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1

Properties of the Atmosphere

1.1 The Atmosphere

The gaseous envelope surrounding the Earth is called the atmosphere. There is no defined upper limit to the atmosphere, but much of this study is limited to the first 60,000 ft where most aviation activity is conducted.

1.2 Gas Composition

Gases are found in the atmosphere in the following proportions by volume:

Nitrogen	78%
Oxygen	21%
Other gases	1% (eg argon, carbon dioxide, water vapour)

Oxygen is essential for the sustenance of life and the combustion of materials. In the context of aviation, oxygen is required for the combustion of fuel, a deficiency of this gas resulting in incomplete burning and reduced engine efficiency.

Water vapour is present in the atmosphere in varying proportions, and is responsible for the weather around the earth, which in turn affects aircraft operations and performance. Additionally the presence of water vapour may cause icing of the airframe or engine which may impair an aircraft's performance.

1.3 Regions of the Atmosphere

The atmosphere is divided into a number of layers:

(a) *The Troposphere* – where temperature decreases with increase of height. In this region nearly all significant weather occurs.

(b) *The Tropopause* – the upper limit of the troposphere where temperature stops decreasing with an increase of height. The tropopause is therefore the *upper limit of significant* weather, the *first point of* *lowest temperature*, and additionally it is the region for *maximum wind* strengths.

The height of the tropopause varies with latitude, season of the year, and prevailing weather conditions with the result that it is usually higher in low latitudes, in summer and in fine weather.

Typical heights for the tropopause are:

Latitude	*Tropopause Height*	
Equator	16–17 km	53,000–57,000 ft
45°N/S	10–12 km	33,000–39,000 ft
Poles	7½–9 km	25,000–29,000 ft

(c) *The Stratosphere* extends from the tropopause to approximately 50 km amsl, and is characterised by the temperature being steady or increasing with height.

(d) *The Mesosphere* extends from 50 km to 80 km. The temperature generally decreases with height.

(e) *The Thermosphere* or *Ionosphere*, where temperature increases with height.

1.4 Temperature

(a) Units

The temperature scales most commonly used are Celsius or Centigrade, Fahrenheit and Kelvin or Absolute.

The first two scales are based on the melting point of ice, being 0°C and 32°F respectively, and the boiling point of water, being 100°C or 212°F.

Being a form of energy, heat is related to the random movement of molecules in a substance. If heat is reduced, the molecules become less active. The minimum temperature to which a substance can be reduced is approximately -273°C, and this is known as Absolute zero, or 0°K. Correspondingly, the melting point of ice is equivalent to 273°K and the boiling point of water to 373°K.

To convert from one temperature scale to another, the following formulae may be used:

$$F = \frac{9C}{5} + 32$$

$$C = \frac{5}{9}(F - 32)$$

$$K = C + 273$$

(b) Temperature Variation in the Troposphere

At ground level, in general, the temperature increases with decrease of latitude.

With increasing altitude, the conductive and convective effects from the earth are reduced so that temperature will usually decrease with height up to the tropopause. See Fig. 1-1.

Typical values of temperature found at the tropopause are:

Latitude	*Temperature*
Equator	-80°C
45°N/S	-56°C
Poles	-45°C

There is, therefore, a reversal of temperatures with latitude in comparison to those found at ground level. This is partly because the tropopause is higher at the equator and the temperature decrease is effective over a greater height.

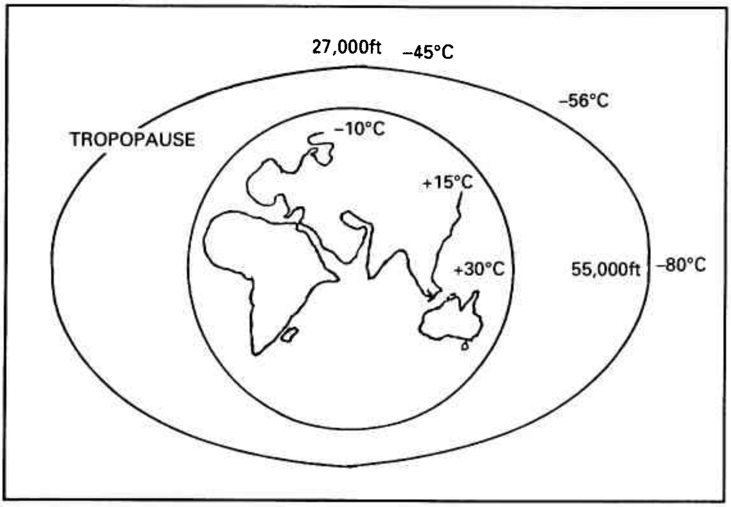


Figure 1–1

(c) Lapse Rates

The temperature decrease with increase of height is referred to as lapse rate.

A representative value of 2°C/1000 ft is a typical value for the troposphere, and this figure is used as the reference for the Jet Standard Atmosphere (JSA).

The International Standard Atmosphere (ISA) uses the comparable value of 1.98°C/1000 ft.

For meteorological purposes, differentiation between dry (that is, not saturated) and saturated adiabatic lapse rates is made, and the values of 3°C/1000 ft and 1.5°C/1000 ft respectively are used. The difference of lapse rate for saturated air is caused by the release of latent heat during condensation, thus reducing the temperature change.

(d) Temperature and Aircraft Performance

At a given pressure, an increase of temperature results in a reduction of density.

Firstly, considering airframe performance, a reduction of density (ρ) reduces lift (L). This may be counteracted by increasing the true airspeed (v) to achieve the required amount of lift (L):

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

where: C_L = coefficient of lift Thus: $C_L \ \tfrac{1}{2} \rho V^2 S$

and S = surface area

The dynamic pressure is gained at the expense of an increased take-off run, cruising TAS or landing run according to the stage of flight.

On the credit side, drag (D) reduces with increase of temperature:

$$D = C_D \ \tfrac{1}{2}\rho V^2 S$$

A piston engine's performance is related to the temperature of the air being drawn into the cylinder head. The higher the temperature, the lower the density and weight of fuel/air mixture that can be burnt in the combustion chamber. The power output of the engine therefore falls with increase of temperature.

For a propulsion system, piston or jet,

Thrust = Mass of air × Acceleration to which air is subjected

Thus an increase of temperature will reduce the mass flow and, therefore the thrust.

1.5 Pressure

(a) Definition

Pressure is the force exerted on a unit area, ie:

$$\text{Pressure} = \frac{\text{Force}}{\text{Area}} = \frac{\text{Mass} \times \text{Acceleration}}{\text{Area}}$$

In the atmosphere, pressure is caused by the mass of the gaseous molecules acting under the force of gravity on a given area. As all molecules act under gravity then the pressure can also be considered to be the weight of a column of air on a unit area.

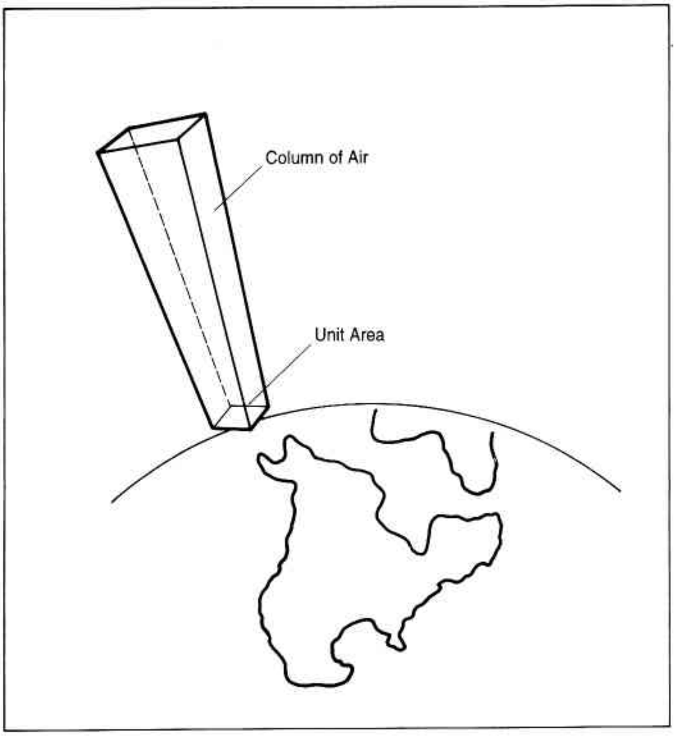


Figure 1–2

(b) Units

The metric units of pressure are dynes per square centimetre, where the dyne is the force required to accelerate 1 gram by 1 centimetre per second.

The System International units of pressure are Newtons per square metre, where the Newton is the force required to accelerate 1 kilogram by 1 metre per second. The Newton is therefore, equal to 10^5 dynes.

Although largely obsolete, the Imperial system of units is still encountered, and pressure is expressed in pounds per square inch.

In meteorology the unit of pressure is the millibar (mb), which is equivalent to 1000 dynes per square centimetre.

Before the introduction of the millibar, meteorological pressure was measured in terms of the length of a column of mercury in a barometer that the weight of the atmosphere could support.

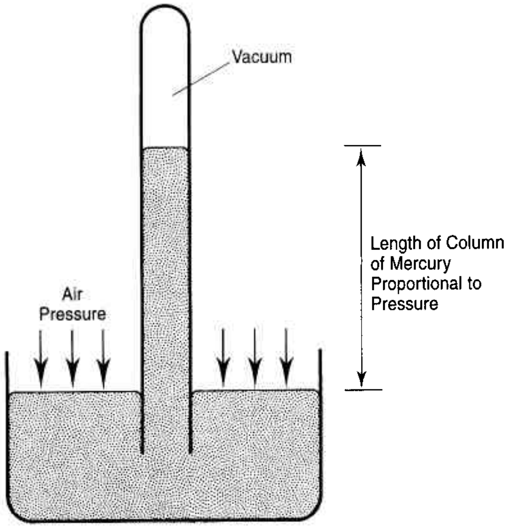


Figure 1–3

(c) Variation of Pressure in the Atmosphere

At sea level, pressure generally varies between 950 and 1050 mb. In tropical revolving storms and tornadoes, however, pressures may fall much lower.

With increasing altitude the mass of overlying air decreases and so the pressure falls. Pressure values of the International Standard Atmosphere are given below:

Altitude (ft)	*Pressure* (mb)	*Pressure* (psi)	*Pressure* (in HG)	*Pressure* (mm HG)
40,000	187.6	2.72		
30,000	300.9	4.36		
20,000	465.6	6.75		
10,000	696.8	10.11		
0	1013.25	14.7	29.92	760

From the table it should be noted that at about 18,000 ft, the pressure is half the sea level value.

Also, it should now be apparent that the rate of pressure decrease with height is not constant. In the first 10,000 ft, the pressure falls at a rate of approximately 1 mb per 30 ft but between 30,000 ft and 40,000 ft the pressure decrease is closer to 1 mb per 88 ft.

(d) Pressure Altitude

The altitude at which a given pressure occurs in the International Standard Atmosphere is called the pressure altitude.

If, for example, the pressure at the top of Mount Everest were determined as 300.9 mb, then the pressure altitude would be 30,000 ft.

Assuming the same mean sea level conditions, and two columns of air of the same height, but differing temperatures, then the cold air would have a greater mass than the warm air due to the density difference. The pressure of the atmosphere, however, is caused by the mass of overlying molecules on a unit area. The pressure above the column of warm air is therefore higher than that above cold air. Because a higher pressure is found at a lower level, then the pressure altitude above warm air is lower than the pressure altitude above cold air. Alternatively it can be expressed that the true altitude of an aircraft is more than that indicated (assuming the correct mean sea level pressure has been set on the subscale) above warm air, and less than that indicated above cold air. (Fig 1-4)

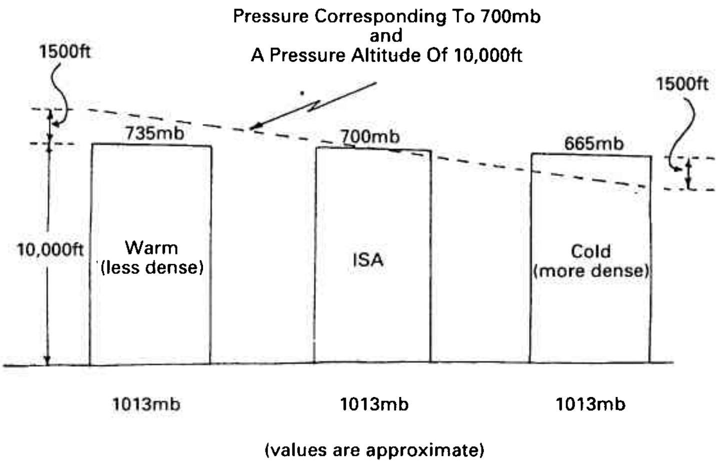


Figure 1–4

1.6 Density

(a) Definition

Density is the mass per unit volume of a substance, at a specified temperature and pressure.

$$\text{Density} = \frac{\text{Mass}}{\text{Volume}}$$

(b) Units

Density is expressed in grams, or kilograms per cubic metre for metric or SI units, respectively. The Imperial units are pounds per cubic feet.

Factors affecting density when considering a gas are:

$$\text{Density} = \frac{\text{Pressure}}{\text{Gas constant} \times \text{Absolute temperature}}$$

For a given temperature, therefore, an increase of pressure increases density, or, at a given pressure, a decrease in temperature increases density.

(c) Variation of Density in the Atmosphere

At sea level, densities vary between 1.20 and 1.55 kg per cu m, the higher values being usually associated with the colder temperatures of higher latitudes, and the lower values typical of Equatorial latitudes.

Air at lower levels in the atmosphere is compressed by the mass of the overlying air. With increasing altitude, the overlying mass reduces and air can now expand, resulting in further reduction of pressure.

With increasing altitude the temperature also decreases, but at a rate lower than the pressure. Density, therefore, decreases with height.

Density values of the International Standard Atmosphere are shown below:

Altitude [ft]	*Density* [kg/cu m]	*Density* [lb/cu ft]
40,000	0.302	0.019
30,000	0.458	0.029
20,000	0.653	0.041
10,000	0.905	0.056
0	1.225	0.077

At about 22,000 ft, the density is half the sea level value.

We have already seen that density at sea level tends to be higher at the Poles than at the Equator. However, at 26,000 ft, the density value is similar at all latitudes.

(d) Variation of Density with Humidity

The total pressure of the atmosphere is equal to the sum of the individual pressures of the gases. The pressure of moist air is less than that for dry air, and so humidity decreases the total pressure. From the gas equation, it can be seen that the reduction in pressure results in a lower density. The greater the humidity, the lower the density.

(e) Density Altitude

This is defined as the altitude in the International Standard Atmosphere at which a given density is found.

Aircraft performance is largely dependent on density altitude as opposed to true or pressure altitude.

(f) Density and Performance

The effects of density on lift, drag, power and thrust have been considered in the section concerning temperature.

There are, however, additional effects of density performance.

Above about 300 kt TAS, air becomes significantly compressed, and locally increases the density. At much higher speeds this may give a marked increase in drag, and when increasing altitude, this can offset the otherwise reducing drag value.

A similar compressibility effect increases drag on a propeller blade, reducing its efficiency, particularly at higher altitudes.

A jet engine's performance, however, is enhanced by this compressibility effect as mass flow is improved.

(g) Air Density and the Human Body

The reduced density of air with increasing altitude means that in a given volume of air breathed in, the oxygen content has decreased. Above 10,000 ft this reduction leads to hypoxia, its effects ranging from lack of judgement to sleepiness or collapse, according to height.

At night, the reduced intake of oxygen impairs night vision at altitudes of 4,000 ft and above.

To counter these problems, aircraft operating above 10,000 ft must have an enriched oxygen supply, either in conjunction with a pressurised cabin, or through face masks. At night, ideally, oxygen should be available from ground level upwards.

1.7 Performance Ceilings

(a) Service Ceiling

This is defined as the altitude at which the rate of climb of an aircraft falls to a specified figure, usually 100 ft per minute.

(b) Absolute Ceiling

The absolute ceiling is the altitude at which the rate of climb of an aircraft falls to zero.

(c) Piston-Engined Aircraft

For such aircraft, operating under 26,000 ft, then the improved atmospheric density found in winter in high latitudes will give the highest ceiling.

(d) Jet-Engined Aircraft
As most jet-engined aircraft operate above 26,000 ft, then the best performance ceiling will be found at the highest tropopause and lowest temperature, ie in summer, and at low latitudes.

1.8 The Gas Laws

Introduction
Whilst air is not an ideal gas, it does conform, within close limits, to the results of Boyle's and Charles' laws.

(a) Boyle's Law
The volume (V) of a given mass of gas at constant temperature is inversely proportional to pressure (P):

$$V \times \frac{1}{P} \text{ or } PV = \text{constant}$$

This can be expressed in the form:

$$P_1 V_1 = P_2 V_2$$

(b) Charles' Law
The volume of a given mass of gas at constant pressure, increases by 1/273 of its volume at 0°C for every 1°C rise in temperature:

$$V \times K \text{ or } \frac{V}{K} = \text{constant}$$

The alternative expression below is also useful:

$$\frac{V_1}{K_1} = \frac{V_2}{K_2}$$

(c) Combined Boyle's and Charles' Law Equation
The results of both laws may be combined in one equation, expressing the behaviour of a gas under varying conditions of pressure, volume and temperature:

$$\frac{P_1 V_2}{K_1} = \frac{P_2 V_2}{K_2}$$

1.9 The International Standard Atmosphere

In order to provide a datum for aircraft performance comparison, and instrument calibration, an assumed set of conditions has been determined. Whilst representative, these conditions do not necessarily reflect actual conditions in the atmosphere. The values used are listed below:

(i) Temperature 15°C at msl, and decreasing at 1.98°C per 1,000 ft to 36,090 ft (11 km) where the temperature remains constant at -56.5°C until 65,617 ft (20 km).

(ii) Pressure 1013.25 mb at msl.

(iii) Density 1.225 kg/cu m at msl.

1.10 Speeds

(a) *Indicated Airspeed* (IAS) The dynamic pressure of air against a vehicle, or indicated airspeed, is equal to $\frac{1}{2}\rho V^2$, where ρ = density, and V = true airspeed. An airspeed indicator, calibrated to ISA, mean sea level conditions records the dynamic pressure as a speed. If, for example, the indicated reading were 200 kt, then it means that the dynamic pressure is the same as it would be at a true air speed of 200 kt at standard conditions at mean sea level.

(b) *Rectified Airspeed* (RAS) The indicated airspeed, corrected for instrument and position errors (IE and PE).

(c) *Equivalent Airspeed* (EAS) The rectified airspeed corrected for compressibility (C). It should be noted that compressibility is always a subtracted quantity.

(d) *True Airspeed* (TAS) The equivalent airspeed corrected for density.

(e) *Calibrated Airspeed* (CAS) Some airspeed indicators are corrected for mean sea level compressibility. Calibrated airspeed is the value of this reading, corrected for instrument and position errors.

(f) *Mach Number* (Mn) Mach number is the ratio of TAS to the local speed of sound (LSS).

Chapter 1: Test Yourself.

1 With increasing altitude pressure decreases and:

a) temperature decreases at the same rate as pressure reduces.
b) temperature decreases but at a higher rate than pressure reduces.
c) temperature decreases but at a lower rate than pressure reduces.
d) temperature remains constant to 8,000 ft.

Ref para 1.4

2 Density = :
a) $\frac{\text{Mass}}{\text{Volume}}$
b) $\frac{\text{Volume}}{\text{Mass}}$
c) Volume × Mass
d) Mass × Temperature

Ref para 1.6

3 Total pressure of air will:
a) not be affected by temperature.
b) increase with increased humidity.
c) reduce with increased humidity.
d) not be affected by moisture.

Ref para 1.5

4 A reduction in air pressure results in:

a) no significant change in density.
b) a reduction in density.
c) an increase in density.
d) erratic variations in density.

Ref para 1.6

5 The absolute ceiling of an aircraft is the altitude at which the:

a) rate of climb falls to zero.
b) rate of climb falls to 50ft/min.
c) rate of climb falls to 100ft/min.
d) rate of climb has a negative value.

Ref para 1.7

2

Aerodynamics – Basic Principles of Airflow

2.1 Atmospheric Pressure

In the previous chapter it was shown that the atmosphere exerts pressure at all times. This type of pressure, which exerts a force on all bodies, is called *static pressure* and acts equally in all directions. When air is in motion, however, it possesses an additional energy (kinetic energy) due to the fact that it is moving, and the faster it moves the more kinetic energy it has. If moving air is now brought to rest against some object, the kinetic energy is turned into pressure energy. This pressure on the surface of the body which causes the moving air to stop is called *dynamic pressure*. The value of dynamic pressure depends on the density of the air and its speed and may be expressed as:

$$\text{Dynamic pressure} = \tfrac{1}{2}\rho V^2$$

This is an important equation which affects all aerodynamic studies. As shown in Fig 2-1 any object in still air will experience static pressure in all directions but an object which is moving, or is placed in a moving airstream, will experience an additional pressure due to the moving air being brought to rest.

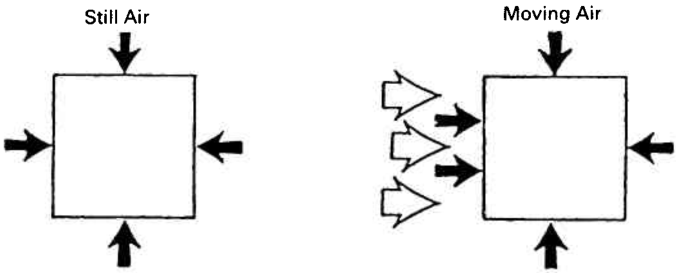


Figure 2–1

If the speed of the moving air is comparatively slow, say 100 kt, the dynamic pressure exerted by it is quite small in relation to the static pressure at sea level. In fact the dynamic pressure will only amount to less than 2% of the static pressure. If, however, the speed is increased to, say 450 kt, the dynamic pressure rises considerably, to about 30% of the static pressure. It is important to note that at low speeds the density of the air is not significantly affected by these changes in pressure and the air can be considered as an incompressible fluid. At high speeds, however, say in excess of 300 kt, this assumption can no longer be made and the changes in density due to compressibility become significant.

2.2 Streamline Flow

It is useful to illustrate the path followed by air when it passes around fixed objects and the idiom used is that of streamlines. A streamline is the path traced out by a single particle of airflow such that this particle does not cross the path of any other. This can be illustrated by dropping dye into a stream of water and watching the visible path of the dye when it moves with the water. Streamlines are illustrated in Fig 2-2.

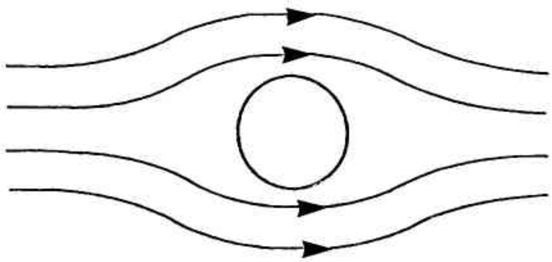


Figure 2-2 Streamlines

2.3 Flow Continuity

When water flows down a tube the principle of continuity of flow applies and the mass flow in the tube is the same at any point along its length. This rule applies even if the tube is not of constant diameter and this is clearly shown in the diagram at Fig 2-3. The mass flow at A, B and C is the same so if the density of the water is p the cross sectional area of the tube 'a' and the speed of the water is 'v' then:

$$\text{Mass flow} = pav$$

The continuity theorem states that the mass flow at any point A = the mass flow at point B = the mass flow at point C.

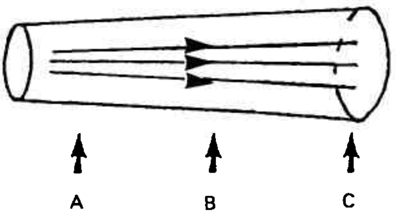


Figure 2–3 Mass Flow

2.4 Venturi Effect

In a venturi tube, that is a tube that has a constriction in it, as shown in Fig 2-4, the previous rule still applies; mass flow is always a constant even if the tube is not of constant diameter. If, therefore, the pressure is measured at points 1 and 2 in the venturi, it can be said:

$$p_1 a_1 v_1 = p_2 a_2 v_2$$

Considering the fluid as incompressible, then as the cross sectional area a_1 is considerably bigger than the cross sectional area a_2 the speed v_1 must be less than the speed v_2. In other words, as the flow passes through

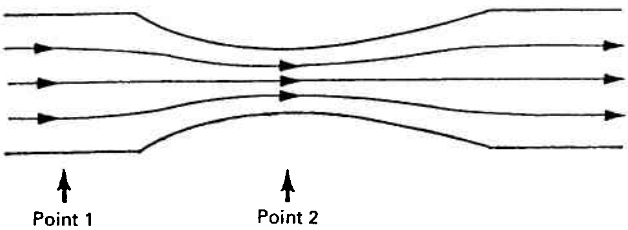


Figure 2–4 Venturi

the venturi the speed of the fluid increases. This can often be seen when watching the flow of a river through the arch of a bridge observing how the water speeds up as it flows through the arch or constriction. The streamlines associated with this flow are shown in Fig 2-4. It can be seen that the streamlines draw closer together as they pass through the venturi.

Moving away from the concept of the tube to that of an aircraft wing, as illustrated in Fig 2-5 it can be seen that due to the curvature of the wing on its upper surface a venturi has been created between the upper surface and the undisturbed air some distance above it. The streamlines will be similar to those in Fig 2-4 and, of course, the flow of the air will be increased in speed as it passes through the venturi.

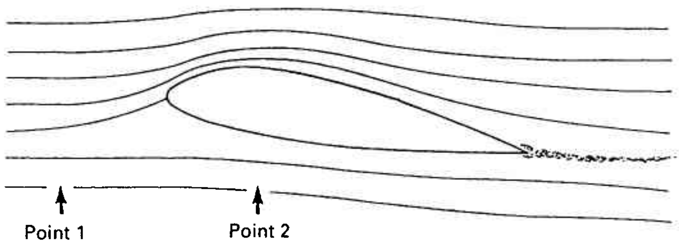


Figure 2–5 Aerofoil

(a) Bernoulli's Theorem.

During the last century Bernoulli put forward his theorem stating that the total pressure (ie static + dynamic) in a fluid is constant if no work is done by it or on it.

$$\text{Total pressure } H = S + \tfrac{1}{2}pV^2 = \text{constant.}$$

Referring back to Fig 2-5 and looking at the point ahead of the wing marked 1 we can find the total pressure at this point:

$$H_1\ S_1 + \tfrac{1}{2}p_1V_1^2.$$

Similarly the total pressure at the point marked 2 can also be expressed as:

$$H_2 = S_2 + \tfrac{1}{2}p_2V_2^2$$

However, Bernoulli's theorem states that the total pressure in a fluid is constant, therefore these two expressions must equal each other. Therefore:

$$S_1 + \tfrac{1}{2}p_1V_1^2 = S_2 + \tfrac{1}{2}p_2V_2^2$$

Considering the density to be a constant factor and knowing the speed at point 1 is less than the speed at point 2, it follows that the pressure at point 1 must be higher than the pressure at point 2. To put it differently there is a reduction in pressure over the upper surface of the wing as a result of Bernoulli's Theorem. It is this reduction in pressure over the upper surface of the wing of an aircraft that creates lift and is the reason an aircraft can fly.

2.5 Stagnation

Referring to Fig 2-6 note the flow of air around an object. Notice how the air divides – some flows over the top of the wing and some below it and right in the centre, at the leading edge of the wing, the air is brought completely to rest at point A. This point is called the stagnation point and it is where the full dynamic pressure plus whatever static pressure is effective at the time will be felt.

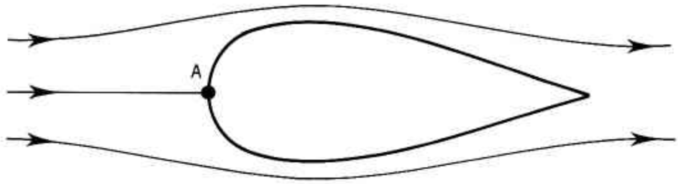


Figure 2–6 Stagnation Point

2.6 Measuring Airspeed

The principle of the stagnation point is used in the measurement of airspeed. Air is directed from a pitot tube facing into the airflow to a flexible diaphragm in the airspeed indicator. This flexible diaphragm, in the form of a capsule, in fact is a stagnation point and will feel the full effect of dynamic pressure. Static pressure is fed to both sides of the capsule so that it cancels out. The resultant movement of the diaphragm can be taken by a suitable linkage to a dial, thus indicating airspeed. It should be noted that the airspeed indicator is in fact a dynamic pressure indicator but is calibrated suitably in knots. As it measures dynamic pressure directly it is extremely useful when flying the aircraft as most aerodynamic functions of the aircraft are directly related to dynamic pressure. For instance, the stalling speed of an aircraft is always measured in indicated airspeed and remains, for the same weight, pretty well a constant figure regardless of altitude. No mention has been made yet of compressibility and in fact this should be taken into account. The airspeed indicator reading (corrected for instrument and position errors), when corrected for compressibility at all speeds is called equivalent air speed (EAS).

Chapter 2: Test Yourself.

1 The airflow over the upper surface of a cambered wing:

a) increases in velocity and pressure.
b) increases in velocity and reduces in pressure.
c) reduces in velocity and pressure.
d) reduces in velocity and increases in pressure.

Ref para 2.4

2 As the camber of an aerofoil section is increased:

a) velocity of the airflow is decreased.
b) pressure over the upper surface is decreased.
c) pressure over the upper surface is increased.
d) pressure over the upper surface remains the same for any camber.

Ref para 2.4

3 The stagnation point on an aerofoil in flight is:

a) located at the point of deepest section.
b) air at rest at the section leading edge.
c) air at rest between the trailing edge streamlines.
d) air at rest on the upper surface of the wing.

Ref para 2.5

4 The stagnation point is:

a) static pressure plus dynamic pressure.
b) static pressure minus dynamic pressure.
c) static pressure only.
d) dynamic pressure only.

Ref para 2.5

5 In general terms 'Lift' is a result of:

a) an increase of pressure under the wing.
b) a reduction of pressure over the wing upper surface.
c) a reduction of pressure over the upper and lower surfaces.
d) an increase of pressure above and below the wing.

Ref para 2.4

3

Aerodynamics – Aerofoils and Actions (Definitions)

3.1 Chord Line

The chord line of an aerofoil is the straight line joining the leading edge to the trailing edge. It is normally used as a reference line when measuring the angular position of the wing related to the airflow. Fig 3-1

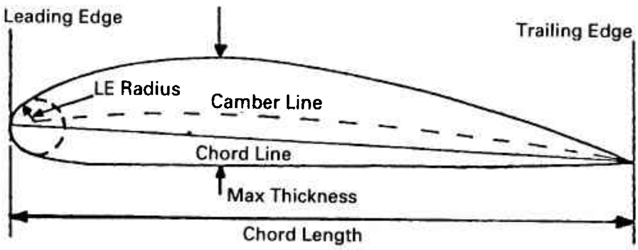


Figure 3–1

3.2 Mean Camber Line

A line which joins the leading edge to the trailing edge such that it is equidistant from the upper surface and lower surface of the aerofoil. If it is curved the aerofoil is described as cambered. Fig 3-1

3.3 Thickness/Chord Ratio ("Fineness Ratio")

This is the ratio of the maximum thickness of the cross section to the chord, and is usually expressed as a percentage. Fig 3-1

3.4 Angle of Attack

The angle of attack is the angle between the chord line of the wing and the direction of the relative airflow. Fig 3-2

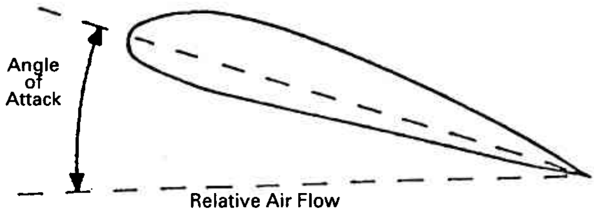


Figure 3–2

3.5 Angle of Incidence

This is the angle at which the aerofoil is attached to an aircraft fuselage when the aircraft is in rigging position.

The term rigging position is essentially an aircraft engineer's term which means the aircraft is jacked clear of the ground and is laterally and longitudinally in the attitude it would possess in level flight.

3.6 Wash Out

A decrease in wing angle of incidence from root to tip.

3.7 Wash In

An increase in angle of incidence from root to tip.

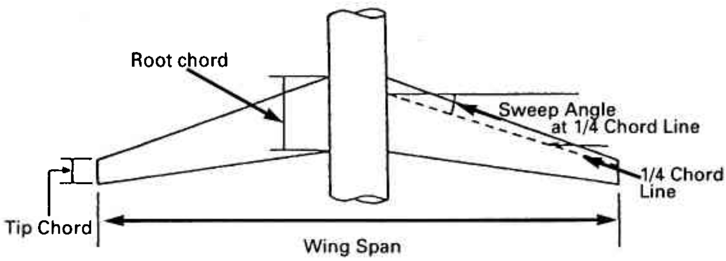


Figure 3–3

3.8 Wing Area

The area enclosed by the wing outline and extending through the fuselage to the centreline.

3.9 Mean Chord (Geometric)

The wing area divided by the span.

3.10 Taper Ratio

The ratio of the root chord to tip chord. Fig 3-3

3.11 Aspect Ratio

The ratio of the wing span to the mean chord, or alternatively span^2 to wing area.

3.12 Wing Loading

The weight of the aircraft divided by the wing area.

3.13 Sweep Angle

The angle between the lateral axis and the ¼ chord line (may be referred to as the leading edge). Fig 3-3

3.14 Dihedral

The upward inclination of the wing to the plane through the lateral axis. Fig 3-4

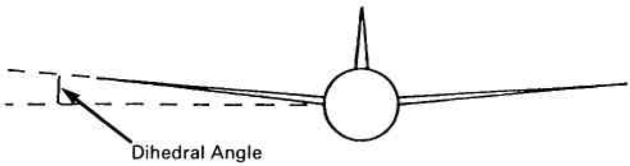


Figure 3–4

3.15 Anhedral

The downward inclination of the wing to the plane through the lateral axis. Fig 3-5

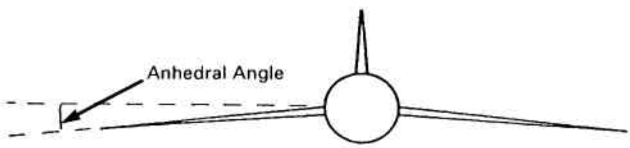


Figure 3–5

3.16 Axes and Flight Controls (Primary Controls)

(a) Elevators

The elevator is attached to the trailing edge of the tailplane and controls the pitching moment about the lateral axis. A backward movement of the control column moves the elevator up and causes the aircraft nose to pitch up. Fig 3-6

Pitching Rotation about the Lateral axis.
Control by Elevators (longitudinal control)

Figure 3–6 Pitching Control by Elevators

(b) Ailerons

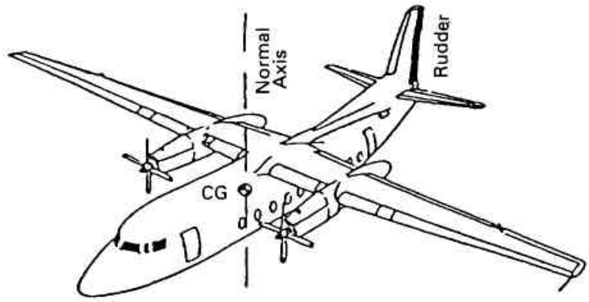
The ailerons are attached to the outboard trailing edges of the wings or mainplanes and controls the rolling motion about the longitudinal axis. If the control column is moved to the right the right aileron moves up and the left aileron down, causing a roll to the right. Fig 3-7

Rolling Rotation about the Longitudinal axis.
Control by Ailerons (Lateral control)

Figure 3–7

(c) Rudder

The rudder is attached to the rear edge of the fin and causes the aircraft to yaw about the normal axis. Movement of the right rudder pedal forward moves the rudder to the right causing the aircraft to yaw to the right about the normal axis. Fig 3-8



Yawing Rotation about the Normal axis
Control by Rudder (Directional control)

Figure 3–8

3.17 Alternative Forms of Control

(a) Stabilator or All-Moving Tail

Sometimes used in place of separate elevator control.

