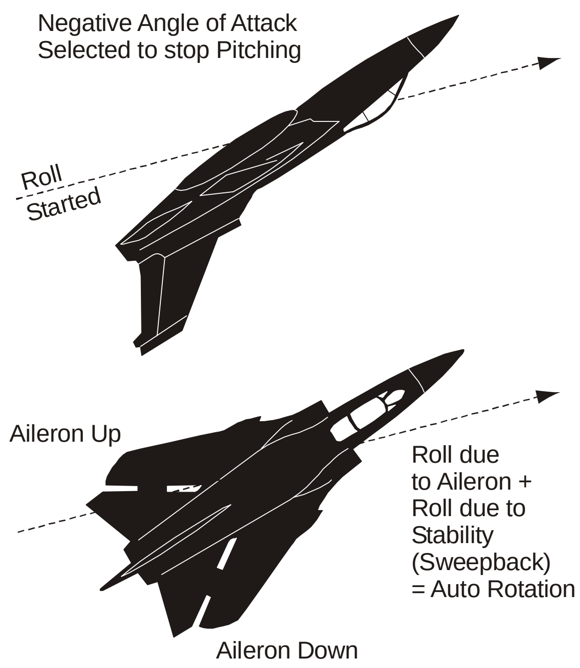
The difference in lift between the upper and lower wings will tend to oppose the roll, i.e. it will be stabilizing in influence. Only a badly designed aircraft would diverge in this situation. On the other hand, if an aircraft with a nose-down principal axis is flying at high speed, or if an aircraft is flying in a negative g condition, its inertial axis may well point below its flight path, in which case if it rolls through 90º, the differential lift from the wings as it passes the 90º position will tend to increase the rate of roll and autorotation may follow, as shown in Fig 56. Autorotation is not often encountered in this way at high speed, but it is quite common when rolling from the inverted position at the top of a loop.

1-21 Fig 56 Autorotation (Nose-down Principal Axis)



118. This problem is only apparent where the manoeuvre has been incorrectly completed by rolling off too early and using large aileron deflection. If an aircraft with a large B:A ratio is rolled off the top of a loop with a negative angle of attack, it will tend to roll about its inertial axis (Fig 57). The enhanced lateral stability adds to the roll due to aileron and autorotation may result. In this condition the aircraft has very little forward speed and a very high rate of roll. Weathercock stability is often poor at low air speed, and the high rates of roll that result from this situation may lead to a yaw divergence and spin. This form of cross-coupling is common to many aircraft.

1-21 Fig 57 Autorotation from Loop



CHAPTER 22 - DESIGN FOR HIGH SPEED FLIGHT

Introduction

1. The aim of this chapter is to give a general insight into the problems which confront the designers of high-speed aircraft and the various methods available for overcoming or reducing them; the great variety of planforms and profiles precludes a more detailed examination. References to many of the design features considered here can be found in the earlier chapters on the basic principles of flight, but some of the relevant sections and diagrams are repeated here for the convenience of the reader.

Design Requirements for High-speed Flight

2. For flight at transonic and supersonic speeds, aircraft must incorporate certain design features to minimize the effects of compressibility; unless the extra drag due to compressibility is kept to a minimum, the maximum speed would be seriously affected. Furthermore, the design features must overcome unacceptable changes in stability or control which may be encountered in transonic and supersonic flight.

3. The design features selected for a particular aircraft depend on its intended role. The following list indicates some of the possible important requirements for current Service aircraft:

a. High critical drag rise Mach number, (M_{CDR}).

b. Low drag coefficient at supersonic speeds.

c. High maximum and low minimum speed.

d. Adequate stability and control throughout the speed range.

e. Good manoeuvrability at high speeds and altitudes.

f. Stable platform for weapon release.

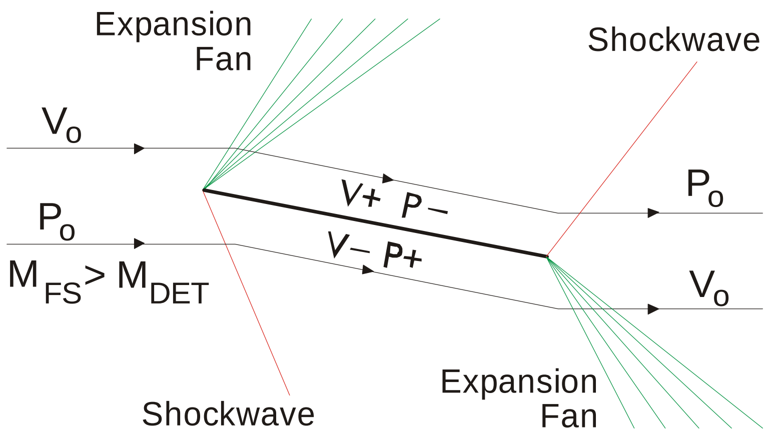
4. The success of an aircraft generally rests on the aerodynamic efficiency of the wings, hence it is with wing design that this chapter is primarily concerned. Since the wing cannot be divorced from the remainder of the airframe, a brief reference is made to the more general design features of the body/wing combinations.

Wing Sections

5. The ultimate choice of wing section rests on both supersonic and subsonic considerations. As these are not fully compatible, the result, in practice, is a compromise. The following consideration of wing sections is developed from the theoretical optimum wing section to the practical design.

6. **The Flat Plate**. A low thickness/chord (t/c) ratio wing is required for high-speed flight, hence the optimum supersonic aerofoil section employs the lowest t/c ratio possible; the flat plate satisfies this requirement. If a flat plate is placed in supersonic flow at a positive angle of attack, the flow pattern shown in Fig 1 is produced.

1-22 Fig 1 The Flat Plate

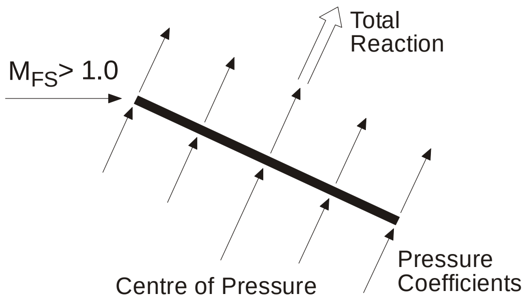


The flat plate in a supersonic flow has two particular characteristics:

a. It has a uniform decrease in pressure over the top surface and a uniform increase in pressure on the under surface, thus producing a uniform pressure differential (Fig 2).

b. The centre of pressure on the plate is in the mid-chord position at all angles of attack (Fig 2).

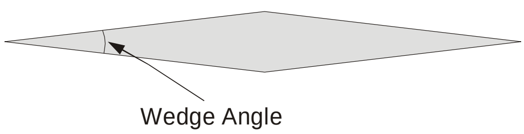
1-22 Fig 2 Pressure Coefficients on a Flat Plate



The flat plate is the most efficient aerofoil section for supersonic flight and is thus suitable for guided missiles. However, the plate is unsuitable for piloted aircraft since there is normally a requirement for some thickness to the wing to accommodate control systems. Wing strength is also necessary to avoid aero-elastic problems, and the physical construction of the wing requires a certain thickness. Therefore some other section must be employed for piloted aircraft.

7. **The Symmetrical Double Wedge**. If a wing must have thickness, then the next optimum shape for supersonic flight is the diamond or symmetrical double wedge, with as low a t/c ratio as possible, as in Fig 3.

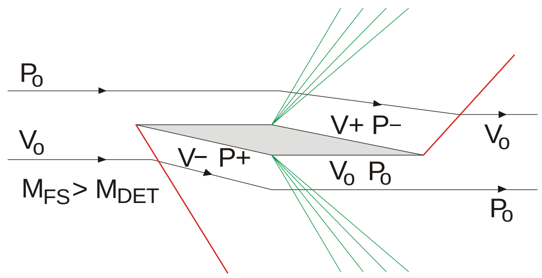
1-22 Fig 3 Double Wedge Aerofoil



The double wedge is advantageous for supersonic flight because some of the shockwaves may be avoided. It has an optimum angle of attack for the best lift/drag ratio, viz half the wedge angle.

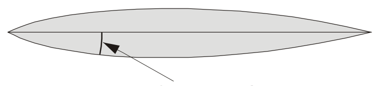
The flow pattern resulting when the double wedge is placed in supersonic flow at the optimum angle of attack is shown in Fig 4.

1-22 Fig 4 Double Wedge Aerofoil at Optimum Angle of Attack



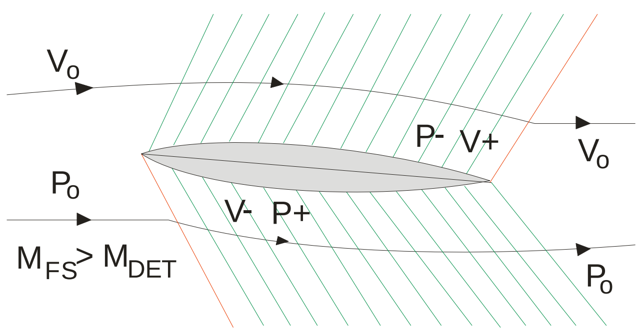
At this optimum angle of attack, only two shockwaves exist;there is an expansion over the top rear surface and a compression under the bottom front surface. The centre of pressure is again in the mid-chord position. If the angle of attack is varied, either two further shockwaves appear, or the existing ones intensify with the drag increasing at a greater rate than the lift. Although the double wedge wing is efficient in supersonic flight, the sharp corners present difficulty in subsonic flight. Piloted aircraft must be able to fly subsonically, if only during take-off and landing and therefore some other section must be employed.

1-22 Fig 5 Bi-convex Aerofoil



8. **The Symmetrical Bi-convex Wing**. The Symmetrical bi-convex aerofoil section in Fig 5 is a compromise between the optimum supersonic and subsonic requirements; it has curves for low speed work, and it has a low t/c ratio for high-speed flight. Large movements of the centre of pressure are avoided and the wing can be made of sufficient strength. As the surfaces are curved, the pressure changes produced are not sudden as in the optimum supersonic sections. In supersonic flight, when the angle of attack is increased to the surface angle, the shockwaves are reduced to a minimum of two as in Fig 6.

1-22 Fig 6 Angle of Attack Equal to Surface Angle



Thickness/Chord Ratio

9. The t/c ratios of modern aircraft wing sections fall into three main categories depending upon the design speed range, ie:

a. **Subsonic**. Subsonic sections have t/c ratios of 15% to 20%, with a point of thickness at approximately 30%, chord.

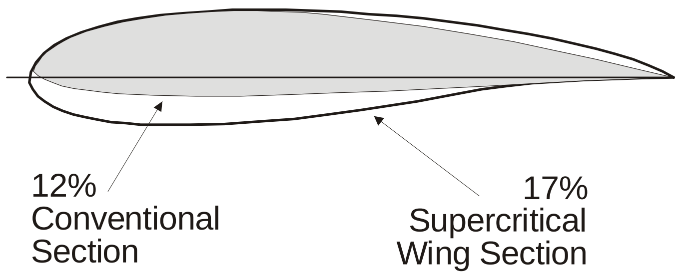
b. **Transonic**. Wings designed for operation within the transonic region, usually have a t/c ratio of 10% or less. However, the supercritical wing (see para 10) has t/c ratios as high as 17%.

c. **Supersonic**. Supersonic wings have t/c ratios of 5% or less. Additionally, the section should have a sharp leading edge to reduce M_{DET} and the bow shockwave intensity. It is of interest to note that the trailing edge of a high-speed wing can be thick and blunt. This effectively reduces the t/c ratio, as the wing behaves as a thinner section with the rear part chopped off.

The Supercritical Wing

10. The supercritical wing differs from the conventional aerofoil section in having a relatively flat upper surface, and a more pronounced curvature on the lower surface. Because the airflow does not achieve the same increase of speed over the flattened upper surface as with a conventional wing, the formation of shockwaves is delayed until a higher M_{FS} with a consequent increase in M_{CDR}. There is also a significant reduction in the intensity of the shockwaves when they do form. Some of the lift lost by the flattened upper surface of the supercritical wing is recovered by a pronounced reflex camber at the trailing edge (Fig 7).

1-22 Fig 7 Supercritical Wing



11. Other obvious features of the 17% t/c ratio wing are the large section depth and the relatively blunt, rounded leading edge. The section depth has advantages of increased storage space for fuel, undercarriage, etc, and also offers the possibility of reducing the number of externally carried stores with the benefit of drag reductions. Another advantage of the deep section is that a high wing strength can be combined with a light structure. The rounded leading edge should prevent leading edge separation and so provide reasonable low speed handling characteristics.

12. This type of wing could be used to increase performance in one of two ways as described below:

a. **Increased Payload**. By using present day cruising speeds, the fuel consumption would be reduced, thus allowing an increase in payload with little or no drag increase over a conventional wing at the same speed.

b. **Increased Cruising Speed**. By retaining present day payloads and adopting a thinner section supercritical wing, the cruise Mach number could be increased with little or no increase in drag.

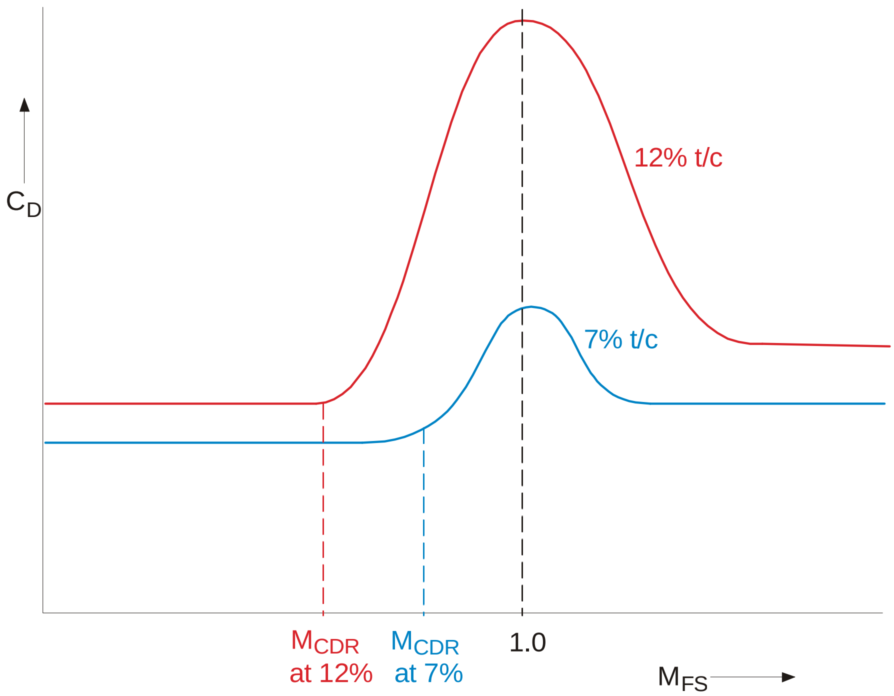
To make full advantage of the benefits by the supercritical wing, it must be embodied in a supercritical design, i.e. the wing/body combination must also be supercritical. This implies accurate area-ruling, and the elimination of any high speed points on the airframe, e.g. cockpit canopies, etc.

Low Thickness/Chord Ratio Wings

13. On a low t/c wing, the flow acceleration is reduced, thus raising the value of M_{CRIT} for the wing, eg if M_{CRIT} for a 15% t/c wing is M 0.75, then M_{CRIT} for a 5% t/c wing will be approximately M 0.85. Not only is the shockwave formation delayed by the thinner wing, but also the intensity of the waves when they do form is correspondingly less. This reduces the drag caused by energy losses through the shockwaves, and also reduces the boundary layer separation drag because the weaker adverse pressure gradient across the wave may be insufficient to cause separation; any separation that does occur will cause a less severe drag rise, as the shockwaves move rapidly to the trailing edge. To prevent the wing shockwave from forming simultaneously along the whole length of the wing, the t/c ratio must reduce from root to tip. The aim is to achieve the first shockwave formation at the wing root.