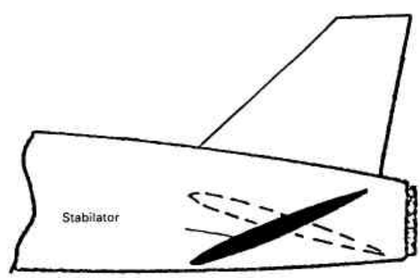


Figure 3–9 Stabilator

(b) Spoilers

May be used instead of or in addition to ailerons. When the spoiler is operated it causes a loss of lift on the side it is raised, thus causing a roll to that side. Movement of the control column to the right causes the right spoiler to rise but the left spoiler to remain retracted.

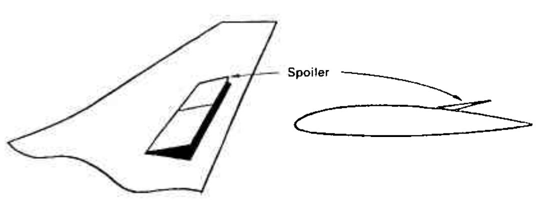


Figure 3–10 Spoiler

Chapter 3: Test Yourself.

1 The thickness/chord ratio of a wing is also known as the:

a) aspect ratio.
b) mean chord ratio.
c) Fineness ratio.
d) incidence ratio.

Ref para 3.3

2 The angle of attack of an aerofoil section is the angle between the:

a) chord line and the mean chord line.
b) chord line and the relative airflow.
c) underside of the surface and the relative airflow.
d) mean camberline and the relative airflow.

Ref para 3.4

3 The Mean Chord (Geometric) is the:

a) wing area divided by the span.
b) ratio of root chord to tip chord.
c) ratio of the wing span to the mean chord.
d) wing area multiplied by the span.

Ref para 3.9

4 A High Aspect Ratio wing is a wing with:

a) long span, long chord.
b) long span, short chord.
c) short span, short chord.
d) short span, long chord.

Ref para 3.11

5 The angle between the lateral axis and the ¼ chord line is known as:

a) the dihedral angle.
b) the sweep angle.
c) the incidence angle.
d) the chord angle.

Ref para 3.13

4

Drag

4.1 Introduction

It is convenient to study the subject of drag under two distinct headings:

Profile Drag or 'Zero Lift Drag'
Induced Drag or 'Lift Dependent Drag'. (See Chapter 6)

4.2 Profile Drag (Parasite Drag)

Profile drag is discussed under three sub-headings:
(i) Skin Friction Drag
(ii) Form or Pressure Drag
(iii) Interference Drag.

(a) Skin Friction and Boundary Layer
Consider a flat smooth surface over which an airstream is flowing. What may seem to be a smooth surface to an observer, will, to a molecule of air, seem a very rough one. Air is a viscous medium, and any surface subjected to a moving airstream will inevitably have, through viscous adhesion, a minutely thin layer of air at its surface which has zero relative velocity.

Succeeding layers adjacent to the surface will, through the same viscous action, be subject to retardation, but to a lesser degree with increasing distance (albeit a very small one) from the surface. A point is therefore reached where the airflow will be unaffected, and its velocity will be that of the 'free stream' airflow.

This layer of air from the surface where there is zero velocity, to the point where there is no retardation, is referred to as the 'Boundary Layer' and is normally defined as the region in which the velocity of flow is less than 99% of the free stream value.

The boundary layer exists in two forms: (a) Laminar Flow, and (b) Turbulent Flow.

Physical laws dictate that at some point along a surface which is subject to a moving airstream, the flow will change from laminar to turbulent. This point is of importance in the study of drag, the significant feature being that the drag is greater in the turbulent layer than in the laminar.

The main variables which dictate the change from the laminar state to the turbulent are: (a) Velocity of flow, (b) Viscosity of the fluid, or air, (c) Size of the object.

Generally speaking, the transition point for an aerofoil section will be at the point of maximum section depth where the velocity of flow is greatest (refer to Venturi effect). As can be clearly seen, it pays to maintain laminar boundary layer flow as long as possible over an aerofoil section in order to reduce drag, and to keep the surface as smooth as possible.

One method of ensuring a greater percentage of laminar flow is to maintain an increasing depth of section as far back from the leading edge as possible, thereby locating the point of maximum velocity farther aft. This results in a wing section known as a *laminar flow wing*; a description which is, of course, only partially true; Fig 4-1 indicates non-laminar and laminar sections.

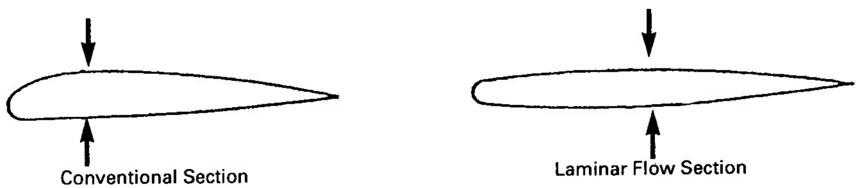


Figure 4–1 Conventional and Laminar Flow Sections

Figure 4-2 shows some important features of the transition from laminar to turbulent flow, these being:

(i) The depth of the laminar layer typically given as 0.07in.
(ii) The depth of the turbulent layer typically given as 0.7in.
(iii) The velocity gradients of the two layers being different leads to the greater shearing or friction effect occuring in the turbulent layer.

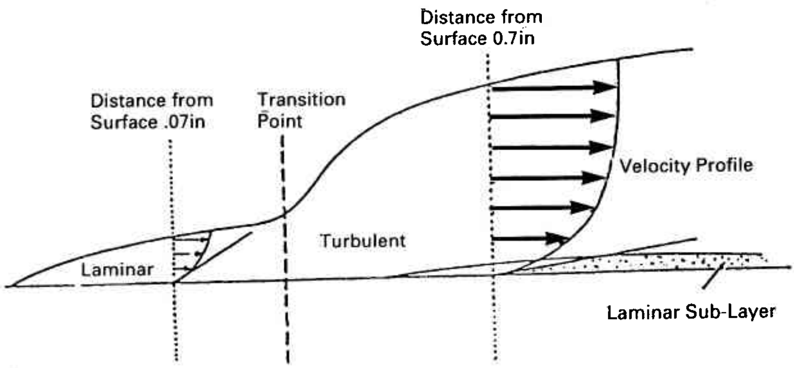


Figure 4–2 The Boundary Layer

(b) Form or Pressure Drag

When an object is placed in a viscous fluid, such as air, which is moving relative to the object, it will experience a resistance owing to the formation of vortices which create turbulent as opposed to streamlined flow.

If we regard a flat plate at right angles to an airflow (Fig 4-3) as being an extreme case, the kinetic energy of the airstream is largely brought to rest and converted to pressure energy: the diagram also shows the point 'S' which is referred to as the stagnation point. The pressure brought to rest, is referred to as the 'Dynamic Pressure'; it is of considerable importance, and is the pressure experienced by any object when a moving airstream is brought to rest: it is quite distinct from static pressure. The formula for dynamic pressure is $\frac{1}{2}\rho V^2$ where ρ = air density and V is velocity.

It may be seen from Fig 4-3, that the flow behind the plate is composed of vortices, and since these have low pressure in the centre, we now have high pressure in front and low behind the plate: this results in a drag force in the direction of the moving airstream.

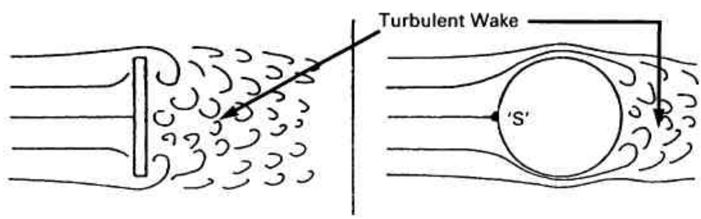


Figure 4–3 Turbulent Wake

(c) Reduction of drag with streamlining

It is clear from the extreme case of the flat plate at right angles to the airflow, that it represents the maximum generation of vortices and turbulence; in other words, maximum resistance or drag. The production of vortices require the expenditure of energy in order to generate them, and this of course, is wasteful. By substituting a cylindrical section for the plate, as in Fig 4-3, we produce a less abrupt change in the path which the airflow is trying to follow. In this case, fewer vortices are generated; there is less difference in pressure from the front to the rear of the shape, and a degree of 'streamlining' has been achieved.

Taken a step further, reference to Fig 4-4 will show a more streamlined shape as in a symmetrical aerofoil section. This shape allows the airflow a much more gradual passage from the front of the section to the rear than in the case of the cylinder. The end result therefore, of streamlining, is to produce much less vortex generation, reduced turbulence, and greatly reduced drag.

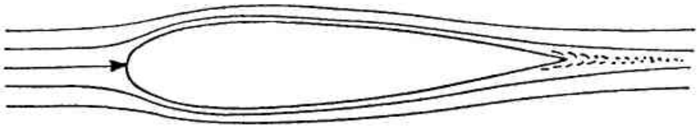


Figure 4–4 The Streamlined Aerofoil

Although by careful streamlining considerable reductions in the form drag are achieved there is a limit to extension of the method. The 'fineness ratio' of an aerofoil section is a measure of its streamlining, and is defined as the thickness to chord length ratio. Figure 4-5 shows a section of conventional ratio, but if this ratio is too great, the resulting very thin section leads to attendant constructional difficulties.

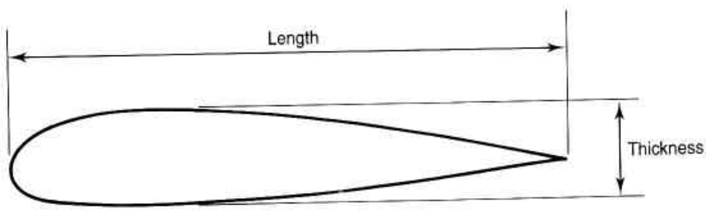


Figure 4–5 A Conventional Fineness Ratio

(d) Interference drag
On a complete aircraft, the total drag is greater than the sum of the values for the individual parts of the aircraft. This additional drag is the result of 'flow interference' in such areas as wing/fuselage, wing/nacelle junctions, and in fact any areas where such junctions exist.

The interference leads to modifications of boundary layers (discussed later) and creates greater pressure differences between fore and aft areas on the surfaces concerned, this in turn leading to greater total drag. This drag can be reduced in value by careful fairing or the addition of fillets in the areas concerned.

(e) The Drag Formula
It is found by experience that, within certain limitations of flow velocity, the resistance of an object in a moving airstream is proportional to:
(i) The shape of the object and frontal area
(ii) The square of velocity
(iii) The density of the fluid

As a basic formula this is written as $R \propto \rho V^2 S$ or $R = K\rho V^2 S$

In Fig 4-3, clearly, not all the air is being brought to rest by the plate, as some of it is seen to be flowing round the edges. This means that the full conversion of kinetic energy to pressure energy is not realised. For this reason, the value of 'K' in the second formula will vary according to the shape of the object and its associated system of vortices: the value of 'K' is found by experiment.

The importance of the unit of dynamic pressure has already been emphasised, and the above basic formula is now modified by its inclusion.

The new formula therefore, becomes:

$C_D \frac{1}{2}\rho V^2 S$ where C_D is the 'coefficient of drag', and S the wing area. The unit of dynamic pressure $\frac{1}{2}\rho V^2$, is very often written simply as 'q' because of its frequent use.

As a point of interest, the value of K in the basic formula is about 0.6 for a flat plate, but since $C = 2K$, we now have a value of 1.2. Other values of the drag coefficient that are of interest are:

A cylindrical section – 0.6
A streamlined section – 0.06
A pitot tube has a value of unity.

To conclude, the combined drag due to skin friction, form drag and interference drag under the heading of 'Profile drag', increases in the manner shown in Fig 4-6.

The subject of Induced Drag or 'lift dependent drag' is discussed in the chapter on Lift.

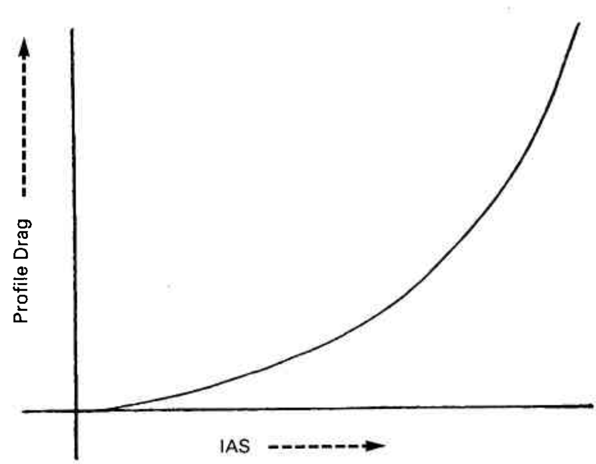


Figure 4-6 Increasing IAS (indicated air speed) results in increasing profile drag

Chapter 4: Test Yourself.

1 In level flight a section of the flow ahead of a given point over the upper surface of the wing is laminar, that point is termed the:

a) C of P.
b) separation point.
c) laminar point.
d) transition point.

Ref para 4.2 (a)

2 With increasing speed in level flight:

a) induced drag increases.
b) profile drag increases.
c) profile drag remains constant.
d) induced drag remains constant.

Ref para 4.2 (e)

3 Generally speaking, the transition point for an aerofoil section is the:

a) point of maximum section depth.
b) separation point.
c) point of greatest pressure.
d) leading edge.

Ref para 4.2 (a)

4 To ensure a greater percentage of laminar flow is achieved over the wing upper surface:

a) the section maximum depth should be as near to the leading edge as possible.
b) the section maximum depth should be as near to the trailing edge as possible.
c) the section maximum depth should be at the ¼ chord.
d) the section should be of a bi-convex shape.

Ref para 4.2 (a)

5 As the angle of attack of a wing is increased:

a) the C of P moves aft.
b) the boundary layer thickens.
c) the boundary layer becomes thinner.
d) the boundary layer thickness will remain the same.

Ref para 4.2

5

Lift

5.1 Introduction

It has been shown that if a streamlined body is placed in a moving airstream it produces drag, a force in the direction of the airflow. It should be noted that the streamlined body we were examining was symmetrical in shape. This drag force was the total force produced by the streamlined body. If we now incline the streamlined body at a small angle to the airflow the total force is now no longer in the direction of the airflow and this is illustrated in Fig 5–1. The total force can now be resolved into two forces, drag and the one at right angles to it, lift.

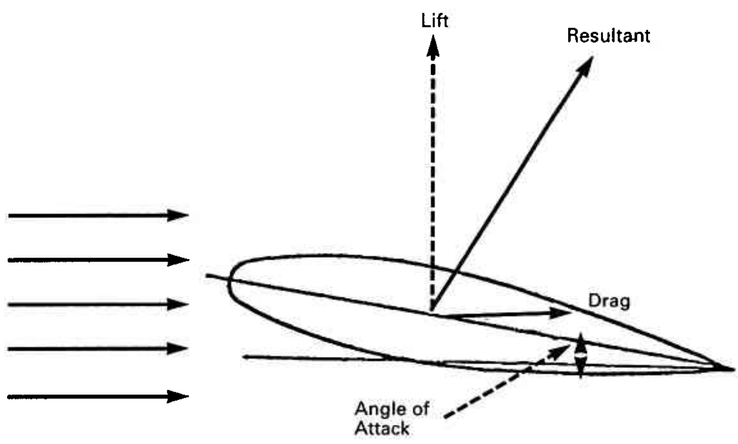


Figure 5-1

5.2

The diagram may give the impression that the lift and drag forces are approximately equal, but it has only been drawn this way for the sake of clarity. An aerofoil section in fact, produces lift many times greater than the value of drag it also produces. In Chapter 2 Bernoulli's theorem indicated that there will be a reduction in pressure over the upper surface of the wing; this reduction provides approximately two thirds of the lift produced by a wing. The general pressure distribution over the surfaces of a wing at a small angle of attack is illustrated in Fig 5-2.

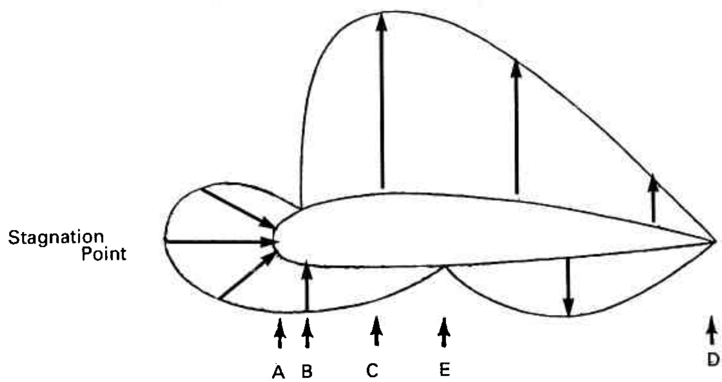


Figure 5–2

(a) Pressure distribution

The upper surface of the wing produces a considerable reduction in pressure but the lower surfaces produce a mixture of increase and decrease in pressure as well. The detail of the diagram shows that at the leading edge of the wing, point A, the full pressure is felt, this being the stagnation point. As the air moves over the upper surface of the wing, towards station B, it is approaching an area of lower pressure and at station B the pressure is just atmospheric or static. Past station B the pressure steadily reduces until it reaches its minimum value at C as indicated by the longest vector, and after C as the air moves towards the trailing edge of the wing the pressure, although below static pressure, is now gradually increasing. The fact that the air travelling from C towards D at the trailing edge is now moving against an adverse pressure gradient is of considerable importance when we come to discuss stalling. On the undersurface of the wing at point A the pressure was above static, in fact the full dynamic pressure was felt there and to some extent an increase in pres-

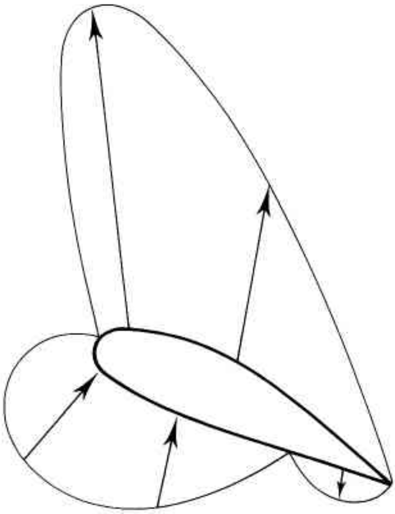


Figure 5–3

sure is felt on the undersurface of the wing up to about point E. Thereafter the wing undersurface produces a small venturi of its own which gives a reduction in pressure, and in order to limit this reduction the undersurface of the wing is given considerably less curvature than the upper.

The pressure distribution as shown in Fig 5-2, is for a comparatively small angle of attack, say about 4°. Changes in the angle of attack of the aerofoil produce very considerable changes in the pressure distribution and Fig 5-3 illustrates the pressure pattern at a high angle of attack, say about 12°.

(b) Pressure gradients

The most obvious difference between this diagram and Fig 5-2 is the change of shape of the below static pressure on top of the wing. The main feature of this new shape is that the point of minimum pressure is very much nearer the leading edge of the wing than it was before. This means that the air travelling from C to the trailing edge of the wing has to deal with a very much longer and larger adverse pressure gradient. The only means available to the air to travel against this adverse pressure gradient is its own kinetic energy – its energy of motion – and if that adverse pressure gradient proves to be too great for the kinetic energy of the air, the flow will in fact break away from the wing. This situation is called a stall and is dealt with in a later section. On the undersurface of the wing the effect of the increase in pressure is enhanced, thus providing more lift

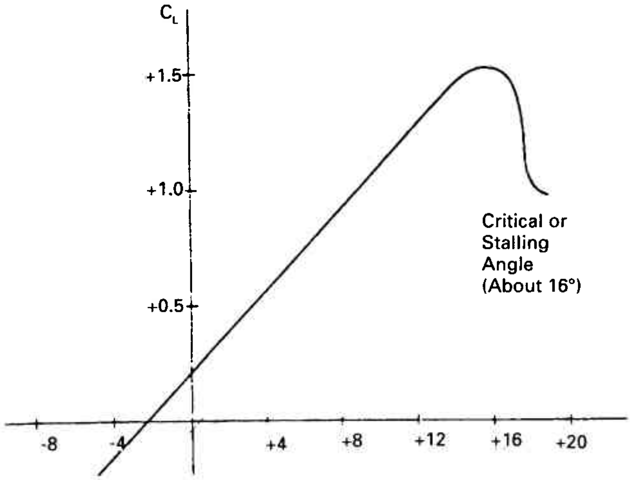


Figure 5–4

and the small amount of negative pressure towards the trailing edge has been reduced. The overall effect of the increase in the angle of attack is to increase lift but this process can only be carried out to a certain point and when this point is reached, the wing stalls. The relationship between the angle of attack and lift is illustrated in Fig 5-4. It can be seen that there is a steady increase in lift as the angle of attack increases and then a sudden decrease at the stalling angle which occurs at about 16°.

(c) Lift Equation

The basic factor controlling the value of lift is dynamic pressure. The equation for this, as already noted, is:

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

The size of a wing will obviously affect the amount of lift produced and this must therefore be added to the equation:

$$\text{Lift} = \tfrac{1}{2}\rho V^2 S, \text{ where S is the wing area.}$$

The shape of a wing will also influence the amount of lift that can be generated and this produces a factor, dependent upon the cross-sectional area of the wing, called the coefficient of lift, C_L. As demonstrated with angle of attack this will have an influence upon the amount of lift gener-

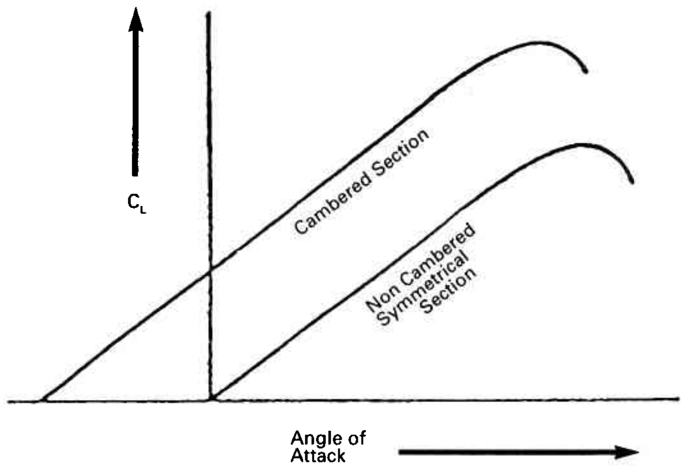


Figure 5–5

ated. The coefficient of lift is in fact a derivative of the wing shape and its angle of attack. The full lift equation can therefore be written:

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

The shape of the lift curve for any wing will be more or less the same but it should be noted that the higher the camber of the wing the greater the lift it will develop. This is illustrated in Fig 5-5 where a cambered section is compared to a symmetrical section. A point of interest is that although the cambered section still generates lift at a zero angle of attack the symmetrical section does not.

(d) Lift/Drag Ratio

The total resultant force derived from airflow over a wing can be resolved into two forces, lift and drag. The whole object of the exercise is of course to produce lift and in an ideal situation would be done without incurring drag. Unfortunately, this is never possible but it is of great importance to know the ratio between lift and drag so that the aircraft can be designed to provide the maximum amount of lift for the minimum amount of drag. Lift and drag vary with the angle of attack and the variations of these two are shown in Fig 5-6 (a) and (b).

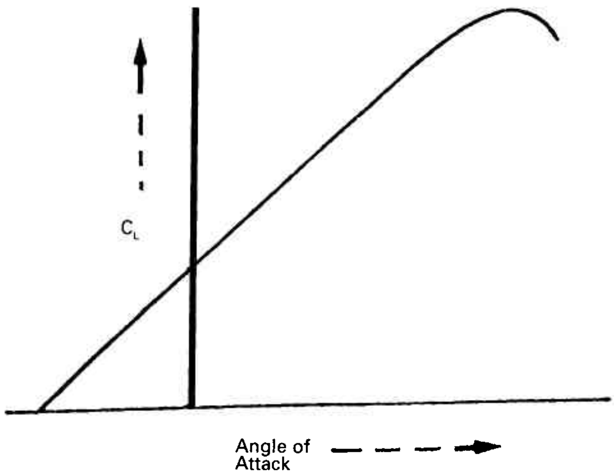


Figure 5–6 (a)

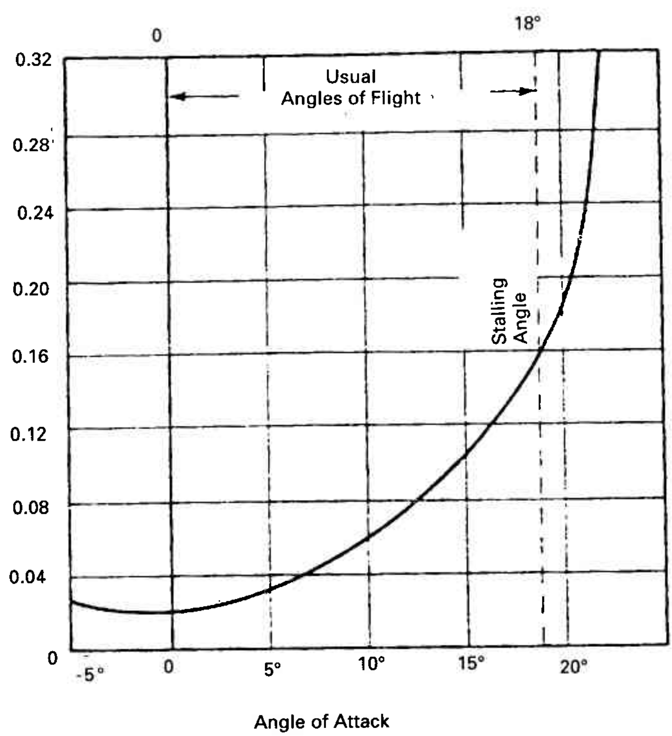


Figure 5–6 (b)

If these two figures are combined mathematically they produce a curve as shown in Fig 5-7. It can be seen that there is a steady increase in the lift/drag ratio, which is what is desirable, until an angle of attack of about 4°. Thereafter the situation deteriorates as the lift/drag ratio lessens until, at an angle of attack of around 15°, it tails right off, this being the stalling angle. The highest point on this curve where we are getting the largest amount of lift for the smallest amount of drag, occurs at about 4° and this is therefore the optimum angle of attack. Obviously, the combination of most lift for least drag is the most efficient and why aircraft are usually flown at the optimum angle of attack.

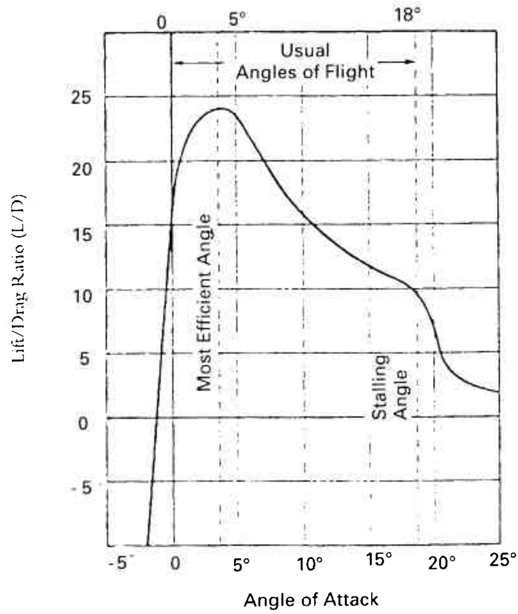


Figure 5–7

(e) Movement of the Centre of Pressure

In Chapter 4 the centre of pressure was defined as that point on the chord line through which the lift can be considered to act. The vector representing lift through the centre of pressure passes through the point of minimum pressure on the upper surface of the aerofoil. This is illustrated in Fig 5-8.

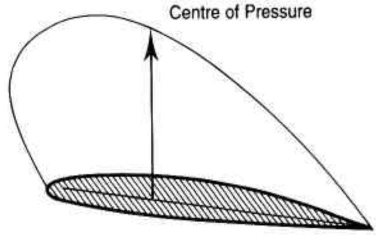


Figure 5–8

(f) Spanwise Distribution of Pressure

The amount of lift produced by the upper surface of the wing will gradually decrease from root to tip. This means that although the pressure on top of the wing is all below static pressure, it is much lower near the root than it is near the tip. On the underside of the wing the reverse applies and the pressure near the root is much higher than it is near the tip. Looked at in plan view, this will cause the air flowing over the upper surface of

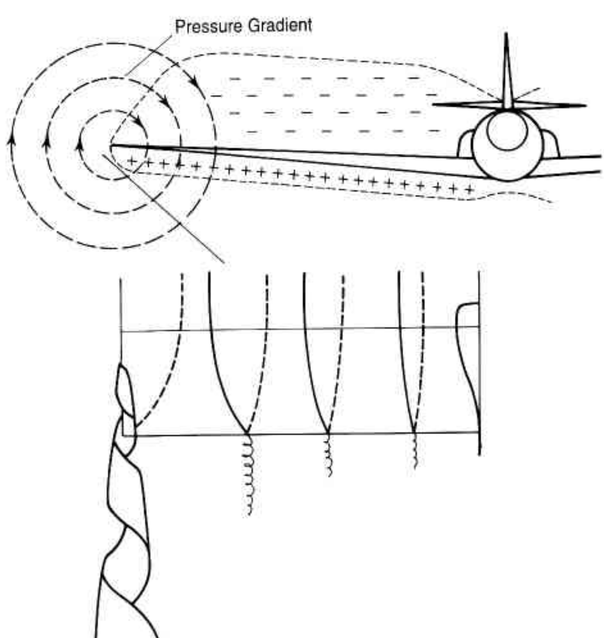


Figure 5–9

the wing to be deflected inwards and the air flowing over the underside of the wing to be deflected outwards. This is illustrated in Fig 5-9.

When the two airflows meet at the trailing edge of the wing they are moving in different directions and the result is to form a sheet of vortices. If one were to be able to see the air and stand behind the trailing edge of the wing, the vortices on the right-hand wing would be rotating anti-clockwise and on the left-hand wing rotating clockwise. The result of these vortices is to impart a downward velocity to the airflow. This downward movement of the air as it passes over the trailing edge of the wing is called downwash.

Chapter 5: Test Yourself.

1 For a cambered wing section the zero lift angle of attack will be:

a) positive.
b) 4°.
c) zero.
d) negative.

Ref para 5.2 (c)

2 If the angle of attack of a wing is increased in flight the:

a) C of P will move forward.
b) C of G will move aft.
c) C of P will remain in the same place.
d) C of P will move aft.

Ref para 5.2 (a)

3 When maintaining level flight an increase of speed will:

a) have no effect on the C of P position.
b) cause the C of P to move forward.
c) cause the C of P to move aft.
d) cause the C of G to move forward.

Ref para 5.2 (c)

4 For the same angle of attack a cambered wing will produce:

a) less lift than one with no camber.
b) more lift than one with no camber.
c) the same lift regardless of camber.
d) less lift and drag than one with no camber.

Ref para 5.2 (c)

5 The Lift/Drag ratio of a wing section at its stalling angle is:

a) moderate.
b) of a negative value.
c) low.
d) high.

Ref para 5.2 (d)

6

Induced Drag

6.1 Introduction

Pressure distribution over the upper and lower surfaces of the wing was examined in Chapter 5. As high pressure exists underneath the wing and low pressure on top of the wing, the one place where these pressures will attempt to equalize is around the wing tip. The high pressure underneath the wing moves upwards towards the low pressure on the upper surface and in doing so assumes a rotary motion. This rotary motion spirals back from the wing tip, moving in an anticlockwise direction from the right-hand wing tip as viewed from behind and in a clockwise direction from the left-hand wing tip. Energy is required to produce this rotational vortex from each wing tip and this energy can come only from thrust. The vortices therefore create drag and this drag is called *induced drag.*

6.2 Drift effect

The larger the lift being produced by the wing, the bigger the pressure difference between the lower and upper surfaces. The larger the pressure difference the stronger the vortex produced and it can therefore be said that induced drag is proportional to lift. In straight and level flight lift must equal weight, so if weight is increased then lift must be increased and therefore induced drag will be larger. The same is also true for a turn

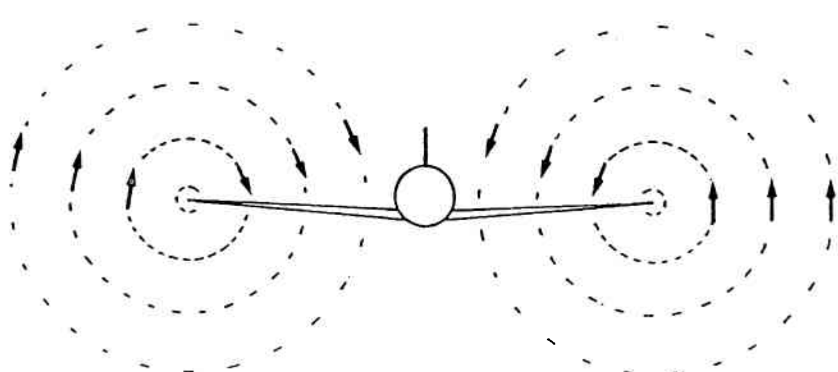


Figure 6–1 Tip Vortices

where lift must be increased, producing more induced drag although the gravitational weight has not been changed. Vortices are shown diagramatically in Fig 6-1.

6.3 Downwash

The effect of the vortex is to deflect the air downwards as it passes over the trailing edge of the wing, in other words producing downwash. As the maximum strength of this movement is close to the vortex, as one moves from the wing tip towards the fuselage the downwash steadily decreases. Thus for a given strength of vortex, the larger the wing span the less will be the effect of this downwash velocity.

The angular deflection of the airflow will depend on the speed. For a given downwash velocity the deflection angle will be greater at low speeds than at high speeds, as shown in Fig 6-2.

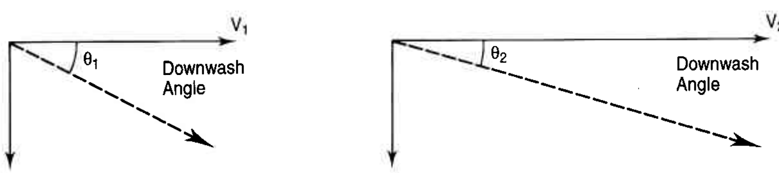


Figure 6-2 Downwash Angle

The total reaction force of a wing is at right angles, not to the initial direction of the airflow, but to the resultant between the original direction and the final direction. It will be readily seen that the more the final flow is deflected downwards – in other words the bigger the downwash – the more the total reaction is tilted rearwards, and this is clearly illustrated in Fig 6-3. The actual usable lift in level flight has to be perpendicular. This leaves a small rearward component of the total reaction force and this is induced drag.

From Fig 6-3 it will be seen that the larger the lift component the bigger will be the rearward component D_i, induced drag. Induced drag is in fact proportional to lift^2.

6.4 Span effect

The wing span of the aircraft has a marked effect on the amount of induced drag. The strength of the vortex diminishes from the wing tip towards the fuselage and therefore the downwash created by it also diminishes. For a given strength of tip vortex, therefore, the longer the wing span the lower will be the average downwash and the lower the induced drag. For a given amount of lift, the longer span and short tip chord

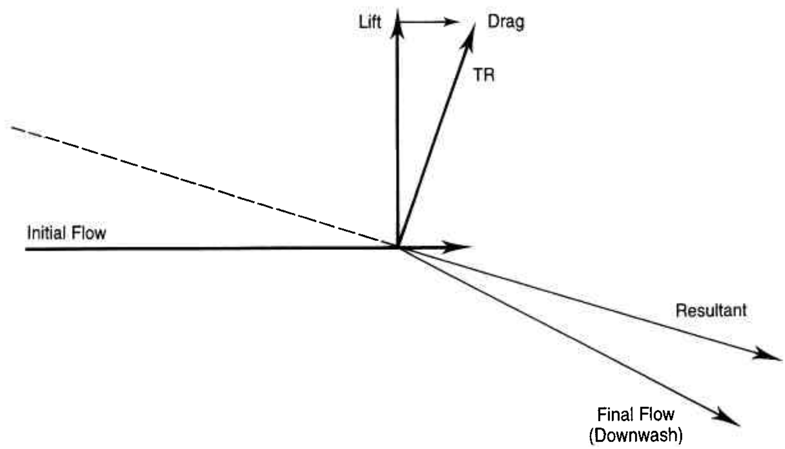


Figure 6–3 Downwash Angle

produces a weaker vortex than a wing with a short span and long tip chord and will therefore give less induced drag. In other words, the higher the aspect ratio, the lower the induced drag.

6.5 Summary of effects

The amount of induced drag created by a wing depends upon the amount of downwash and we saw from Fig 6-2 that the slower the speed the bigger the angular deflection downwards of the air. This therefore means that the induced drag is largest at low speeds and decreases as the speed increases. We can say from this that induced drag varies inversely as the square of the speed or, induced drag is proportional to $\frac{-1}{\text{speed}^2}$.

To summarise the effects of induced drag, then:

(a) Induced drag increases with an increase in weight.
(b) Induced drag decreases with wing span, that is, high aspect ratio reduces induced drag.
(c) Speed increases, induced drag decreases.

Several deductions can be made from this summary. It becomes apparent that gliders and sailplanes having very long, narrow wings – wings with a high aspect ratio – and normally flying at very low speeds when induced drag is at its highest, therefore benefit from high aspect ratio wings to reduce this drag to a minimum. Conversely, large jet transport aircraft do not usually have high aspect ratio wings and, in addition, are usually of very high weight. From this it is apparent that at low speed they will have very high induced drag.

Further reference to Induced Drag is made in Chapter 10 Wing Planforms.

Key Points To Note

1 Induced Drag is proportional to Lift.
2 Induced Drag is inversely proportional to Speed.
3 Induced Drag is greatest toward the wing tip.
4 Induced Drag is less with greater wing aspect ratio.
5 A High Aspect Ratio Wing has a long span and a short chord.
6 The amount of lift generated by the wing upper surface is greatest towards the wing root.
7 The airflow over the upper surface of the wing in flight tends to flow towards the root.
8 In flight, the angle of attack at which the largest amount of lift is generated for the smallest amount of drag is at approximately 4°. This is known as the optimum angle of attack.
9 It can be said that from an angle of attack of zero up to 4° the Lift/Drag ratio increases. Above 4° it decreases.
10 At zero angle of attack a cambered aerofoil produces some lift and some drag.
11 At zero angle of attack a symmetrical wing produces no lift but some drag.
12 On a wing in flight, $^2/_3$ of the lift is produced by the upper surface and the remainder by the lower surface.

Chapter 6: Test Yourself.

1 Induced drag is:

a) greatest towards the wing root and downwash is greatest at the tip.
b) greatest towards the wing tip and downwash is greatest toward the root.
c) balanced from root to tip.
d) greatest toward the tip and downwash decreases from tip to root.

Ref para 6.5

2 Airflow over the upper surface of the wing generally:

a) flows towards the root.
b) flows towards the tip.
c) flows straight from leading to trailing edge.
d) produces higher pressure than that flowing over the undersurface.

Ref para 6.5

3 Induced drag:

a) increases as the square of the speed.
b) varies inversely as the square of the speed.
c) reduces with increased angle of attack.
d) remains constant for a given speed regardless of angle of attack.

Ref para 6.5

4 For a given angle of attack induced drag is:

a) greater on a high aspect ratio wing.
b) greater towards the wing root.
c) greater on a low aspect ratio wing.
d) balanced across the span of the wing.

Ref para 6.5

5 Induced drag:

a) increases with increase in speed.
b) increases with increase in aircraft weight.
c) reduces with an increase in angle of attack.
d) reduces with altitude at constant I.A.S.

Ref para 6.5

7

Total Drag

7.1 Introduction

In preceding sections it has been shown that the aircraft is subjected to two types of drag, profile drag and induced drag. Profile drag increases with speed and is proportional to the square of the speed, and induced drag decreases with speed and is inversely proportional to the square of the speed. These two curves of profile and induced drag are shown against speed in Fig 7-1. The two curves can be amalgamated to give the total drag curve of the aircraft. The lowest point on this total drag curve gives the speed at which the total drag is a minimum. This speed is called the minimum drag speed, V_{md}. For a constant weight and in straight and level flight the V_{md} will be a constant indicated airspeed for all altitudes. It would be reasonable to assume that one would be better off flying the aircraft at V_{md} because the drag is least at this speed. In practice however, aircraft are not normally operated at this speed because the overall efficiency, especially that of the engine, may be better at a higher speed.

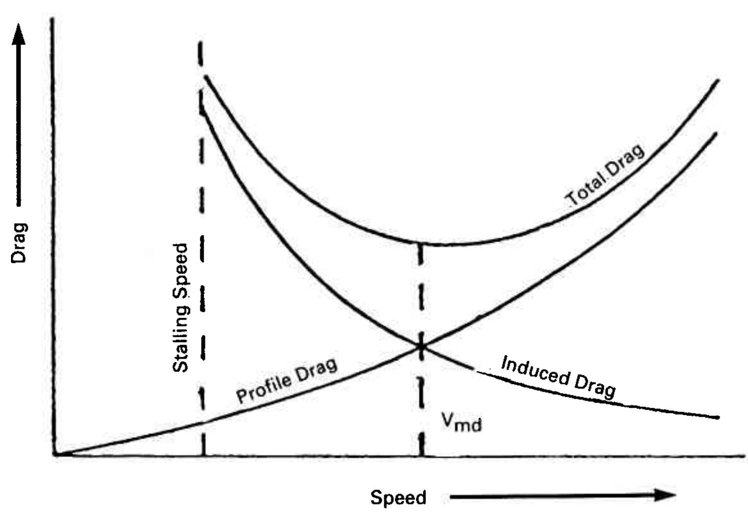


Figure 7–1

It is of some importance when handling an aircraft to know if the speed is stable. By this it is meant that if for some reason the speed increases, perhaps due to temporary turbulence, does the speed tend to decay back to its original value or not? The answer to this question can be found by examining the total drag curve which is shown again in Fig 7-2.

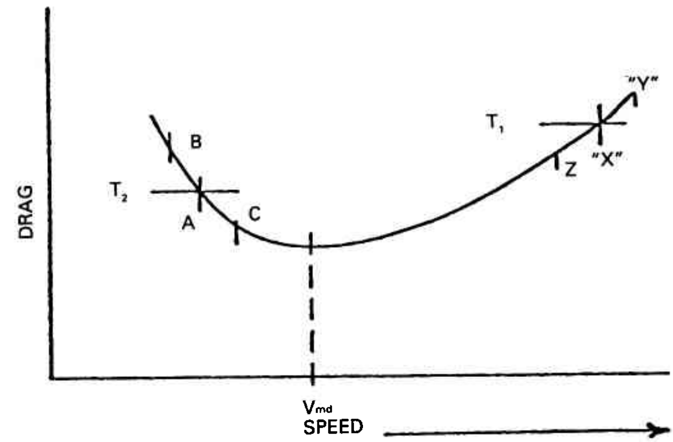


Figure 7–2

Consider an aircraft flying at speed X. In straight and level flight thrust = drag so the thrust required is indicated by the horizontal line T_1. If for some reason the speed increases from X to Y, the thrust remaining unchanged, the drag now exceeds the thrust so the speed will drop back to its original value at X. If, on the other hand, the speed decays to point Z, thrust is now greater than drag and the speed will automatically return to its original value at X. It should be noted that the two speeds quoted here are above V_{md}. On the other side of the curve with the aircraft flying at speed A the thrust level is now T_2. If the speed now reduces to B the drag becomes higher than the thrust and the speed will continue to decay. If, conversely, the speed increases to point C, the thrust becomes higher than the drag and the speed will continue to increase. These second examples are speeds below V_{md}. This simple illustration makes clear why at speeds higher than V_{md} the speed tends to be stable and at speeds below V_{md} the speed is not stable.

This speed instability below V_{md} is most marked on jet transport aircraft. The great weight of some of the larger types of such aircraft today produces very high induced drag values and makes handling on the approach somewhat difficult. Considerable anticipation is required to check either increase or decrease in speed, the whole thing being aggravated to the slow response of jet engines. It would be fair to say that this type of aircraft requires more precise handling on the approach than the piston-engined aircraft and is less forgiving of imprecise handling.

In the section on induced drag it was shown that if weight is increased then induced drag also increases. The minimum drag speed occurs at the point where the curve for profile drag crosses the curve for induced drag. As will be seen from Fig 7-1, at this speed the value of the induced drag is the same as that of profile drag. In other words profile drag equals induced drag and total drag is double the value of either one.

Figure 7-3 illustrates the fact that an increase in aircraft weight will raise the speed at which V_{md} occurs.

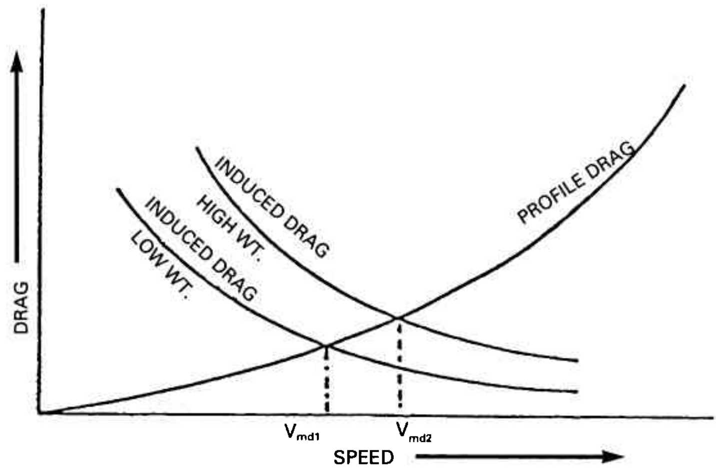


Figure 7–3

It was shown in the previous Chapter that the effect of an increase in aspect ratio is to decrease induced drag. From this it follows that aircraft with high aspect ratios will have a lower V_{md} than aircraft with low aspect ratios.

7.2 Wave Drag

It has been shown that drag is the same at any altitude for a given IAS but an aircraft climbing at this constant IAS has a steadily increasing Mach number. When this Mach number reaches a certain value the drag starts to increase because of compressibility effects. This drag is known as wave drag and its effect on the total drag curve is illustrated in Fig 7–4.

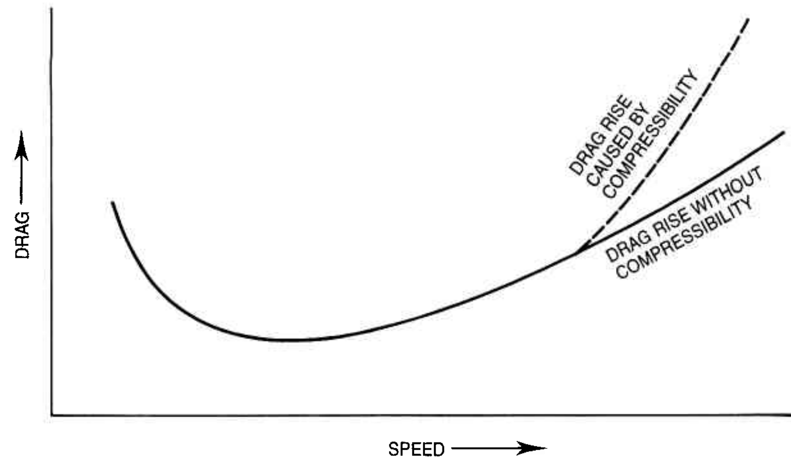


Figure 7–4

7.3 SUMMARY: Check List.

From the previous paragraphs the following has been established:
an increase in angle of attack will produce an increase in lift brought about by the velocity of the airflow over the upper surface of the wing being increased.
An increase in the angle of attack will cause:

(a) The Centre of Pressure to move forward.

(b) The Transition Point to move forward.

(c) The Separation Point to move forward.

(d) The Stagnation Point to move down and aft towards the under-surface of the wing.

The Centre of Pressure will reach its farthest forward point at just below the stalling angle.

Induced Drag is directly related to lift because as the angle of attack is increased the induced drag will increase. Due to the greater pressure difference between the upper and lower surfaces of the wing, the tip vortex (the basic origin of induced drag) will become intensified. For a given speed the greater the angle of attack the greater the induced drag.

It is important to realise that although induced drag increases with increased lift when increasing angle of attack, the increase in lift will always be much greater than drag up to and including the stalling angle.

Remember the stalling angle is the angle above which a given aerofoil will stall.

Induced Drag is influenced by the aspect ratio of the wing, the higher the aspect ratio for a given wing area the less the induced drag produced.

Induced drag is always greatest towards the wing tip where the tip vortex is generated by air flowing from the underside of the wing to the upper surface, where it then flows aft and down behind the wing and tends to converge behind the aircraft. It can also be said the flow on the upper surface of the wing tends to flow aft and towards the wing root, and on the undersurface of the wing, aft and towards the wing tip, as a direct result of the influence of the vortex at the wing tip. The flow towards the root and tip on the upper and lower surfaces cause numerous vortices to form at the trailing edge of the wing.

The greater the chord length at the wing tip the more intense the tip vortex becomes and so the greater the induced drag. Hence a high aspect ratio wing with a long span and a short chord will produce less induced drag than a low aspect ratio wing with a short span and a long chord.

The Lift/Drag ratio of an aerofoil increases rapidly up to approximately 3° to 4° at which angles the lift is some 24 times the drag, the ratio then falls progressively until at the stalling angle, approximately 15°, the lift may only be 10 or 12 times as great as the drag. Above the stalling angle the ratio falls still further until an angle of attack of 90° is reached when lift will be zero.

The best all round angle of attack is 3° to 4° where the Lift/Drag ratio is greatest, and this angle of attack is also known as the optimum angle of attack.

It is also important to note that a cambered aerofoil, even at zero angle of attack will produce some lift and some drag. Even at some negative angles of attack a cambered aerofoil will produce some lift and drag. But remember,a symmetrical aerofoil at zero angle of attack will produce no lift but some drag.

To obtain a good understanding of the Principles of Flight it is important to interrelate the various points that are made at each stage and to avoid considering them as separate entities.

Chapter 7: Test Yourself.

1 With an increase in aspect ratio for a given IAS induced drag will:

a) remain constant.
b) increase.
c) reduce.
d) none of the above.

Ref para 7.1

2 Induced drag:

a) is only equal to profile drag when the aircraft is at rest.
b) is equal to profile drag at the stalling angle.
c) is equal to profile drag at V_{md}.
d) is never equal to profile drag.

Ref para 7.1

3 With an increase in aircraft weight:

a) V_{md} will be at the same value.
b) V_{md} will be at a lower speed.
c) V_{md} will be at a higher speed.
d) total drag will be unchanged.

Ref para 7.1

4 With an increase in aspect ratio the value of V_{md} will:

a) remain the same.
b) be reduced.
c) be increased.
d) none of the above.

Ref para 7.1

5 For a given IAS an increase in altitude will result in:

a) no change in the value of induced drag.
b) an increase in induced drag.
c) a reduction in profile drag.
d) a reduction in induced drag.

Ref para 7.1

8

Stalling

8.1 Introduction

It has already been shown that the lift produced by a wing steadily increases as the angle of attack is increased, but only up to a certain point. Past this angle of attack the lift decreases rapidly and the angle at which this occurs is known as the stalling angle.

8.2 The Determining Factor

A stall is produced when the airflow has broken away from most of the upper surface of the wing. The determining factor in this is the angle of attack: the wing always stalls at a fixed angle, usually in the region of 15°.

8.3 The Cause

The cause of the stall is the inability of the air to travel over the surface of the wing against the adverse pressure gradient behind the point of minimum pressure. Figure 8-1(a) illustrates the pressure distribution over the upper surface of the wing at a small angle of attack, say about 4°. The minimum pressure point is at B, and the air travels from A to B without difficulty as it is moving from high to low pressure. However, from B to C it is being forced to travel from low to high pressure, that is, against an adverse pressure gradient. This poses no problems at low angles of attack because the kinetic energy of the air is adequate to take it to the trailing

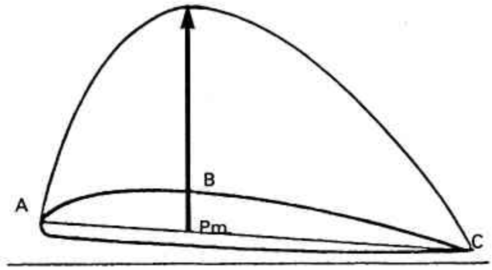


Figure 8–1 (a)

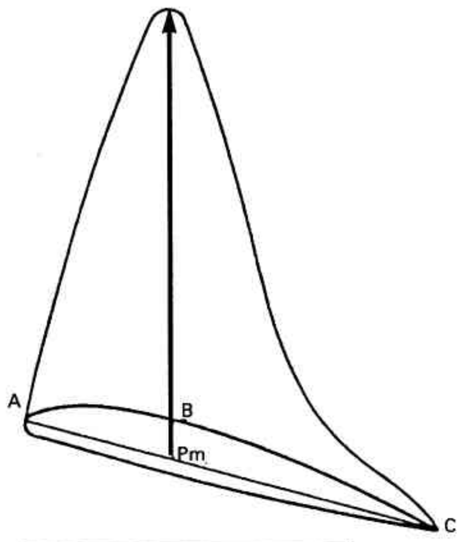


Figure 8–1 (b)

edge. As angle of attack is increased however, the minimum pressure point moves forward and the distance B to C increases until at the stalling angle it covers most of the wing. This is illustrated in Fig 8-1(b). When the angle of attack reaches a certain value the air runs out of kinetic energy and breaks away from the surface of the wing in a random manner. Lift decreases sharply and drag increases considerably.

8.4 Alleviation

Various design features can be incorporated in the wing which will assist in ensuring that the root of the wing stalls before the tip. These are:

(a) The wing may be twisted so that the tip is at a smaller angle of incidence than the root, which will ensure that the root reaches its stalling angle before the tip.

(b) The cross-section of the wing tip may be given a higher camber than the root, which will give it a higher coefficient of lift.

(c) A stall-inducer may be fitted to the wing root as illustrated in Fig 8–2. These strips reduce the effective camber of the root. This reduces its coefficient of lift and will cause it to stall before the tip.

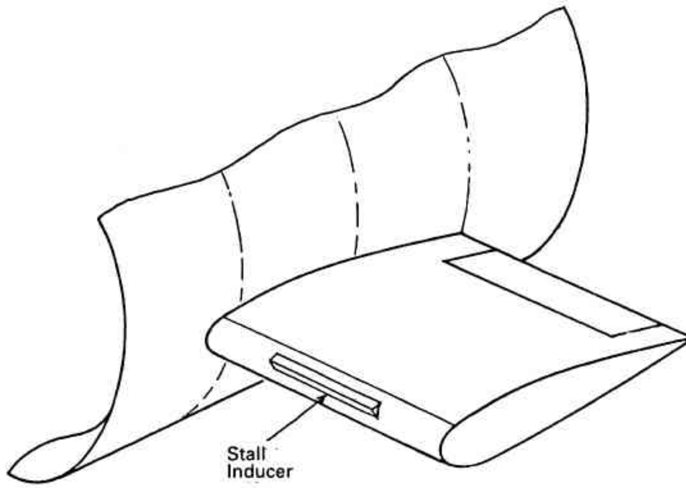


Figure 8–2

8.5 The Effect of Engine Power

If engine power is on there will be a reduction of stalling speed compared with the power-off stalling speed. With propeller-driven aircraft this is due to:

(a) Vertical component thrust

(b) The propeller slipstream over the wings.

8.6 Constancy

In straight and level flight at the stall, for a given wing area, cross-section and weight, the lift is of fixed value. This is a most fortunate occurrence when one considers the lift equation:

$$\text{Lift} = \tfrac{1}{2}\rho V^2 S c_l + \text{angle of attack}$$

As lift at the stall is a fixed value and angle of attack, wing area and coefficient of lift are also constant, the total value of $\tfrac{1}{2}\rho V^2$ must also be constant. $\tfrac{1}{2}\rho V^2$ is dynamic pressure shown on the airspeed indicator and it is for this reason that for a given weight an aircraft will always stall at the same indicated airspeed regardless of height.

8.7 Weight Effect

Any change of weight will require a different value of lift for straight and level flight, an increase in weight requiring an increase in lift. At the stalling angle in level flight, the greater the weight the more the lift required and, therefore, the higher the stalling speed. A useful rule of thumb in this context is that the percentage increase in stalling speed is half the percentage increase in weight. Thus:

Weight 2000 lb, normal stalling speed 100 kt.
Weight 2200 lb, percentage increase 10%, stalling speed increases 5%, ie to 105 kt.

8.8 Loading In Turns

The same effect is produced during manoeuvres which produce a G loading, for instance, turns. During a turn the lift not only has to balance the weight but also the centrifugal force resulting from the aircraft moving in a curved path. Because of this the lift has to be greater than in level flight and, provided the speed is kept constant, the only way that this extra lift can be derived is by an increase in angle of attack. This increase in angle of attack puts the aircraft wing nearer to the stalling angle. The result of having to produce effectively more lift from the wings is that the aircraft's weight appears to be increased, hence the expression G loading. The increase in stalling speed is calculated by taking the normal stalling speed in level flight for the aircraft's weight and multiplying it by the square root of the G loading. For example:

Normal stalling speed 100 kt,
Stalling speed in a 2 G turn = 100 × square root 2
= 100 × 1.4
= 140 kt.

Further details of calculating stalling speeds are given later in this chapter.

8.9 Effect of Shape

A wing does not normally stall over its entire length simultaneously; the stall begins at one part of the wing and then spreads. The main factor governing where the stall begins is the shape of the wing, and will be dealt with in a later section. It is plainly undesirable that a wing stalls from its tip first as this can lead to control difficulties. Any tendency to drop a wing at the stall may well lead to spinning. Further advantages of having a wing stall from its root rather than tip first are that aileron control can be maintained up to the point of full stall and the separated airflow from the wing root will cause buffet over the tail which serves to act as a stall warning.

When the angle of attack increases to high values the upward inclination of the thrust line provides a vertical component which acts in concert with the lift to support the aircraft's weight. The slipstream from the propeller increases the speed of the air flowing over the wing, thus delaying the stall. Caution should be exercised in power-on stalls as their effect may result in a tip stall on a wing which normally stalls from the root.

8.10 The Position of the Centre of Gravity

The stalling speed will be affected by the position of the centre of gravity. If the centre of gravity is forward of the centre of pressure a down-load is required from the horizontal stabilizer. The effect of this is that the lift is supporting not only the weight through the centre of gravity but also the down-load on the tail, therefore the lift will have to be higher and in turn the stalling speed will be higher. The nearer that the centre of gravity approaches to the centre of pressure, the less will be the down-load and the stalling speed will consequently be reduced.

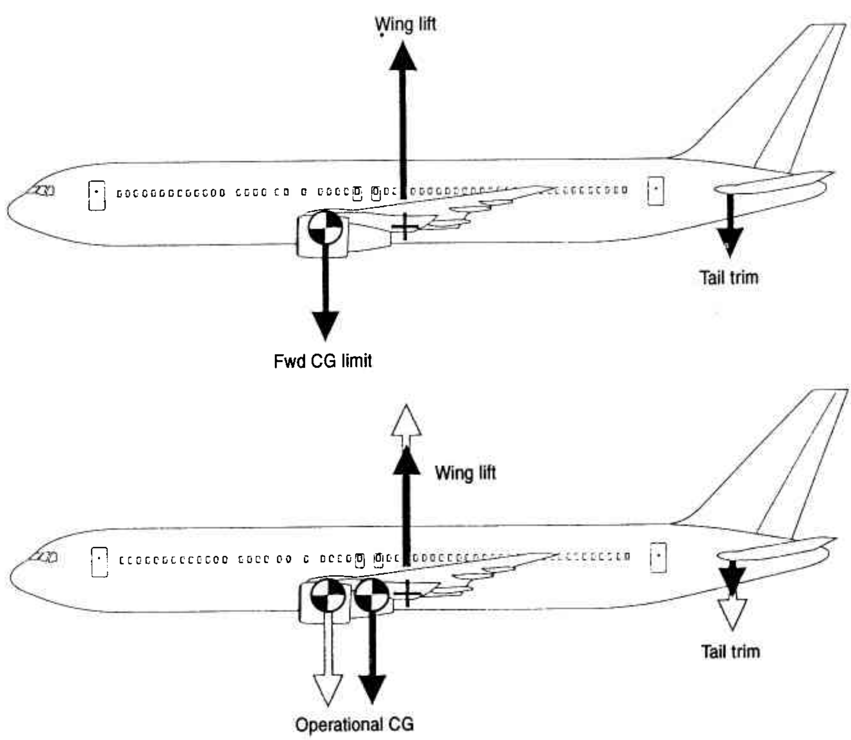


Figure 8–3 The location of the centre of gravity affects the tail loading and hence the stalling speed.

8.11 Icing

The effect of ice formation on a wing is to corrupt the camber of the wing and so considerably to reduce the coefficient of lift. This can be brought about by extremely thin layers of ice – even hoar frost – and the utmost care must be taken to de-ice the wings of an aircraft prior to takeoff if there is any suggestion that ice may be present on the wings. The drastic effect of ice in reducing the coefficient of lift and, as a result, causing the stalling speed to be much higher than normal, is illustrated in Fig 8-4.

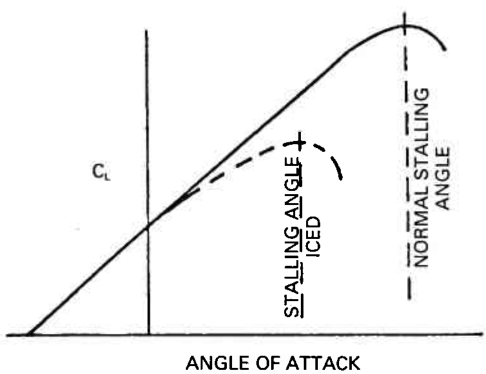


Figure 8–4

8.12 Stall Warning Devices

It is not normal to have an angle of attack indicator on the flight deck; it is usual instead to have some form of stall warning alarm operated by a switch which is sensitive to angle of attack. The warning may take the following forms:

(a) A visual warning, example a flashing light.

(b) Audible warnings, example a horn or stick knocker.

(c) A stick shaker.

8.13 Spinning

Following a stall involving a wing drop, a spin may develop. Referring to the diagram in Fig 8-5, the wing which drops increases its effective angle of attack due to having acquired a downward velocity. This increase in angle of attack causes a further decrease in lift and an increase in drag. The upgoing wing, however, experiences a decrease in angle of attack and an increase in lift. As the lift has been reduced on the downgoing wing it will continue to drop and any attempt to raise it by the use of ailerons merely aggravates the situation because it will increase the angle of attack still further. At the same time the increase in drag on the downgoing wing, coupled with a decrease in drag on the upgoing wing, will produce a yawing moment towards the dropped wing. From this it can be seen that the aircraft will roll and yaw towards the dropped wing, and this motion may be self-sustaining. If it is self-sustaining, the motion is described as a spin. Spinning is discussed in detail in Chapter 9.

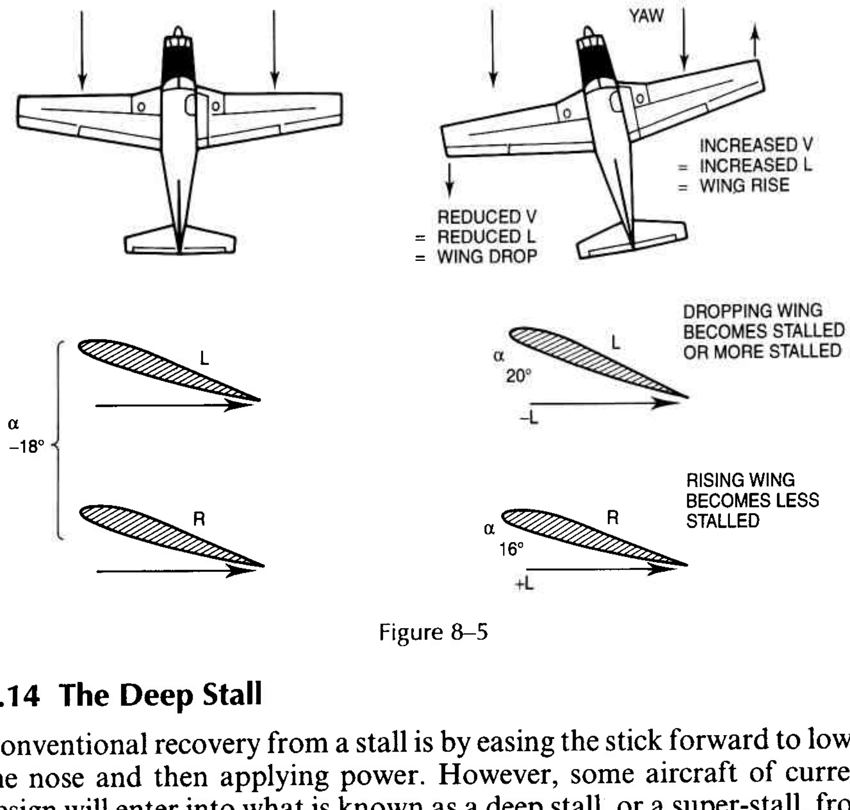


Figure 8–5

8.14 The Deep Stall

Conventional recovery from a stall is by easing the stick forward to lower the nose and then applying power. However, some aircraft of current design will enter into what is known as a deep stall, or a super-stall, from which normal recovery is not possible. Broadly speaking, these aircraft have swept back wings, high speed wing sections and a high T-tail.

The airflow following a stall in a conventional aircraft is illustrated in Fig 8-6. It can be seen that although the air has broken away in a random manner from the upper surface of the wing, the horizontal stabilizer and the elevators are still in undisturbed air. The result of this is that the horizontal stabilizer will produce a sharp nose down pitch which may be assisted by application of elevator.

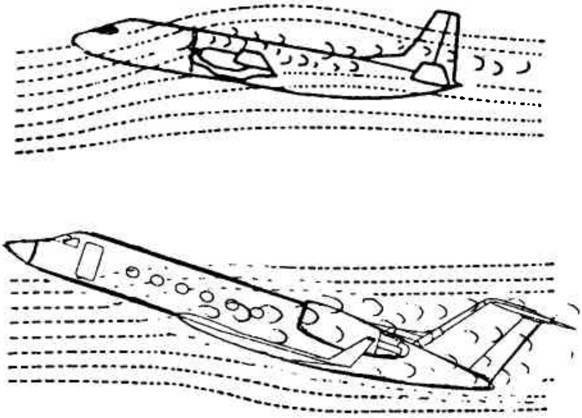


Figure 8-6

This can be contrasted with the state of affairs when an aircraft with a high T-tail is stalled. This time the separated air from the wings, following the stall, entirely covers the horizontal stabilizer and elevators, virtually reducing their effectiveness to nil. In the case of aircraft with sweep back on the wings, the wing itself may develop a nose up pitching moment after the stall. This is due to the tendency of a swept wing to stall at the tip and so cause the centre of pressure to move forwards. The situation is often aggravated because the aircraft has now acquired a vertical downward velocity which will progressively increase the angle of attack way beyond the stalling angle. Finally, this type of aircraft is often equipped with rear-mounted engines and the effect of turbulent air entering the engine intakes may be to cause them to flame out, causing a complete loss of power.

Obviously an aircraft with these characteristics cannot be permitted to stall. When such an aircraft is first built, it is equipped with a tail-mounted parachute for use in test flying to bring the nose down in the event of it entering a super-stall. For general airline operation, aircraft of this type are fitted with equipment called a stick pusher. This is actuated by an angle of attack sensor on the fuselage (usually de-iced) which senses that the angle of attack is approaching the stall. Signals are then sent to an electro-hydraulic system, the rams of which physically push the control stick forward, thus preventing the aircraft from entering the stall.

8.15 Detail Calculations and Factors Affecting Stalling Speed and Stalling Angle

Calculation of the stalling speed:

During level flight, lift is exactly equal and opposite to the weight.

Therefore: (i) Lift = Weight

The lift formula is: (ii) Lift = $C_L \tfrac{1}{2}\rho V^2 S$

It stands to reason that when the C_L is maximum, V must be a minimum value (low speed, high angle of attack).

This minimum value of velocity is, therefore, the stalling speed (V_s), when the C_L is at maximum value.

Therefore: (iii) Lift = Weight = $C_L(\max)\ \tfrac{1}{2}\rho V_s^2\ S$

So, rearranging the formula, it becomes:

(iv) Weight = $C_L(\max)\ \tfrac{1}{2}\rho V_s^2\ S$

Thus, to obtain the V_s (stalling speed), the formula is so rearranged:

Therefore (v) $\dfrac{\text{Weight}}{C_L(\max)\ \tfrac{1}{2}\rho S} = V_S^2$

($C_L \tfrac{1}{2}\rho s$ transposed)

Normal stall speed (vi) $V_S = \sqrt{\dfrac{\text{Weight}}{C_L(\max)\ \tfrac{1}{2}\rho S}}$

Factors affecting the stalling speed of an aircraft.

1. Weight
2. Load Factor
3. Wing Area
4. Change in $C_L(\max)$
5. Power and Slipstream

1. Weight

Any change in the weight of an aircraft will affect the stalling speed. It will be noted from the formula:

$$V = \sqrt{\frac{\text{Weight}}{C_L(\text{max})\ \tfrac{1}{2}\,\rho\, S}}$$

that if the weight increases, the
division thereof by C_L(max) S results in
an increased stalling speed (V_s).

2. Load Factor

Any manoeuvre that requires an increase in total lift without a corresponding increase in wing area, must increase the effective total weight acting on the aerofoils.

This apparent weight increase is known as a load factor, which is defined as the ratio of the load acting on the aircraft during the manoeuvre to the loading acting on the aircraft in straight and level flight.

$$\text{Load Factor} = \frac{\text{Total Lift}}{\text{Aircraft Weight}} = \frac{\text{Total Weight}}{\text{Actual Weight}}$$

As demonstrated in the previous paragraph, any increase in weight results in a higher stalling speed. This new stalling speed may be calculated from the following formula:

$$\text{New } V_s = \text{Old } V_s \times \sqrt{\text{Load factor}}$$

3. Wing Area (S)

Where increased wing area is obtained by the use of Fowler flaps, the division of a given weight by an increased value of (S) results in a lower value of V.

4. Change in C_L (max)

The use of flaps increases the C_L of that wing. Once again, the division of a given weight by a larger value of C_L results in a lower stalling speed. This is the advantage of the use of flap during the landing manoeuvre because it permits the original value of lift to be retained at a lower speed. It is particularly useful in the lowering of the approach speed.

5. Power and Slipstream

When power is applied at the stall, the already nose-high attitude produces a vertical component of lift. This consequently reduces the work load (ie weight) of the wings and allows a much lower stalling speed to be attained. The slipstream at high power settings provides an extra boost to the stagnating airflow over the aerofoil and thus controls the boundary layer. See Fig 8-7.

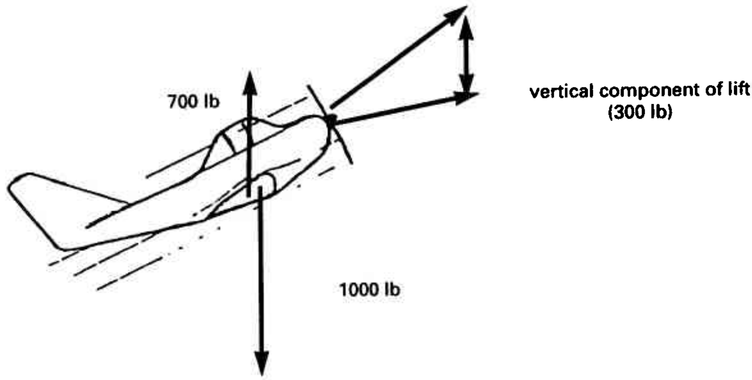


Figure 8–7

8.16 Wing Tip Stalling

An aircraft wing is designed to stall progressively from the root section to the tips. The reasons for this are as follows:

1. An early buffeting is induced over the tail sections.
2. Aileron effectiveness is maintained up to the stalling angle of attack.
3. Large rolling moments of the aircraft are prevented in the event of one wing tip stalling before the other.