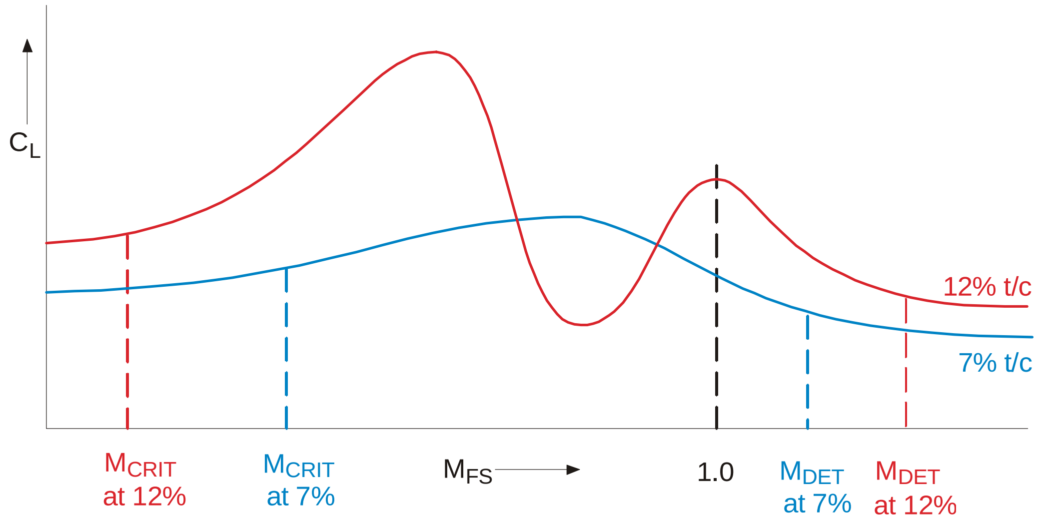
14. The graph at Fig 8 compares the C_D of a 7% t/c ratio straight wing with that for a 12% t/c ratio straight wing at increasing Mach number. It can be seen that the C_D peak of the thinner wing is only one third the value of that for the thicker wing, and also the M_{CDR} of the former is increased. As the shockwaves are less intense and move faster to the wing trailing edge, the shock stall is avoided.

1-22 Fig 8 Effect of Thickness/Chord Ratio on C_D



15. A similar comparison of C_L for the two t/c ratios is given in Fig 9, and this shows that the fluctuations in C_L are smoother and of decreased amplitude on the thinner wing. This has the effect of giving the CP a more predictable rearward movement which, in turn, improves the stability of the aircraft over the transonic speed band. Another important advantage can be seen from Fig 9, ie the transonic speed band is reduced for the thinner wing; M_{CRIT} is increased, and M_{DET} reduced.

1-22 Fig 9 Effect of Thickness/Chord Ratio on C_L



16. The use of a low t/c ratio wing section incurs some disadvantages:

a. The values of C_L at low subsonic speeds are reduced (see Fig 9), thus causing problems at approach speeds. For a given wing loading, the stalling speed increases as the thickness/chord ratio is reduced.

b. On thin wings, particularly when swept, it is difficult to avoid aero-elastic problems due to wing flexing and twisting.

c. Limited stowage space is available in thin wings for high lift devices, fuel, undercarriage, etc.

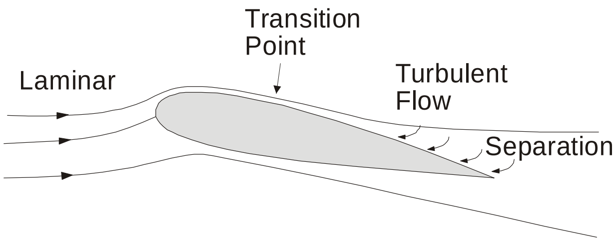
d. Leading edge separation is present on thin wings even at low angles of attack, particularly when the leading edge is sharp. In some configurations, however, this separation is controlled and can be made to increase the lift by the subsequent vortex formation.

Leading Edge Separation

17. The adoption of the thinner wing and smaller leading edge radius has resulted in a type of stall which has assumed great importance in recent years; this is the stall which follows flow separation from the leading edge. This may happen with a 12% t/c ratio wing, but will almost certainly occur on an 8% t/c ratio wing.

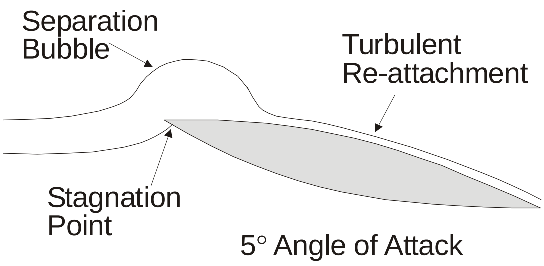
18. On the subsonic type of aerofoil, the classic stall starts with separation of the turbulent boundary layer near the trailing edge (Fig 10). The stall is a comparatively gentle process with buffeting providing plenty of pre-stall warning. As the stall progresses, the point of separation moves slowly forwards, towards the transition point. On the subsonic type of aerofoil there is no possibility of the laminar flow separating.

1-22 Fig 10 The Low Speed Stall



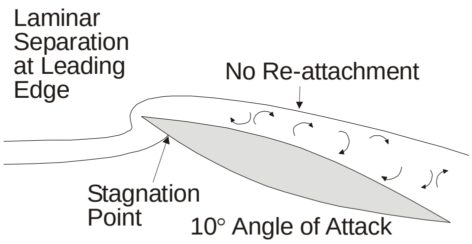
19. As the angle of attack of a thin wing is increased, the stagnation point moves to the undersurface. The subsonic flow has to turn an extremely sharp corner and this results in separation of the boundary layer laminar flow at the leading edge with turbulent re-attachment behind a 'bubble'. Fig 11 shows an aerofoil at 5º angle of attack.

1-22 Fig 11 Formation of a Separation Bubble



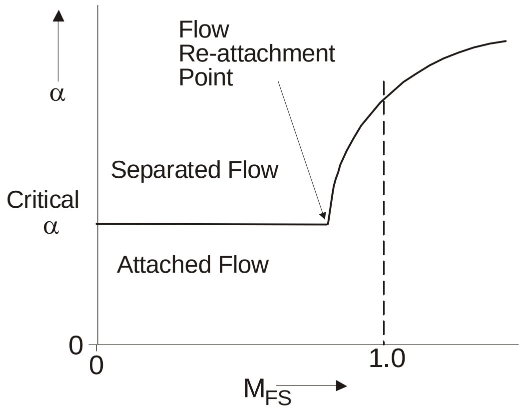
20. As the angle of attack increases further, the separation bubble spreads chordwise until it bursts at the stalling angle. Laminar flow separation from the leading edge, without re-attachment, results. Fig 12 shows this separation occurring at 10º angle of attack.

1-22 Fig 12 Laminar Flow Separation



21. The leading edge stall therefore presents a serious problem since no pre-stall warning is given. But, unlike a conventional stall, the flow can be re-attached with an increase in speed while maintaining the same angle of attack. After re-attachment has occurred, the critical angle of attack increases with speed. This is shown at Fig 13.

1-22 Fig 13 Variation of Critical Angle of Attack with M_{FS}



Points of Maximum Thickness

22. The position of the maximum thickness point of a wing also has an influence on the drag produced. Thin wings are usually associated with a maximum thickness point at mid-chord; this has the effect of reducing the leading edge angles to a minimum, with a corresponding reduction in shockwave intensity. Additionally, the transition point of the boundary layer is well aft and the initial shockwave formation is well back towards the trailing edge. Less of the wing surface is affected by conditions behind the wave and the shockwave travels over a smaller chord distance in the transonic region. The optimum maximum thickness position for minimum drag is 50% chord. However, when the point is between 40% and 60% chord, the increase in C_D above the optimum is negligible.

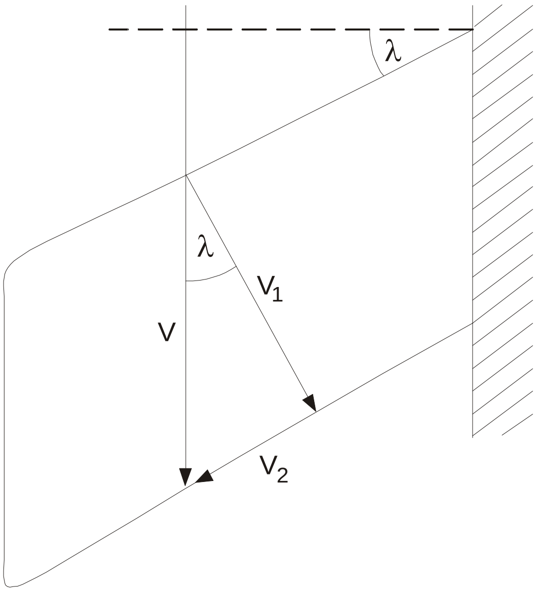
SWEEPBACK

Sweepback and its Advantages

23. In Fig 14, the flow (V) across a swept wing is divided into two flows, V_1 is the component at right angles to the leading edge and V_2 is parallel to the leading edge. As the flow component V_2 has no effect on the flow across the wing, the component V_1 will produce the entire pressure pattern over the wing. So it is the component of the flow normal to the leading edge that affects the values of M_{CRIT} and M_{CDR}, and a higher M_{FS} can be reached before the M_L normal to the leading edge reaches 1.0. The theoretical increase in M_{CDR} is a function of the cosine of the angle of sweep (λ) such that:

$$M_{CDR}\ (\text{Swept}) = \frac{M_{CDR}(\text{Unswept})}{\cos \lambda}$$

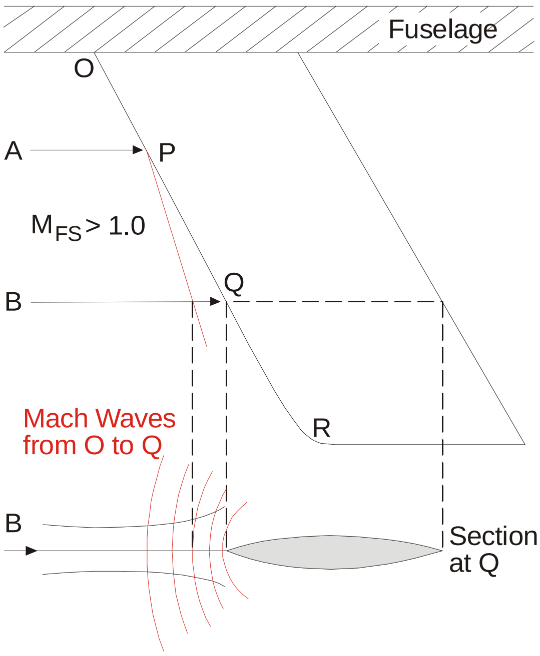
1-22 Fig 14 Effect of Sweepback on M_{CRIT}



λ = Angle of Sweep
V_1 = M_{CRIT}(Straight) = V cos λ
V = M_{CRIT}(Swept)
V_2 = Flow Component Parallel to Leading Edge

24. To see why the component V_1 normal to the leading edge determines whether the flow V will behave in a subsonic fashion, and undergo smooth, gradual accelerations despite the flow speed being supersonic, consider Fig 15.

1-22 Fig 15 Mach Waves on a Swept Wing



Two streamlines at some supersonic speed are shown approaching the leading edge of a swept wing. Points O, P, Q and R are stagnation points on the leading edge. Considering streamline A in isolation, it will be brought to rest and compressed at P and will give rise to a Mach wave which will lie at the appropriate Mach angle. Note that it lies ahead of the leading edge and streamline B passes through it before arriving at point Q. Since there is a small pressure rise across the Mach line, streamline B will experience a small deceleration. Between points P and Q there are a large number of points, all sources of Mach waves, through each of which streamline B will have to pass and, in so doing, will be progressively decelerated before finally coming to rest at Q. Streamlines above and below the streamline Q will be deflected as they pass through the part of the Mach waves oblique to the streamlines; this divides the supersonic flow to allow the wing to pass. There is now supersonic flow behaving in a subsonic manner and a subsonic type of wing section could be used.

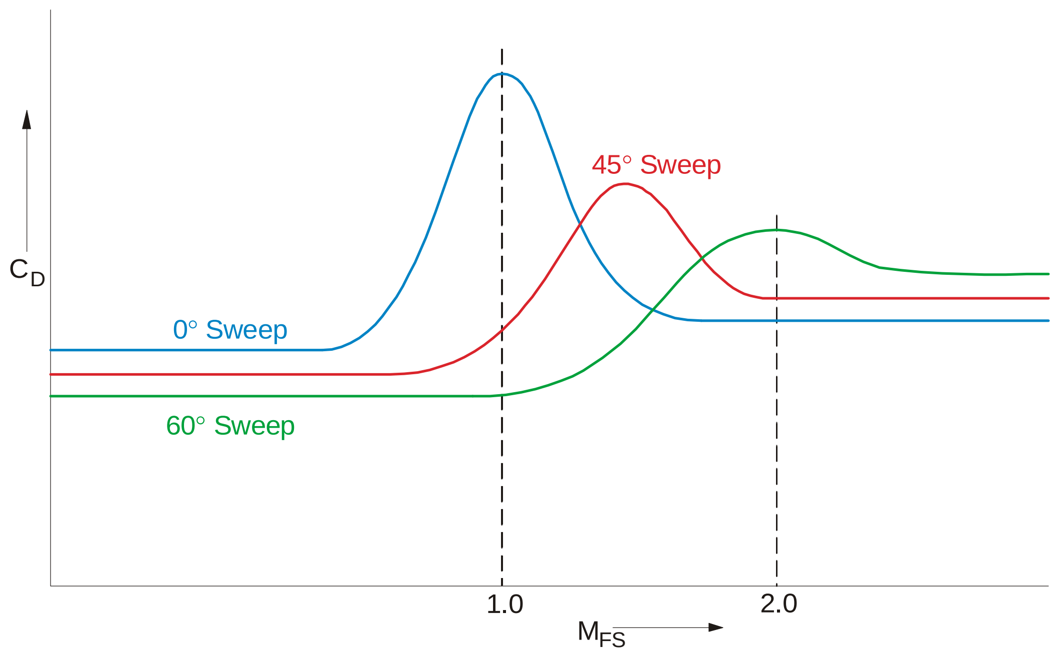
25. The Mach waves affecting streamline B will not be restricted to those originating between P and Q; Mach waves originating at an infinite number of points on the leading edge of the wing inboard point Q will have their effect. By the same argument, all streamlines outboard of the wing root at O will be modified to some extent before they reach the leading edge. Consider the flow at or close to O. Since this point is the wing/fuselage junction, there is no wing up-stream and the flow will not be affected until it reaches the leading edge, where it will be suddenly compressed giving rise to a shockwave. The wing in this region, more accurately described as close to the plane of symmetry, behaves as if it were unswept and this is one of the reasons for the reduction of the theoretical benefits of sweepback.

26. When the Mach lines lie along the leading edge, no pre-warning is possible and, since the flow will behave in a supersonic manner, a supersonic leading edge is essential. Exactly the same argument can be used for points behind the leading edge. When considering the rest of the wing, the sweep of the line of maximum thickness and the sweep of the trailing edge are important. It follows that, certainly with a delta planform, their sweep must be less than that of the leading edge.

Effect of Sweepback on Lift and Drag

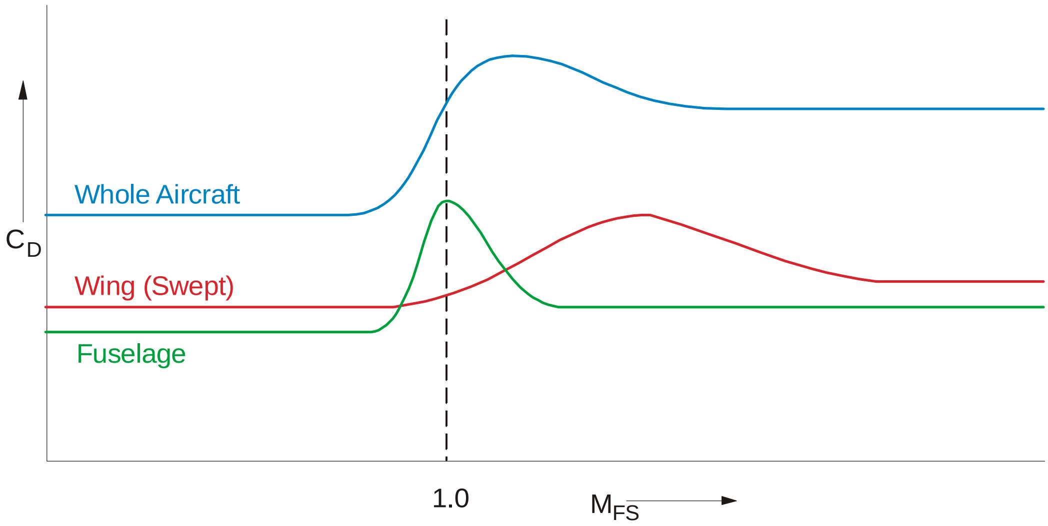
27. Fig 16 shows the effect on the drag coefficient as the angle of sweepback is increased at constant angle of attack. It can be seen that at the higher Mach numbers a straight wing has the smaller drag coefficient.