Methods used in the prevention of tip stalling:

1. *Washout*:
This is a progressive reduction of wing incidence from the root to the tip. This results in the wing root reaching the critical angle of attack before the tip.

2. *Root spoilers*:
This method employs a triangular-section strip fixed to the leading edge of the wing near the root. At high angles of attack, the airflow is obstructed in following the contour of the leading edge and this results in a breakdown of the airflow whereby an early stall is induced at the wing root.

3. *Change of aerofoil section*:
The aerofoil section may be gradually changed by decreasing the camber slightly at or near the tips, or by sweepback. This results in a slight

decrease in lift at the tips thus giving an aerofoil with more gradual stalling characteristics from the root to the tip. The effect of sweepback is to increase the stalling angle.

4. *Slats and Slots*

By employing slats and slots on the outboard sections of the wing, the effective angle of attack at that part of the wing is decreased. Thus, when the root section reaches the critical angle of attack, the tip sections remain unstalled.

Note: Tapering the aerofoil from root to tip gradually reduces the C_L towards the tips; this in itself reduces the high rolling moment which would occur if the one tip stalled before the other.

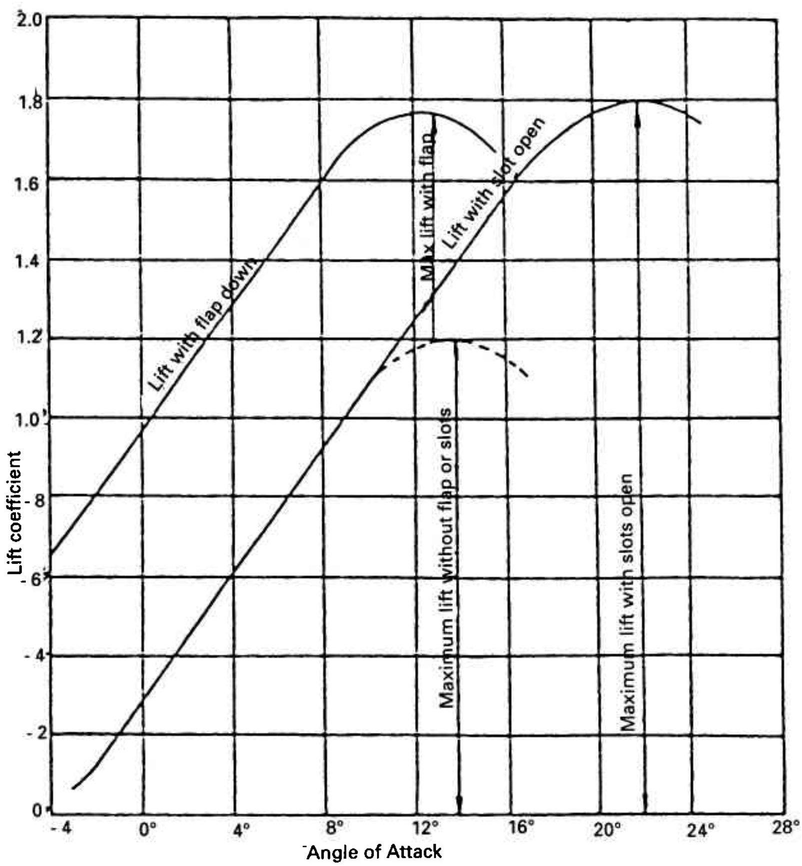


Figure 8–8 Effect of Flaps and Slots on Maximum lifts at Stalling Angle.

8.17 The effect of aspect ratio on the stalling angle:

Note: When referring to stalling angle, it is that angle with the horizon as viewed abeam by the pilot from the flight deck.

As discussed under wing tip vortices, the net direction of the airflow is altered.

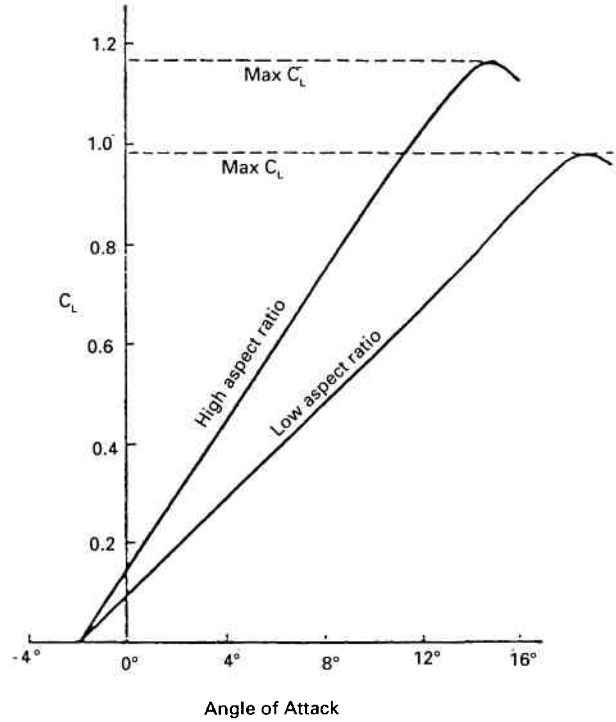


Figure 8-9: Effect of Aspect Ratio on the C_L(max).

Aircraft having high aspect ratios (long span and short chord) have very little induced downwash and, therefore, the net direction of the airflow remains largely unaltered. Conversely, aircraft with low aspect ratio wings (broad tips) induce a large amount of downwash which alters the net direction of the airflow significantly.

Because of this altered airflow, low aspect ratio wings have significantly higher stalling angles than do wings of high aspect ratio. (See Fig 8-10).

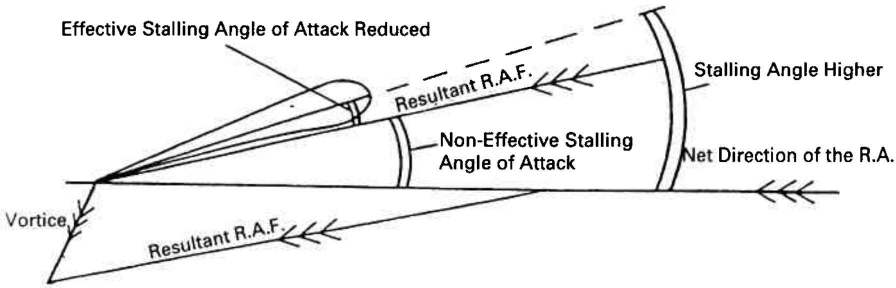


Figure 8–10

Note: This explains why rectangular wings usually stall from the root to the tip. The effective stalling angle of attack is reduced at the tips due to the presence of the wing tip vortex resulting in the net direction of the airflow being altered.

8.18 The Effect of Sweepback on the stalling angle:

In itself, a swept wing has a low aspect ratio and thus the presence of wing tip vortices are marked and give rise to a downwash that alters the net direction of the relative airflow. Since an aerofoil stalls when the critical angle between the chord line and the relative airflow is exceeded, the presence of the downwash alters this relative airflow and, having a downward component, results in the stalling angle being higher when the critical angle of attack is reached.

Swept wings therefore, have higher stalling angles than those of unswept wings (Fig 8-11).

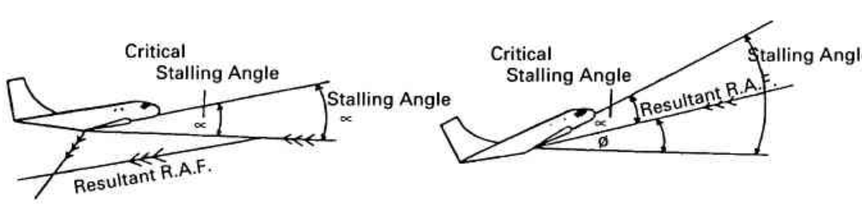


Figure 8–11

8.19 The effect of Flap on the stalling angle:

With each successive increase of flap, the characteristics of the aerofoil are changed, ie the chord line assumes a steeper inclination, being the straight line from leading edge to trailing edge. The critical stalling angle (about 15 degrees) is therefore reached with little or no inclination of the longitudinal axis of the aircraft (ie aircraft in straight and level attitude). Any further increase in flap setting in this attitude would result in the critical stalling angle of attack being exceeded. To prevent this, the aircraft would have to be placed in a nose down attitude, thereby reducing the critical angle of attack to within limits (about 15 degrees). Fig 8-12.

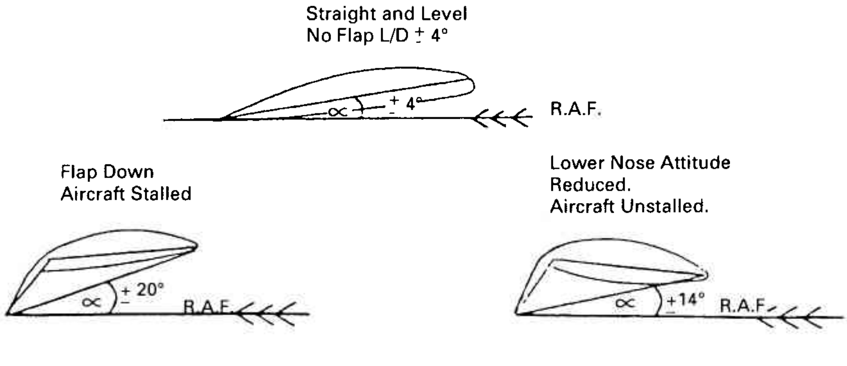


Figure 8–12

Thus, the effect of flap reduces the stalling angle although the critical angle of attack remains about 15 degrees.

Note: The stalling angle, or level flight stalling angle, is increased when leading edge flaps are employed.

Further reference to wing planforms and their stall characteristics are discussed in Chapter 10.

Key Points to Remember

1 With increased angle of attack, whilst maintaining level flight, induced and profile drag will increase.

2 Whilst maintaining level flight, at a constant altitude, an increase in speed will result in a reduction in induced drag.

3 Profile drag equals induced drag at V_{md}.

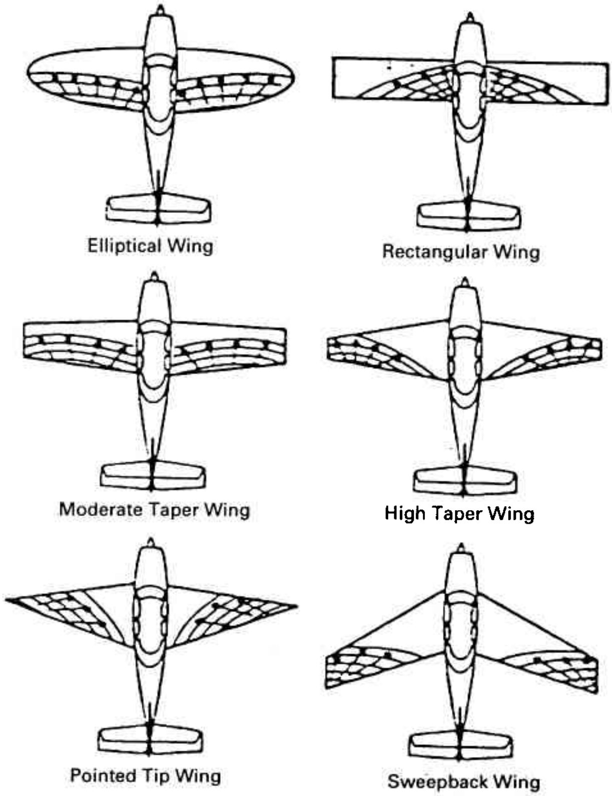


Figure 8–13 Wing Planforms (Exaggerated) and their stall patterns.

4 Profile drag is proportional to speed.

5 Induced drag is inversely proportional to speed.

6 With an increase of weight of the aircraft the V_{md} will increase.

7 With an increase in altitude the stalling angle will remain the same.

8 With an increase of speed the stalling angle will remain the same.

9 Washout of a wing will ensure the root of the wing stalls first.

10 Stall inducers may be fitted to the leading edge of the wing root to ensure that the wing stalls at the root first.

11 Stall warning sensors are normally fitted at, or near, the wing leading edge.

12 If the centre of gravity is forward of the centre of pressure a down load will exist on the tailplane.

13 If the centre of gravity is forward of the centre of pressure the stalling speed will be increased.

14 The nearer the centre of pressure is to the centre of gravity the lower the stalling speed will be.

Chapter 8: Test Yourself.

1 At the point a wing enters a stalled condition:

a) Lift and Drag rapidly reduce.
b) Lift slowly reduces and Drag rapidly increases.
c) Lift reduces sharply and Drag considerably increases.
d) Lift rapidly reduces and drag increases slightly.

Ref Para 8.3

2 A wing will stall:

a) at the stalling angle.
b) at the optimum angle of attack.
c) just below the stalling angle.
d) just above the stalling angle.

Ref para 8.1

3 With increasing altitude the angle at which a wing will stall:

a) remains the same.
b) reduces.
c) increases.

Ref para 8.2

4 As the angle of attack of a wing is increased in level flight:

a) the C of G moves aft and the C of P forward.
b) the C of P and transition point move forward.
c) the C of P moves aft and the separation point forward.
d) the C of P moves forward and the stagnation point aft over the upper surface.

Ref para 8.3

5 Stall inducers may be fitted to a wing:

a) at the tip to cause the root to stall first.
b) at the root to cause the tip to stall first.
c) at the root to cause the root to stall first.
d) at the tip to cause the tip to stall first.

Ref para 8.4

9

Spinning

9.1 Introduction

Spinning is a complex subject to explain in detail and cannot be described in general terms which are true for all types of aircraft. One type of aircraft may behave in a certain manner in a spin whilst another type will behave quite differently under the same conditions. In the example given in this chapter the spin is taken to be deliberately induced, erect and to the right.

9.2 Phases of the Spin

The full spin manoeuvre consists of three fundamental phases:

(a) The incipient spin.

(b) The fully developed spin.

(c) The recovery.

(d) The steady erect spin.

(a) The Incipient Spin
A necessary ingredient of a spin is the aerodynamic movement known as autorotation. This is basically the rotational movement of the aircraft about its normal axis, and it leads to an unsteady motion which is a combination of:

i) The ballistic path of the aircraft, which is dependent on the entry attitude.

ii) An increasing angular velocity generated by the autorotative rolling moment and the drag induced yawing moment.

(b) The Steady Spin
The incipient stage of the spin may continue for some two to six turns after which the aircraft will settle down into a steady stable spin. There will be some sideslip and the aircraft will rotate about all three axes. In most cases this stable condition, the steady spin, is characterised by a steady rate of rotation and a steady rate of descent.

(c) The Recovery

The recovery is initiated by the pilot's operation of the controls first to oppose the autorotation and then to reduce the angle of attack so as to unstall the wings. A steep dive thereupon ensues from which the aircraft may be normally returned to straight and level flight.

(d) The Steady Erect Spin

During rotation the aircraft will describe a ballistic trajectory the character of which will be dependant upon 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 rates both of descent and of rotation about the axis of the spin. This will occur if the aerodynamic and inertia forces and moments 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.

9.3 Motion of the Aircraft

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

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

ii) 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.

The combination of these motions results in the aircraft descending in a vertical spiral or helix. The helix angle is usually small – generally less than 10°. Fig 9-1 shows the motion of the aircraft in a spin.

As the aircraft always presents the same face to the axis of the spin, it follows that it must be rotating about a vertical axis passing through the centre of gravity at the same rate as the CG is rotating about the spin axis.

The angular velocity may be resolved into components of roll, pitch and yaw with respect to the axes of the aircraft itself. In the spin shown in Fig 9-1b the aircraft is rolling right. For convenience the direction of the spin is defined by the direction of yaw.

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

(a) Longitudinal Axis Vertical

When the longitudinal axis is vertical the angular motion will be a roll.

(b) Lateral Axis Vertical
For the aircraft to present the same face to the axis of the spin, 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 its normal axis at the same rate at which it rotates about the axis of rotation. Thus the angular motion is all yaw.

Although these examples are hypothetical and may not be possible in practical terms, they illustrate the relationship between the aircraft's 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:

(i) The rate of rotation of the aircraft about the spin axis.

(ii) The attitude of the aircraft. This is usually defined in terms of the pitch angle and the wing tilt angle. Wing tilt angle (which must not be confused with bank angle), involves simultaneous displacement about the normal and the longitudinal axes.

The aircraft's attitude in the spin also has an important effect on the sideslip present, as shown in Fig 9-1c. If the wings are level, there will be

Axis (Symbol)	**Longitudinal** (x)	**Lateral** (y)	**Normal** (z)
Positive Direction	Forwards	To right	Downwards
Angular Velocity			
Designation	Roll	Pitch	Yaw
Symbol	p	q	r
Positive Direction	to right	nose-up	to right
Moments of Inertia	A	B	C
Moments			
Designation	rolling moment	pitching moment	yawing moment
Symbol	L	M	N
Positive Direction	to right	nose-up	to right

Table 1: Sign Conventions Used in this Chapter.

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.

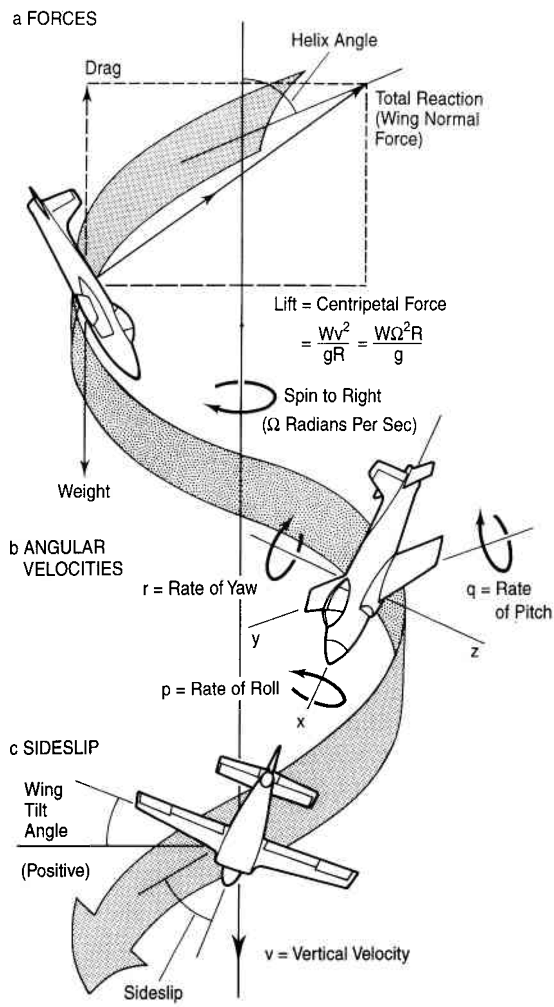


Figure 9–1 The motion of an aircraft in an Erect Spin to the right

9.4 Balance of Forces in the Spin

Only two forces are acting on the centre of gravity while it is moving along its helical path, as can be seen in Fig 9-1 a:

a) Weight

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.

Since the weight and 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. 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.

If the wings are level (lateral axis horizontal), then from the balance of forces in Fig 9-1 a:

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

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

b. $$\text{Lift} = \text{Centripetal force}$$

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

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

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

If the wings are not level, 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.

9.5 Effect of Attitude on Spin Radius

If for some reason the angle of attack is increased by a nose-up change in the aircraft's attitude, Fig 9-2, the vertical rate of descent V will decrease because of the higher C_D. 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. It can also be shown that an increase in pitch increases the rate of spin, which will decrease R still further.

The two extremes of aircraft attitude possible in the spin are shown in Fig 9-2. The actual attitude adopted by an aircraft will depend on the balance of moments.

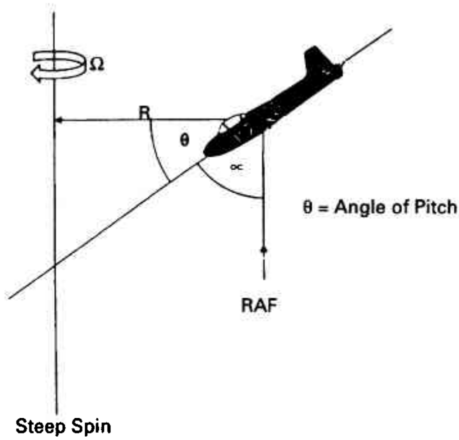


Figure 9-2 Simplified diagram of Pitch Attitude.

The effects of pitch attitude are summarised below:

An increase in pitch (ie a flatter spin) will:

a Decrease the rate of descent.

b Decrease the spin radius.

c Increase the spin rate.

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

9.6 Angular Momentum

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. 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 effects of mass distribution in order to understand the spinning characteristics of different aircraft and the effect of the controls on recovery from the spin.

Moment of Inertia (I)
To predict the behaviour of a rotating system it is necessary to comprehend the nature of inertia moments. A moment of inertia expresses not only the amount of a 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. Figure 9-3 illustrates how the distribution of mass affects angular momentum.

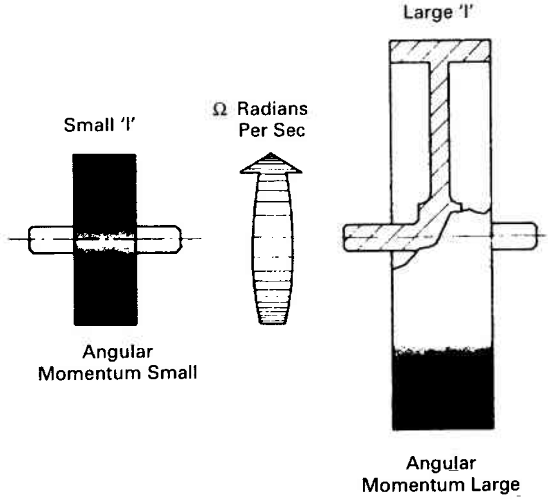


Figure 9-3 Two rotors of the same weight and angular velocity

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

Longitudinal Axis.
The distribution of the 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 wing fuel tanks will have a large value of A, particularly if the fuel tanks are close to the wing tips. On some aircraft types some fuel may also be stored in fuselage fuel tanks, and this combined with a lower aspect ratio will result in a reduction of A for such aircraft types.

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 mass being distributed along the whole length of the fuselage with a consequent increase in the value of B.

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. These moments of inertia measure the mass distribution about the body axes and are decided by the design of the aircraft. It is thus implicit that the values of A, B and C for a particular aircraft will be changed if the disposition of equipment, freight and fuel is altered.

9.7 Inertia Moments in a Spin

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:

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

Pitch
The imaginary concentrated masses of the fuselage as shown in Figure 9-4 tend to flatten the spin.

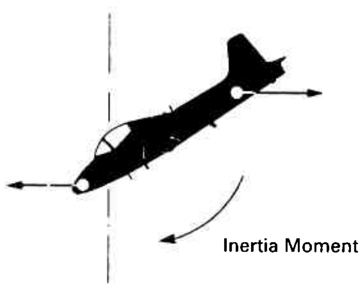


Figure 9–4 Inertia pitching moment.

Yaw

The inertia couple is complicated by the fact that it is comprised of two opposing couples caused by the wings and fuselage, as shown in Fig 9-5. Depending on the dominant component, the couple can be of either sign and of varying magnitude. The inertia yawing moment can be expressed as:

$N = (A - B)pq$, is negative and thus anti-spin when $B > A$; positive and pro-spin when $A > B$. The B/A ratio has a profound effect on the spinning characteristics of an aircraft.

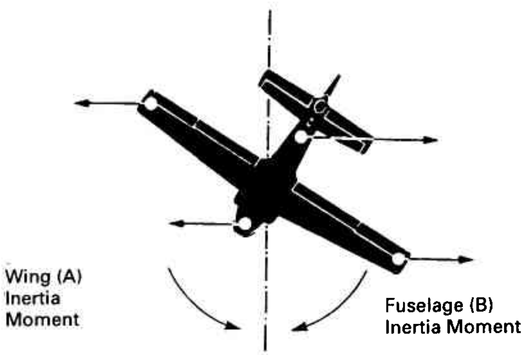


Figure 9-5 Inertia yawing moments.

9.8 Aerodynamic Moments

At this stage it is necessary to examine the contributions made by aerodynamic factors in the balance of moments in roll, pitch and yaw. These are discussed separately below.

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. Even at angles of attack above the stall, this still remains true and the dihedral effect induces a rolling moment in the opposite direction or sense to the sideslip. In the spin the relative airflow is from the direction of the outer wing (outward sideslip) and the result is a rolling moment in the direction in which the aircraft is spinning; this contribution is therefore pro-spin.

(b) Autorotative Rolling Moment

It can be 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:

(i) The difference in speed of the wings
Lift of the outer wing is increased and that of the inner wing decreased inducing a pro-spin rolling moment.

(ii) Differences in angle of attack of the wings
In a spin the direction of the free stream airflow is practically vertical whereas the direction of the wing motion due to the yaw is parallel to the longitudinal axis. The yawing velocity therefore changes not only the speed but also the angle of attack of the wings. Fig 9-6 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 outer wing C_L is increased and the C_L of the inner wing decreased, thus producing another pro-spin rolling moment.

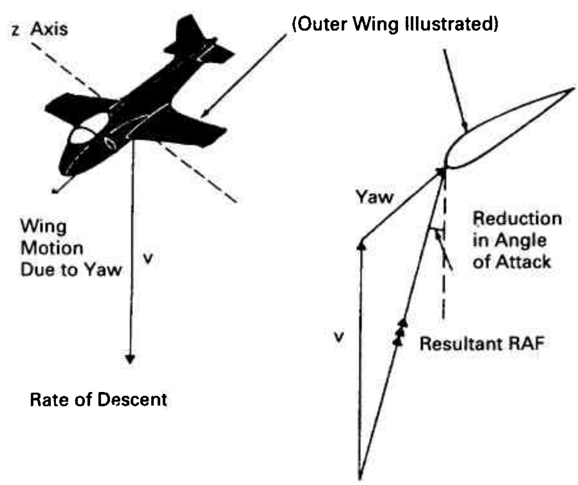


Figure 9-6 Change in angle of attack due to yaw
(outer wing)

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

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:

(e) Positive Longitudinal Static Stability
In a spin the aircraft is at a high angle of attack and is therefore disturbed in the nose-up sense by 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.

(f) Damping in Pitch effect
When the aircraft is pitching nose up the tailplane is moving down and its angle of attack is increased. The pitching velocity therefore produces a pitching moment in a nose down sense. The rate of pitch change in a spin is, however, usually very low and consequently the damping in pitch contribution is small.

(g) 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.

Aerodynamic Yawing Moments
The overall aerodynamic yawing moment is made up of a large number of separate elements, some resulting from the yawing motion of the aircraft and some arising out of the side slipping motion. The main contributions to balancing the moments about the normal axis to produce a steady rate of yaw are as follows:

(h) Positive Directional Static Stability
When sideslip is present the keel surfaces (Fin and Fuselage) aft of the CG produce an aerodynamic yawing moment tending to turn the aircraft into alignment with the sideslip vector (ie directional static stability or weathercock effect). This is an anti-spin effect, the major 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. Outward sideslip, however, usually produces a net yawing moment towards the outer wing, ie in the 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 very small.

(j) Damping in Yaw effect

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.

Fuselage strakes, see Fig 9-7, are useful devices for improving the characteristics in a spin on some types of 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. Combining fin and tailplane into a V or Butterfly tail has occasionally been used to improve spin recovery and has the additional advantages of lighter construction and less drag.

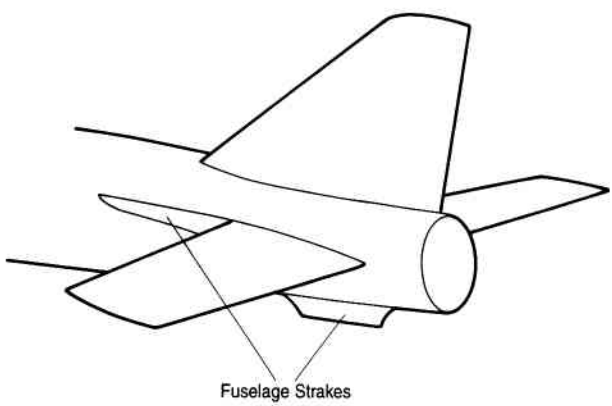


Figure 9-7 Fuselage Strakes.

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 it is usual during recovery to pause after applying out of spin rudder so that the anti-spin yawing moment may take effect before down elevator is applied.

Balance of Moments

It can be seen that the balance of forces in a spin has a strong influence on the rate of descent. It does not, however, determine the rate of rotation, wing tilt or the 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 spin of a spinning aircraft can be determined only by applying specific numerical values of the aircraft's aerodynamic and inertia data to the general relationships discussed below.

Rolling Moments

The balance of rolling moments in an erect spin is:

a. *Pro-spin*: The following aerodynamic rolling moments in an erect spin are:
   (i) Autorotative rolling moment.
   (ii) Rolling moment due to sideslip.
   (iii) 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 rolling moment effect due to sideslip. This in turn, would increase 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.

Pitching Moments

It has been previously stated 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 equalibrium can usually be achieved over a wide range. Increase in pitch will cause an increase in the rate of rotation (spin rate). This in turn will decrease the spin radius.

Yawing Moments

The balance of yawing moments in an erect spin is:

a. *Pro-spin:*

   (i) Yawing moment due to applied rudder.

(ii) A small contribution from the wing, due to yaw, is possible at large angles of attack.
(iii) Yawing moment due to sideslip (vertical surfaces forward of CG).
(iv) Inertia yawing moment, $(A - B)pq$, if $A > B$.

b. *Anti-spin:*

(i) Inertia yawing moment, if $B > A$.
(ii) Yawing moment due to sideslip (vertical surfaces aft of the CG).
(iii) 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.

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 (incidence, spin rate and attitude). For aircraft in which B is much larger than A, the inertia yawing moment can be large and, thus difficult to balance. This could be the cause of the oscillatory spin often found with such aircraft.

Yaw and Roll Axis
The requirements of balance about the yaw and roll axes greatly limit the range of incidence in which spinning can occur, and determine the amount of sideslip and wing tilt involved. The final balance of 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.

9.9 Spin Recovery

Effect of Controls in Recovery from a Spin
The relative effectiveness of the three controls in recovery from a spin will now be considered. Recovery is achieved by stopping the rotation and this in turn is achieved 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, the rudder is not the only means by which yawing

may be induced by the pilot. Once the rotation has stopped the incidence is reduced and the aircraft recovers.

The control movements which experience has shown to be generally most favourable to 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 become the primary control although, in the final analysis, it is its effect on the yawing moment which makes it work.

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

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 will, in turn:
   - (i) Cause a small increase in the anti-spin inertia rolling moment, (C-B)rq, (C>B) and thus help to restore balance about the roll axis.
   - (ii) 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.

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.

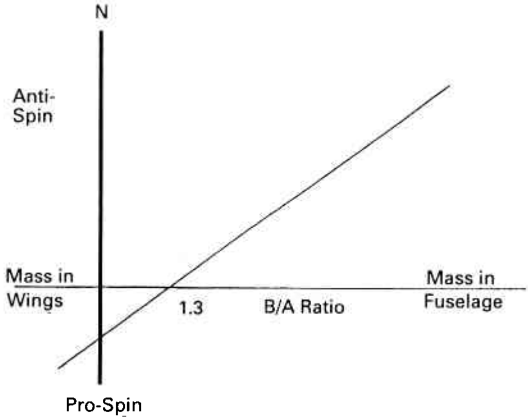


Figure 9-8 Yawing Moment (N) per degree
of Aileron

Some aircraft have their B/A ratio changed in flight through consumption of stores and fuel. 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 during a spin to maintain ailerons neutral to avoid an unfavourable response which may delay or even prohibit recovery.

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

Effect of Elevators

It has already been stated 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:

(i) Inertia Yawing Moment (A-B)pq. If B > A, the inertia yawing moment is anti-spin. A reduction in q will make the inertia 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.

(ii) 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 9-9) and rate of roll. The rate of rotation about the spin axis will increase.

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.

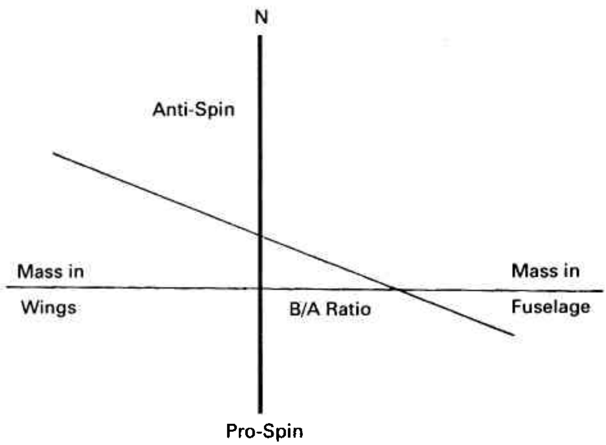


Figure 9-9 Yawing Moment (N) per degree of Down Elevator

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

The rudder is nearly always effective in producing an anti-spin aerodynamic yawing moment although 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.

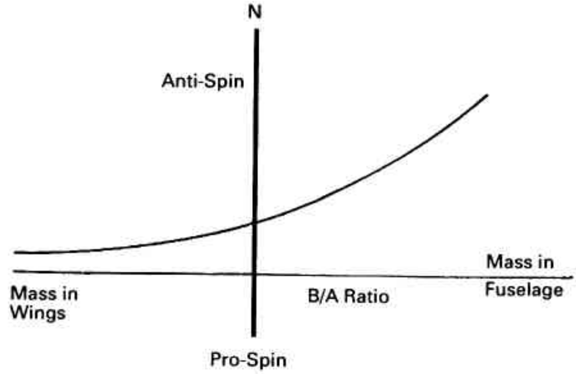


Figure 9-10 Yawing Moment (N) per Degree of Anti-spin Rudder

Inverted Spin

Figure 9-11 shows an aircraft in an inverted spin but following the same flight path as in Fig 9-1. 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.

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

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

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.

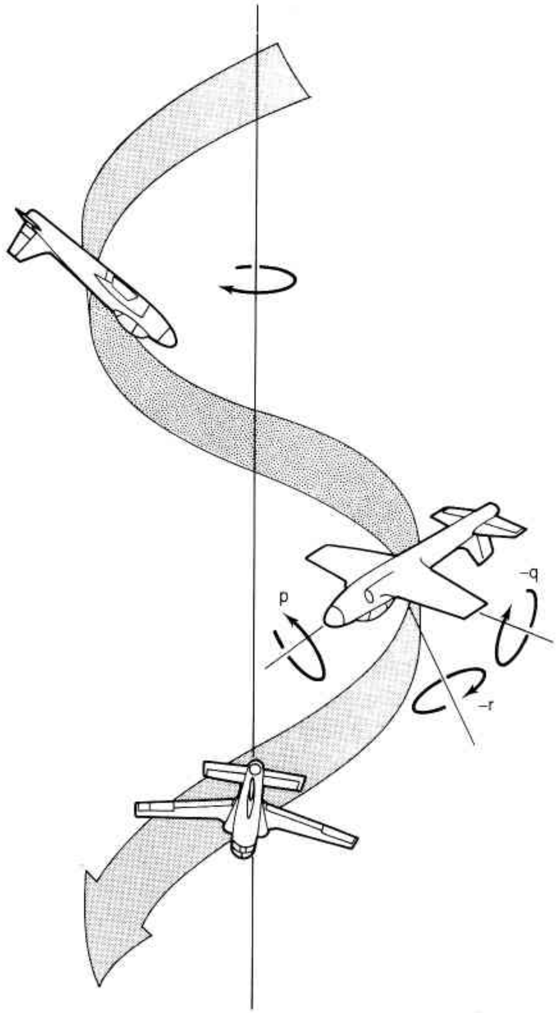


Figure 9-11 The Inverted Spin.

(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 may be less favourable and may even become pro-spin.

Oscillatory Spin
A combination of high wing loading and high B/A ratio makes it difficult for such 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 in each turn; the aircraft seems to wallow in the spin.

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. 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, (eg in-spin aileron or anti-spin rudder) will increase the violence of the oscillations.

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

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
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.

9.10 Gyroscopic Cross-coupling Between Axes

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 visualised as acting rather in the manner of the bob-weights of a governor.

Another, and more versatile, explanation of the cross-coupling effects can be given by analogy with gyroscopic precession, regarding the aircraft as a rotor.

Inertial Moments in a Spin

The inertial moments generated in a spin are essentially the same as the torque exerted by a precessing gyroscope. Figures 9-12, 9-13 and 9-14 illustrate the inertial or gyroscopic moments about the body axes. These effects are described below:

(a) Inertial Rolling Moments (Fig 9-12)

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

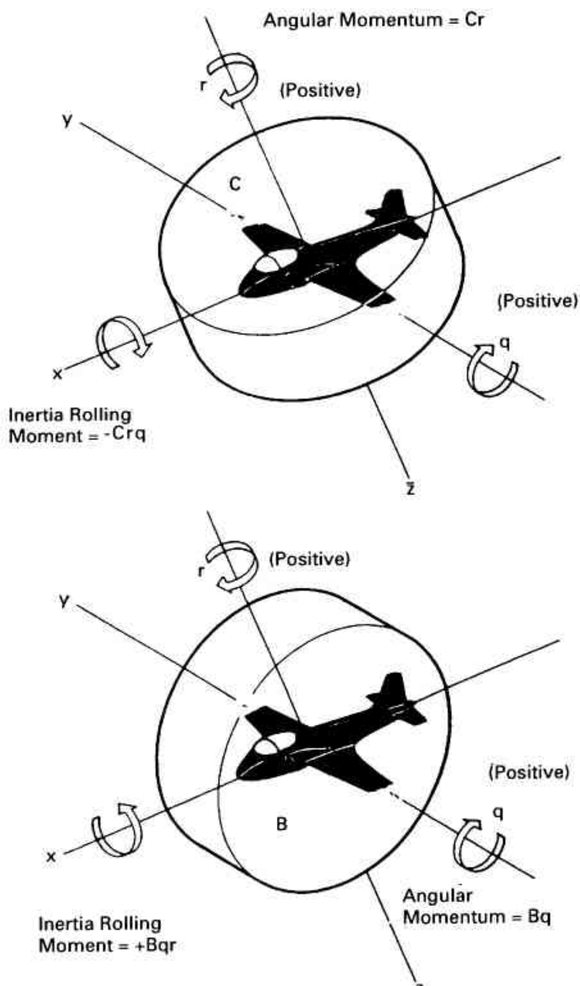


Figure 9-12 Total Inertia Rolling Moment.

momentum in the pitching plane is in a pro-spin sense and equal to +Brq. The total inertia rolling moment is therefore equal to (B-C)rq, or since C > B: -(C-B)rq.

(b) Inertial Pitching Moments (Fig 9-13)

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

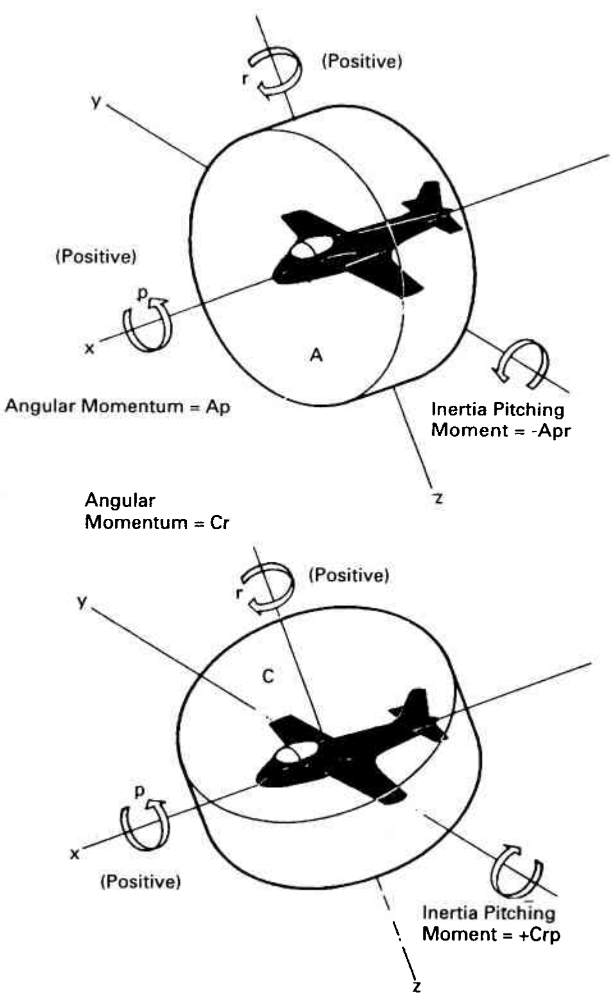


Figure 9-13 Total Inertia Pitching Moment.

(+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.

(c) Inertial Yawing Moments (Fig 9-14)

Regarding the aircraft as 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

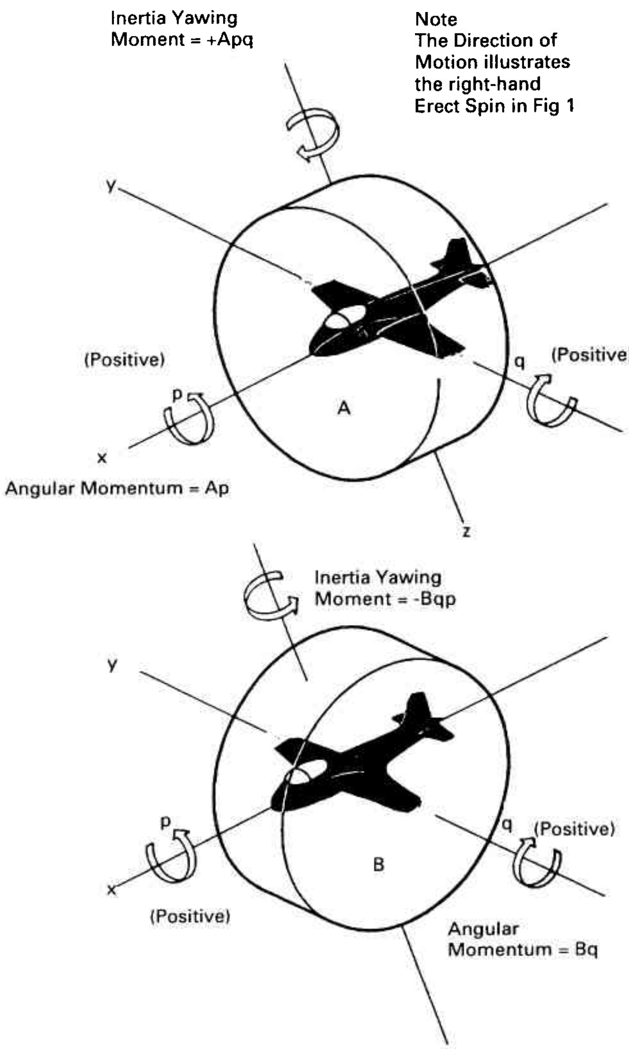


Figure 9-14 Total Inertia Yawing Moment

rotor will generate a torque tending to precess the rotor about the normal axis in the same direction as the spin. This inertial yawing moment is equal in value to +Apq where the positive sign indicates a pro-spin 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 inertial yawing moment in an anti-spin sense equal to -Bpq. The total inertial yawing moment is therefore equal to (A-B)pq, or if B > A: -(B-A)pq.

Key Points to Note:

1. A necessary ingredient of a spin is the aerodynamic movement known as autorotation.
2. Autorotation basically is the continuous rotation of the aircraft about its normal or vertical axis, (or spin axis). This will normally be coupled with a rolling moment.
3. In most cases a steady spin is qualified by a steady rate of rotation and a steady rate of descent.
4. Recovery from a spin is initiated by the pilot's control operation, first to oppose the autorotation and then to reduce the angle of attack.
5. In a spin, the angle of attack of the inner wing is greater than the angle of attack of the outer wing.
6. In a spin, the inner wing is fully stalled and the outer wing partially stalled.
7. An increase in pitch in a flat spin will:
   (i) Decrease the rate of descent
   (ii) Decrease the spin radius
   (iii) Increase the spin rate
8. Even when a wing is stalled the ailerons produce a rolling moment.
9. In a spin, the aircraft is at a high angle of attack.
10. Shielding of the fin by the tailplane can considerably reduce the effectiveness of the fin as a stabilising surface during a spin. This has led to the employment of a V, or Butterfly Tail on some aircraft as an aid to better spin recovery.

Chapter 9: Test Yourself.

1 When recovering from a spin the pilot should first:

a) reduce the angle of attack.
b) oppose the autorotation.
c) increase the angle of attack.
d) increase the drag yawing moment.

Ref para 9.10

2 In a spin:

a) the angle of attack is the same on both wings.
b) the angle of attack is greater on the inner wing.
c) the angle of attack is greater on the outer wing.
d) both inner and outer wings are fully stalled.

Ref para 9.10

3 At high angles of attack in the spin:

a) elevators act in the opposite sense.
b) ailerons act in the normal sense.
c) ailerons are totally non effective.
d) rudder control is lost.

Ref para 9.10

4 The centripetal force in a spin is a component of:

a) weight only.
b) weight and centrifugal force.
c) lift.
d) lift and weight.

Ref para 9.10

5 In a spin:

a) both wings are fully stalled.
b) the outer wing is fully stalled and the inner wing partially stalled.
c) both wings are partially stalled.
d) the inner wing is fully stalled and the outer wing partially stalled.

Ref para 9.10

10

Wing Planforms

10.1 Introduction

The previous chapters have considered the basics of lift, drag, stalling and spinning and explained the causes of these phenomena. It is however, also necessary to study another important aspect of the design of wings, ie the planform. The planform is the geometrical shape of the wing as viewed from above; it largely determines the amount of lift and drag obtainable from a given wing area, and has a marked effect on the value of the stalling angle of attack.

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

10.2 Aspect Ratio

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

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

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

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.

10.3 Aspect Ratio and Induced (Vortex) Drag

The origin and formation of trailing edge and wing tip vortices was explained earlier and it was shown that induced downwash was the cause of induced drag. The induced drag produced by a wing is proportional to the lift generated.

The amount of induced drag under a given set of conditions can be found from the formula:

$$\text{Induced drag} = \frac{kC_L^2qS}{\pi A},$$

$$\text{Where } \frac{C_L^2}{\pi A} = \text{the coefficient of induced drag,}$$

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

From the formula it can be seen that induced drag is inversely proportional to aspect ratio. A graph showing the curves of two different aspect ratio wings plotted against C_D and angle of attack is given in Fig 10-1.

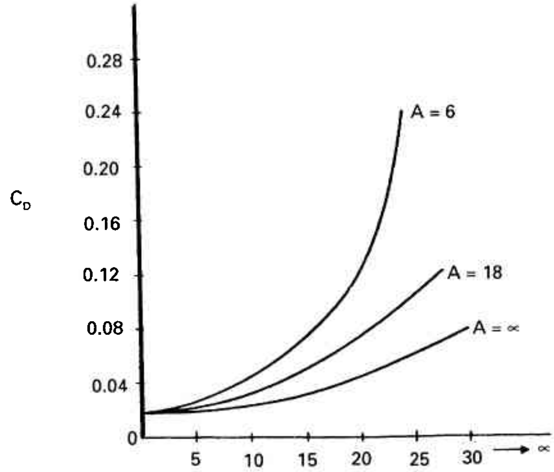


Figure 10-1 Effect of Aspect Ratio on C_D

10.4 Aspect Ratio and Stalling Angle

From the previous paragraph it can be seen that as the aspect ratio is decreased, so the induced drag is increased. It was also stated that the stall occurs when the effective angle of attack reaches the critical angle. Thus for a given aerofoil section the higher the aspect ratio, the lower is the stalling angle of attack. For a simple example, consider a wing in level flight: if there were no induced downwash (and hence no induced drag) then the wing would stall when the angle of attack reached its critical angle relative to the 'horizontal' total airstream past the wing.

However, all wings have vortices and so induce a downward component in the direction of the total airflow; thus the lower the aspect ratio, the larger the vortices and the greater the induced downwash. The stall will therefore occur when the effective angle of attack, which now has a downward component, reaches the critical angle. In the two cases which have been considered, the stalling angle of the wing with no induced drag is the lower by, approximately, the angular degree of the induced downwash.

The reduced effective angle of attack of very low aspect ratio wings can delay the stall considerably. Some delta wings have no measureable stalling angle up to 40° 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 in fact be far from a true stall and still be generating appreciable lift.

10.5 Use of High Aspect Ratio

While a high aspect ratio wing will minimize induced drag, long thin wings conversely increase weight and have relatively poor stiffness characteristics. Also the effects of vertical gusts on the airframe are aggravated by higher 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 use of low aspect ratio planforms.

10.6 The Effects of Taper

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. Certain local effects, however, take place throughout the span of the wing and these are due to the distribution of area throughout the span. The typical lift distribution is arranged in some elliptical fashion.

The natural distribution of lift along the span of 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. The effect of the elliptical planform is illustrated in Fig 10-2 by the plot of local lift coefficient C_L to wing coefficient, $^{C_l}/_{C_L}$, against semi-span distance. The elliptical wing produces a constant value of $^{C_l}/_{C_L} = 1.0$ throughout the span from root to tip. Thus, the local section angle of attack α_o and local induced angle of attack, α_i 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.

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_i and α_o, a large, local α_i reduces the local α_o 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 10-2 which depicts a local coefficient at the root almost 20% greater than the wing lift coefficient.

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 and 5% higher induced drag than an elliptical wing of the same aspect ratio.

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 towards the tip. In the immediate vicinity of the pointed tip an upwash is encountered which indicates that negative induced angles of attack exist in that area. The resulting variation of local lift coefficient shows low C_l at the root and very high C_l at the tip. The

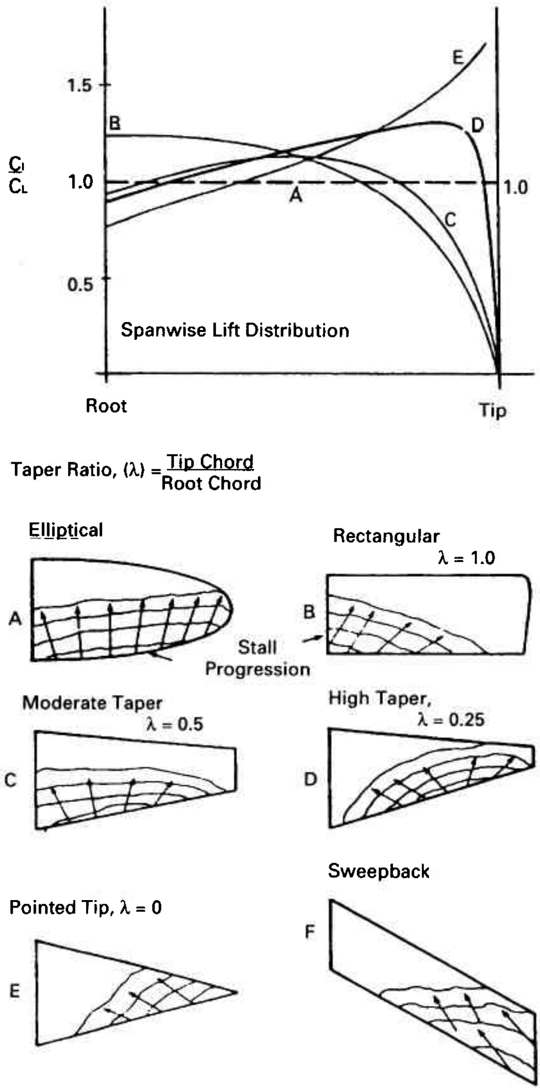


Figure 10-2 Lift Distribution and Stall Patterns.

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 $^{C_l}/_{C_L}$ throughout the span of the wing of taper ratio = O is shown on the graph of Fig 10-2. As with the rectangular wing, the non-uniformity of downwash and lift distribution result in the inefficiency of this planform. For example, a pointed wing of A=6 would have 17% higher induced angle of attack and 13% higher induced drag than an elliptical wing of the same aspect ratio.

Between the two extremes of taper will exist planforms of more tolerable efficiency. The variations of $^{c}/_{c_L}$ 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 are nearly identical. A wing of A=6 and taper ratio =0.5 has only 3% higher α_i and 1% greater C_{D_i} than an elliptical wing of the same aspect ratio.

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 aerodynamic efficiency, however, the planform is tailored by wing twist and section variation to obtain as near as possible the elliptic lift distribution.

10.7 Stall Patterns

An additional outcome of planform area distribution is the stall pattern of the wing. The desirable stall pattern of any wing is one where the stall begins at the root sections first. The advantages of the root stalling 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, in fact, be difficult to obtain with certain wing configurations. The types of stall pattern inherent with various planforms are illustrated in Fig 10-2. 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 the stall at essentially the same wing angle of attack and the stall will both 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. This stall pattern is of course, 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, however, the rectangular wing finds limited application, chiefly 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 that of 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 that 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 an aircraft which is definitely subsonic in performance.

(f) Sweepback applied to a wing planform alters the lift distribution in a way similar to decreasing the taper ratio. The full significance of sweepback are discussed in the following paragraphs.

10.8 Sweepback

Swept-back Leading Edges

This type of planform is used on high speed aircraft and may take the form of a swept-back wing, or of a delta with or without a tailplane. The reason for the use of these planforms is their low drag at the higher speeds. The high speed/low drag advantages are however, gained at the cost of a poorer performance at the lower end of the speed scale.

Effect of Sweepback on Lift

If a straight wing is changed to a swept planform, with similar parameters of area, aspect ratio, taper, section and washout, the $C_{L_{max}}$ is reduced. This is due to premature flow separation from the upper surface at the wing tips. For a sweep angle of 45°, the approximate reduction in $C_{L_{max}}$ is around 30%. Figure 10-3 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.

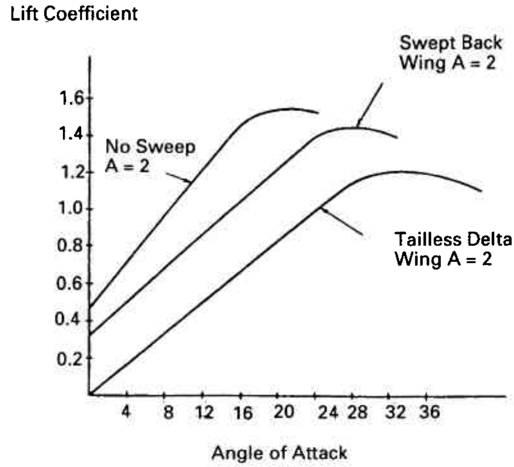


Figure 10-3 Effect of Planform on C_{Lmax}

The main reasons for the lowering of the C_L slope is best explained by examination of Figs 10-4 and 10-5. From Fig 10-4 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.

Figure 10-5 shows that an increase in fuselage geometric incidence $\Delta\alpha$ 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 it has already been 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.

Considering Figure 10-3, 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_{Lmax}.

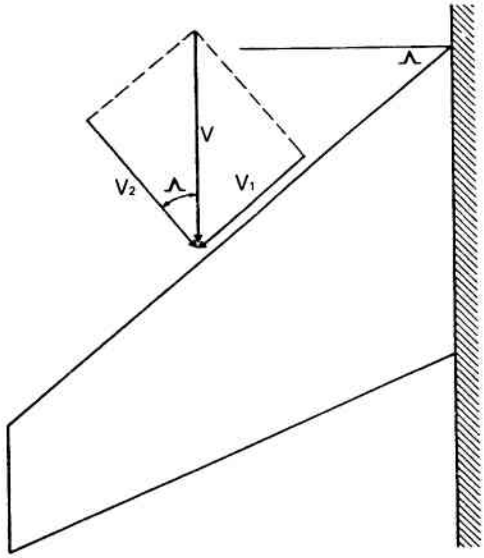


Figure 10-4 Flow Velocities on a Swept Wing

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 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 15°, and the angle of attack at touch-down will therefore have to be something of this order. Figure 10-3 shows that the C_L corresponding to this angle of attack is lower than the C_{Lmax} for each wing. 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. Figure 10-6 shows typical planforms for the three types of wing under consideration, with areas adjusted to give the same stalling speed. The much larger area of the delta wing is evident.

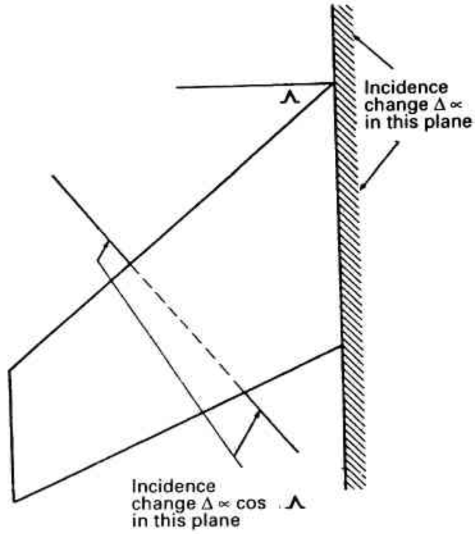


Figure 10-5 Effect on Angle of Attack by Incidence Change

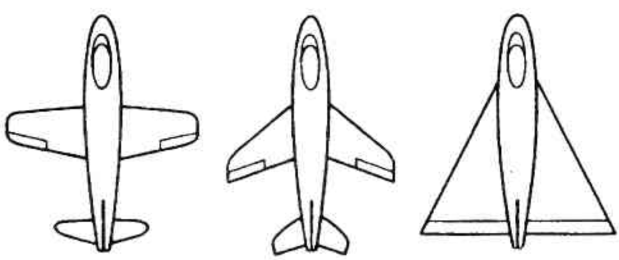


Figure 10-6 Planform Areas giving a Common Stalling Speed

Effect of Sweepback on Drag

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, C_L is reduced by sweeping the wing, 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.

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 that 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, with a large increase in power needed to restore equilibrium. In fact, the stage may be reached where even the use of full power is insufficient to prevent the aircraft from descending rapidly.

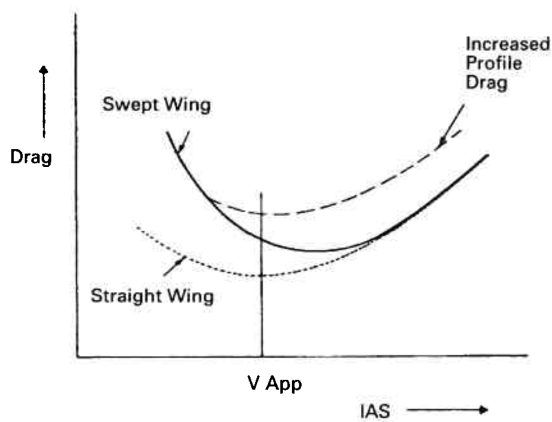


Figure 10-7 Improvement in Approach Speed Stability

On some aircraft this problem is overcome by employing high drag devices, such as airbrakes or drag-chutes, to increase the profile drag. This results in a flatter drag curve with the minimum drag speed closer to the approach speed, see Fig 10-7. A further advantage is that more power is required on the approach, which on turbojet aircraft, means better engine response.

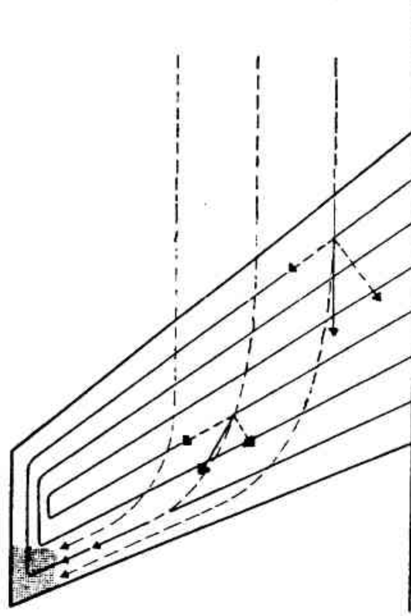


Figure 10-8 Outflow of Boundary Layer.

Effect of Sweepback on Stalling

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. The pressure distribution on a swept wing is shown by isobars in Fig 10-8. The velocity of the flow has been shown by two components, one at right angles and the other 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, therefore the result is that the actual velocity is reduced (as it is over an unswept wing) and also directed outwards towards the tips.

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.

The spanwise drift initiates 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_{Lmax} over the wing tips.

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 on 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 10-9 and the sequence of the vortex development and its effect on the airflow over the wing is shown in Fig 10-10. From the latter 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 10-11).

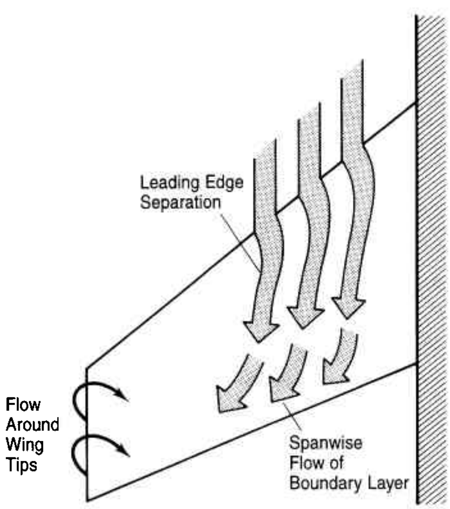


Figure 10-9 Vortex Development.

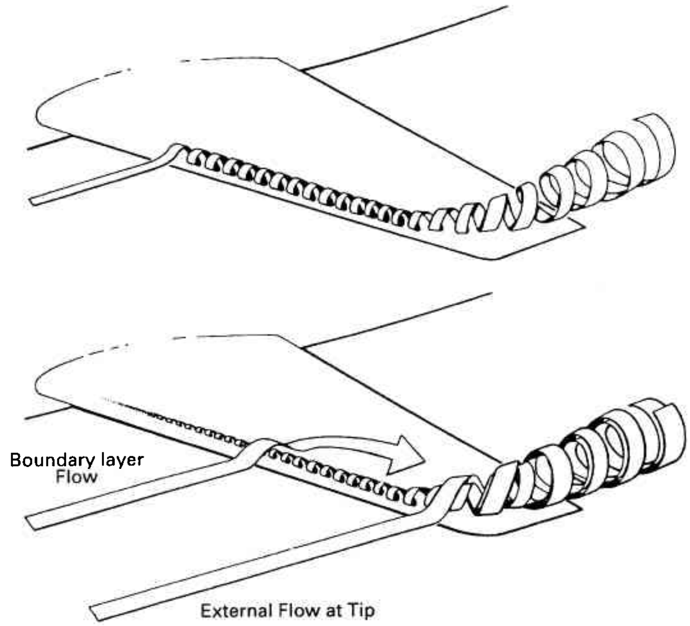


Figure 10-10 Formation of Ram's Horn Vortex

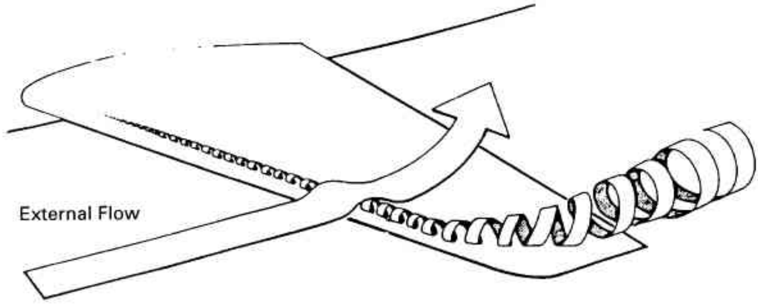


Figure 10-11 Influence on External Flow.

The spanwise flow of the boundary layer increases as angle of attack is increased. This causes the vortex closer inboard to become detached from the leading edge (see Fig 10-12). 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.

10.9 Alleviating the Tip Stall

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.

(b) Leading Edge Slots
These have the effect of re-energizing the boundary layer.

(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.

(d) Boundary Layer Blowing
High velocity air is injected into the boundary layer to increase its energy.

(e) Vortex generators
The purpose of these devices is to re-energise the boundary layer by making it more turbulent. The increased turbulence results in high-energy air in layers immediately above the seriously 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.

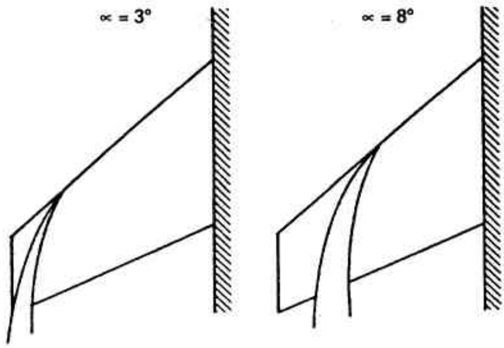


Figure 10-12 Shift of Ram's Horn Vortex

(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. 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:

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

(ii) The 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 farther forward and, when the tip eventually stalls, the forward shift in CP is less marked, therefore 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 in so far 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 an extended leading edge, the effect being to intensify the inboard vortex behind the devices to create a stronger restraining effect on boundary layer outflow. The choice whether to use either or both of these devices lies with the designer and depends on the desired flight characteristics of the aircraft.

10.10 Pitch-up

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.

Centre of Pressure Movement

When the swept-back wing is installed, 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 10-13) is reduced and a nose-up pitching moment results which aggravates the tendency.

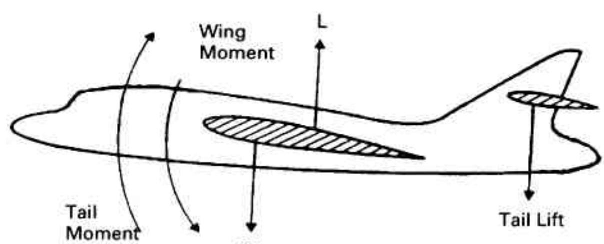


Figure 10-13 Nose-up Pitching Moment Resulting from Tip Stalling

Change of Downwash over the Tailplane

Figure 10-14 shows that the maximum downwash from a 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 the downwash changes with angle of attack are less marked.

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. Therefore in Fig 10-15, the points A and B rise through the same distance and the points C and D rise through a same distance but one that is greater than A and B. Therefore C rises farther than A and there is a consequent loss in incidence at this section. This aeroelastic effect is termed 'washout due to flexure', and it is obviously greatest at the wing tips.

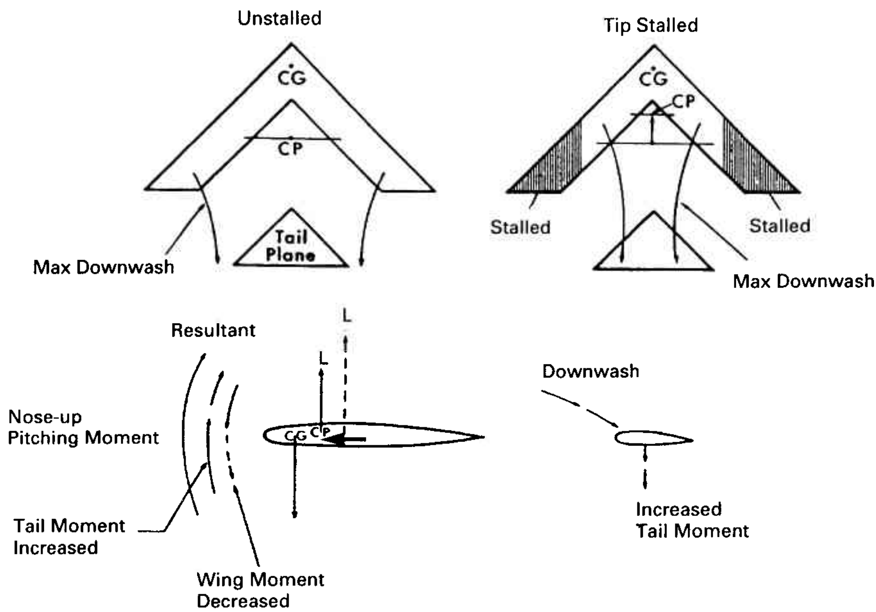


Figure 10-14 Variation of Downwash

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.

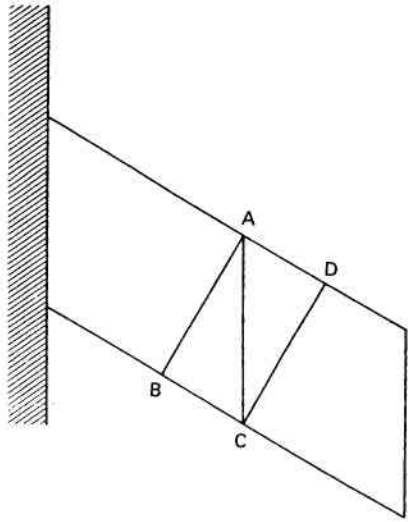


Figure 10-15 Washout due to Flexure

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 incidence is decreased giving rise to a pitch-up tendency.

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 farther 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 incurring accelerations due to g force, the onset of pitch-up can be violent and sudden, corresponding to the rate of spread of the stall.

The Crescent Wing
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.

10.11 Forward Sweep

General

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.

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

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.

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 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
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 increased 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 itself leads to increased effective angle of attack at the tips. The result of this aeroelastic 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 made possible controlled wing twist under load; so allowing the effect described to be eliminated.

Vortex Generation

Figure 10-16 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.

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.

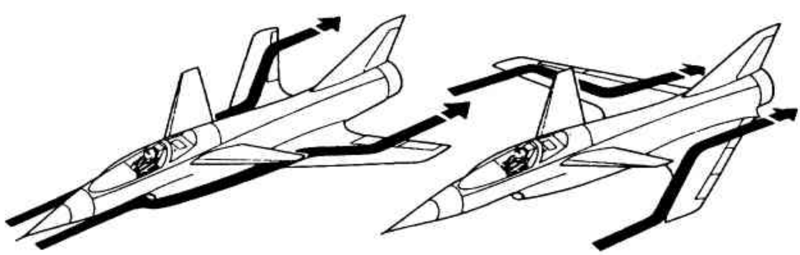


Figure 10-16 Comparison of Ram's Horn Vortex Behaviour

Stalling

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 tailplane would tend to lie in a vortex, the popular option is to combine forward sweep with a canard foreplane. Downwash from a carefully placed canard can delay root stall, and even the vortices from the canard can be used to energise the airflow over inboard sections of the wing, maintaining lift up to higher angles of attack.

The root-stall characteristics give better 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.

10.12 Delta Wings

Tailless Delta

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_{Lmax} is reduced

(b) The stalling angle is increased

Reduction of C_{Lmax}

The chord line of a 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 when the trailing edge is moved from one angular position to another, the effective aerofoil section of the wing has been changed.

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 both of the low-pressure area over the upper surface of the wing and the high-pressure area below, thereby lowering the C_L.

The curves of Fig 10-17 show that any section with a raised trailing edge must suffer a decreased C_{Lmax} compared to the basic section.

Increase in Stalling Angle

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 given delta wing is used without a tailplane, 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 tailplane.

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 and the direction of the airflow remains constant when at maximum C_L irrespective of the setting of the hinged trailing edge. Figure 10-18 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.

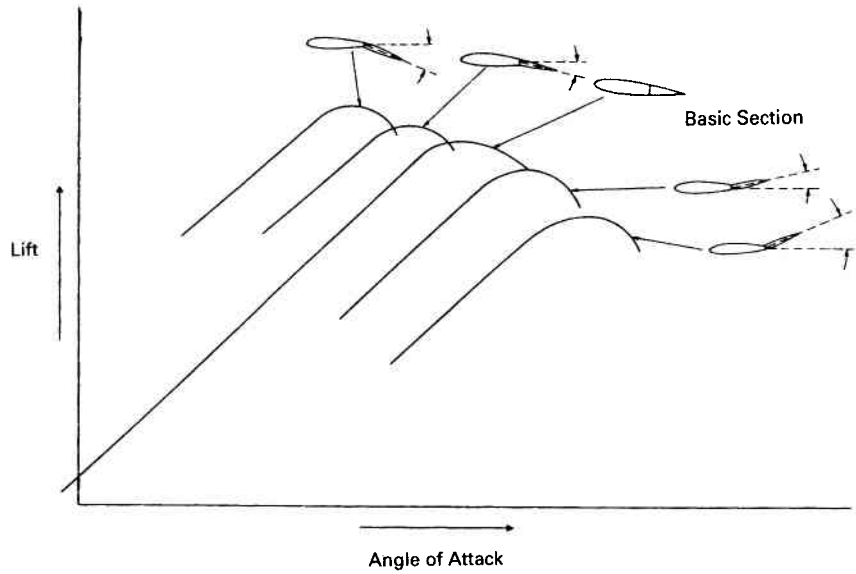


Figure 10-17 Effect of Hinged Trailing Edge on $C_{L_{max}}$ and Stalling Angle.