1-22 Fig 16 Effect of Sweepback on C_D



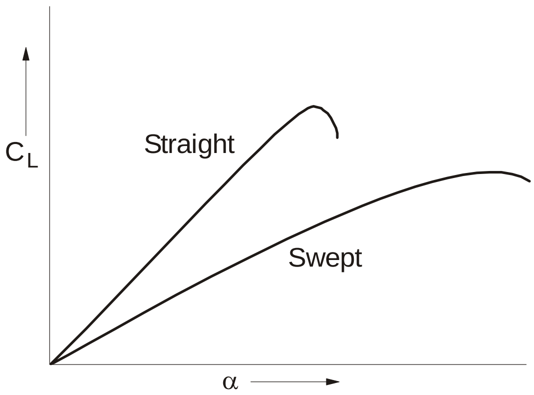
28. If the total aircraft drag is considered, the variation of the drag coefficient will be as shown in Fig 17. By delaying the wing peak C_D rise until after the fuselage peak C_D at M = 1.0, the aircraft's total C_D increase is spread over a large range of Mach numbers and the maximum value is less than if both fuselage and wing reached their peaks at the same speed.

1-22 Fig 17 Total Aircraft Drag and Mach Number



29. Lift can be treated in the same way as in para 23, i.e. only V cos λ, will produce lift. This results in the lower slope shown in Fig 18.

1-22 Fig 18 Effect of Sweepback on C_L



Reduced Advantages of Sweepback

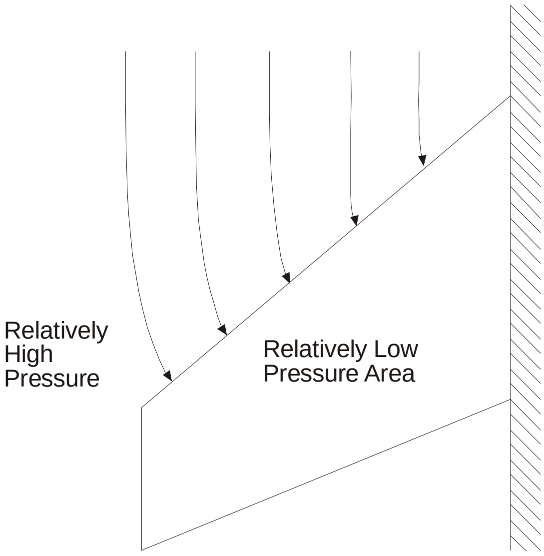
30. In paras 23 to 26 the theoretical advantages of sweepback were explained. In practice, the advantage is only about half the theoretical value. The reason for this is the result of two effects:

a. The change in the free-stream flow at the wing tip.

b. Compression at the wing/fuselage junction.

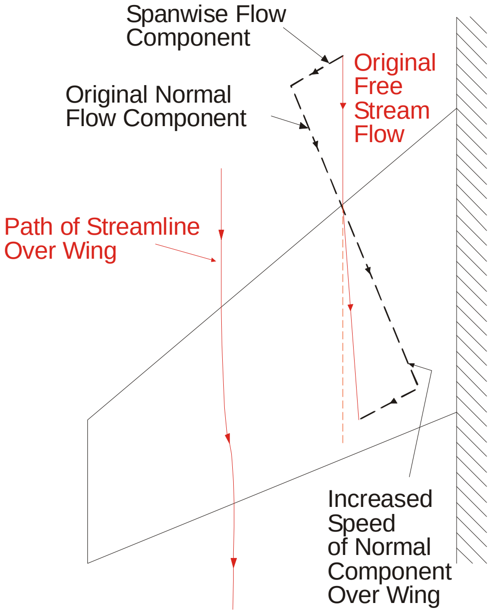
31. **Change in Free Stream Flow at the Tip**. In the wing tip area, there is a change in direction of the free stream flow due to the influence of a pressure gradient from the free stream pressure to the reduced pressure on top of the wing. The modified flow, shown in Fig 19, effectively reduces the sweepback angle causing the onset of compressibility problems at a lower M_{FS}.

1-22 Fig 19 Change in Free Stream Flow



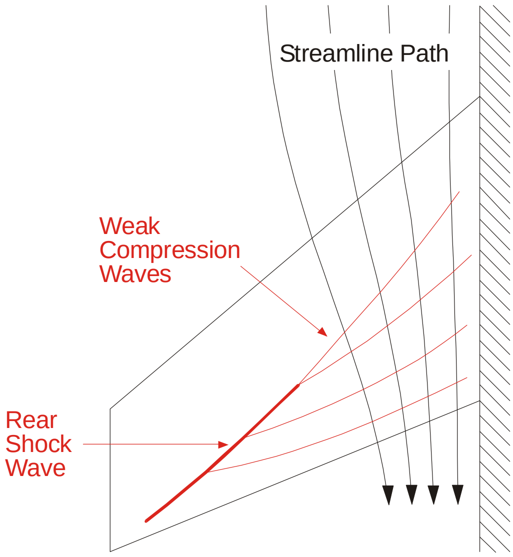
32. **Compression at the Wing/Fuselage Junction**. To understand the formation of the rear shockwave, it is necessary to examine the flow at the wing root. As the air passes over the wing the component of flow normal to the leading edge is accelerated by the shape of the wing, but the spanwise component remains unaltered. This results in the streamlines being deflected towards the wing root as shown in Fig 20. As the velocity of the normal component of the flow over the wing changes, the streamlines follow a curved path. The effect is maximum where the acceleration is greatest, and therefore occurs most noticeably near the wing root where the t/c ratio is a maximum. The streamline at the root, however, is constrained by the fuselage and will itself affect the curvature of adjacent outboard streamlines. When the flow passes the point of maximum thickness, the velocity decreases and recompression takes place.

1-22 Fig 20 Inboard Flow Over a Swept Wing



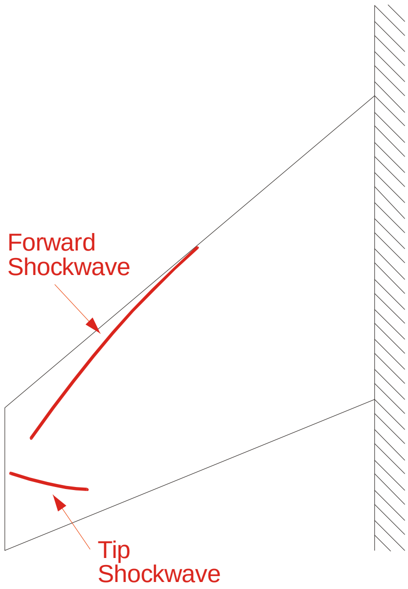
At the wing root, the recompression is diffused, but the compression waves may coalesce along the wing and form the rear shock as shown in Fig 21.

1-22 Fig 21 Compression at the Wing Root



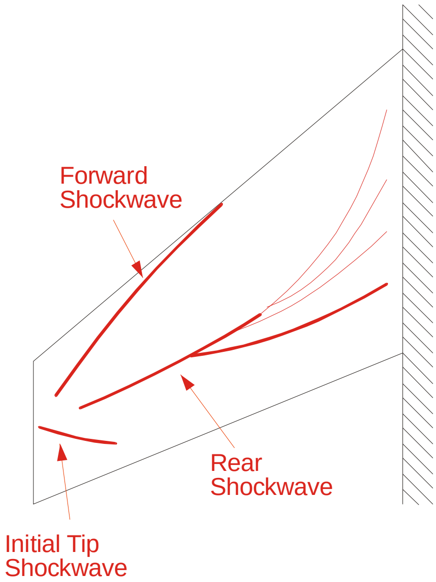
These two effects cause shockwaves to form at a lower M_{FS} than the theoretical M_{FS} for a given angle of sweepback. The effectively reduced angle of sweepback caused by the inboard deflection of the streamlines results in the formation of the initial tip shockwave and the forward shockwave as in Fig 22.

1-22 Fig 22 Position of Forward and Initial Tip Shockwaves



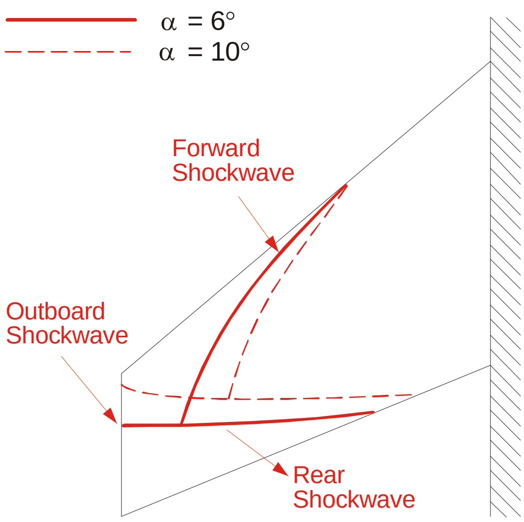
The combined effects shown in Figs 19 to 22 produce a shockwave pattern over a swept wing as shown in Fig 23.

1-22 Fig 23 Shockwave Pattern on a Swept Wing



With increasing speed, the tip shockwave merges with the rear shockwave and the forward shockwave joins the rear shockwave to form the outboard shockwave giving the pattern shown in Fig 24.

1-22 Fig 24 Shockwave Pattern with Increased Speed and Angle of Attack



As the shockwaves combine, the total intensity increases as the sum of individual intensities. If no flow separation has occurred with the original individual shockwaves, it almost certainly will when the outboard shockwave forms. An increase in angle of attack will cause the outboard shockwave to move forward and inboard as shown in Fig 24. The separated flow behind this shockwave usually causes loss of lift at the wing tip which, in turn may cause pitch-up during manoeuvre at transonic and low supersonic speeds.

High Alpha Problems of Swept Wing Aircraft

33. The use of sweepback to obtain the advantage described above creates several undesirable handling characteristics. The main ones are listed below and described, with their remedies, in the subsequent paragraphs:

a. Low slope of C_L / α curve.

b. Very high drag at large angles of attack.

c. Tip stall.

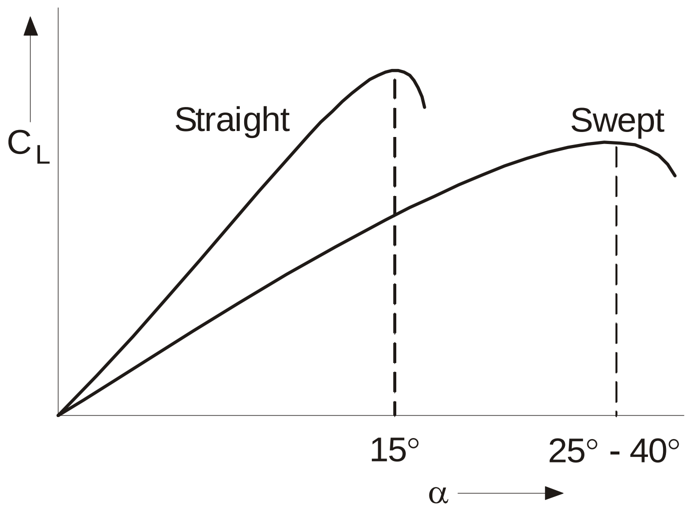
d. Pitch-up.

e. Wing flexure.

f. Dutch roll.

34. **Low Slope of CL /α Curve**. The strong downwash over the centre sections of a swept wing effectively reduces the angle of attack in that region. It also delays flow separation until a very high angle of attack is reached (see Fig 25). For a given value of C_L, the higher angle of attack for the swept wing incurs problems of visibility at low speeds, particularly on the approach to a landing.

1-22 Fig 25 Comparison of Straight and Swept Wing C_L Curves



The simplest method of overcoming this problem is to use a lower angle of attack corresponding to a lower value of C_L, and accept the resulting higher approach speeds. The problem then becomes one of stopping the aircraft safely, by the use of expensive braking devices, or using longer runways. An alternative is to use the higher angle of attack and lower speeds and have long undercarriage legs to give the necessary ground clearance in the landing attitude; this may also require a droopable nose section to improve the visibility from the cockpit. A further solution is the use of variable incidence wings, the higher incidence being used for landing, giving a high angle of attack for the wing, while leaving the fuselage in a more acceptable attitude for landing and low speed handling.

35. **High Drag at High Angles of Attack**. Swept wing aircraft suffer a large increase in lift dependent drag at high angles of attack, mainly because of the development of the ram's horn vortex. The combination of effects which causes this vortex formation is described in Volume 1, Chapter 9, para 34 et seq. As the ram's horn vortex accounts for a major proportion of the drag at high angles of attack, any measure which delays its onset or limits its subsequent growth is beneficial. The design methods used to achieve these objects usually involve one or more of the following: checking the

boundary layer outflow; reducing leading edge separation; limiting the inboard development of the vortex. The boundary layer outflow is reduced by:

a. Wing fences, which not only provide a physical barrier to the outflow, but also induce a small fore and aft vortex which carries the boundary layer rearwards with it. Inboard of the fences, the boundary layer is thickened, thus reducing the slope of the local lift coefficient curve (C_1/α) and increasing the wing root stalling tendency. The overall effect is, however, beneficial in that the stalling tendency is reduced at the more critical tip regions.

b. Leading edge slats or slots, which increase the velocity of airflow over the wing surface and re-energize the boundary layer.

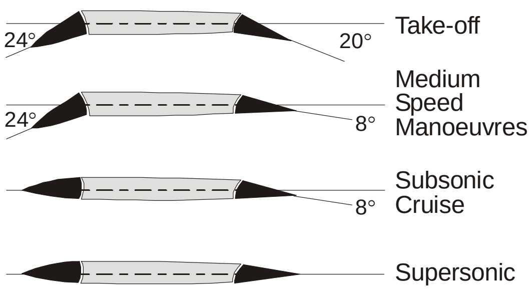
c. Boundary layer control by sucking or blowing.

d. Vortex generators, which re-energize the boundary layer by mixing in the fast moving air above the boundary layer.

Reduction in leading edge separation is achieved by drooping the leading edge. Examples of both fixed and movable leading edges are illustrated below:

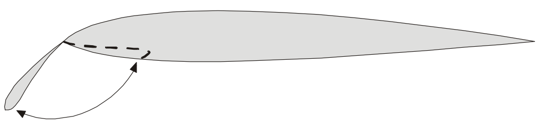
e. The droopable leading edge operates through some 24º (Fig 26). This is used in conjunction with the trailing edge flap to give the optimum wing camber for the phase of flight.

1-22 Fig 26 Droopable Leading Edge



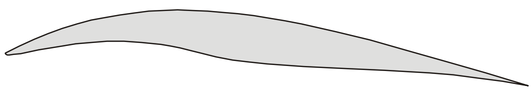
f. The Kruger flap (Fig 27) is used on many swept wing transport aircraft.

1-22 Fig 27 Kruger Flap



g. The Concorde used a permanently drooped, or cambered, leading edge (Fig 28).

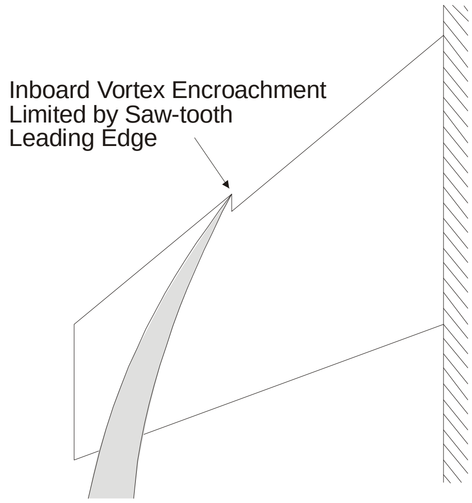
1-22 Fig 28 Cambered Leading Edge



The inboard development of the ram's horn vortex is limited by:

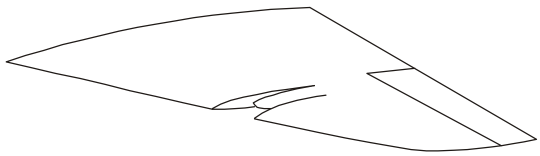
h. A saw-tooth leading edge (Fig 29), which provides a discontinuity in the leading edge; this tends to anchor the vortex at that point and prevents further inboard encroachment. By acting as a vortex generator, the saw-tooth also reduces boundary layer outflow. The reduced t/c ratio of the tip area has a beneficial effect on tip stalling tendencies (see para 37).