It is however, 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 10-18 that the angle is greater. Figure 10-18 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.

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.

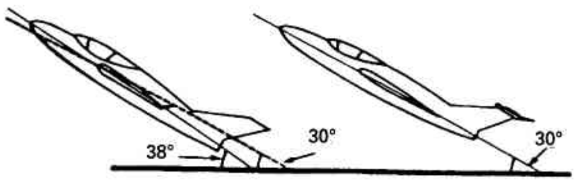


Figure 10-18 Comparison of Stalling Angle.

The C_L Curve

Reference to Fig 10-3 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_{Lmax}, 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

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_{Lmax}. By careful design, however, the vortex can be controlled and used to advantage.

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:

(i) 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 thereby improved.

(ii) 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.

10.13 Polymorphic Aircraft

General

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.

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 thereby be swept back when the aircraft is being flown at high speeds and swept forward again when flying at low speeds. Such aircraft are often labelled 'swing wing'.

Stability and Control Problems

When the wing of an aircraft is swung 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 CG 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 form 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.

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.

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, however, 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.

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

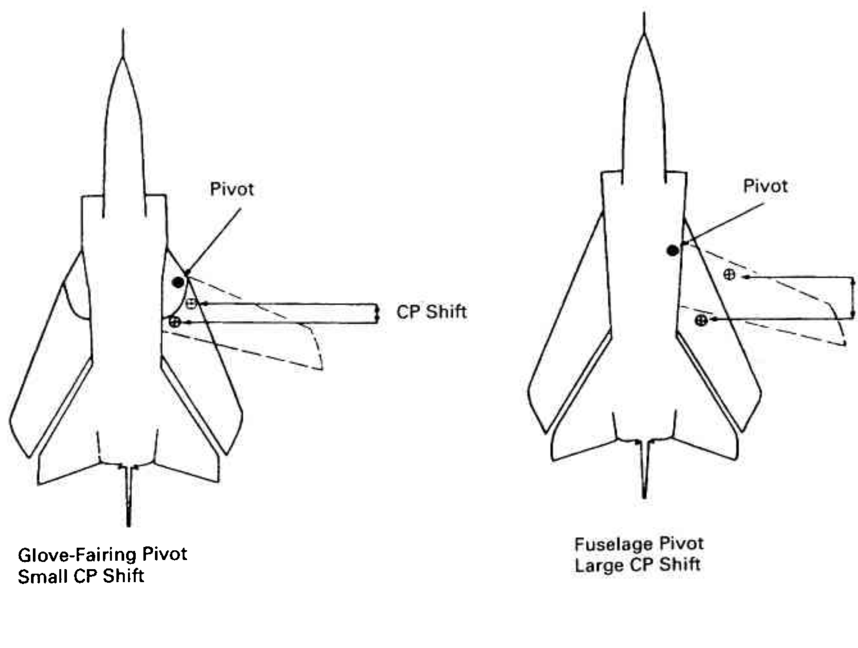


Figure 10-19 Movement of CP

span at low-speed, minimum-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 as indicated in Fig 10-19.

10.14 Canard Design

A canard-type configuration is one which has a foreplane located forward of the wing instead of the more conventional tailplane. 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 foreplane of smaller area than that of a tailplane. The trim drag problem will also be reduced because, at high speeds, an up-load will be required on the foreplane to trim the aircraft. There are, however, certain disadvantages with this layout:

(a) Stalling problems
On a 'conventional' 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 foreplane 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 tail layout, so the airflow from the foreplane 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.

10.15 Summary

Planform Considerations

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.

Aspect ratio (A) is found by dividing the square of the wing span by the area of the wing:

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

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 40°).

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

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

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.

The factors effecting pitch-up are:

(i) Longitudinal instability.

(ii) Centre of pressure movement.

(iii) Change of downwash over the tailplane.

(iv) Washout due to flexure.

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.

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.

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_{Lmax} is reduced.

(b) The stalling angle is increased.

Vortex lift has the following characteristics:

(i) Leading edge flow separation causes the CP to be situated nearer to midchord.

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

The canard configuration has the following advantages and disadvantages:

Advantages

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

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

(c) The stability and trim requirements can be satisfied with a smaller foreplane area.

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

Disadvantages

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

(ii) If the foreplane stalls first control is lost.

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

Chapter 10: Test Yourself.

1 Swept wings are used on some aircraft types to:

a) delay M_{crit}.
b) give improved low speed handling.
c) produce greater lift for a given wing area.
d) reduce lateral stability

Ref para 10.8

2 Induced drag is:

a) proportional to aspect ratio.
b) inversely proportional to aspect ratio.
c) inversely proportional to lift.
d) proportional to speed.

Ref para 10.5

3 The higher the aspect ratio of a wing:

a) the greater the induced drag.
b) the greater the rigidity of the wing.
c) the lower the stalling angle.
d) the greater the vortex drag.

Ref para 10.5

4 Spanwise movement of airflow on a swept wing may be reduced by:

a) high cambered wings.
b) non slotted trailing edge flaps.
c) increased angle of incidence.
d) wing fences.

Ref para 10.8

5 As a swept wing passes through the transonic speed range:

a) a nose up pitching moment is experienced.
b) the wing C of P moves forward.
c) the wing C of G moves aft.
d) a nose up trim will be required.

Ref para 10.10

11

Flight Controls

11.1 Introduction

The purpose of flight controls is to enable the aircraft to be rotated about its three axes. Control in pitch is exercised by elevators which move the aircraft about its lateral axis, control in roll by the ailerons which move the aircraft about its longitudinal axis and control in yaw by the rudder which moves the aircraft about its normal, or vertical, axis. Controls usually take the form of hinged aerofoils mounted on the trailing edge of the wing, the horizontal stabiliser or tailplane, and the rudder normally attached to the trailing edge of the fin. When they are moved they alter the effective camber of the section to which they are attached and therefore alter the amount of lift being generated. Within reason, controls are positioned as far away as possible from the axis of rotation about which they are effective, so that they create the largest moment for the least amount of force.

When a control surface is deflected the forces acting on it try to return it to the neutral position. The total returning force is the lift force on the control surface multiplied by the distance of the centre of pressure of the control surface to the hinge. This force is called the hinge moment and is shown in Fig 11-1.

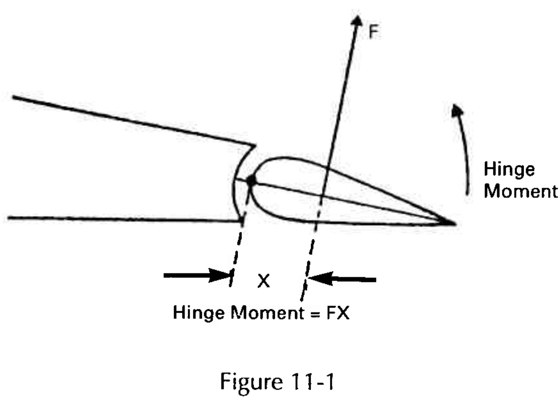


Figure 11-1

Obviously this hinge moment has to be opposed by some force if the control is to remain deflected, and this force is supplied by the pilot through the control column or rudder bar. The degree of lift force generated by a control surface will depend on the square of the speed, and as the speed increases it can reach considerable magnitude. Because of this, on any but very low speed aircraft the amount of control force required will be far too high for easy operation of the controls without some form of assistance. This assistance is called aerodynamic control balancing. Various forms of aerodynamic balancing are used, and they all operate on the principle of either reducing the hinge moment or producing a force which will help to balance the hinge moment by acting in the opposite direction. The various types of aerodynamic balancing used on current aircraft are discussed in the following paragraphs:

11.2 Inset Hinge

This type of aerodynamic balancing is commonly used on modern aircraft and achieves its reduction of control column loading by positioning the hinge so that part of the control surface leading edge moves in the opposite direction to the remainder of the control surface. Fig 11-2 shows an example of this type of aerodynamic balance.

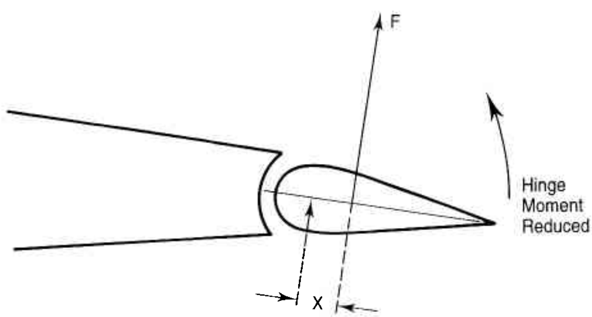
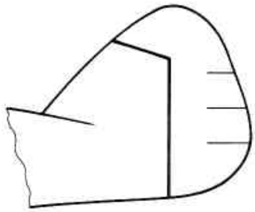


Figure 11-2

Care must be taken in the design of this type of balance to ensure that the centre of pressure is not too near the hinge line. When a control surface is deflected its centre of pressure moves forward, and if the margin between the centre of pressure and the hinge line is too small it is possible that the centre of pressure will move forward of the hinge line and so lead to the surface overbalancing.

11.3 Horn Balance

In this type of system a portion of the control surface itself acts ahead of the hinge line, so producing a force in opposition to the hinge moment. Such a balance is shown in Fig 11-3.

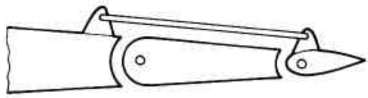


Horn Balance

Figure 11-3

11.4 Balance Tab

This type of system has the same effect as the horn balance but is produced by attaching a small aerofoil to the trailing edge of the control surface and is arranged so that when the control surface is moved the balance tab is automatically moved in the opposite direction mechanically. (Fig 11-4)



Balance Tab

Figure 11-4

The balance tab, although quite small, is acting at a considerable distance from the hinge line of the main control surface, and therefore produces a considerable assistance in moving it. There is some loss of overall effect of the control in this system and also a small drag penalty is incurred. It should be noted that when the main control surface is deflected the tab moves in the opposite direction, but, the chord line of the tab remains parallel to the chord line of the fixed surface as shown in Fig 11-4. This balanced motion achieves the required objectives whilst avoiding any excessive drag penalties.

11.5 Servo Tab

This type of tab is used on larger subsonic aircraft as an aid to the pilot in overcoming heavy control loads. With this type of system when the control column is moved the tab is moved by the control input which then causes the airflow to be deflected which in turn moves the control surface. The system is shown in Fig 11-5.

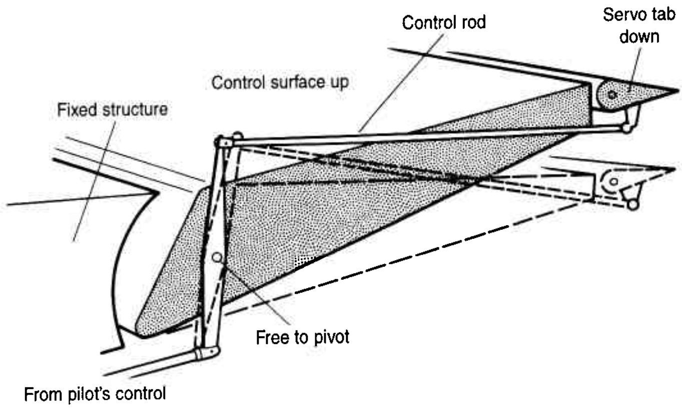


Figure 11–5 Servo tab operation

11.6 Anti-Balance Tab

This type of tab is used to increase the load on the control column when the control surface is deflected. It operates automatically in a similar way to a balance tab but in the opposite direction. The anti-balance tab is often used to increase the 'Feel' in a control system. Fig 11-6 shows an example of an anti-balance tab.

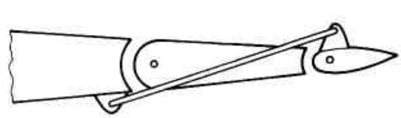


Figure 11-6

The types of aerodynamic balance discussed so far are quite suitable for slow and medium speed aircraft but when considering the requirements of high speed aircraft they are no longer of very much value. The problems involved in flying at high true airspeeds and high Mach numbers make it virtually impossible for satisfactory control to be achieved by manual controls. In view of this, power operation of all control surfaces is desirable. In a conventional control system the control column forces felt by the pilot conveys a considerable amount of information on control deflection but as there is no direct connection between the control column and the control surface in a powered control system, the 'Feel' felt on the control column due to the air loads acting upon it will not be felt as in a manual system. In order to maintain accurate control of the aircraft and to prevent overstressing of the aircraft artificial feel must be provided. As the control surface is being operated by hydraulic power the tabs as previously mentioned will have little or no effect. An example of a simple power operated control system is shown in Fig 11-7.

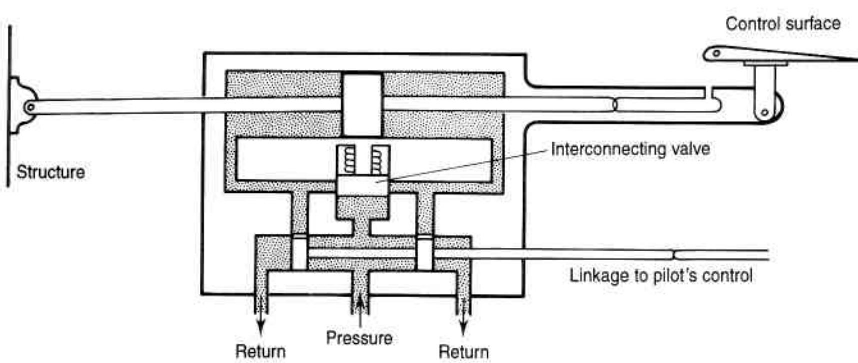


Figure 11-7

11.7 Internal Balance

Although fairly commonly used, this form of aerodynamic balance is not obvious because it is contained within the contour of the parent control surface. When the control is moved, a pressure difference is generated 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 11-8.

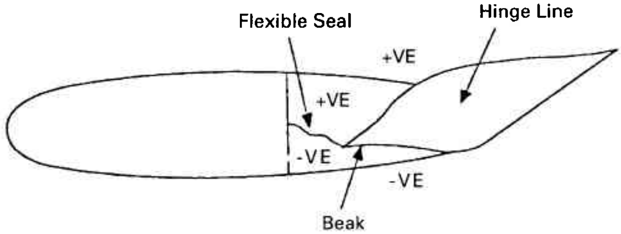
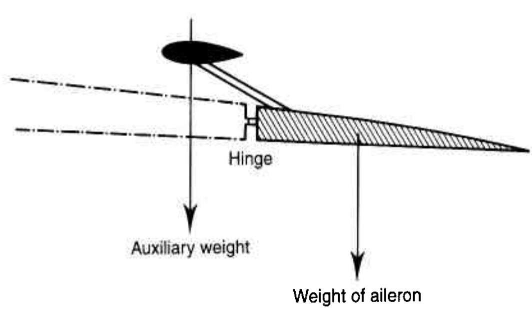


Figure 11-8

11.8 Mass Balance

Problems can arise with oscillatory movements of the control surface caused by variation in the moment of the control surface centre of gravity about the hinge. This variation can be brought about through the flexing of the entire structure when a load is applied to it. It is possible under some circumstances for these oscillations to be divergent and cause complete failure of the structure.

This form of oscillation is called 'Flutter', and as the main factor involved is the moment of the centre of gravity about the control surface hinge, the possibility of it being reduced by moving the centre of gravity nearer to the hinge line must be considered. This is usually achieved by adding weights to the control surface so positioned that they act in front of the hinge line and thereby move the control surface centre of gravity to, or just in front of, the hinge line, as shown in Fig 11-8. On modern aircraft the mass balance weights are normally housed inside the control surface structure.



Mass balance

Figure 11-9

It was stated previously that each set of controls operates around one of the three axes of movement.

Elevators	control the aircraft about the lateral axis, that is in *pitch*.
Ailerons	control the aircraft about the longitudinal axis, that is in *roll*.
Rudder	controls the aircraft about the normal or vertical axis, that is in *yaw*.

Unfortunately the operation of some controls causes additional movement about another axis, and the most obvious example of this is the effect of ailerons which are designed to give a rolling moment about the longitudinal axis. In order to achieve this, one aileron must be lowered and the other one raised, and the one which is lowered will always cause additional drag and so produce a yawing moment in the opposite direction to the intended turn. This effect is called 'adverse aileron yaw', and it is most marked when aileron deflection angles are large, usually at low speeds. There are two methods of correcting the condition, the first being differential ailerons. In this system (Fig 11-10a) the up-going aileron moves through a greater angle than the down-going and the drag is, therefore, balanced on each side of the aircraft. The other method is by use of what is called a Frise aileron, (see Fig 11-10b).

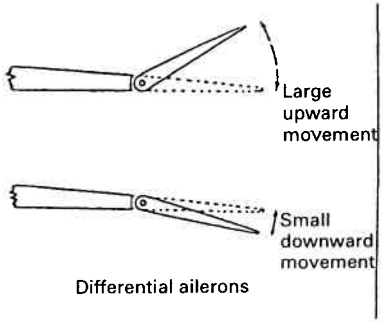


Figure 11-10 a

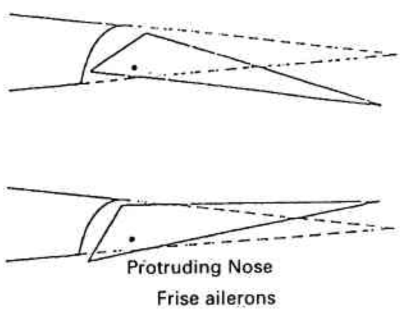


Figure 11-10 b

The excess drag from ailerons is generated by the down-going aileron so in the Frise system the up-going aileron is mechanically arranged to project below the undersurface of the wing when raised. This creates the additional drag to balance the increased drag of the down-going aileron. Frise ailerons are not in general use today as, in particular, they are most unsuited to high speed aircraft.

11.9 Adverse roll tendency due to rudder deflection

Reference to diagram (a) of Fig 11-11, shows that if the rudder is deflected to the right, the lift force generated by the fin and rudder will yaw the aircraft to the right. The fin will have a centre of pressure as does the wing, and depending on the vertical distance between the fin centre of pressure and the centre of gravity on the longitudinal axis of the aircraft, there will be a tendency for the aircraft to roll to the left in this case.

A tall fin will clearly produce a greater tendency to roll than a short one, (diagram (b) Fig 11-11).

The phrase 'tendency to roll' is emphasised, in view of the fact that the roll is normally totally masked by the extra lift from the faster moving wing. Clearly this masking effect will depend upon the span of the aircraft; a greater span producing more lift, (diagram (c) Fig 11-11).

If, on certain aircraft (eg Britannia and Belfast), there is a noticeable roll due to yaw, this can be eliminated by interlinkage of ailerons and rudder. It is therefore, most unlikely that any pilot would ever notice any adverse roll when yawing the aircraft.

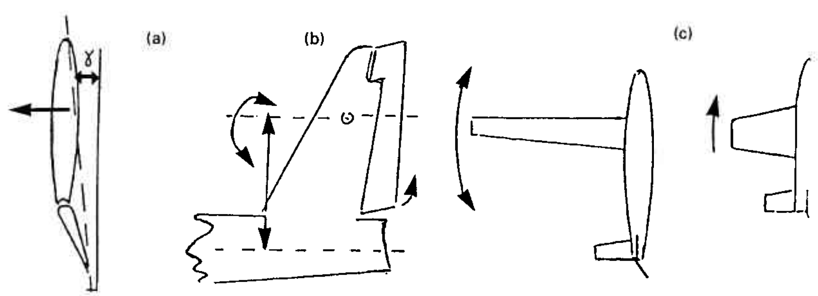


Figure 11-11

The wing structure of an aircraft is flexible and the varying loads brought about by operating a control surface will tend to twist or bend the structure. This effect is particularly noticeable with ailerons, and when an aileron is deflected downwards the resultant force on the aileron passes through its centre of pressure as illustrated in Fig 11-12.

This force has a moment about the main structural member of the wing which will cause the wing to twist, decreasing its angle of incidence. This will, of course, decrease the lift being produced by the wing when the required effect of deflecting the aileron down is to increase the lift. If the wing twist is such that these two forces cancel each other out then

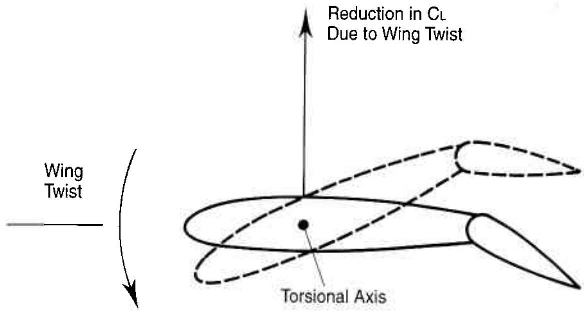


Figure 11-12

the aileron will have no effect, and if the process is continued further the application of aileron will produce a roll in the opposite direction to that intended. The force generated by the aileron is proportional to the square of the speed, so this effect will be most marked at high speeds. In fact, the effect can be so marked that on some high speed transport aircraft above a certain speed the ailerons are locked and roll control is vested in spoilers alone.

One of the major problems associated with the use of the conventional elevator is that the application of trim in the longitudinal plane reduces the effect of the elevator control. Trim tabs will be dealt with in the next section, but suffice it to say here that to allow an aircraft to be flown straight and level a certain amount of continuous deflection of the elevator may be necessary, this deflection being maintained by a trim tab. Whichever way the elevator is deflected its full range of movement will be consequently reduced in this direction.

On large jet transport aircraft the changes in longitudinal trim due both to use of fuel and to speed changes can be large, and a conventional elevator and trim tab system is not suitable. It is replaced by a horizontal stabiliser which can move in its own right. Operated by electro-hydraulic jacks it produces a very powerful leverage about the lateral axis to cope with the very large trim changes involved. It has the immense advantage that, whatever position it is set in, it leaves full elevator control available.

These days, the idea of the all-moving tail plane has been adopted for use on low speed aircraft as well. The additional force that it generates enables a smaller horizontal stabiliser to be used with a consequent reduction in weight and drag.

Mention was made earlier of a type of control called a spoiler. As shown in Fig 11-13 spoilers are panels in the upper surfaces of the wing that are hinged at their leading edges and can be opened and shut so that, when open they reduce the amount of lift being generated by the wing.

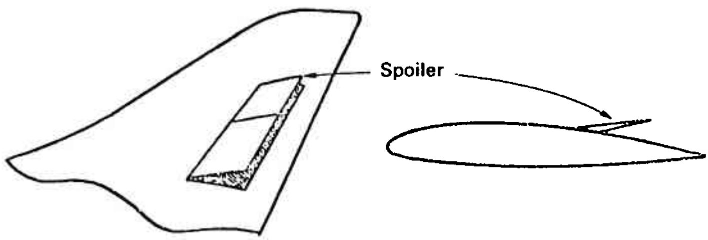


Figure 11-13 Spoilers

Spoilers have three normal uses:

(i) When operated through small angles independently on one wing or the other they can be used to augment aileron control, or even replace it.

(ii) On landing, after the aircraft has actually contacted the runway, if the spoilers on both wings are fully opened the lift is completely destroyed; This is called 'lift dumping'. It enables the aircraft to sit more firmly on the ground and thus reduce the distance required to stop and it is also of considerable value in crosswind landings.

(iii) Lift dumping in high level flight.

Modern jet transport aircraft have extremely low drag coefficients in the cruising configuration. This makes them somewhat reluctant to slow down from high speeds, and on some aircraft airbrakes are fitted.

An airbrake usually consists of a flat section which can be raised into the airflow so that it creates the maximum amount of drag whilst at the same time not damaging the lift. Airbrakes are not all that common however, and resort usually has to be made to other means of slowing down rapidly when required. One of the most commonly used methods is the lowering of the main gear to create additional drag, but on some aircraft reverse thrust from the engines may be selected in the air as well as on the ground.

Some aircraft, notably delta wing types, have a system whereby the ailerons and elevators are combined into one control called an elevon. When the control column is moved backwards or forwards both surfaces move up and down together, but when the control column is moved sideways one elevon comes up and the other goes down. Another combination of controls is that of elevator and rudder to produce a V or butterfly tail. This is then called a ruddervator.

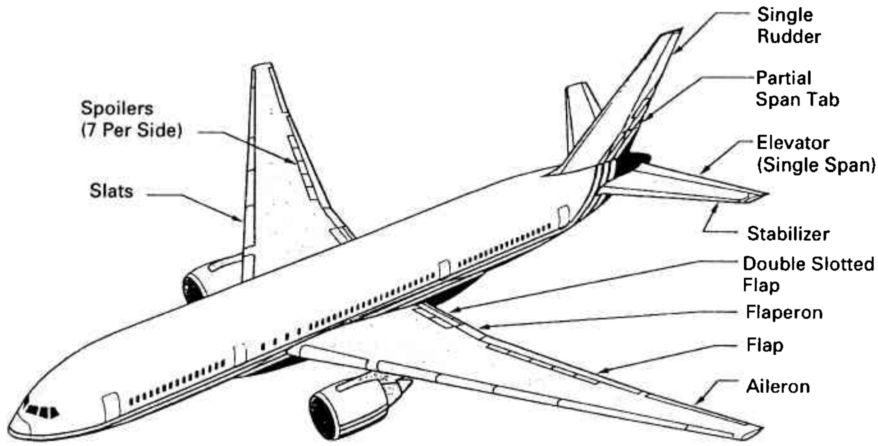


Figure 11-14 Large Jet Control Surfaces and High Lift Devices

In the example aircraft shown in Fig 11-14 all primary control surfaces are operated by hydraulic activators which, in turn, are controlled by electronic signals originating from the flight deck controls and the auto flight systems.

Aileron operation is confined to the lower airspeed envelope and at high speed the ailerons are locked out of action and lateral (Roll) control is a function of the spoilers and flaperons. This later function is to reduce wing flexure at high speed.

Some aircraft employ two sets of ailerons, one outboard for lower speeds and the other inboard for high speed. In such cases the outboard ailerons are locked out at high speeds.

Chapter 11: Test Yourself.

1 Control about the lateral axis is achieved by the:

a) rudder.
b) ailerons and rudder.
c) ailerons.
d) elevators.

Ref para 11.8

2 The purpose of a balance tab is to:

a) increase control surface feel.
b) reduce the load on the control surface.
c) reduce the load at the control column.
d) reduce control surface flutter.

Ref para 11.4

3 When a control surface is deflected which is fitted with horn balance:

a) the horn is located aft of the hinge line.
b) the horn moves in the opposite direction to the surface.
c) the horn acts aft of the hinge line.
d) the control column loads will be increased.

Ref para 11.3

4 On a differential aileron control system the:

a) up and down going ailerons move through the same angle of deflection.
b) up-going aileron moves through a greater angle of deflection.
c) up-going aileron leading edge protrudes below the wing undersurface.
d) down-going aileron moves through a greater angle of deflection.

Ref para 11.8

5 Frise ailerons are fitted to:

a) increase the rate of roll.
b) reduce control column loads.
c) combat adverse aileron yaw.
d) prevent overloading of the control system.

Ref para 11.8

12

Tabs

12.1 Introduction

Tabs are small aerofoil sections hinged to the trailing edges of control surfaces. The main purposes for which they are used are:

(a) Trimming

(b) Aerodynamic balancing

(c) Servo operation

For aircraft in flight to be in equilibrium, the moments about each of the three axes of the aircraft must balance. If they do not balance then an additional force must be supplied by deflection of the controls to keep the aircraft in equilibrium. It is most undesirable that continuous control surface deflection be applied at the control column because of the physical effort involved, and to overcome this problem trimming tabs are provided. The action of a trim tab is best understood by considering the situation with an aircraft which tends to fly nose down continuously. To

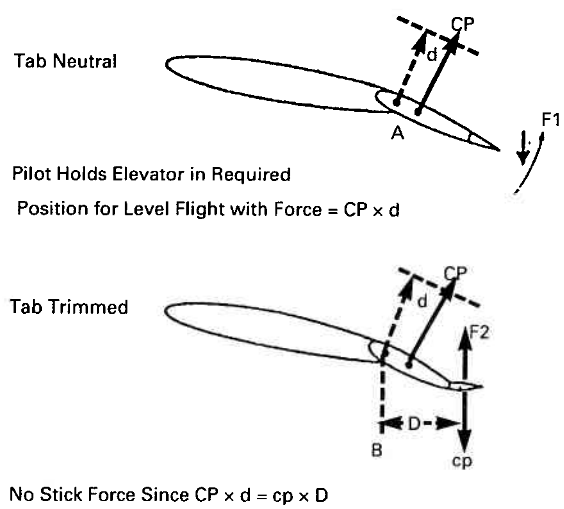


Figure 12-1 Principle of the Tab

correct this the elevator must be deflected upwards and maintained in this position. To bring this about, the trim tab attached to the trailing edge of the elevator is deflected downwards as shown in Fig 12-1. This diagram indicates that the total force exerted downwards by the elevator is F1 x d, the distance of the elevator's centre of pressure from its hinge, A. The trim tab, having been moved in the opposite direction, exerts a force F2 upwards through its centre of pressure, and although this force is smaller than F1 it has a much longer arm from the hinge of the main control, B, therefore its total moment is able to balance out that of the main control. It should be noted that the action of the trim tab also slightly reduces the effect of the main control surface. The final force exerted is F1-F2 and this will necessitate a slightly larger deflection of the main control surface with a subsequent small increase in drag.

Trimming tabs are normally controlled either by trim wheels in the cockpit or, alternatively, by electrical switches activating motors. These controls are usually arranged so that they act in a natural sense, that is to say with the control wheel moved in the direction in which the aircraft is required to be trimmed, ie aircraft nose high, move the top of control wheel forward towards the nose to bring it down. On some light aircraft fixed trimming tabs may be fitted and they consist of small sheets of metal which can be bent permanently to correct known out-of-trim forces, but they cannot, of course, be adjusted in flight. As with any other aerodynamic control surface the effect of a trim tab is proportional to the square of the speed. At high speeds very small trim adjustments will achieve the desired effect whereas at low speeds a considerable movement may be required. Fig 12-2(a) and 12-2(b).

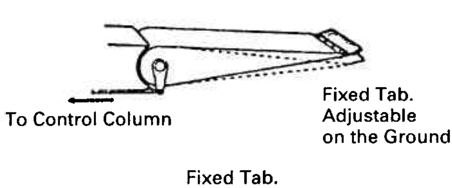


Figure 12-2 (a)

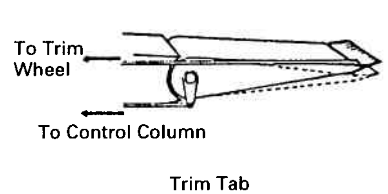


Figure 12-2 (b)

Balance tabs are a form to ease the load on the controls. They are mechanically arranged to move in the opposite direction to the main control surface, as illustrated in Fig 12-3(a). The operation of this type of tab is completely automatic, and as can be seen from the diagram it produces a force in the opposite direction to the main control surface but at the cost of producing a small reduction in control effectiveness.

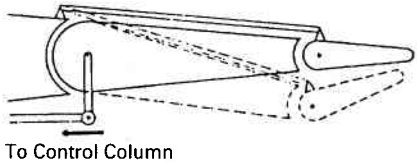


Figure 12-3 (a)

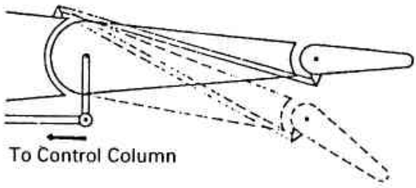


Figure 12-3 (b)

In some aircraft, far from requiring assistance in moving a control surface against the aerodynamic loads, the hinge moment is too small. This results in very low loads at the control column, a lack of feel and the possibility of over-stressing the airframe due to an excessive inadvertent deflection of the control surface. It occurs because of the hinge being too close to the centre of pressure of the control surface. In order to improve the situation an anti-balance tab is fitted which operates in the same direction as the control surface, such as illustrated in Fig 12-3(b).

On some aircraft, aerodynamic balance is not considered necessary at low speeds but is required at higher speeds when the aerodynamic loads increase considerably. A type of balance tab to deal with this situation is called a spring tab. The basis of the system is illustrated in Fig 12-4. The movement of the control column is transmitted to a lever pivoted on the main control surface but not directly operating it. Operation of the surface is through springs, and at low aerodynamic loads the movement of this pivot arm is transmitted to the main control surface through the springs, and no alteration in the geometry between the pivot and balance tab takes place. When the aerodynamic loads increase, however, transmission of control column movement, via the pivot arm, to the control surface compresses the spring. This upsets the geometry of the system and brings into operation the balance tab on the trailing edge, thus giving some assistance in moving the control surface.

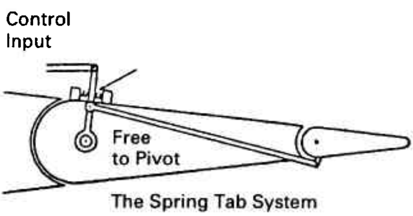


Figure 12-4

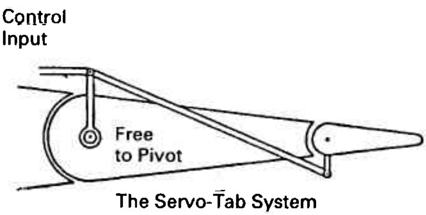


Figure 12-5

When manual controls are used to operate very large control surfaces the loads involved, even with the assistance of balance tabs, may be unacceptable. Under these circumstances servo tabs are used to operate the control surfaces. A servo tab is a small aerofoil section, once again attached to the trailing edge of the main control surface, which is directly operated by the control column. In this system there is no direct connection between the control column and the main control surfaces, the control column only operating the servo tabs. As the system depends entirely upon aerodynamic forces, any movement of the control column when the aircraft is on the ground will produce no control surface movement, only the servo tabs will move. This system is illustrated in Fig 12-5. Care must be exercised in pre-flight checks for full and free movement of control surfaces when servo tabs are used.

In the section on flight controls, mention was made of the variable incidence horizontal stabiliser. In this system the incidence of the entire horizontal stabiliser is changed, as required, to adjust for longitudinal trim requirements instead of using a conventional trim tab. The horizontal stabiliser is pivoted about its central point and moved by electro-hydraulic jacks. The change in longitudinal trim which this system can exert, compared to an ordinary trimming tab, is very considerable, and in view of this precautions have to be taken against the result of a runaway in the operating system, winding the horizontal stabiliser either fully up or fully down. This is usually achieved by having the left and right halves of the horizontal stabiliser entirely separate from each other and each being operated by two separated systems. This will obviate the possibility of both sides of the stabilizer running away together and also produces a multiple redundancy in the event of partial failure of one of the electro-hydraulic systems. This form of longitudinal trimming is a necessity on large jet aircraft where the conventional system of elevator and trim is unable to cope with the large trim changes brought about by, firstly, large centre of gravity movements; secondly, wide operating speed range and, finally, high lift devices. It will be remembered that as the elevator is not deflected in any way for the purposes of trimming, the full range of control is always available from it.

On supersonic aircraft, for instance, Concorde, the deflection of control surfaces at high subsonic and supersonic speeds is undesirable due to the aerodynamic effects involved. One way of overcoming this problem is to use fuel in various tanks to trim the aircraft. When passing from subsonic to supersonic speeds there is a marked rearward movement of the centre of pressure which can cause a severe out-of-trim situation. This can be overcome by pumping fuel to move the centre of gravity to be coincident with the centre of pressure, and in Concorde this is in fact done automatically.