1-22 Fig 29 Saw-tooth Leading Edge



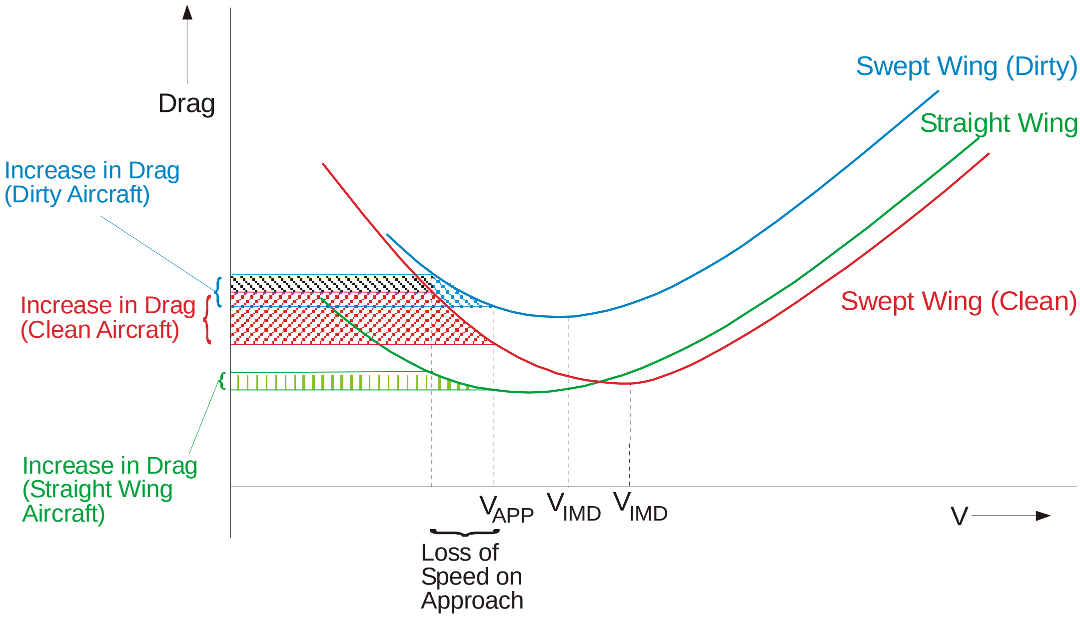
i. A notched leading edge (Fig 30) performs the same function as the saw-tooth, except that it has no effect on the t/c ratio. However, this device is sometimes combined with an increased t/c ratio wing section outboard of the notch.

1-22 Fig 30 Notched Leading Edge



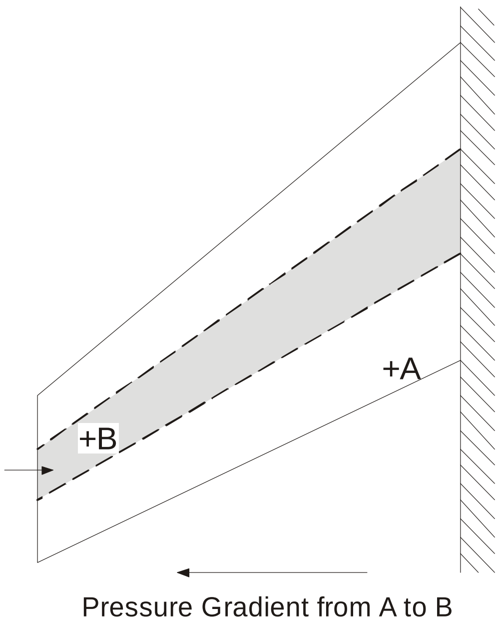
36. **Modified Drag Curve for Approach Speed Stability**. Fig 31 compares the drag curves for a straight wing and a swept wing aircraft, both in a clean configuration. Assuming a common approach speed (V_{APP}), it can be seen that, for the same value of speed loss on the approach, the swept wing aircraft suffers a greater drag penalty than its straight wing equivalent. If the zero lift drag of the swept wing is increased (by lowering large flaps, extending airbrakes or deploying a drag parachute), the total drag curve can be modified to approximately the same profile as that for the straight wing aircraft, although at the penalty of higher drag values. The effect is to reduce the drag rise following a loss of speed on the approach, thus making the approach speed less unstable. A further benefit from the higher total drag is that higher power settings are needed at lower speeds and the turbojet engine response is much quicker from these settings than from near idling.

1-22 Fig 31 Modification to Drag Curve for Swept Wing Low Speed Stability



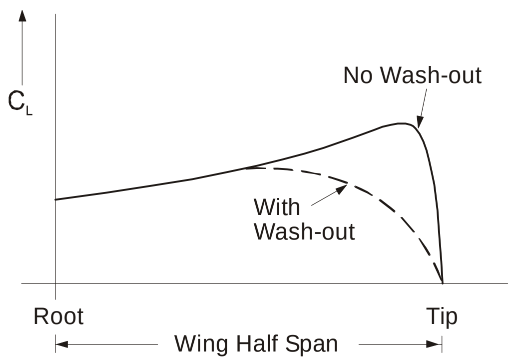
37. **Tip Stall**. The development of tip stalling is covered in Volume 1, Chapter 9, paras 32 to 36. The alleviation of tip stalling is achieved by reducing the boundary layer outflow. It follows, therefore, that all of the methods described above for reducing the ram's horn vortex effect will also be successful in varying degrees in preventing tip stalling. In Fig 32, the shaded portion of the wing represents the area of minimum pressure over the wing top surface, with the pressure increasing both ahead of and behind that area.

1-22 Fig 32 Pressure Gradient on a Swept Wing



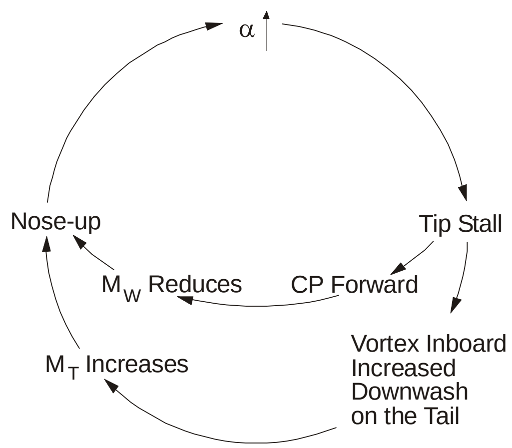
Between the two points A and B, a pressure gradient exists; it is this gradient which gives the outflow component to the wing. If the pressure gradient is reduced by reducing the C_L in the area of B, the outflow component is decreased. The reduction in lift at B is achieved by washout from wing root to tip, and its effect is shown in Fig 33.

1-22 Fig 33 Effect of Washout on Spanwise Loading



38. **Pitch-up**. Pitch-up is a form of longitudinal instability. In this case, the behaviour of the aircraft following an increase in angle of attack is for the aircraft to continue to diverge in pitch, thus further increasing the angle of attack. The sequence of events following a change in angle of attack form the self-perpetuating cycle shown in Fig 34. M_W is the pitching moment of the wing (normally nose-down), and M_T the pitching moment of the tail (normally nose-up). The effectiveness of any attempt to interrupt the cycle will be influenced by the rate of change of pitch which has developed before the control column is moved forward. Aircraft having large inertia in pitch may, in extreme cases, overshoot the demanded increase in angle of attack by a large margin. Therefore anticipation is required of the pilot, especially at high angles of attack, otherwise the aircraft may pitch-up beyond the stalling angle of attack. On some T-tail aircraft, pitch-up can lead to a deep stall attitude (see para 41).

1-22 Fig 34 Pitch-up Cycle

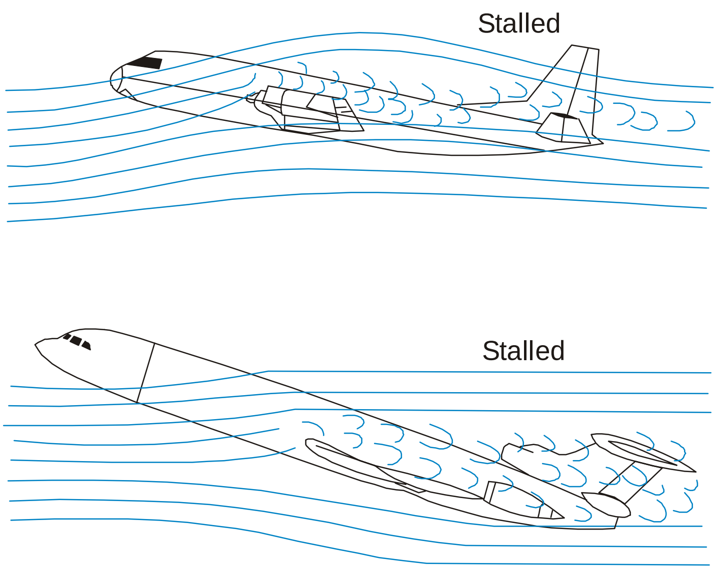


39. **Wing Flexure**. A thin, swept wing can flex under load if it has insufficient stiffness. Upward flexing of the wing causes the angle of attack to be reduced along the wing, with the maximum effect occurring at the wing tips. The reduced lift on the outboard wing section causes the CP to move forward and this, in turn, may induce pitch-up. The geometry of this washout due to flexure is described in Volume 1, Chapter 9, para 42. To minimize the effect, a swept wing must be made stronger, and therefore heavier, than an equivalent straight wing. A delta planform almost eliminates this problem while still retaining the advantages of sweepback.

40. **Dutch Roll**. All types of aircraft can suffer from Dutch rolling. This oscillatory yawing and rolling motion, which varies from predominantly roll (wing rock) to predominantly yaw (snaking), may occur when lateral stability overcomes directional stability (see Volume 1, Chapter 17, paras 65 to 67). Methods of reducing Dutch roll include the use of autostabilizers and the reduction of the excessive lateral restoring moment inherent in a swept wing by setting the wings at a small anhedral angle.

41. **Deep Stall**. Some of the characteristics of swept wings (i.e. tip stall and pitch-up) increase the possibility of entering the deep stall region. Recovery from a stall is dependent on the pitching moment from the wing and the pitching moment from the tailplane. In the case of a straight wing aircraft at the stall, the CP tends to move rearwards, giving a nose-down pitching moment. On a swept wing, however, the tendency is for the CP to remain forward, possibly giving a nose-up pitching moment. All now depends on the tailplane, which, if set low relative to the mainplane, will be working clear of the separated flow from the mainplane and will therefore provide a nose-down pitching moment and give a conventional reaction to the stall. If, however, the tailplane is set high (i.e. T-tail), then increasing the angle of attack reduces the vertical separation between mainplane and tailplane and, at the stall, the tailplane is immersed in the separated mainplane wake and therefore suffers a reduction in effectiveness (see Fig 35). The nose-up pitching moment of the mainplane may be sufficient to hold the aircraft in its stalled condition. The rate of descent which develops aggravates the condition by increasing the angle of attack even further and may make subsequent recovery impossible. While being unsuitable from the deep stall and pitch-up aspects, the T-tail configuration is dictated by the mounting of engines on the rear fuselage.

1-22 Fig 35 Tailplane Position at the Stall



When mounted on top of a swept fin, the tailplane has the advantage of operating over a longer moment arm. The tailplane also acts as an endplate to the fin and increases the effective aspect ratio of the fin. Where the T-tail is used, an artificial stall reaction is usually designed into the control system in the form of a stick shaker, stick pusher or both. Both are sensitive to pitching rates and angle of attack; the stick shaker provides the stall warning (sometimes accompanied by an audible warning), and then, if this is ignored by the pilot, at a slightly higher angle of attack the stick pusher simulates the conventional nose-down pitching at the stall by pushing the stick forward. Some aircraft, particularly those involved in the investigation of stall characteristics, are equipped with a drag parachute or rocket in the tail. When used in a deep stall, the chute or rocket provides the essential nose-down pitch moment to enable recovery to be made.

Modification to the Swept Planform

42. Para 32 explained the formation of the shockwave pattern on a swept wing. It is possible to design a wing which is completely free of shockwaves at a supersonic M_{FS}. The shockwaves are eliminated by:

a. Increasing the sweep of the wing root.

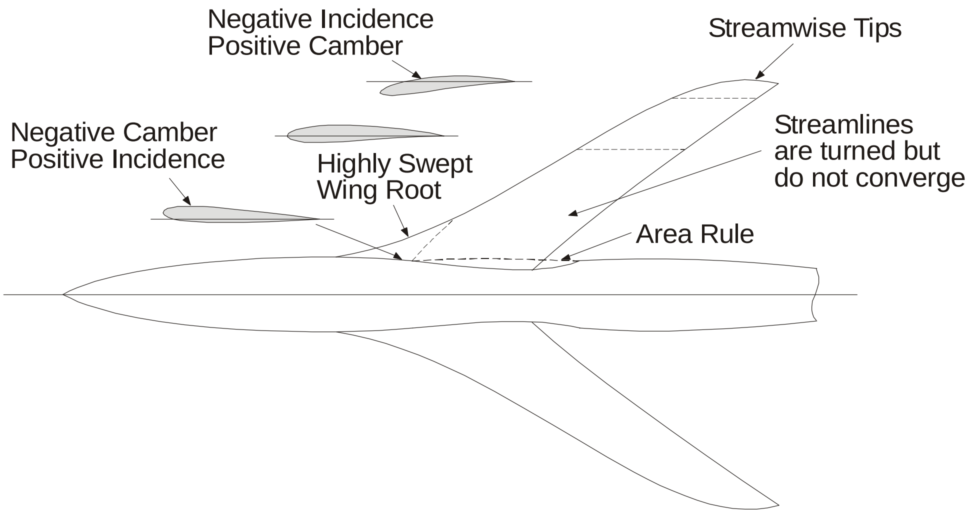
b. Increasing the sweep of the wing tip.

c. Fuselage 'waisting' (area rule).

d. Suitable wing twist and camber.

The combined effects are illustrated in Fig 36 and described below.

1-22 Fig 36 Ideal Supersonic Wing



43. **Wing Root Sweep**. It has already been seen (para 32) that the wing close to the junction with the fuselage behaves as if it were unswept and gives rise to a shockwave at the wing root. This effect is countered by increasing the sweepback of the wing root as shown in Fig 36.

44. **Wing Tip Sweep**. The free stream flow at the wing tip is influenced by the disposition of the areas of low and high pressure (Fig 19) and results in an effective reduction in the wing sweepback and the formation of the tip shockwave. This effect is alleviated by rounding off the wing tips so that the deflected flow meets the leading edge at a constant angle. Such a shape is called a streamwise or Kuchemann tip.

45. **Fuselage Waisting**. Fuselage waisting prevents the re-compression waves produced at the root (Fig 21) from coalescing along the span, and this prevents the formation of the rear shockwave. Fuselage waisting is also a feature of area ruling which is discussed in para 62 et seq.

46. **Wing Twist and Camber**. Wing twist and camber control the spanwise and chordwise lift distribution and thus eliminate local high lift spots which could stall or develop shockwaves before the rest of the wing.

OTHER PLANFORMS

Delta Wings

47. There are many types of delta wings, the three main parameters being sweep of the leading edge, sweep of the line of maximum thickness and sweep of the trailing edge. The most important feature is to sweep the leading edge inside the Mach cone.

48. The delta wing reduces wave drag by the methods discussed under sweepback; there is also a reduction due to the very low aspect ratio which gives a low thickness chord ratio.

49. For a brief consideration of the delta, a straight unswept trailing edge is assumed. It is, in effect, a swept wing with the rear portion between the tips and fuselage filled in. Assuming that the delta wing has the same advantages as the swept wing, the additional comments may be made:

a. The long chord allows a good wing thickness for engine and fuel installations yet maintains a low thickness/chord ratio.

b. The torsional stress of a swept wing is relieved. More structural rigidity along the flexural axis can be achieved and hence flexing under load is reduced. Stability in manoeuvre is improved.

c. The tip stalling tendency associated with swept wings is alleviated. The low aspect ratio reduces the boundary layer outflow and any increased loading at the tips can be minimized by decreasing the thickness/chord ratio in that area.

d. An increased wing area reduces the wing loading thus permitting high altitude flight and lower stalling speeds. It also creates more skin friction. A very high stalling angle is produced.

e. At the higher angles of attack, the vortices above the wing are far more pronounced. This vortex is a combination of chord-wise growth of the separation bubble and inflow due to the low aspect ratio.

f. A large chord leads to an increase in skin friction drag and also requires long intakes and/or jet pipes, etc.

Slender Deltas

50. The salient feature of the Concorde is its slender delta planform which provides low drag at supersonic speeds because of its low aspect ratio (slightly less than 1.0), great structural strength for durability, and plenty of stowage for fuel, etc.