12.2 Control locks

All aircraft require some mechanical means of locking the controls in the neutral position when on the ground to prevent possible damage from gusts in high winds when parked. On smaller aircraft these may take the form of wood or metal devices which can be slid over the control surfaces to lock them in position and prevent their movement. On larger aircraft it is common practice to have internal locking systems, but obviously either type of control lock must be removed prior to take-off. In this context it should be borne in mind that external locks fitted to a servo tab operating system will permit the movement of the control column on the flight deck with the locks in place as only the tabs are being operated, therefore, on aircraft with this type of equipment it is essential to confirm that any external locks fitted have been removed. Most modern aircraft with hydraulically operated controls have integral hydraulic locks.

Chapter 12: Test Yourself.

1 The effectiveness of a trim tab is:

a) proportional to the speed.
b) proportional to the square of the speed.
c) inversely proportional to the square of the speed.
d) not proportional to the speed in any way.

Ref para 12.1

2 If the hinge moment of a control surface is too small:

a) balance tabs may be required.
b) horn balance may be required.
c) an anti balance may be required.
d) mass balance will be required to reduce control column loads.

Ref para 12.1

3 A fixed trim tab may be adjusted by:

a) an engineer only.
b) a type rated pilot.
c) a test pilot only.
d) any qualified pilot.

Ref para 12.1

4 If an aircraft is flying nose heavy, to return the aircraft to level flight requires:

a) elevator trim wheel aft, tab up movement.
b) elevator trim wheel forward, tab up movement.
c) elevator trim wheel aft, tab down movement.
d) elevator trim wheel forward, tab down movement.

Ref para 12.1

5 When a control surface is deflected the anti-balance tab will:

a) move in the opposite direction and increase control column loads.
b) move in the opposite direction and decrease control column loads.
c) move in the same direction and increase control column loads.
d) move in the same direction and reduce control column loads.

Ref para 12.1

13

High Lift Devices

13.1 Introduction

High lift devices are incorporated on aircraft wings to reduce the distance required for take-off and landing. The distance used by an aircraft either to take off or land depends on the speed involved, and this speed in its turn is related to the stalling speed of the aircraft. An aircraft cannot approach to land at a speed below its stalling speed, therefore the higher the stalling speed the longer the distance required to complete the landing run. The same applies for take-off, the aircraft not being able to leave the ground until it has achieved flying speed, ie above the stalling speed, therefore the lower the stalling speed under these circumstances the less distance is required. All high lift devices produce the same effect, that is to increase the coefficient of lift of the wing. The methods used for increasing the C_L are:

(a) Flaps

(b) Slats

(c) Boundary layer control

A flap is a hinged aerofoil section which can be mechanically lowered either from the trailing edge or the leading edge of a wing. The effect of lowering a flap is to increase the overall camber of the wing and thus increasing the coefficient of lift. Some types of flap also increase the wing area, thus augmenting the additional camber and producing even more lift.

13.2 Types of Flap

There are many different types of flap in common use and some of the more usual ones are shown below.

(a) Plain flap.
The plain flap (Fig 13-1) is an aerofoil section merely hingeing down from the trailing edge of the wing.

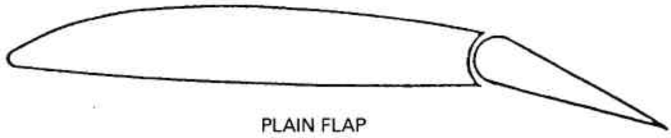


Figure 13-1

(b) Split flap.
As illustrated in Fig 13-2, the split flap hinges down from the undersurface of the trailing edge. This has the advantage that the camber of the upper surface of the wing is not disturbed but, at the same time the flap will produce a considerable amount of drag.

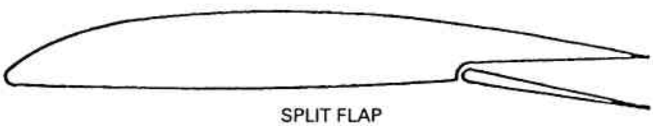


Figure 13-2

(c) Slotted flap.
One of the problems with flaps in general is that at large flap angles the air tends to separate away from the flap upper surface, thus reducing its effectiveness. This can be minimised by putting a small slot between the trailing edge of the wing and the leading edge of the flap, as illustrated in Fig 13-3. The slot produces a venturi which speeds the air up, thus giving it more kinetic energy and enabling it to follow the contour of the flap farther rearwards before breaking away.

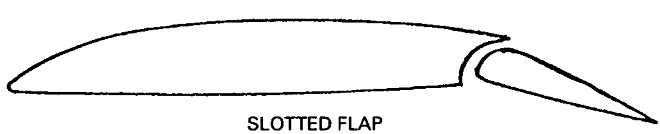
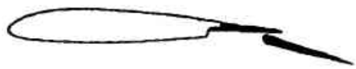


Figure 13-3

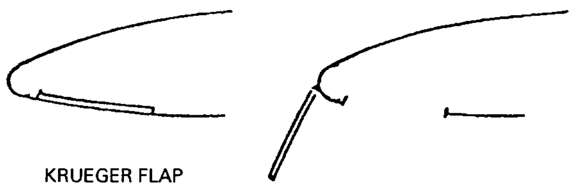
(d) Fowler flap.
The Fowler flap, in addition to moving downwards, also moves rearwards in sections when it is lowered. Whilst increasing the camber of the wing this also enlarges the wing area, and will result in a very large increase in the coefficient of lift. It is quite usual for Fowler flaps also to have slats, and this type is illustrated in Fig 13-4.



FOWLER FLAP

Figure 13-4

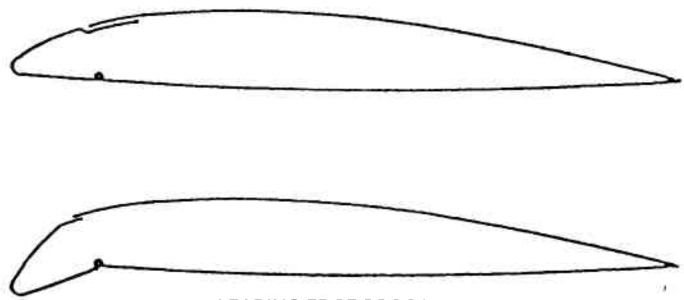
(e) Krueger flap.
This is a leading edge flap which increases the leading edge camber, and is illustrated in Fig 13-5.



KRUEGER FLAP

Figure 13-5

(f) Leading edge droop.
In this system the entire leading edge of the wing is mechanically lowered, as shown in Fig 13-6. This has the effect of increasing the leading edge camber. As can be imagined, the mechanism for lowering the leading edge must be extremely complicated, and this type of high lift device has not found general favour.



LEADING EDGE DROOP

Figure 13-6

The increase in camber caused by the lowering of flaps produces more lift from the given wing section. If we consider straight and level flight, on lowering of flaps the greater lift will enable either the angle of attack to be reduced or the speed to be reduced. Generally speaking, a compromise is reached between these two factors and the speed is considerably reduced with a small reduction in the angle of attack. The effect of lowering flaps is not constant from zero angle down to their full deployment. A selection of flaps down to approximately 30° will give a very large increase in lift for a comparatively small penalty in drag, but further lowering, to say 60°, will not produce much increase in lift but will produce a considerable increase in drag. When considering the distance required for take-off one might first feel that the lowest speed for take-off would give the shortest distance, the whole effect being achieved by large flap angle.

Unfortunately, as already mentioned, a large flap angle incurs a very high drag penalty which reduces the acceleration of the aircraft, so you would end up with a rather long distance before you could reach the unstick speed. A lower flap angle would give a higher unstick speed, but, with less drag, better acceleration would be achieved. In practice, a compromise is reached between these two limits and a flap setting of somewhere in the region of 10° to 12° is used for take-off.

The distance required to land depends on the touch-down speed. The lowest speed will be given by selection of full flap, this giving the lowest stalling speed. In addition the selection of full flap will produce a very considerable amount of drag which will assist in decelerating the aircraft on landing.

13.3 Leading edge slots

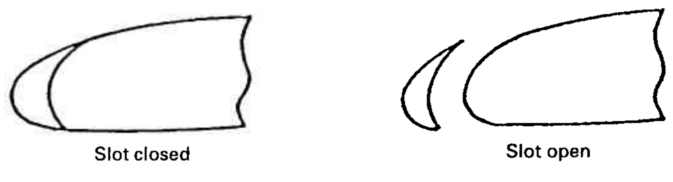


Figure 13-7

The leading edge slot, when opened, prevents the airflow from breaking away at the normal stalling angle. This allows the wing to be used at higher angles of attack, giving higher C_L and so lower speeds. See also Figs 13-14 & 13-15.

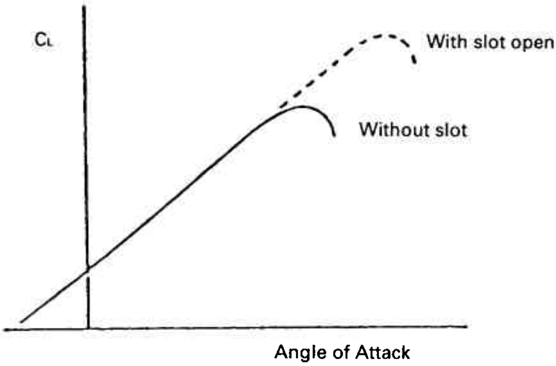


Figure 13-8

13.4 Slats

Slats are lift augmentation devices that take the form of a small auxiliary aerofoil, highly cambered, adjacent to the leading edge of a wing (forming a *slot*), usually along the complete span. They are adjustable, control being either automatic or manual by the pilot. The effect on the C_L and angle of stall/attack may be seen in Fig 13-9, C_L being increased by approximately 70%, and angle of stall by some 10°.

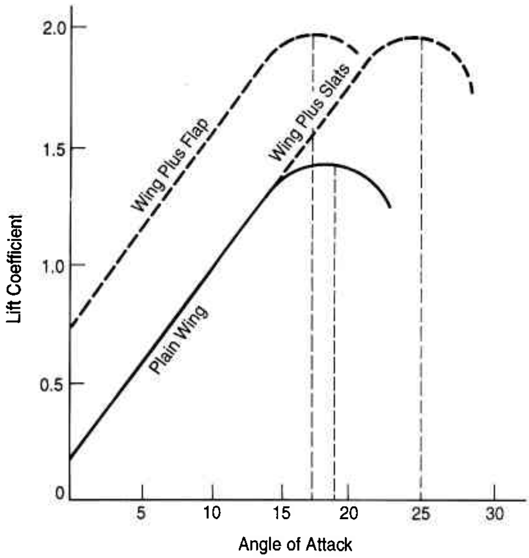


Figure 13-9

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 is generating a high lift coefficient because of its marked camber. The resultant action, aerodynamically, is to flatten the marked peak of the low pressure envelope, changing it to one with a more gradual gradient, as may be seen in Fig 13-10. This flattening means that the boundary layer does not undergo the sudden thickening due to negotiating the steep pressure gradient that existed behind the former peak, so retaining its energy and enabling it to penetrate almost the full chord before separating.

Figure 13-10 also shows that although the pressure distribution is flatter, the area of the low pressure region is unchanged or even increased. The passage of the boundary layer over the wing is assisted by the air flowing through the slot (between slat and leading edge) accelerating through the venturi effect, thus adding to the kinetic energy and so helping it to penetrate against the adverse pressure gradient.

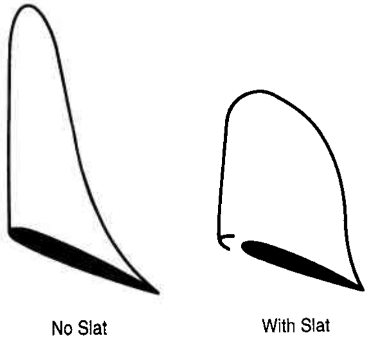


Figure 13-10 Effect of Slats on Pressure Distribution

To summarise the foregoing thus far, slats have the effect of:

(a) Delaying separation until an angle of attack of 25° to 28° is attained, during which time:

(b) Lift coefficient has increased by approximately 70%. It should be evident that the stalling speed of a slatted wing is significantly reduced, eg if an unslatted wing stalls at 100 kt, its fully slatted counterpart would stall at approximately 80 kt. The exact amount of reduction depends on the length of leading edge covered by the slat, and the chord of the slat.

Automatic slats are often located at the wing leading edge in front of ailerons in order to increase the stalling angle by being automatically extended when the aileron moves down. Normally the action of a down going aileron will reduce the stalling angle of the wing at that point.

13.5 Slat Control

Since slats are of use only at high angles of attack, some method must be used to fair the slats with the leading edge, thus precluding increased drag at normal flight configuration.

If the slats are small and the drag is negligible they may be fixed, ie non-automatic. Large slats, however, are invariably of the automatic type. They are usually of the mechanical control, hydraulically actuated kind, their selection being mechanically matched to the selection of flap, the linkage being such that slats are extended before flap and before the speed reaches that used for approach and landing. The reverse occurs on take-off, when slats are fully in only after flap is up, and at the correct airspeed.

In the event of malfunction either of flaps or slats, it is usual to be able to 'split' the linkage between the two, thus isolating the inoperative control, allowing the serviceable unit to operate normally. On some aircraft the stall sensing unit may be used to extend slats only if the sensor is activated by approach to the stall angle. Figure 13-11 illustrates a typical slat segment of the kind more common to aircraft with swept back wings.

Figure 13-11

13.6 The Boundary Layer

This may be best described as the layer of air extending from the surface to the point where no drag effect is discernible, or, that region of flow in which the speed is less than 99% of the free stream flow, and it usually exists in two forms – laminar and turbulent. Figure 13-12 illustrates the boundary layer.

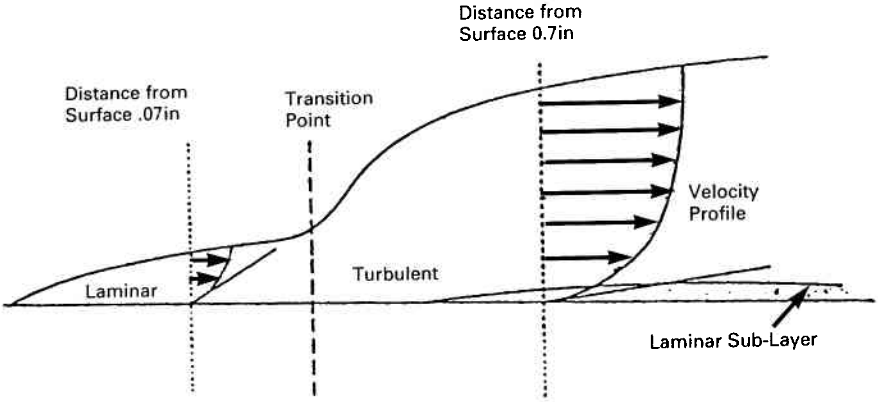


Figure 13-12

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 13-10 it may 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.

It can be seen that the nature of the boundary layer is a controlling factor in the determination of surface friction drag, but more important still, the nature of the boundary layer also 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.

The boundary layer cannot be eliminated entirely, though some measure of control of it may be afforded by wing devices, one already dealt with being LE slots (Fig 13-7), which have the effect of re-energising the boundary layer. Others are:

(i) Boundary layer fences to restrict the boundary layer outflow. They also check the spanwise growth of the separation 'bubble' along the leading edges.

(ii) Boundary layer suction; suitably placed suction points draw off the weakened layer so that a new high energy layer can take its place.

(iii) Boundary layer blowing; high velocity air injected into the boundary layer to increase its energy.

(iv) Vortex generators; these re-energise the boundary layer, and are usually positioned ahead of control surfaces.

(v) LE extension, also known as 'sawtooth' LE – restricts the outward flow of the boundary layer.

(vi) LE notch, has the same effect as LE extension. These forms (v) and (vi), are dealt with in a later chapter.

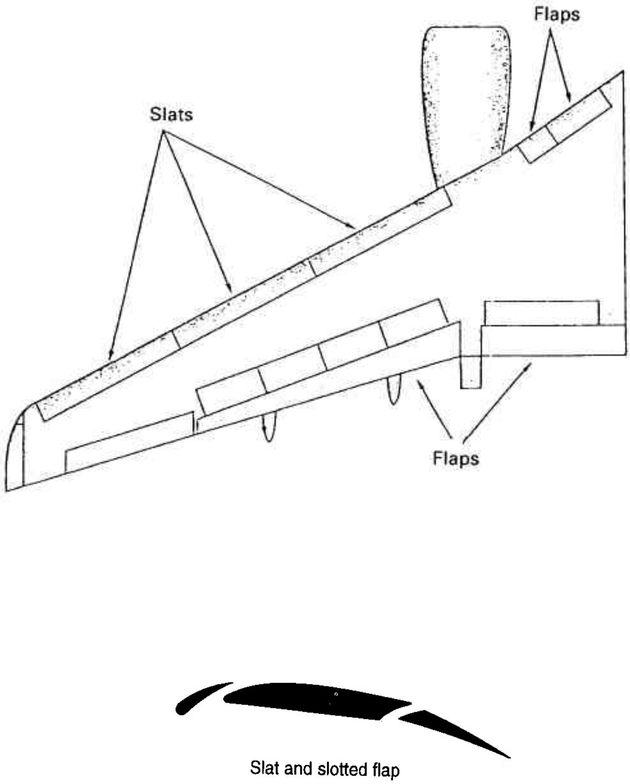


Figure 13-13 Slat and Slotted Flap Combination

13.7 Slat and Slotted Flap Combination

The combination shown at Fig 13-13 will provide a 75% increase of maximum lift with a basic aerofoil angle at max lift of 25°. This will provide more control of the boundary layer with an increase of camber and wing area. The pitching moment that a trailing edge flap will produce on its own can be neutralised.

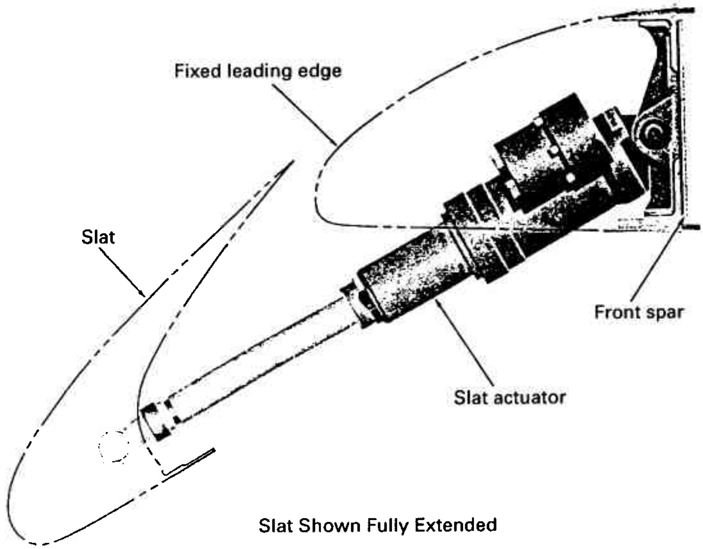


Figure 13-14

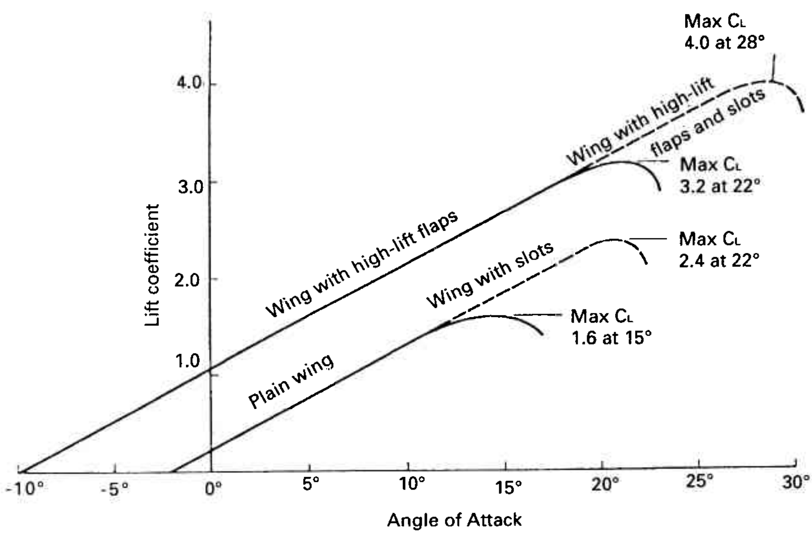


Figure 13-15

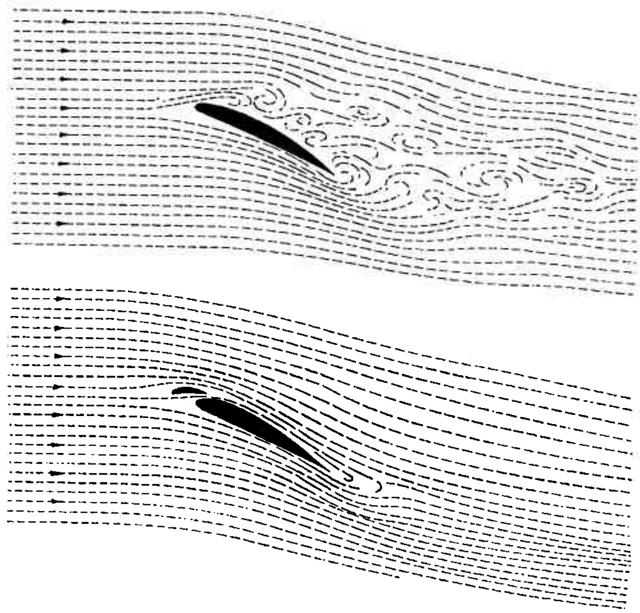


Figure 13-16 Effect of slot on airflow over an aerofoil at large angle of attack

Chapter 13: Test Yourself.

1 A Fowler flap is one which when selected:

a) increases camber and wing area.
b) increases wing area only.
c) increases camber only.
d) dumps lift only.

Ref para 13.2

2 When the angle of attack of a wing is increased:

a) the boundary layer becomes thinner.
b) the transition point moves aft.
c) the boundary layer thickens.
d) boundary layer thickness is unaffected.

Ref para 13.6

3 As the angle of attack of a wing is increased in level flight:

a) laminar flow at the front of the wing becomes turbulent at the separation point.
b) laminar flow at the front of the wing becomes turbulent at the transition point.
c) boundary layer separation at the leading edge.
d) boundary layer separation will not occur below the stalling angle.

Ref para 13.6

4 Vortex generators employed on a wing:

a) ensure that the root end stalls first.
b) ensure that the tip stalls first.
c) are located near the trailing edge.
d) are normally positioned ahead of control surfaces.

Ref para 13.6

5 A vortex generator is designed to:

a) enhance wing tip vortices.
b) re-energise the boundary layer.
c) delay M_{crit}
d) increase air pressure.

Ref para 13.6

14

Stability

14.1 Introduction

The study of aircraft stability can be extremely complex, so for the purpose of this chapter the subject will be greatly simplified. Stability is first defined in general terms and it will then be seen how the aircraft designer incorporates stability into an aircraft.

14.2 Definitions

To quote Newtons's first law again, 'a body will tend to remain in a state of rest or of uniform motion unless disturbed by 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. This motion may best be considered under two headings, static stability and dynamic stability:

14.3 Static Stability

Static stability describes the immediate reaction of the body following disturbance. (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. The examples shown should help to make this clear. Note that in colloquial usage positively stable and negatively stable are usually *stable* and *unstable* respectively.

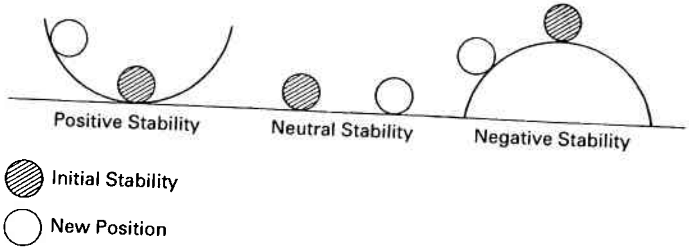


Figure 14-1 Static Stability Analogies.

In order to relate the response of a body to its initial state of equilibrium it is useful at this stage to use the analogy of the 'bowl and ball' in illustration (see Fig 14-1). 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 farther away from its original position it has negative stability, and if the ball tends to remain in its new position it has neutral stability.

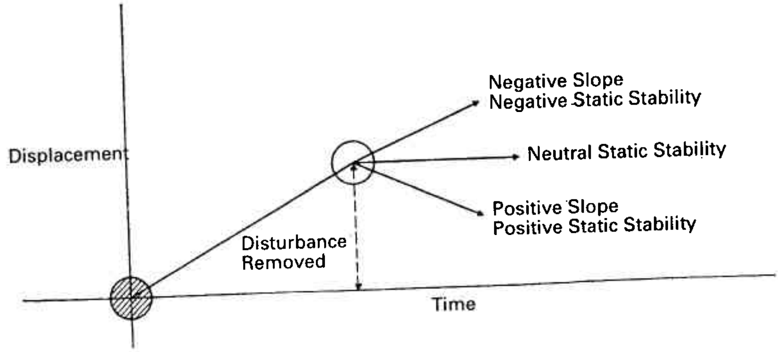


Figure 14-2 Graph of the Degrees of Stability

The concept of stability 'degree' can be expressed more usefully in the graphical form of Fig 14-2. Displacement, plotted on the vertical axis, may refer to any system, eg distance, moments, volts, etc. No scale is given to the horizontal axis which may vary from microseconds to hours, or even years.

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.

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)

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.

14.4 Directional Stability

A simple approach both to directional and to 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; these may be referred to as the vertical stabilizers. Figure 14-3 shows how a displacement in yaw through an angle β, resulting in sideslip, produces a restoring moment and, therefore, positive directional (static) stability. Two points are worth noting:

(a) The dart rotates about its centre of gravity (CG).

(b) The momentum of the dart momentarily carries it along the original path, ie the relative airflow RAF is equal and opposite to the velocity of the dart.

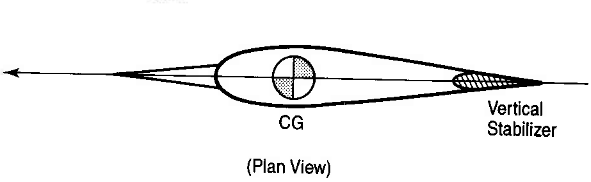


Figure 14-3 The Positive Stability of a dart

An aerodynamic shape like a fuselage may be unstable. Reference to Fig 14-4 shows that this occurs when the centre of pressure (CP) is in front of the CG.

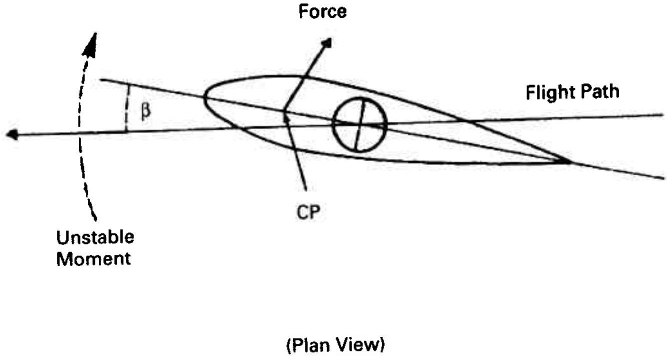


Figure 14-4 The Negative Static Stability of a Streamline Body When CP is ahead of CG.

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 14-5). 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'.)

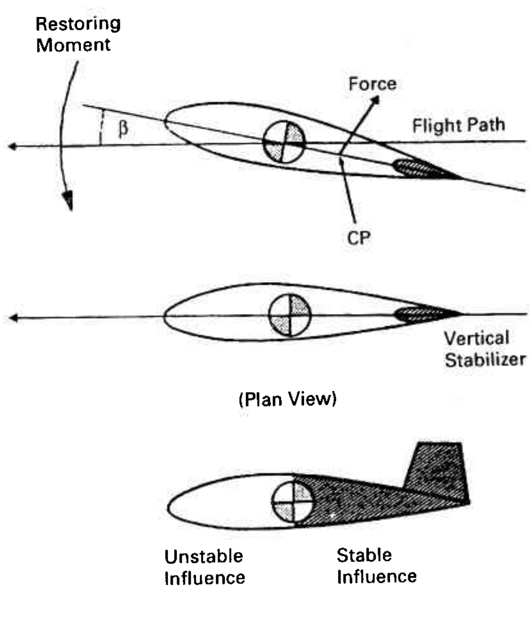


Figure 14-5 Positive Static Stability with the addition of a fin

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.

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 sideforce 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.

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; rearward movement will decrease the directional stability.

Longitudinal Stability
The analogy of the dart can be used with advantage 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. On an aircraft, the tailplane and elevators perform the functions of a horizontal stabilizer and the conclusions reached will be equally valid. For simplicity, the explanation is limited to stick-fixed static stability, ie elevators locked.

Figure 14-6a 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 d_L and increase the wing pitching moment by the amount d_Lx. The result is to worsen the nose-up displacement: an unstable effect. In the figure at b, the CP is aft of the CG and the wing moment resulting from a displacement in pitch will be stabilizing in its 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 character.

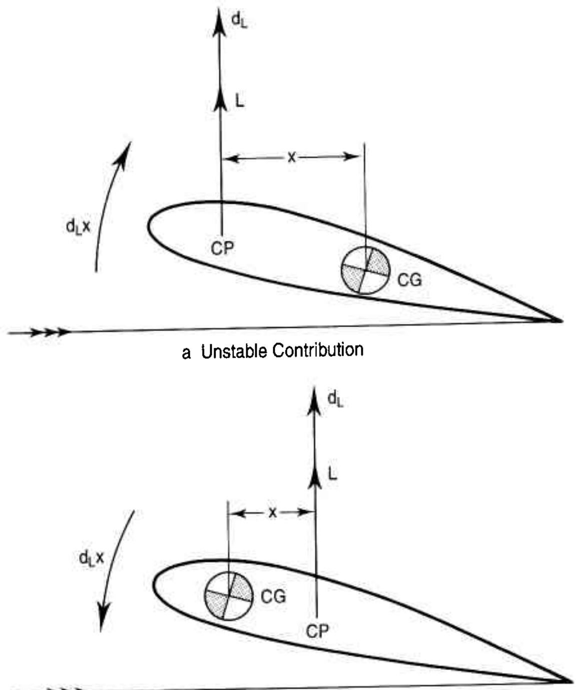


Figure 14-6 Variations in the Position of CP and CG.

Taking the worst case, therefore, the wing may have an unstable influence and the horizontal stabilizer must be designed to overcome this.

The simplified diagram in Fig 14-7 illustrates the growth of 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 all other) contributions for positive static longitudinal stability.

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, ie (Total $\text{Lift}_{\text{tail}}$)y – (Total $\text{Lift}_{\text{wing}}$)x = net pitching moment.

The main factors which affect longitudinal stability are:

(a) Design of the tailplane.

(b) Position of the CG.

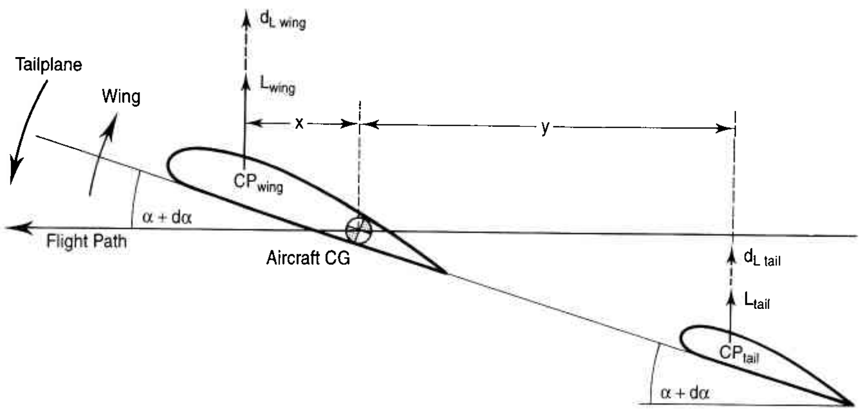


Figure 14-7 Changes in Forces and Moments due to a Small Nose-up Displacement ($d\alpha$)

Design of the Tailplane

The whole tailplane is an aerofoil and the lift force resulting from a change in angle of attack will be proportional to the $C_{L\ tail}$ 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 = $C_{Lwing}qS$ and the total lift produced by the tail = $C_{Ltail}qS$.

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 x 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. 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 angle of downwash 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 farther forward, thereby increasing the moment arm.

Position of the CG

The position of the CG may be marginally under the control of the pilot of the aircraft. From Fig 14-7 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, ie the stability that 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).

Neutral Point

Every aircraft Flightcrew Manual gives 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 flightcrew 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', ie stick free.

CG Margin (Stick Fixed)

The larger the tail area, the larger the tail moment, and so the farther 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 the degree of longitudinal stability. The greater the CG margin, the greater the stability, eg 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.

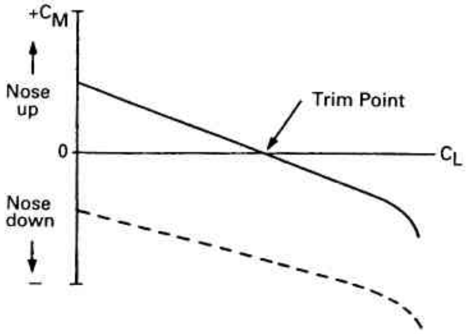


Figure 14-8 Pitching Moment Curves and Trim Point

14.5 Trim Point (Stick Fixed)

Figure 14-8 (the broken line) shows a curve of aircraft pitching moment coefficient. C_M, about the CG vs C_L. The zero-lift pitching moment, C_{MO}, is negative for most aerofoil sections. The negative slope of the curve denotes stability, eg a pitch-up, increasing the C_L, generates a nose-down restoring moment about the CG. The curve will however, never cross the positive horizontal axis, which means that there is no value of C_L at which the aircraft will be in trim (where $C_M = Ø$). If C_{MO} can be made positive by introducing a nose-up pitching moment, then the curve is raised (the solid line) and the aircraft can be brought into trim. This can be achieved by setting the tailplane at a lower angle of incidence than the mainplane to generate a down-load.

Longitudinal Dihedral
The difference between the two settings is known as longitudinal dihedral, but has no effect on the basic stability of the aircraft. Varying the tailplane incidence only shifts the trim point. As the C_L vs angle of attack curves of the mainplane and tailplane may be regarded as straight lines (up to the stall), the variation in lift per degree alpha change, does not depend on the initial incidence settings nor on their difference.

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
In text books on stability it is usual to find that the aerodynamicist writes of the 'aerodynamic centre' (AC) rather than of 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.

Stick-Free Longitudinal Stability
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. Under these conditions, with the tail moment reduced, the balance between the tail and wing moments is changed and, therefore, the position for the CG, about which the moments are equal, will be farther forward, because the less effective tail requires a longer moment arm. That is, the neutral point is farther 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)
In the preceding paragraphs the longitudinal static stability was discussed with respect to a disturbance in incidence 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. 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 ($n = 1$).

If an aircraft is trimmed to fly straight and level (the initial condition, Fig 14-9a), 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 14-9b), 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.

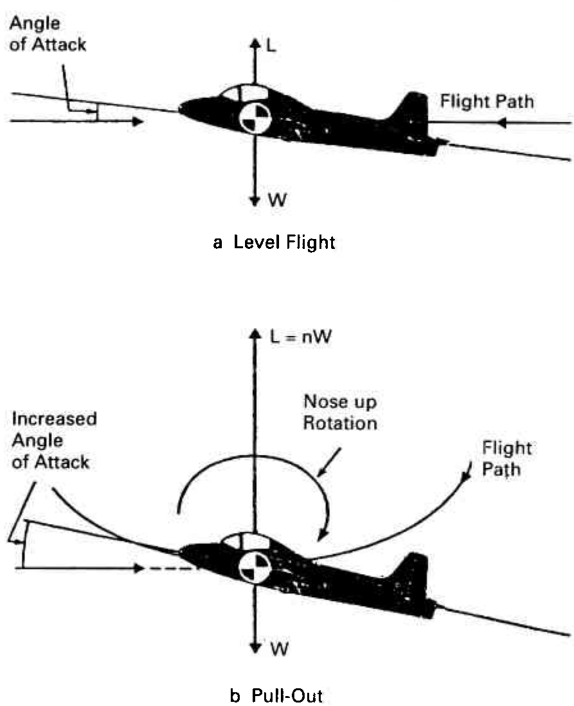


Figure 14-9 Forces Acting on an Aircraft in a Steady Manoeuvre.