51. The BAe-Aerospatiale Concorde was designed to cruise in the region of M 2.2 because this is the highest speed at which cheap and well known aluminium alloys may be used for the main structure, whilst withstanding the kinetic heating (limited to 400 K). A L/D ratio of about 7.5:1 (essential for range) is obtained at M 2.2, whilst the subsonic L/D ratio is about 12.5:1.

52. The low aspect ratio and sharp leading edge produce leading edge separation at a low angle of attack. Previously, the vortex so produced had been an embarrassment because it was unstable, varied greatly with angle of attack, caused buffet, increased drag and decreased the C_{Lmax}. However, with careful design the vortex can be controlled and used to advantage. By having a sharp leading edge and drooping it by varying amounts from root to tip, such that it presents the same angle to the local flow, attached flow is assured up to a particular small angle of attack above which the flow separates from root to tip with a resulting vortex all along the leading edge. Subsequent increases in angle of attack do not significantly change the pattern of flow until an angle of about 40º is reached.

Crescent Wings

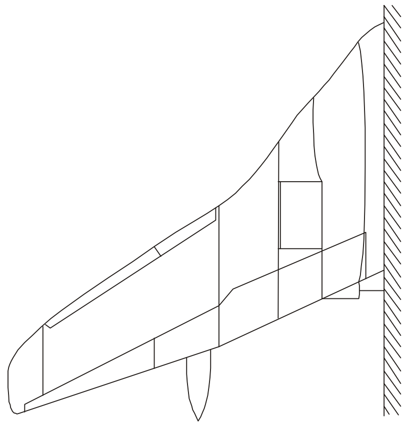
53. The crescent planform employs two methods of raising the critical Mach number and reducing the C_D rise, ie a combination of sweepback and thickness/chord ratio. At the root section where the wing is thickest, the angle of sweep is greatest; as the t/c ratio is reduced spanwise, so is the angle of sweep. The crescent wing planform shown in Fig 37 has three advantages:

a. The critical drag rise Mach number is raised.

b. The peak drag is reduced.

c. Tip stalling is prevented since the outboard sections of the wing are virtually unswept; hence there is less outflow of boundary layer which gives a more uniform spanwise loading.

1-22 Fig 37 The Crescent Wing

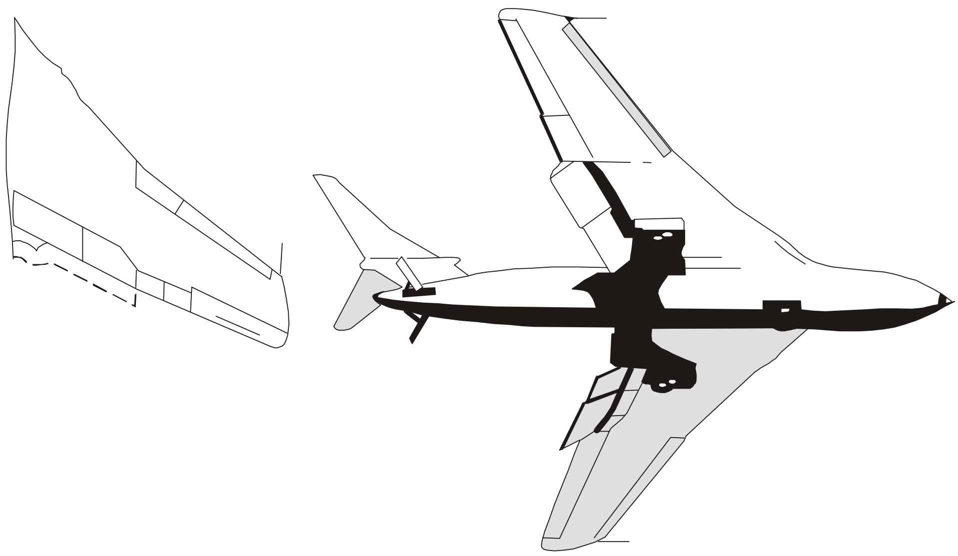


Polymorph Wings

54. Polymorphism is a change in wing planform. The most common current methods of achieving this are described below:

a. The Fowler flap (Fig 38a) is a very efficient method of altering the wing geometry; extension of the flap alters both the wing chord and camber and thus improves the low speed handling characteristics of high-speed wing sections. It is used extensively on modern aircraft.

1-22 Fig 38 Examples of Polymorphic Variable Geometry
Fig 38a Fowler Flap - Increasing Area and Effective Camber



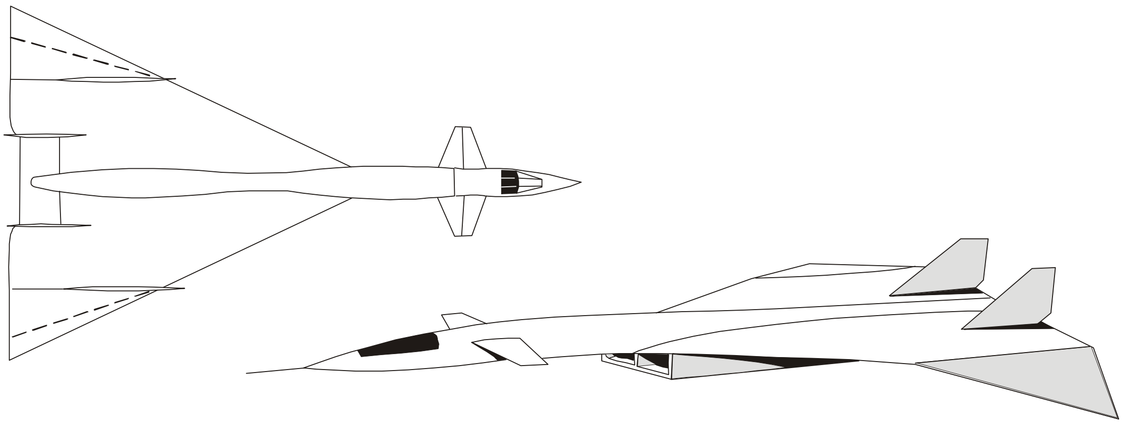
b. Tip droop (Fig 38b) performs three functions:

(1) It improves stability and manoeuvrability in supersonic flight by increasing the effective fin area.

(2) It controls the movement of the CP; the CP is moved forwards at supersonic speeds, thus reducing the nose-down pitching moment and, consequently, reducing the trim drag.

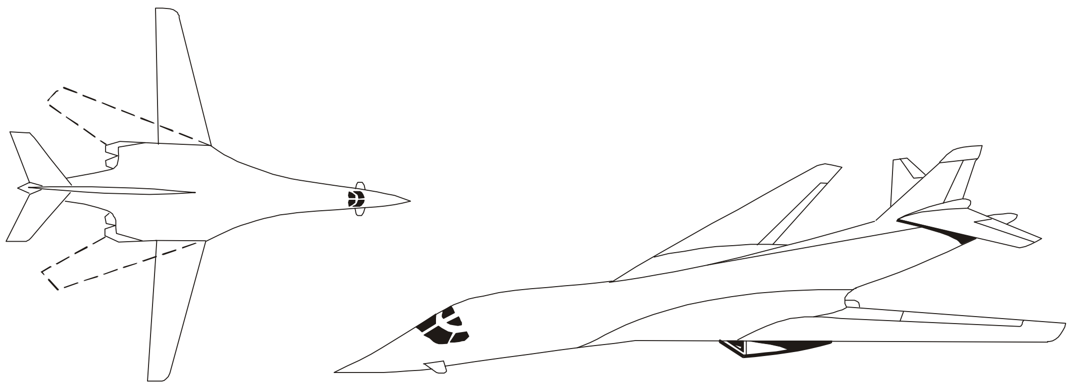
(3) At supersonic speeds, the drooped tips increase the effect of compression lift by partially enclosing the high underbody pressure (see Volume 1, Chapter 21, Fig 33).

Fig 38b Tip Droop



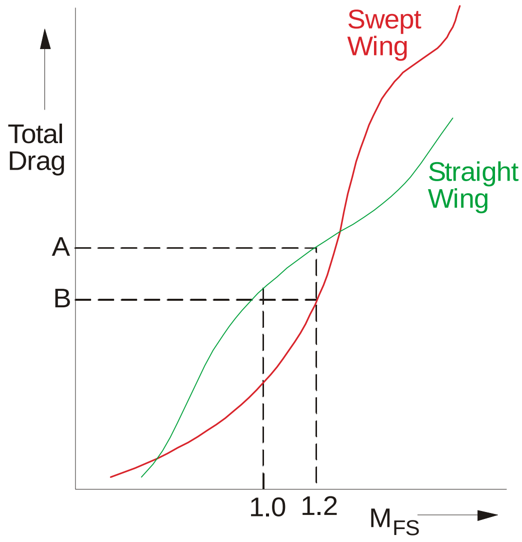
c. Variable sweep or swing-wing (Fig 38c) has been adopted for a number of high-speed aircraft. The advantages gained from variable sweep are examined below.

Fig 38c Variable Sweep



55. In Fig 38c the two extremes of wing position can be seen; fully swept, the aircraft has all the advantages of a swept wing for high-speed flight; fully forward the aircraft is an efficient subsonic aircraft with all the advantages of a straight winged aircraft. Fig 39 illustrates the difference in drag between a straight and a swept winged aircraft. Taking two extremes, a straight winged aircraft would require thrust to equal 'A' amount of drag for level flight at M = 1.2. However, if the wing were swept, it would require thrust to overcome 'B' amount of drag to fly at the same speed. Therefore the engine in a variable-geometry (VG) aircraft can be smaller than for a fixed wing aircraft.

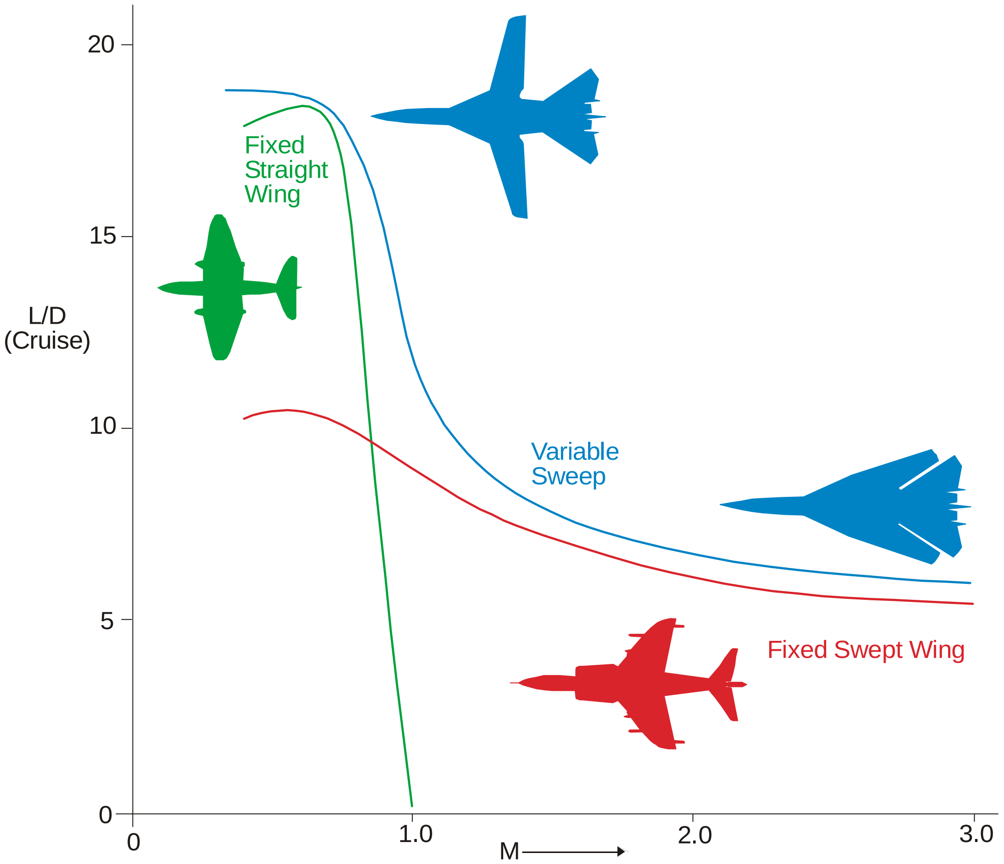
1-22 Fig 39 Drag Comparison Between Straight and Swept Wings



56. As the aircraft with 'B' amount of drag accelerates, the drag can be kept constant as the wing position is progressively changed from the straight to the swept position. In practice, the speed at which the wings would be swept back would be about M_{CDR}.

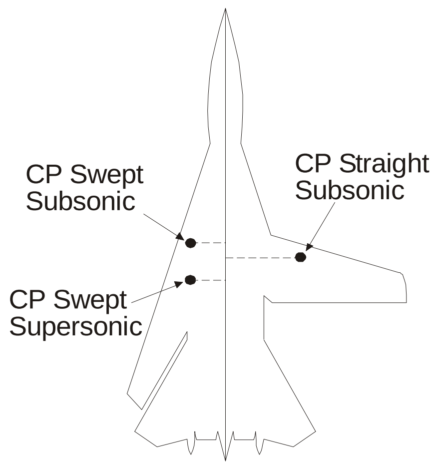
57. Fig 40 shows a graph of the L/D ratios for a straight and swept wing aircraft. When fixed wing aircraft have to operate 'off-design', they are inefficient and so the fuel consumption increases; a VG aircraft can operate over a wide speed range and still remain efficient.

1-22 Fig 40 Lift/Drag Ratios for Straight and Swept Wings



58. If the wings are arranged so that they move within a swept root stub or "glove", i.e. only part of the wing moves, then the aft movement of the CP is reduced as the aircraft becomes supersonic (Fig 41).

1-22 Fig 41 Movement of the Centre of Pressure in a Variable Geometry Aircraft



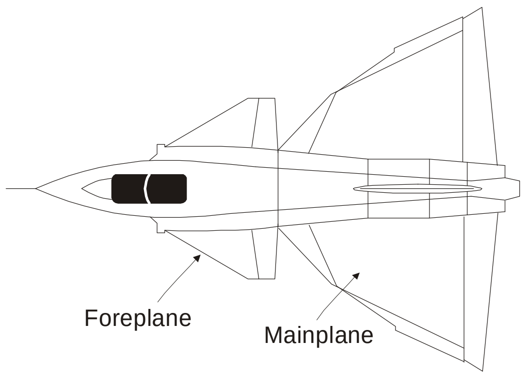
Canard Designs

59. An example of a Canard design is given in Fig 42. This design can be of two kinds:

a. The foreplane acting as a pure trimmer; the manoeuvre in pitch is then controlled by elevons on the mainplane.

b. The foreplane acting as a surface which has the same functions as a conventional tailplane.

1-22 Fig 42 Canard Configuration



60. The main advantages of a Canard design are:

a. The up-force on the foreplane generally contributes to the lift and therefore:

(1) For a given speed, a lower angle of attack is needed on the mainplane; this results in less drag.

(2) There is a lower overall stalling speed which gives a better short field performance.

b. It may be possible to give the foreplane a long moment arm, thus requiring a smaller control deflection.

61. The two main disadvantages are:

a. For a stable configuration, the canard must always be rigged so that the foreplanes stall before the main wings.

b. There may be a problem with directional stability because the vortices shed from the tips of the foreplanes react on the fin.

BODY/WING COMBINATION

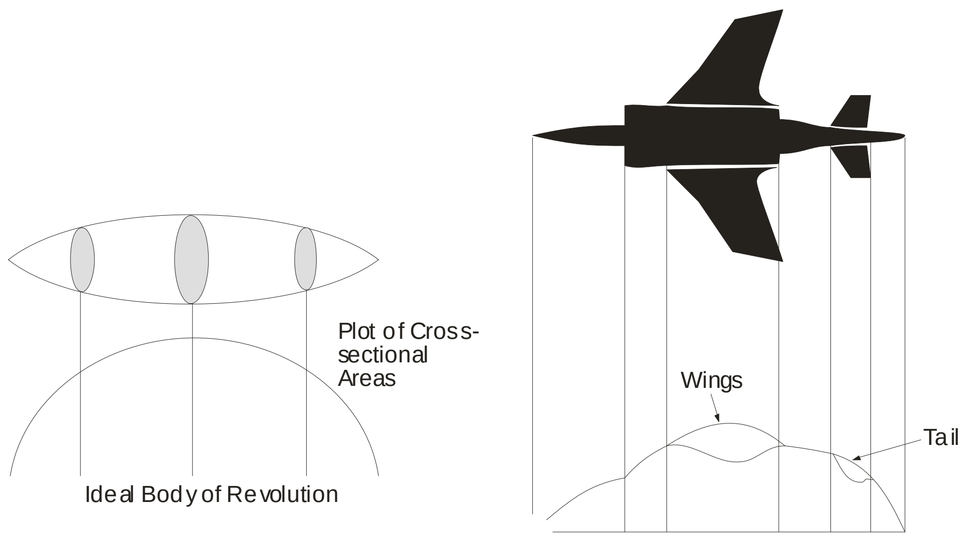
Area Rule

62. In the explanation of wave drag (Volume 1, Chapter 21) it was mentioned that, if the intensity of the shockwaves could be reduced, the drag would be reduced. This was achieved by reducing the thickness of the wing. The same reasoning follows for a fuselage, except that there must be minimum cross-section for the accommodation of a crew, equipment, weapons, etc.

63. The design feature used for mating the various parts of the aircraft together to produce minimum drag is called the Area Rule.

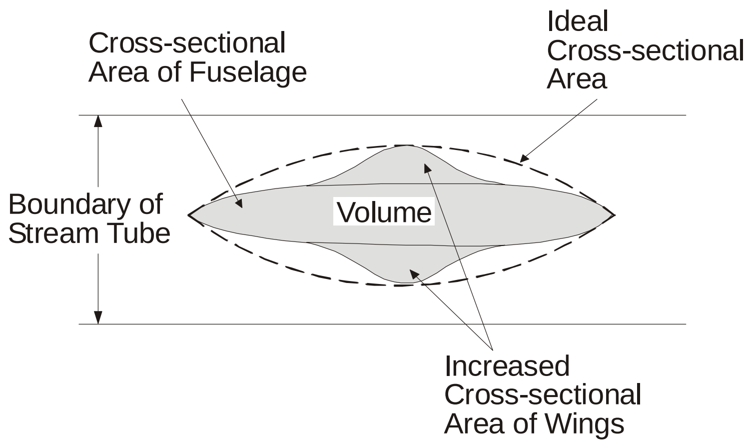
64. On theoretical grounds it can be shown that there is a certain ideal shape for a body of specified volume or frontal area which has minimum drag at a specified transonic or low supersonic speed. A graph distribution of its cross-sectional area along its longitudinal axis is a smooth curve. The area rule states that, if an irregular body such as an aircraft has the same cross-sectional area distribution as this ideal body, it will also have the minimum possible drag. Fig 43 illustrates this principle.

1-22 Fig 43 Principle of Area Rule



65. To see why it is possible to equate the drag of an aeroplane with that of a body of revolution, consider the stream tube of a diameter greater than the span of the aircraft in Fig 44. The flow within the tube will be distorted because of the presence of the aircraft but the distortion will be smoothed by pressure changes along the circumference of the stream tube and by the tendency of all streamlines to resist displacement. If the diameter of the stream tube is to remain constant, then the acceleration within the tube will be a function of the rate of change of cross-sectional area of the aircraft. A body of revolution having the same variation of cross-sectional area will produce the same flow pattern as the aircraft and a typical body for a non-area ruled aircraft is shown in Fig 44.

1-22 Fig 44 Area Rule



The presence of the 'lump', corresponding to the aircraft's wings will obviously mean high velocities round it and a very strong shockwave behind it. If the area is increased as shown by the dotted line, while keeping the same maximum thickness, then the acceleration will be decreased with a corresponding decrease in drag.

66. Area ruling may be achieved in several ways:

a. By 'waisting' the fuselage.

b. By 'thickening' the fuselage ahead of and behind the wings (see Fig 43).

c. By combining a flat-sided fuselage with highly swept wings.

d. By fitting pods on the trailing edge of the wings (this also "corners" the boundary layer).

The effect of good cross-sectional distribution can be seriously reduced if care is not taken to reduce interference drag.

Interference Drag

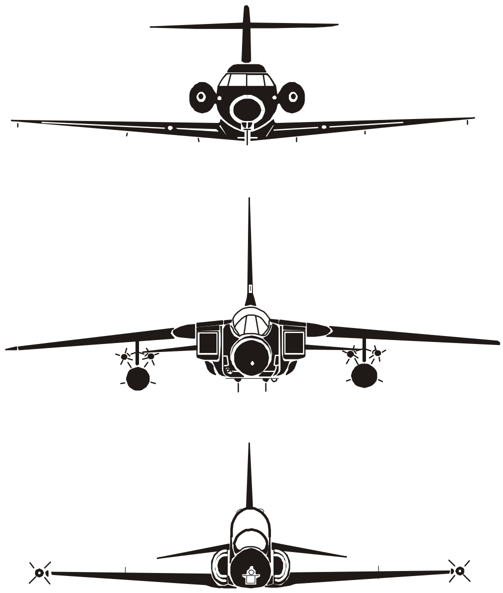
67. Interference drag is caused when two flows around an aircraft meet. If the flow at the body/wing junction is considered, the speed of the flow over the wing is much greater than that along the fuselage. When the flows mix, the speeds of the two flows change so that the flow over the wing is reduced and the pressure pattern is changed in such a way that the wing root produces comparatively less lift than the rest of the wing.

68. On a swept wing this change in pressure pattern has the effect of reducing the sweep of the wing at the root. This effect, explained in para 36, has an adverse effect on transonic flight.

69. At the root trailing edge the retarded flow is entering an area of divergence between the fuselage and wing trailing edges. This slows up the flow even more and causes an earlier boundary layer separation, thus increasing the drag.

70. To reduce the interference drag at high speeds, an effort is made to make all component parts meet at right angles. This produces such wing/body junctions as those shown in Fig 45.

1-22 Fig 45 Reducing Interference Drag



Trim Drag

71. In Volume 1, Chapter 21, the origin of trim drag was explained. There are two basic methods of minimizing the effect:

a. Reduce the movement of the CP.

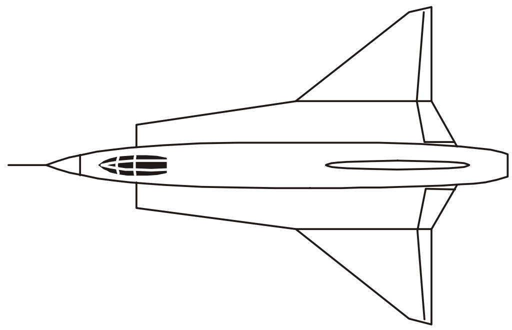
b. Rebalance of forces acting on an aeroplane in supersonic flight so that the supersonic balance is the same as the subsonic balance.

72. **Reducing the CP Movement**. There are four methods of reducing the movement of the CP:

a. **Reflex Camber**. Subsonically, the pressure distribution across the chord of a reflex cambered wing is more uniform than for a section without reflex camber. This means that the CP is further back from the leading edge than for a positive cambered wing; therefore the rearward movement of the CP in supersonic flight is reduced.

b. **Double Delta**. The double delta design (Fig 46) is used as a method of reducing the movement of the CP between subsonic and supersonic flight. At subsonic speeds, the highly swept forward part of the double delta is inefficient and the CP is controlled by the more efficient reduced sweep rear part of the wing. At supersonic speeds, the whole wing becomes efficient and the CP is controlled by the whole, thus countering the rearward movement of the CP.

1-22 Fig 46 The Double Delta Configuration



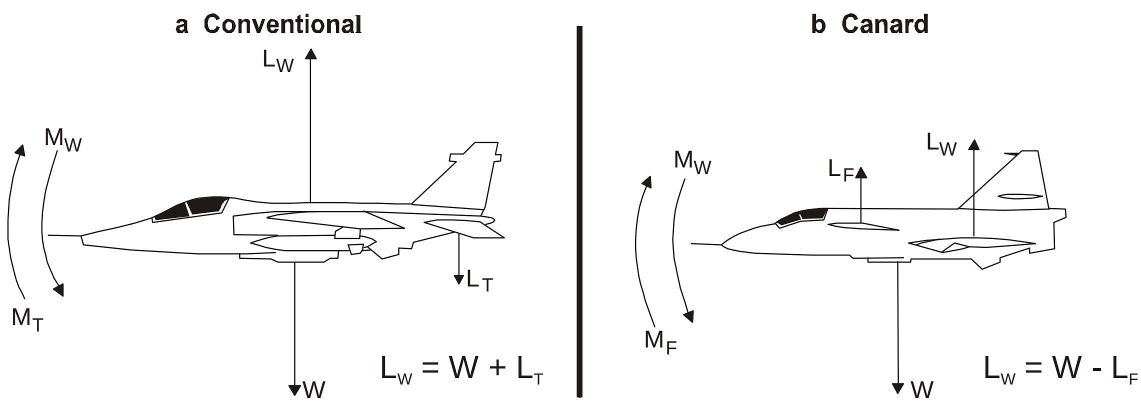
c. **Vortex Core Effect**. As the vortex on top of a slender delta develops across the chord of the wing the pressure at the core reduces from the origin to the trailing edge, and the pressure difference from top to bottom of its wing is greater towards the trailing edge. This again has the effect of positioning the subsonic CP in a rearwards position, reducing the supersonic rearward movement.

d. **Polymorph**. Provided that there is a fixed section of the wing ahead of the moving section, usually a highly swept root section, then there is a reduction in the movement of the CP from the subsonic straight to the supersonic swept position compared to an all-swept aircraft.

73. **CG Movement**. Having accepted a large movement of the CP in supersonic flight, there is currently only one method of rebalancing the forces acting on an aircraft; this is to move the CG. With the rearward movement of CP in supersonic flight, the wing pitching moment (M_W) increases and so the correcting moment from the tailplane (M_T) must also increase, giving two sources of increased trim drag. If M_W could be reduced by moving the CG closer to the CP then the aerodynamic forces required to overcome the increased M_W could be reduced; thus trim drag would be reduced. Many aircraft move fuel to adjust their CG to reduce trim drag.

74. **Balance in Pitch**. In a conventional aircraft (Fig 47a), the pitching moment of the wing (M_W) is balanced by a download on the tailplane; the result is that the wing now has to produce more lift (L_W) to balance the sum of the weight and the elevator download (L_T). The greater lift is produced at the expense of a higher angle of attack and thus more drag.

1-22 Fig 47 Comparison Between Conventional and Canard Trim Methods



In the canard configuration (Fig 47b) the wing pitching moment (M_W) is still nose-down but, since the foreplane is ahead of the CG, the trimming force (L_F) acts upwards and so contributes to the total lift; the result is that the wing lift (L_W) is less than the weight and therefore requires a smaller angle of attack with a corresponding reduction in drag. The final trim drag is dependent only on the efficiency of the foreplane.

Control

75. The traditional simple flap-type control has several disadvantages on high-speed aircraft; these are:

a. Once shockwaves are established on a wing (or tailplane), a control deflection can no longer modify the total flow over the wing. Its influence is limited to aft of the shockwave where the shockwave is located ahead of the control surface or, with the shockwave at the trailing edge, the flow in the immediate vicinity of the control. This results in a loss of control effectiveness at higher speeds.

b. Control effectiveness can be reduced even further if the control is working in a region of shock-induced separation of the boundary layer.

c. When shockwaves are located on or close to the control surface, a deflection of the control will cause them to move. Their movement can significantly change the magnitude of the forces and position of the centre of pressure on the control surfaces; the hinge moments will therefore fluctuate rapidly, and a high frequency 'buzz' will be felt through the controls by the pilot.

76. **Control Developments**. The significant developments in the design of control systems are:

a. Powered controls.

b. All-flying, or slab surfaces.

c. Spoilers, or inboard ailerons.

77. **Powered Controls**. The earliest type of powered system was the power-assisted control. The efforts of the pilot were partially relieved, but a proportion of the force required to move the control was still felt on the stick. The present-day philosophy is to install power-operated controls which make the manual stick forces independent of the actual hinge moment. Such a system, by its irreversible nature, will also prevent the erratic control movements previously mentioned. No external force will move the control surface; moreover, mass balances are unnecessary provided that the controls are never operated manually. Power-operated controls require artificial feel otherwise no resistance to control deflections would be felt by the pilot, with the possible consequence that the aircraft's structural limitations could be exceeded. The most common artificial feel systems, which can be used singly or in combination, are covered in Volume 4, Chapter 4 and are described briefly below:

a. **Spring Feel**. Spring feel is the simplest system; movement of the control compresses the spring, and the pilot experiences a force which is proportional to the strength of the spring and the control deflection, but independent of aircraft speed.

b. **'q' Feel**. In the 'q' feel system, resistance to control movement is provided by either actual or simulated dynamic pressure (ie $\frac{1}{2}\rho_0 Ve^2$ or q) and is thus proportional to Ve^2.

c. **V and V3 Feel**. The 'q' feel system can be modified by suitable springs and linkages to give control feel approximately proportional to Ve and Ve^3. The Ve^3 feel system is used on the rudder controls of a few aircraft to prevent the application of large control deflections and corresponding large side-slip angles at high speeds.

d. **g Feel**. The longitudinal control forces can be modified to react to the application of g, by the inclusion of a bob-weight into the control system. This increases the ratio of stick force/g, and prevents excessive g application during manoeuvres.

78. **Slab Surface**. In the slab surface type of control, the hinged flying controls have been dispensed with and the whole of the flying surface moves. Any change in incidence modifies the flow over the whole surface, giving a much greater aerodynamic response. Although this type of control is most commonly found at the tailplane, slab fins have been used and all-flying wing tips fitted to experimental aircraft.

79. **Inboard Ailerons and Spoilers**. In addition to the disadvantages mentioned above, flap-type ailerons have other undesirable features:

a. At high IAS, large ailerons on a thin swept, high-speed section could cause the wing to twist and possibly produce aileron reversal effects.

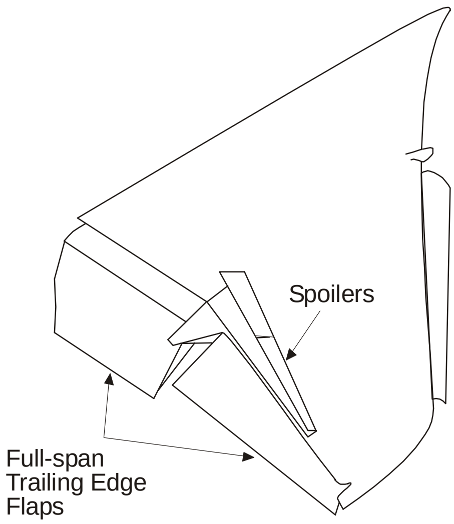
b. Adverse yaw can be significant at either high speed or low speed.

Two methods are used, either separately, or in conjunction, in an attempt to reduce these effects. Inboard ailerons reduce the risk of aileron reversal because of the decreased moment arm and the more rigid wing root section. These ailerons are sometimes supplemented by outboard ailerons for improved low speed handling. Spoilers function by extending into the airflow and causing a breakdown in flow and loss of lift on the required side. The main advantages of this system of roll control are summarized below:

c. The wing trailing edge is left clear for full span flaps (see Fig 48).

d. Because of the lower hinge moments involved, lower control forces are required.

1-22 Fig 48 Spoilers and Full-span Flaps



Ideally, a spoiler should also produce yaw in the direction of the down-going wing (proverse yaw) and thus eliminate the adverse yaw incurred by aileron systems. However, the use of spoilers can incur certain disadvantages which may offset the proverse yaw benefit. These disadvantages are summarized as follows:

e. The effect of the spoiler is non-linear, with a marked reduction in performance at both the high and low ends of the speed scale. At low speeds, ailerons may be required to achieve a safe degree of roll control. Alternatively, a differential tailplane can be used.

f. Where spoilers are used on a swept wing, loss of lift on one wing moves the CP forward and thus introduces a pitch-up moment. This can be removed by suitable interconnections with the elevator, but involves a more complex and heavier control system.

g. At small deflections, the spoiler can cause control reversal. This is because the spoiler acts as a vortex generator and actually increases lift on the wing instead of reducing it. This can be avoided by having a minimum spoiler deflection greater than that which increases the lift, but this can lead to either coarse control for small stick deflections, or a dead range of movement about the stick central position.

CHAPTER 23 - PROPELLER THEORY

Introduction

1. The turbojet produces thrust solely from the jet efflux by accelerating a small mass of air to a high velocity. Given that thrust equals mass $\times$ velocity, the same amount of thrust can be obtained by accelerating a larger mass of air to a lower velocity.

2. The bypass or fan engine is the 'half-way house' between the turboprop and the turbojet. The propeller produces thrust, at subsonic speeds and medium altitudes, more efficiently than the pure jet or the fan-jet because, inter alia, the efficiency of a propulsive system is related inversely to the square of the velocity of the efflux or slipstream.

3. Propellers can be divided into three broad classifications:

a. Fixed pitch.

b. Variable pitch - driven by a piston engine. This type of propeller is governed in flight to an rpm selected by the pilot and may have feathering and reverse thrust capability.

c. Variable pitch - driven by a gas turbine engine. This type of propeller may also have feathering and reverse thrust options.

The aerodynamics are similar for all, but the pitch control is more complex for the turboprop. Hydraulic and mechanical pitchlocks are provided for different operating regimes. In flight, the propeller is governed to run at constant rpm.

4. Mechanical details of propellers and their control mechanisms are examined in Volume 3, Chapter 18.

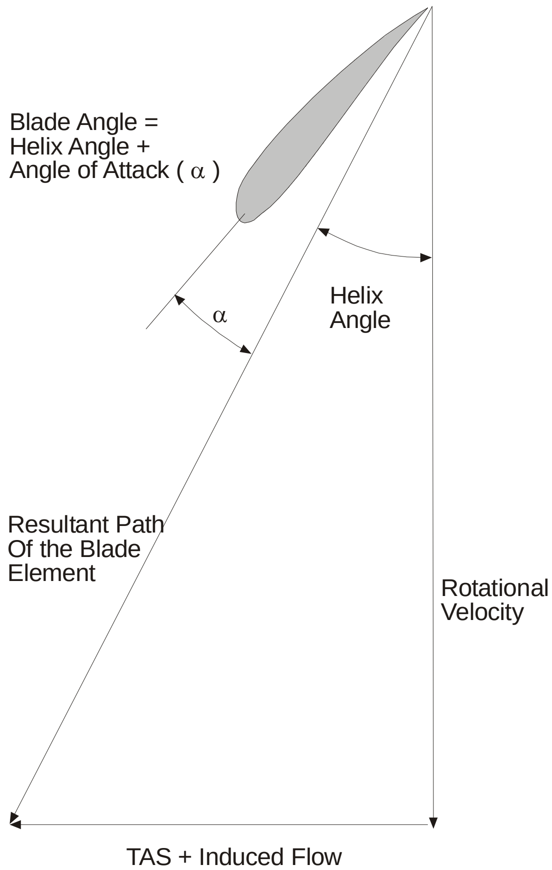
Basic Principles

5. **The Blade Element**. The aerodynamics of the propeller can most easily be understood by considering the motion of an element, or section, of the propeller blade. Because the blade section of a propeller is an aerofoil section, its aerodynamics can be studied in the same way, using the same terms. The helical path of the blade element is compounded of a rotational velocity and a forward velocity.

6. **Rotational Velocity**. When the aircraft is stationary, the motion of the element under consideration is purely rotation. The linear velocity of an element, radius (r) from the centreline, will therefore depend on the angular velocity (N), and the radius, ie rotational velocity = $2\pi rN$. Furthermore, at a given rpm the rotational velocity will increase as r moves towards the propeller tip.

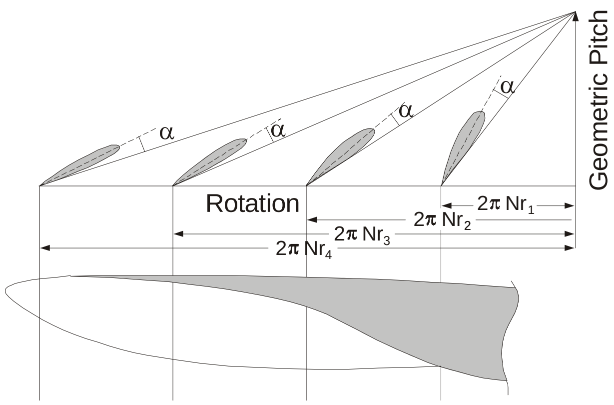
7. **Forward Velocity**. When the propeller is stationary, the motion of the element is entirely due to the forward speed of the aircraft, ie the TAS; but when the blade is rotating, and therefore drawing air through the disc, there is an additional element which is induced flow. For simplicity it will be assumed that the propeller shaft is parallel to the direction of flight. Fig 1 illustrates the resultant path of the blade element under steady cruise conditions. In order to produce the desired aerodynamic force on the element, it is necessary to set the blade at a small positive angle of attack to the resultant relative air flow (RAF). The blade angle, or pitch, of the element is the sum of the helix angle and the angle of attack (α).

1-23 Fig 1 Resultant Motion and Angle of Attack



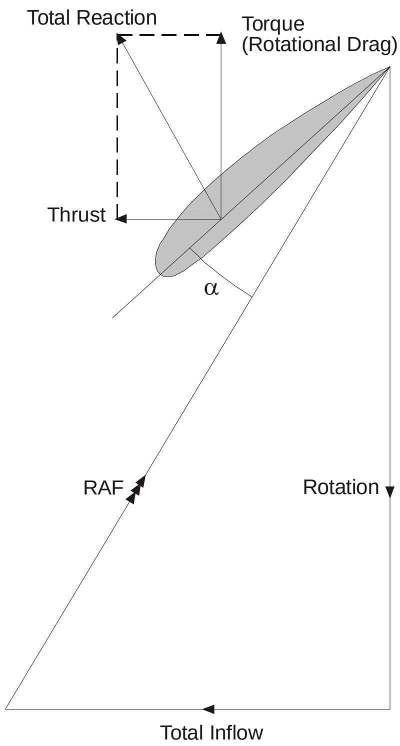
8. **Blade Twist**. Para 6 stated that the rotational velocity depends on the radius of the blade element. It is necessary, therefore, to reduce the blade angle towards the tip in order to maintain an efficient angle of attack along the length of the propeller blade; this is the reason for the helical twist on a blade as shown in Fig 2.

1-23 Fig 2 Blade Twist



9. **Forces on a Blade Element**. The aerodynamic force produced by setting the blade element at a small positive angle of attack (ie the total reaction (TR)) may be resolved with respect to the direction of motion of the aircraft (Fig 3). The component thus obtained which is parallel to the flight path is the thrust force, and that which remains is the propeller torque force. Notice that the propeller torque force is the resistance to motion in the plane of rotation. An effort must be provided to overcome the propeller torque in order to obtain thrust in the same way that the drag must be overcome to obtain lift.

1-23 Fig 3 Forces on a Blade Element



10. **Propeller Efficiency**. The efficiency of any system can be measured from the ratio, Output/Input or, Power out/Power in. The power extracted from this system is the product of Force × Velocity, or Thrust × TAS. The power put into the system is necessary to overcome the rotational drag force and is therefore the product of Propeller Torque (Force) × Rotational Velocity. The efficiency of the propeller can therefore be calculated from the relationship:

$$\text{Propeller Efficiency} = \frac{\text{Thrust} \times \text{TAS}}{\text{Torque} \times 2\pi\text{N}}$$

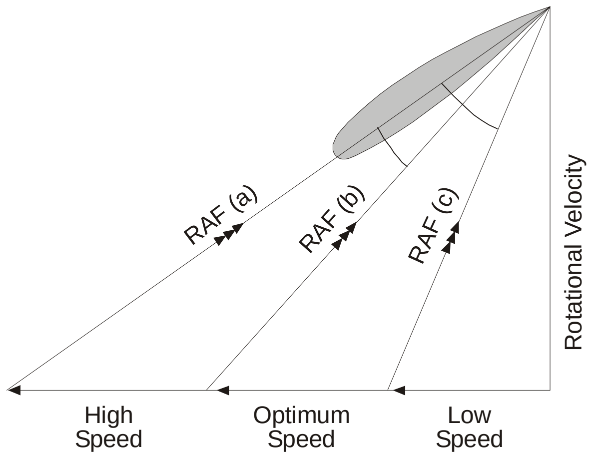
11. **Variation of Propeller Efficiency with Speed**. Fig 4 illustrates a fixed-pitch propeller travelling at different speeds at a constant rpm. If the blade angle is fixed, the angle of attack will change with variations in forward speed. In particular, as speed increases, the angle of attack decreases and with it, the thrust. The effect on the propeller efficiency is as follows:

a. At some high forward speed, the blade will be close to the zero lift angle of attack and thrust will reduce to zero. From the equation in para 10, it will be seen that the propeller efficiency will be zero.

b. There will be only one speed at which the blade is operating at its most efficient angle of attack and where propeller efficiency will be maximum.

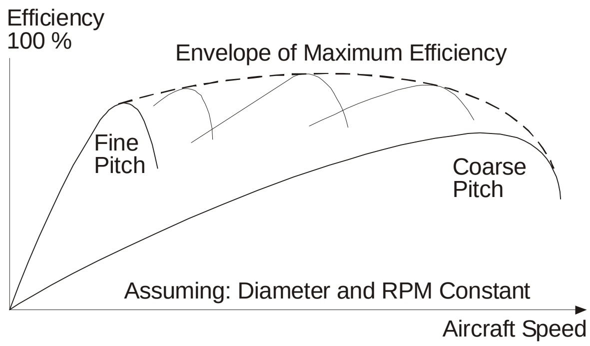
c. At low speeds, the thrust will increase as the angle of attack is increased. Provided that the blade is not stalled, the thrust is very large, but the speed is low and the propeller efficiency is low. Thus no useful work is being done when the aircraft is held against the brakes.

1-23 Fig 4 Effect of Speed on a Fixed Pitch Propeller



These limitations to propeller efficiency led to the development of the two-pitch airscrew. Fig 5 compares the efficiency of a propeller at different pitch settings. The 'fine pitch' position was used for take-off and climb, and the 'coarse' position selected for cruising and high-speed flight. However, it was still basically a fixed-pitch propeller at these settings.

1-23 Fig 5 Efficiency Curves



Propeller Control

12. **Piston Engine**. With fixed pitch, the rpm is dependent on the power setting and the airspeed of the aircraft. Care must be taken to avoid overspeeding the engine in a dive, by throttling back if necessary.

13. **Constant Speed Propellers**. Subsequent development led to the so-called 'constant speed' propeller for piston engines. Power settings for piston engines are defined by rpm and manifold pressure. The constant speed propeller enables the pitch and therefore the rpm, to be selected anywhere between the coarse and fine pitch stops. Once selected, the engine speed is maintained by the pitch control unit (basically, a governor) despite airspeed and power variations. The dotted line in Fig 5 shows the overall improvement in efficiency given by a variable pitch propeller, which can match the maximum efficiency envelope.

14. **Turboprop**. The three types of gas turbine need to be considered:

a. **Fixed Shaft**. The propeller is driven by the shaft carrying the compressor and turbine assembly. The control of fuel flow and propeller rpm must be matched to avoid the risk of surging, over-fuelling or flameout.

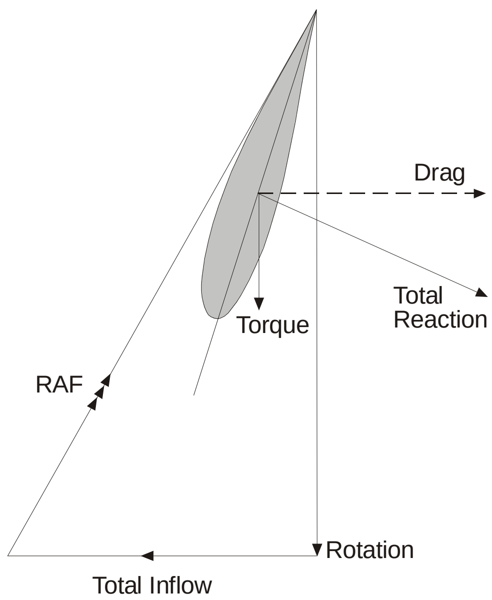
b. **Compound Compressor**. May be regarded as a fixed shaft engine, with similar control of the propeller.

c. **Free Turbine**. Here, the gas generator, comprising the compressor and turbine driving it, is not mechanically connected to the propeller. A separate turbine takes power from the gas generator, to drive the propeller. This gives greater freedom in the control of propeller rpm. The power available is controlled by the throttle or power lever and propeller rpm by a choice of governor settings.

Other Modes

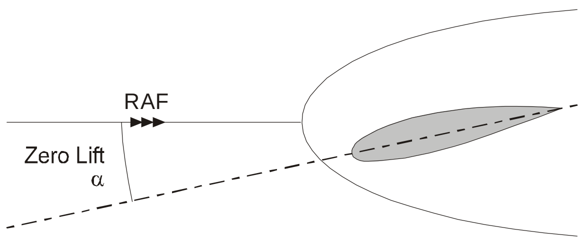
15. **Windmilling**. If the propeller suffers a loss of positive torque, the pitch will fine off in an attempt to maintain the governed RPM selected at the time. The Relative Air Flow (RAF) will impinge on the front surface of the blade section and cause drag and negative torque which will drive the engine (see Fig 6). A very high drag force can be generated if the propeller is directly coupled to the compressor of a gas turbine.

1-23 Fig 6 Forces Acting on a Windmilling Propeller



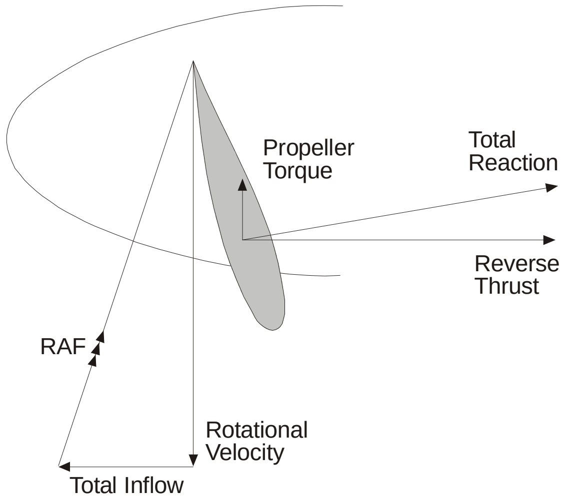
16. **Feathering**. Following engine failure, the windmilling propeller would cause drag which would limit range and degrade performance, perhaps leading to loss of control. Also, by continuing to turn a possibly damaged engine, eventual seizure or fire may result. By turning the blades so that the aggregate effect of the blade section produces zero torque, the propeller is stopped and drag reduced to a minimum, see Fig 7.

1-23 Fig 7 Feathered Blade Section



17. **Reverse Thrust**. If the propeller is turned through the fine pitch stop to around minus 20º and power applied, reverse thrust is obtained. The blade section is working inefficiently, 'up-side-down', analogous with the lift produced from the mainplane in inverted flight. Mechanical devices are used to prevent application of power during the potential overspeed situation as the blade passes beyond the fine pitch stop, until safely in the braking range (see Fig 8).

1-23 Fig 8 Reverse Thrust



Power Absorption

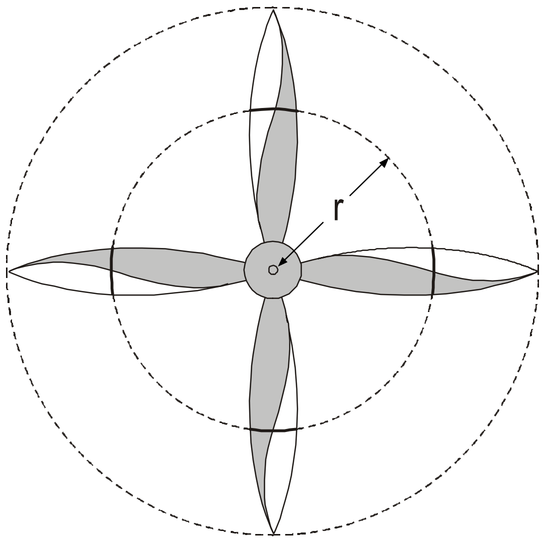
18. The critical factor in matching a propeller to the power of the engine is the tip velocity. Compressibility effects will decrease the thrust and increase the rotational drag, considerably reducing the efficiency of the propeller blade. This consideration imposes a limit on the propeller diameter and rpm, and the TAS at which it can be used.

19. The usual method used to absorb the power of the engine is to increase the 'solidity' of the propeller, i.e. the ratio between that part of the propeller disc which is solid and the circumference at that radius (see Fig 9). Solidity at a given radius is given by the following expression:

$$\frac{\text{Number of Blades x Chord at Radius r}}{\text{Circumference at Radius r}}$$

Note: r is normally equal to 0.7 × tip radius

1-23 Fig 9 Solidity of a Propeller



20. It can be seen that an increase in solidity can be effected by:

a. Increasing the number of blades, or contra-rotating propellers.

b. Increasing the chord of each blade, e.g. 'paddle' blades.

Although the latter is easier, the low aspect ratio of the blades makes a propeller less efficient.

Propeller Twisting Moments

21. Before discussing propeller twisting moments it may be useful to consider centripetal force and centrifugal force.

a. Centripetal force is defined as the force acting on a body, causing it to move in a circular motion that is directed toward the center of axis of rotation.

b. Centrifugal force is defined as the apparent force, equal and opposite to the centripetal force, drawing a rotating body away from the center of rotation, caused by the inertia of the body.

While centripetal force is an actual force, centrifugal force is defined as an apparent force. For example, if one was to spin an object on a string, the string will exert an inward centripetal force on the object causing it to continue in a circular path. At the same time, the mass of the object appears to exert an outward ‘centrifugal’ force on the string. Similarly, with a rotating propeller there is a centripetal force acting towards the center of rotation and an apparent centrifugal force acting opposite to it. Fig 10 illustrates some of the forces acting on a rotating propeller. The force labelled CF, is the apparent centrifugal force as it is opposite to the centripetal force acting towards the center of rotation.

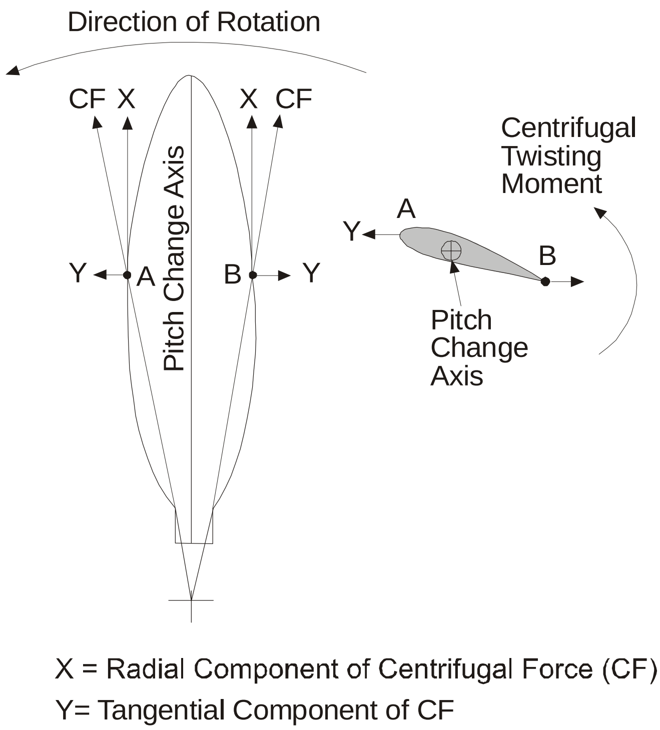
22. Considerable stresses are placed on the propeller and the pitch-changing mechanism in flight. The most important of these are:

a. Centrifugal Twisting Moment (CTM).

b. Aerodynamic Twisting Moment (ATM).

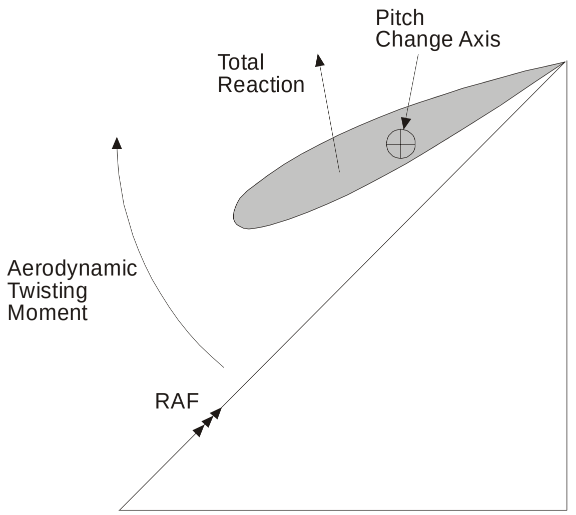
23. **Centrifugal Twisting Moment (CTM)**. Fig 10 illustrates how the apparent centrifugal force on the blade can be resolved into two components. Consider points A and B on the leading and trailing edges respectively. The centrifugal force has one component, X, which is parallel to the pitch change axis and component Y which is at right angles to the pitch change axis. Component X produces tensile stress along the blade but, because points A and B are not in the same plane, component Y produces a moment around the pitch change axis which tends to 'fine' the blade off. This is the centrifugal twisting moment. The effect of the CTM is to make it more difficult for the pitch change mechanism to increase the pitch of the propeller. It can be seen from Fig 10 that a wider blade would increase the moment arm of A and B about the pitch change axis and therefore increase the CTM. If it should be necessary to increase propeller solidity it is therefore preferable to increase the number of blades rather than to increase the blade width.

1-23 Fig 10 Centrifugal Twisting Moment



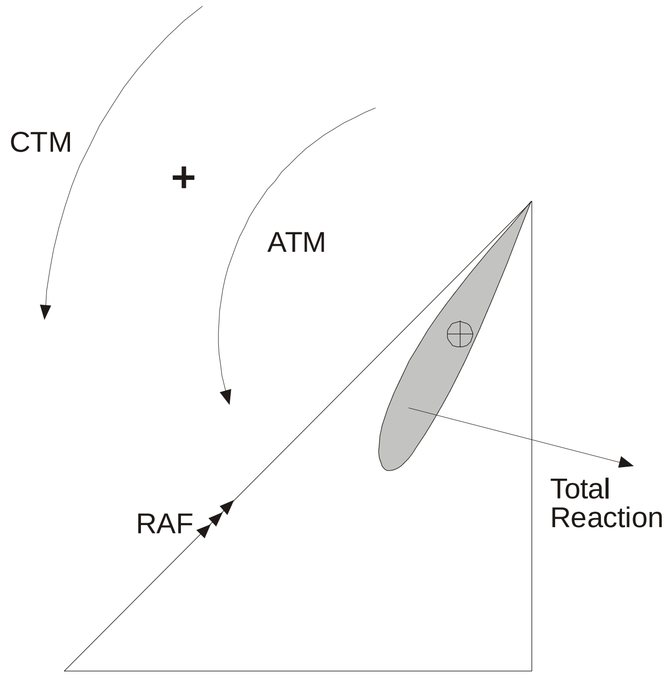
24. **Aerodynamic Twisting Moment (ATM)**. Fig 11 illustrates how the aerodynamic twisting moment is produced. If the pitch-change axis is behind the centre of pressure of the blade, the torque on the blade will be increased by an increase in angle of attack and vice versa. It can be seen that the ATM tends to coarsen the pitch and therefore partially offsets the CTM under normal operating conditions.

1-23 Fig 11 Aerodynamic Twisting Moment



25. **Windmilling Effect on ATM**. When the propeller is windmilling in a steep dive however, the ATM is reversed and acts in the same direction as the CTM (see Fig 12). The cumulative effect may prove a critical factor in the successful operation of the pitch-change mechanism.

1-23 Fig 12 Twisting Moments on a Windmilling Propeller



Swing on Take-off

26. There is often a tendency for a propeller-driven aircraft to 'swing' to one side on take-off. The causes of swing on take-off for a conventional nose-wheel aircraft are:

a. Slipstream effect.

b. Torque reaction.

c. Cross-wind (weathercock) effect.

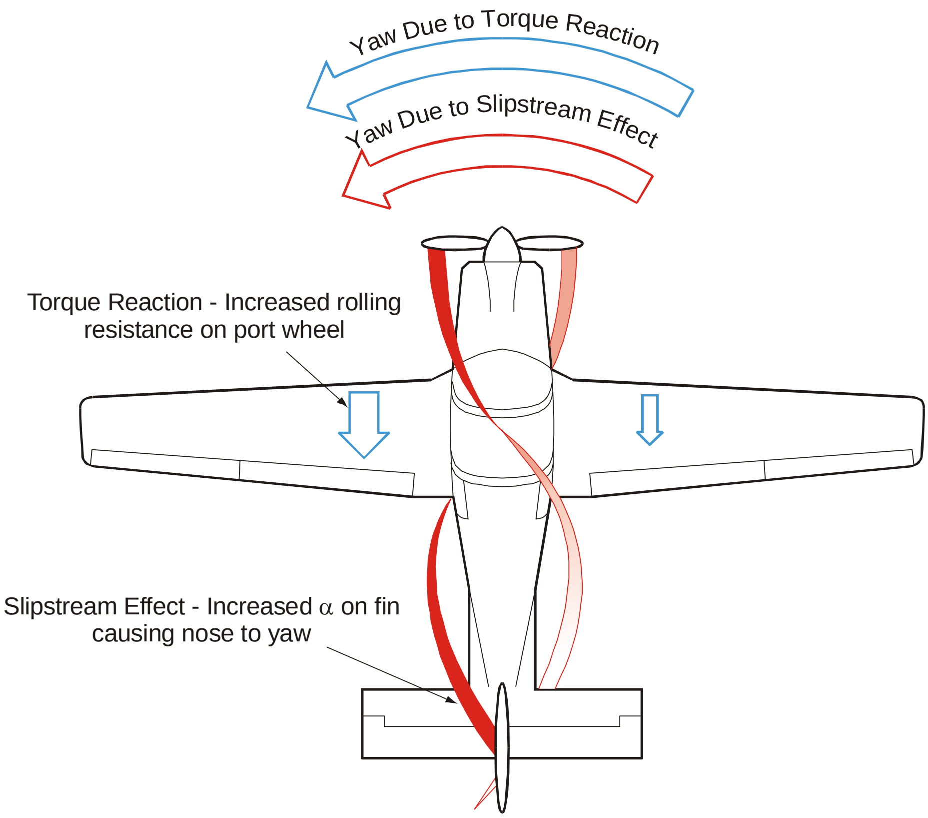
Because the cross-wind effect is not a purely propeller induced cause of swing on take-off, it is not discussed further in this chapter; reference to it will be found in Volume 8, Chapter 13. Fig 13 illustrates the other effects which are described in the following paragraphs.

27. **Slipstream Effect**. A propeller which is rotating in a clockwise direction, viewed from behind, will impart a rotation to the slipstream in the same sense. This rotation produces an asymmetric flow over the fin and rudder such as to induce an aerodynamic force to the right. This, in turn, will cause the aircraft to yaw to the left. A propeller rotating anti-clockwise, viewed from behind, will generate these same forces in the opposite direction producing a yaw to the right.

28. **Torque Reaction**. If the propeller rotates clockwise, viewed from behind, the torque reaction will tend to rotate the aircraft in the opposite sense, ie roll to the left. The rolling motion is prevented by the wheels being in contact with the ground and this results in more weight being supported by the port wheel than the starboard wheel; this will increase the rolling resistance of the port wheel. Consequently the aircraft will tend to swing to the left until the wings take the weight off the main wheels. Once again, for propellers rotating in the opposite direction, a swing to the right will occur.

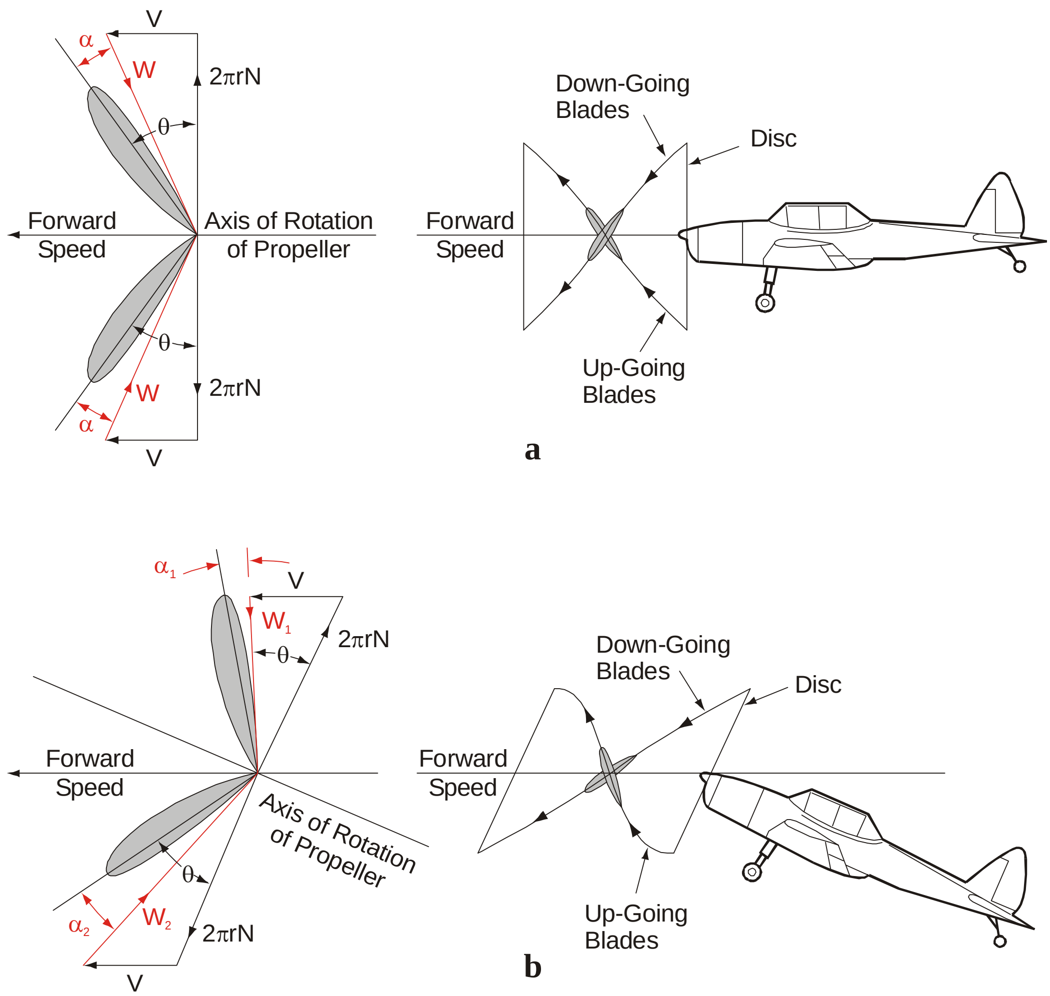
1-23 Fig 13 Slipstream Effect and Torque Reaction

Propeller Rotating Clockwise (viewed from the Cockpit)



29. **Tailwheel Aircraft**. A tail wheel aircraft will be subject to two further effects, both tending to cause the aircraft to yaw on take-off. During the initial part of the take-off run, while the tail is down, the plane of rotation of the propeller will not be at right angles to the direction of forward motion. Fig 14 illustrates the difference in propeller blade angle of attack due to the vector addition of forward speed (horizontal) on rotational velocity (in the plane of the propeller disc). In Fig 14a, where the propeller axis is in the line of flight, the angles of attack and the relative airflows (shown in red) on both propeller sections are equal. Also, the distances travelled in unit time by both the downgoing and upgoing blades are equal. Fig 14b shows, in an exaggerated form, how the angles of attack and resultant velocities are changed when the axis of rotation is inclined. The downgoing blade has a higher angle of attack and is, therefore, producing more thrust than the upgoing blade. Also, the distance travelled in unit time by the downgoing blade is greater than that for the upgoing blade. This means that the downgoing blade has a higher speed relative to the airflow than the upgoing blade and will, for a given angle of attack, again produce more thrust. This is known as asymmetric blade effect. For an anti-clockwise rotating propeller (viewed from the cockpit) the downgoing blade will be on the left of the propeller disc and the aircraft will tend to yaw to the right. This yaw will be in the opposite sense if the propeller direction is reversed. As the tail is raised, asymmetric blade effect will be reduced (to zero when the disc is perpendicular to the horizontal path of the aircraft) but, in so doing, a force is then applied to the top of the propeller disc in a nose down sense. The resulting gyroscopic precessional force (applied 90º in the direction of rotation) will also cause a yaw to the right (for anti-clockwise rotations) and to the left (for clockwise rotations). This is known as gyroscopic effect.

1-23 Fig 14 Tailwheel Aircraft - Asymmetric Blade Effect



r	= Radius	α_1	= Reduced Angle of Attack on Upgoing Blade
N	= Angular Velocity	α_2	= Increased Angle of Attack on Downgoing Blade
V	= Total Inflow	W_1	= Reduced Relative Airflow on Upgoing Blade
θ	= Blade Angle	W_2	= Increased Relative Airflow on Downgoing Blade

30 **Sum of Forces producing Yaw on Take-off**. It will be noted that the effects causing a swing on take-off all produce yaw in the opposite direction to the direction of rotation of the propeller. However, all may be countered by correct application of rudder and become more manageable as speed is increased. The totality of effects will be more noticeable in tailwheel aircraft.