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 14-10); thus the manoeuvre stability is greater than the static stability in level flight.

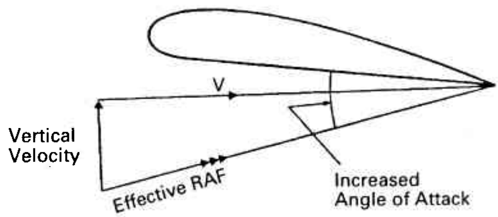


Figure 14-10 Increase in Tailplane Angle of Attack due to its Vertical Velocity

If the aircraft's longitudinal stability is greater in manoeuvre, the position of the CG which achieves neutral stability will be farther 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.

Effect of Altitude

Consider an aircraft flying at two different heights at the same IAS (ie the same value of C_L) and apply the same load factor in each case. Since the TAS is higher at altitude, the rate of pitch of the aircraft decreases

$$\text{(Centripetal force} = \frac{MV^2}{r} = M\varpi V\text{, where } \omega = \text{rate of rotation).}$$

Figure 14-11 shows the decrease in tailplane angle of attack due to the higher TAS and lower rate of pitch. At the same IAS, the higher aircraft has less manoeuvre stability because of the reduction in the tailplane contribution.

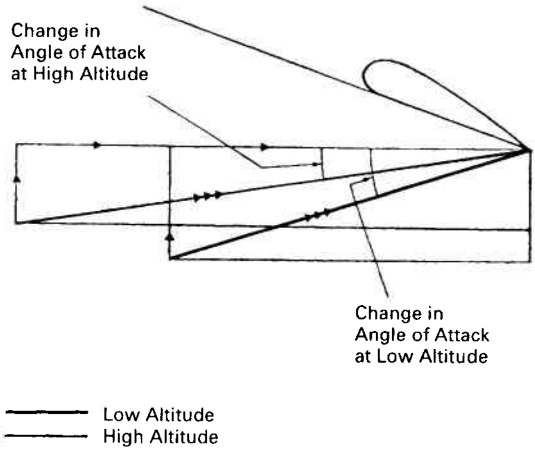


Figure 14-11 Effect of Altitude on the Tailplane Contribution

Lateral Stability (Stick Fixed)

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 14-12). As long as the aircraft is not near the stall the difference in incidence 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. 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 14-12).

The forces arising on the different parts of the aircraft as a result of the sideslip produces 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 14-13b). This effect has a considerable influence on the long-term response (lateral dynamic stability) of the aircraft.

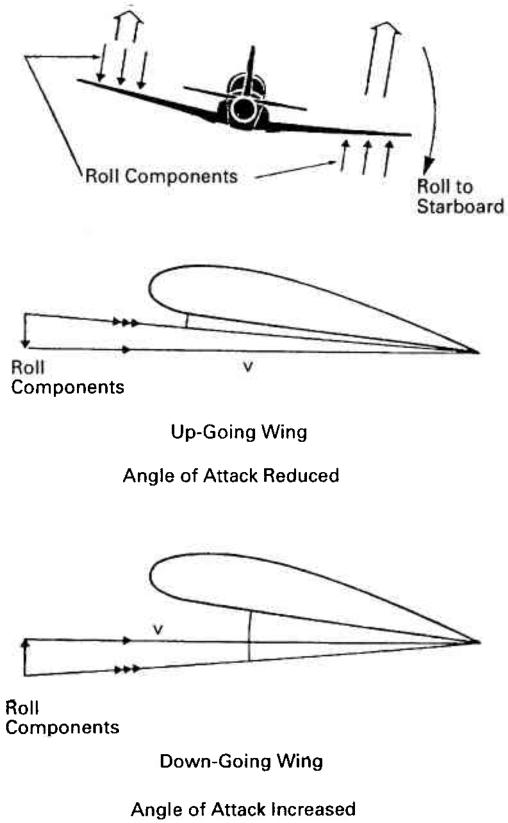


Figure 14-12 Damping-in
Roll Effect

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:
  (i) Dihedral.
  (ii) Sweepback.
(b) Wing/fuselage interference.
(c) Fuselage and fin contribution.
(d) Undercarriage, flap and power effects.

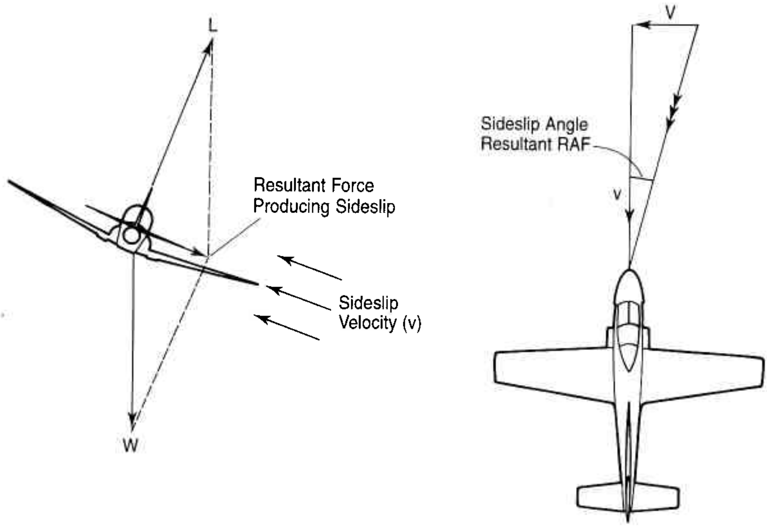


Figure 14-13 Vector Action of Forward and Sideslip Velocities

Dihedral Effect
Dihedral effect can be explained in a number of ways but the explanation illustrated at Fig 14-14 has the advantage of relating dihedral effect to sideslip angle. In Fig 14-14 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). A sideslip to starboard will therefore 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, ie increasing the angle of attack. So long as the aircraft is not flying near the stalling speed the lift will increase.

(b) *Port Wing*. By a similar argument, 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 (ie 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. 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.

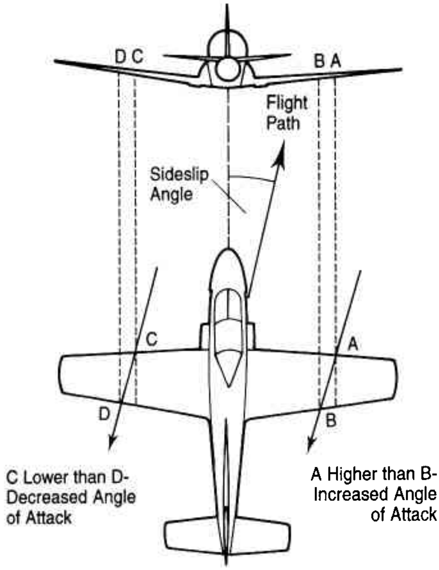


Figure 14-14 Dihedral Effect

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). Figure 14-15 illustrates the principal effects on 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. 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.

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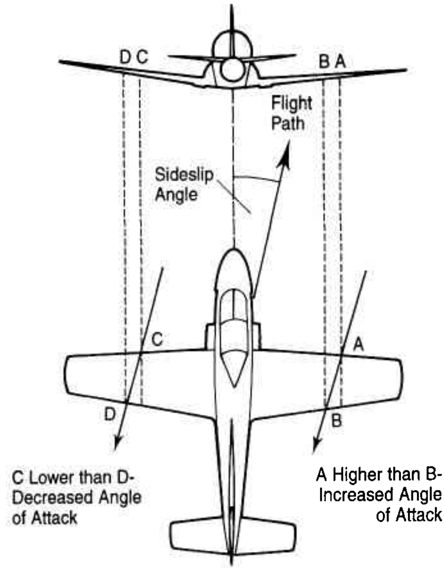

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(b) *Aspect Ratio.* On the leading (low) wing the span is increased and the chord decreased, which is an effective increase in aspect ratio. On the trailing (high) wing, the span is decreased and the chord is increased resulting in a reduction in aspect 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, affects the lift coefficient and also produces a small stable rolling moment in sideslip.

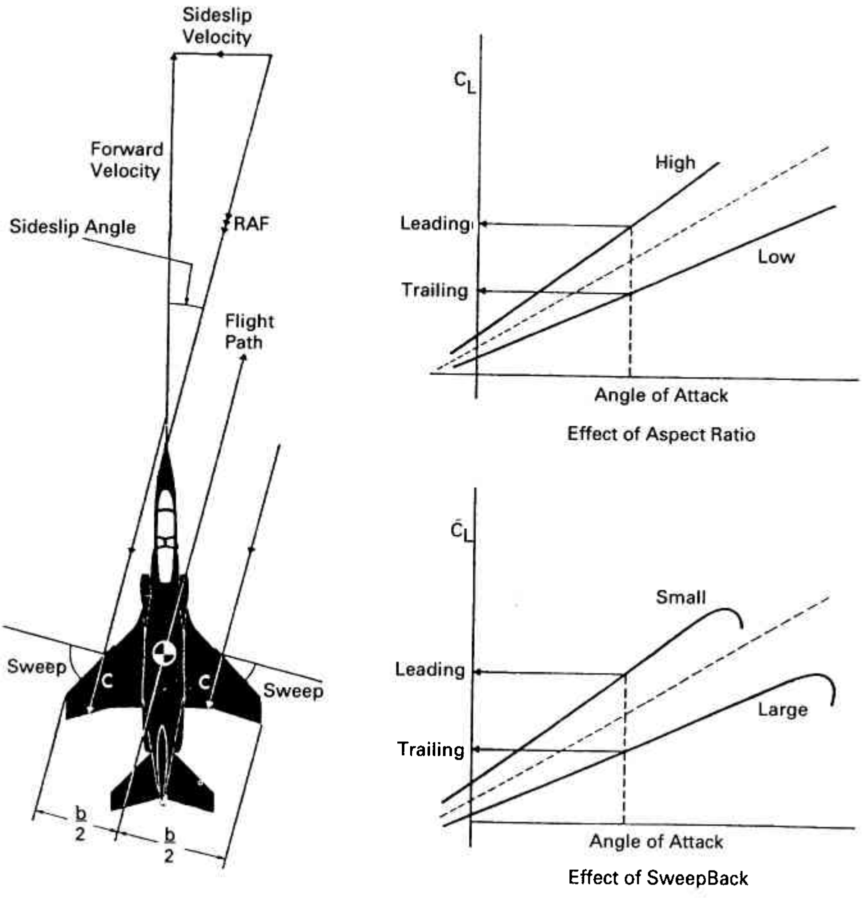


Figure 14-15 Effect of Sideslip on a Swept Planform

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. The contribution of sweep to the lateral (static) stability therefore 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 that for low speeds, therefore the stability at high speeds is much less than that 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.

Handling Considerations

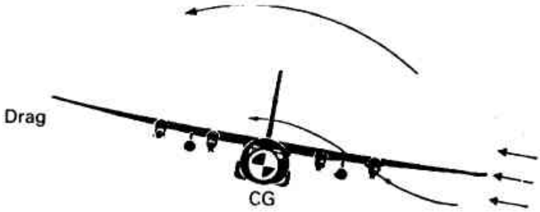
It has been shown that the 'dihedral effect' of sweepback in sideslip produces a strong rolling moment. This has been referred to somewhat imprecisely as roll with yaw. Two applications of this effect at low speeds, where it is strongest, are worth considering:

(a) *Cross-Wind Landings.* After an approach with the aircraft heading into a cross-wind 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.

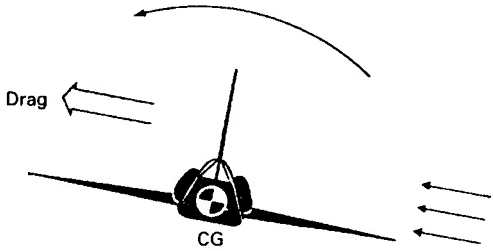
(b) *Wing Drop.* The greater tendency of a swept-wing aircraft to drop a wing at a high angle of attack (aggravated by a steep curved approach) may be further increased by a large deflection of corrective aileron. In such cases the dihedral effect of sweepback may be utilized by applying rudder to yaw 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 incidence and the problem is not as acute as it might have been in the 'transonic era'.

Wing/Fuselage Interference

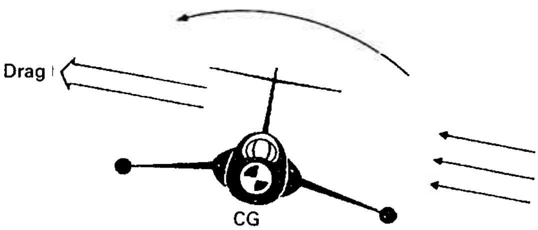
(a) *Shielding Effect* Most aircraft will be affected by the shielding effect of the fuselage. In a sideslip the section of the trailing wing near the root lies in the 'shadow' of the fuselage. The dynamic pressure over this part of the wing may be less than that over the rest of the wing and therefore produces less lift. This effect will tend to increase the 'dihedral effect' and on some aircraft may be quite considerable.



a High Wing



b Low Wing and High Fin



C Tee-Tail

Figure 14-16 Wing/Fuselage Configuration.

(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 around the fuselage, being deflected upwards across the top and downwards underneath. Superimposing a wing in this flow has the following effect, illustrated in Fig 14-16:

(i) *High Wing.* A high-mounted wing 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 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°–3° of dihedral.

(ii) *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°–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.

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 14-16, are:

(a) High wing.

(b) Low Wing and 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.

Slipstream and Flap Contributions

Two important effects which reduce the degree of positive lateral stability are illustrated in Fig 14-17:

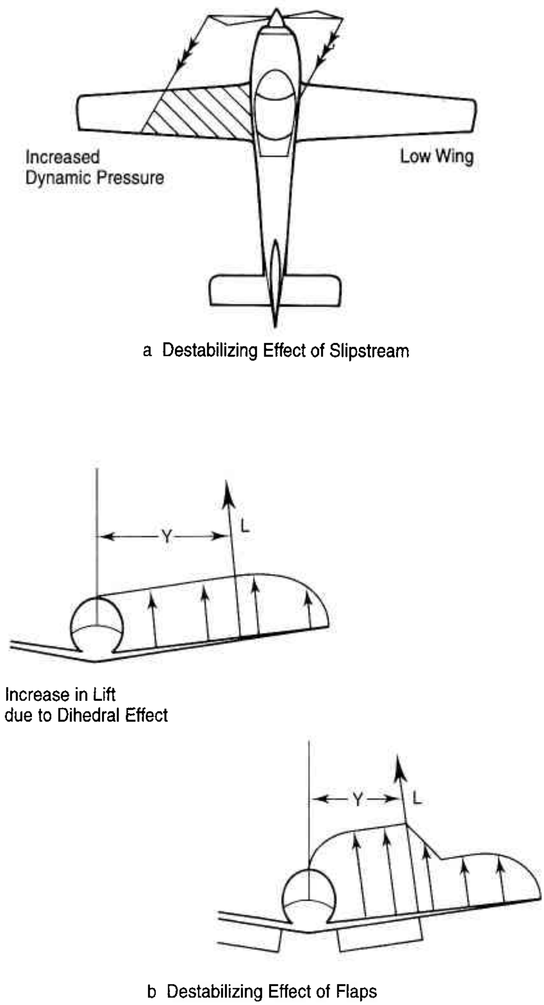


Figure 14-17 Destabilizing effect of Flap and Slipstream.

(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.

Design Problems

It is desirable that an aircraft should have positive lateral static stability. If, however, the stability is too large, it can lead to the dynamic problems listed below, some of which are discussed later:

(a) Lateral oscillatory problems, ie 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.

14.6 Dynamic Stability

General

When an aircraft is disturbed from equilibrium, 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 both in the yawing and the rolling planes.

Some of the factors affecting the long-term response of the aircraft are as follows:

(a) Linear velocity and mass (momentum).

(b) Static stabilities in roll, pitch and yaw.

$$\left.\begin{array}{l}\text{(c) Angular velocities about the three axes.}\\ \text{(d) Moments of inertia about the three axes.}\end{array}\right\}\text{Angular momentum}$$

(e) Aerodynamic damping moments due to roll, pitch and yaw.

Consider a body which has been disturbed from equilibrium 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.

Figure 14-18 illustrates these various forms of dynamic stability; in each case shown, the body has positive static stability.

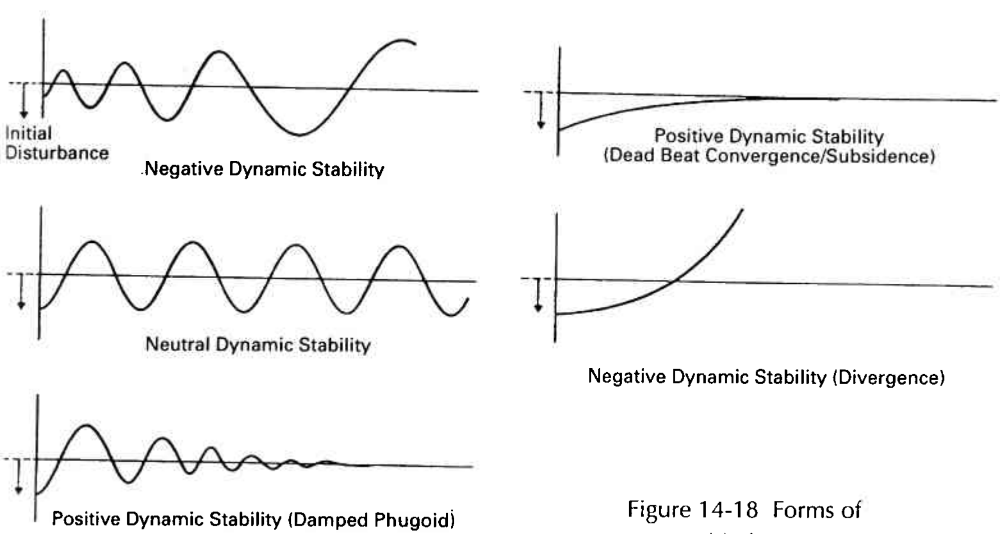


Figure 14-18 Forms of Motion

Dynamic stability is more readily understood by use of the analogy of the 'bowl and ball' described earlier. For example, when the disturbance is removed the ball 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.

If the oscillations are constant in amplitude and time then a graph of the motion would be as shown in Fig 14-19. 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.

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.

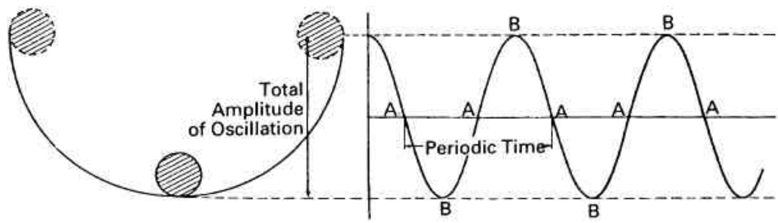


Figure-14-19 Simple Harmonic Motion.

Damping
In the simple analogy given it is assumed that there is no damping in the system; the oscillations will continue indefinitely and at a constant amplitude. In practice, however, there will always be some damping if only because the viscosity of the fluid (air) is a damping factor which is proportional to the speed of 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 14-18 Damped Phugoid). An increase in the damping of the system (eg from a more viscous fluid) will cause the oscillations to die away more rapidly and, eventually, the damping will be such that the oscillation ceases. In this case, after the disturbance has been removed, the mass returns slowly towards equilibrium but does not overshoot it, ie the motion is 'dead-beat' (Fig 14-18 – Positive Dynamic Stability).

Dynamic Stability of Aircraft

Dynamic stability 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 longitudinal (pitching) and lateral (including directional) motions can be neglected. This enables the longitudinal and lateral dynamic stability to be considered separately.

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 (PIO) may develop. The minimum damping specified is that oscillations may decay to one half of their original amplitude in one complete cycle of the motion. Some modern aircraft do not satisfy this requirement and in many cases it has been necessary to incorporate autostabilization systems such as pitch dampers or yaw dampers to improve the basic stability of the aircraft.

Longitudinal Dynamic Stability

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 in pitch can be shown to consist of two separate oscillations of widely differing characteristics; the phugoid and the short-period oscillation, Fig 14-20.

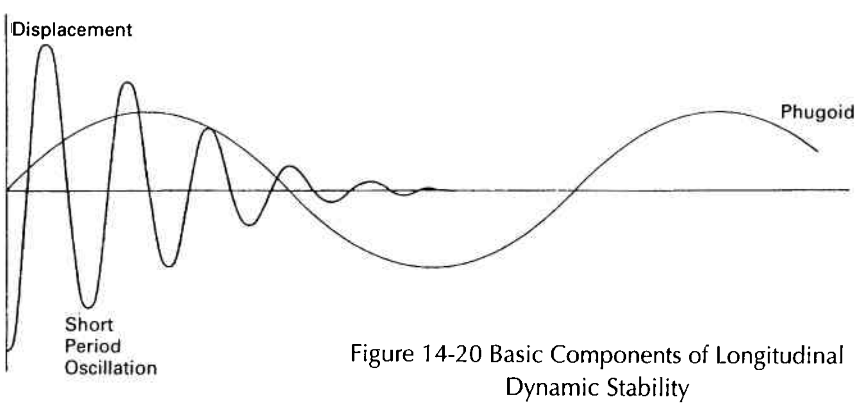


Figure 14-20 Basic Components of Longitudinal Dynamic Stability

Phugoid

This is usually a long period, poorly damped motion involving large variations in the speed and height of the 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 development towards low-drag design has resulted in the phugoid oscillation becoming more of a problem.

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. As stated earlier, the time taken for one complete oscillation will depend upon the static stability, and in this case it is the periodic time of the short-period oscillation.

Stability Factors

The longitudinal dynamic stability of an aircraft, that is, 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.

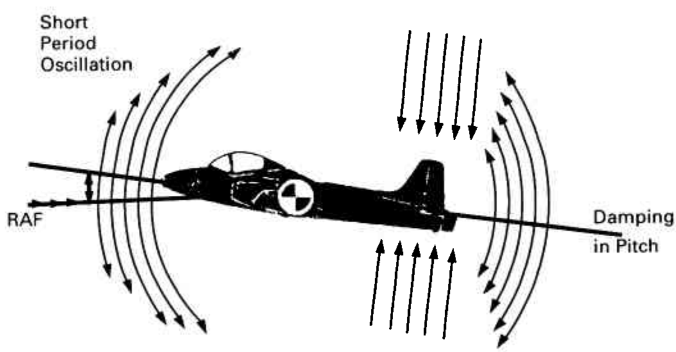


Figure 14-21 Short-period oscillation

Lateral Dynamic Stability
When an aircraft in trimmed level flight is disturbed laterally, the resulting motion consists of the following components:

(a) *Rolling Motion.* Initially the roll will only change the angle of bank, and will be rapidly damped.

(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
The lateral stability of an aircraft depends on the forces that tend to right the aircraft when a wing drops. At the same time however, 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 thereby 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.

If the total rolling moment is strong enough to overcome the restoring force produced by the dihedral and damping in yaw effects, 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.

This form of instability is not very important. 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 and may quickly enter a steep spiral dive; this is due to the interaction of the directional and lateral stability.

Dutch Roll
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 the 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 as follows:

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 the effects is to reduce the lateral stability by setting the wings at a slight anhedral angle.

The lateral dynamic stability of an aircraft is largely decided by the relative effect 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.

14.7 SUMMARY

Static and Dynamic Stability of Aircraft

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.

Stability may be of the following types:

(a) Positive – the body returns to the position 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.

The factors affecting static directional stability are:

(a) Design of the vertical stabilizer.

(b) The moment arm.

The factors affecting static longitudinal stability are:

(a) Design of the tailplane.

- (i) Tail area.
- (ii) Tail volume.
- (iii) Planform.
- (iv) Wing downwash.
- (v) Distance from CP_{tail} to CG.

(b) Position of CG.

- (i) Aft movement of the CG decreases the positive stability.
- (ii) Forward movement of the CG increases the positive stability.

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.

The factors affecting static lateral stability are:

(a) Wing contributions due to:

- (i) Dihedral.
- (ii) Sweepback.

(b) Wing/fuselage interference.

(c) Fuselage and fin contribution.

(d) Undercarriage, flap and power effects.

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

(d) Moments of inertia about the three axes

Angular momentum

(e) Aerodynamic damping moments due to roll, pitch and yaw.

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.

Angular momentum

The lateral dynamic stability of an aircraft is largely decided by the relative effect of:

a) Dihedral effect.

b) Weathercock stability.

Chapter 14: Test Yourself

1 Stability about the normal or vertical axis is provided by:

a) the rudder.
b) the fin and keel surface.
c) the tailplane.
d) the wings and keel surface.

Ref para 14.4

2 Longitudinal stability is increased if the:

a) CP moves forward of the CG.
b) CP acts through the same point as the CG.
c) CG is forward of the CP.
d) thrust acts on a line below total drag.

Ref para 14.4

3 Lateral stability may be increased:

a) with trailing edge flaps lowered.
b) with a high wing.
c) with anhedral wings.
d) with low set wings.

Ref para 14.4

4 Directional stability may be increased with:

a) reduced static margin.
b) pitch dampers.
c) horn balance.
d) yaw dampers.

Ref para 14.4

5 Lateral stability may be increased with:

a) increased dihedral.
b) increased anhedral.
c) lowered trailing edge flaps.
d) yaw dampers.

Ref para 14.4

15

Forces in Flight

15.1 Introduction

The four forces acting in level flight are lift, weight, drag and thrust. The lift acts through the centre of pressure, the weight through the centre of gravity. The drag and thrust act along lines parallel to the longitudinal axis and this is illustrated in Fig 15-1.

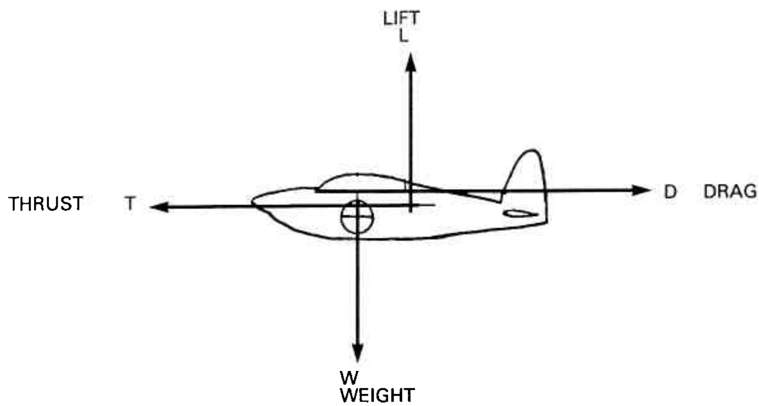


Figure 15-1

For straight and level flight these forces must be in equilibrium but if the points through which these forces act are coincident then the opposing pairs must be equal.

Lift	=	Weight
Thrust	=	Drag

15.2 Pitching Moments

The positions of the CP and CG vary throughout flight, and under most conditions are not coincident, CP varying with angle of attack and CG varying as fuel is used. The result is that the opposing forces (Lift and Weight) set up a couple causing either a nose-up pitch, or a nose-down pitch, depending on the relative positions of CP and CG. This is illustrated in Fig 15-2 and 15-3.

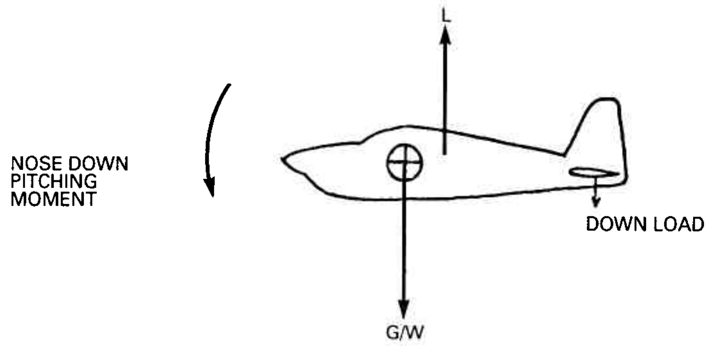


Figure 15-2

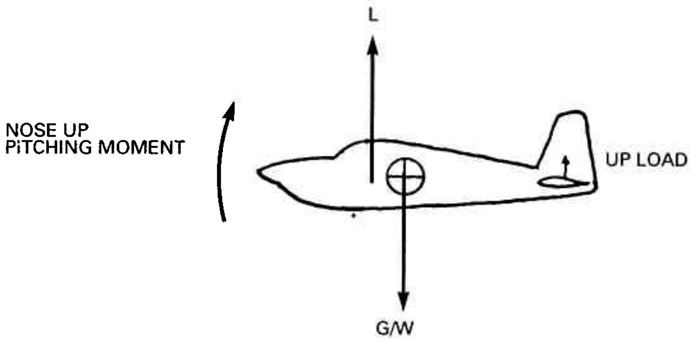


Figure 15-3

Ideally, the pitching moments arising from the Thrust and Drag couples should neutralize each other in level flight, but the ideal is difficult to attain and, as far as possible, the forces are arranged as in Fig 15-1. With this arrangement, the T/D couple causes a nose-up moment, and the L/W couple a nose-down moment, the lines of action of each couple, being such that the strength of each couple is equal. If, now, the engine is throttled back, the T/D couple is weakened, and the L/W couple pitches the nose down.

The tailplane and/or elevator has a stabilizing function in that it supplies the force necessary to counter any residual pitching moments. If any nose-up or nose-down pitch occurs, the elevator deflection can be altered to provide an up or down load to trim the aircraft. This is also shown in Fig 15-2 and 15-3.

If the elevator has to produce a down load balancing force, this effectively increases the aircraft weight. So, to maintain level flight at the same speed, the angle of attack must be increased to maintain lift. The increase in drag is known as trim drag.

The Relationship between Angle of Attack, IAS and Altitude assuming that in level flight lift equals weight then:

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

ie our normal theory of flight formula + angle of attack where:

ρ = density

V = TAS

S = wing area

C_L = a constant coefficient of lift

As for a given weight lift will be constant, then the equation must also be constant. The only variables in the equation are density, TAS and angle of attack. It must be remembered that the expression $\frac{1}{2}\rho V^2$ is dynamic pressure or IAS. In view of this, for a fixed IAS and weight the angle of attack will be constant for any altitude.

Looking at it from a different point of view, if IAS ($\frac{1}{2}\rho V^2$) is increased then, to keep the equation balanced, the angle of attack must be decreased and vice versa. To summarise:

(i) At constant weight and IAS, angle of attack is fixed regardless of altitude.

(ii) If IAS is increased, angle of attack must be decreased and vice versa.

For optimum aerodynamic efficiency, the maximum amount of lift will be produced for the least amount of drag. This, of course, means flying at the maximum lift/drag ratio which has already been shown to occur at a fixed angle of attack, usually around 4°. It was also shown that for a given weight this will represent a fixed indicated airspeed, regardless of height. If however, the weight decreases due to use of fuel, then it will be necessary to decrease the indicated airspeed to maintain the same angle of attack.

15.3 Effects of Climbing, Gliding and Turning

(a) Climbing

During a climb, an aircraft gains potential energy by virtue of elevation, achieved by one or a combination of two means, viz:

(a) Use of propulsive energy above that required for level flight.

(b) Expenditure of the aircraft's kinetic energy.

In a climb, although the weight continues to act vertically downwards the lift does not. The lift is now at right angles to the flight path of the aircraft, and effective weight can now be resolved into two components, one supported by the lift and the other acting in the opposite direction to the flight path, in the same direction as drag. From this, two things can be seen: firstly the lift is now less than that required in straight and level flight, W Cos γ, and secondly, the thrust has to be equal and opposite to the sum of drag and weight components along the flight path $T = D + W \sin \gamma$. This is shown in Fig 15-4. It is still considered sufficiently correct to assume L = D up to about 15° climb angle (Cos 15° = 0.9659, ie the error is less than 2%).

Rate and Angle of Climb

Figures 15-4(a) and (b) show that rate of climb is determined by the amount of excess power, and angle of climb by the amount of excess thrust left after opposing drag.

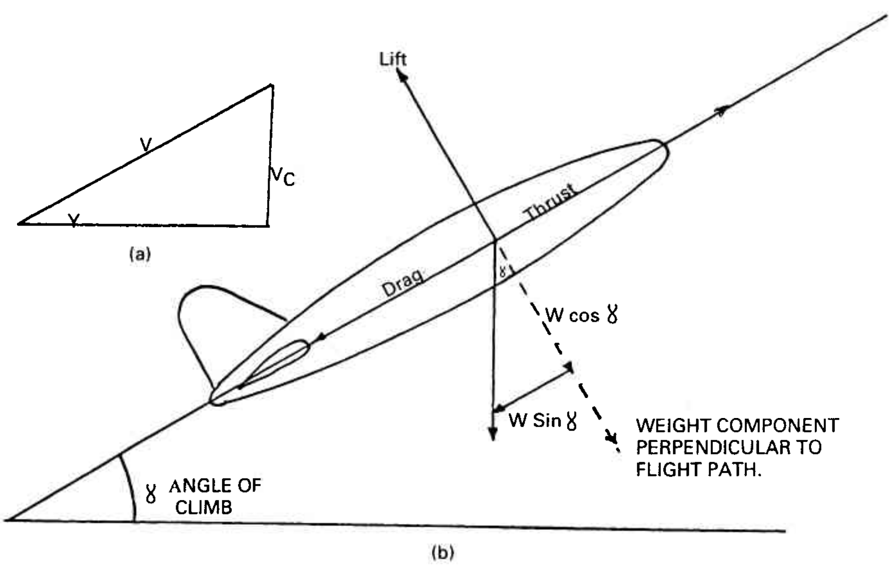


Figure 15-4

Rate of Climb

Fig 15–4 (a) $\quad \sin\gamma = \dfrac{Vc}{V} = \dfrac{\text{Rate of Climb}}{\text{Speed in Climb}}$

Fig 15–4 (b) $\quad \sin\gamma = \dfrac{\text{Thrust} - \text{Drag}}{\text{Weight}}$

Therefore $\quad \dfrac{Vc}{V} = \dfrac{\text{Thrust} - \text{Drag}}{\text{Weight}}$

Therefore $\quad Vc = V \dfrac{(\text{Thrust} - \text{Drag})}{\text{Weight}}$

$$= \frac{\text{Power (Avaliable)} - \text{Power (Required)}}{\text{Weight}}$$

$$= \frac{\text{Excess Power}}{\text{Weight}}$$

or $\quad \dfrac{Vt - Vd}{W}$ $\quad$ where Vt = Thrust Horsepower
Vd = Drag Horsepower

Angle of Climb

From Fig 15-4(b) it can be seen that for the maximum angle of climb, where $\sin\gamma = \dfrac{\text{Thrust} - \text{Drag}}{\text{Weight}}$, the aircraft should be flown at a speed which gives the maximum difference between Thrust and Drag. Alternatively, if climb angle = 0, ie level flight, then

$$\frac{\text{Thrust} - \text{Drag}}{\text{Weight}} = 0$$

But if climb is vertical, ie 90°, then

$$\text{Thrust} = \text{Drag} + \text{Weight}$$

or

$$\frac{\text{Thrust} - \text{Drag}}{\text{Weight}} = 1$$

So, it can be deduced, the factor controlling the angle of climb will be the excess of thrust over drag.

Power Available and Power Required

The thrust power curve for a piston engine differs from that of a jet engine, as shown in Fig 15-5. The main reason for this is that the thrust of a jet remains virtually constant at a given altitude, regardless of speed, whereas the piston engine, under the same set of circumstances and for a given bhp, suffers a loss at both ends of its speed range because of reduced propeller efficiency.

$$\text{THP (avail)} = \frac{\text{Thrust (lb)} \times \text{Speed (fps)}}{550}$$

The horsepower required to propel an aircraft is found by

$$\text{Pwr (reqd)} = \frac{\text{Drag (lb)} \times \text{TAS (fps)}}{550}$$

The curve depicted in Fig 15-5 can be assumed to apply equally to a piston or a jet propelled aircraft, ie the airframe drag is the same regardless of power and speed. The increase in power required at the lowest speed is caused by rapidly rising effects of induced drag.

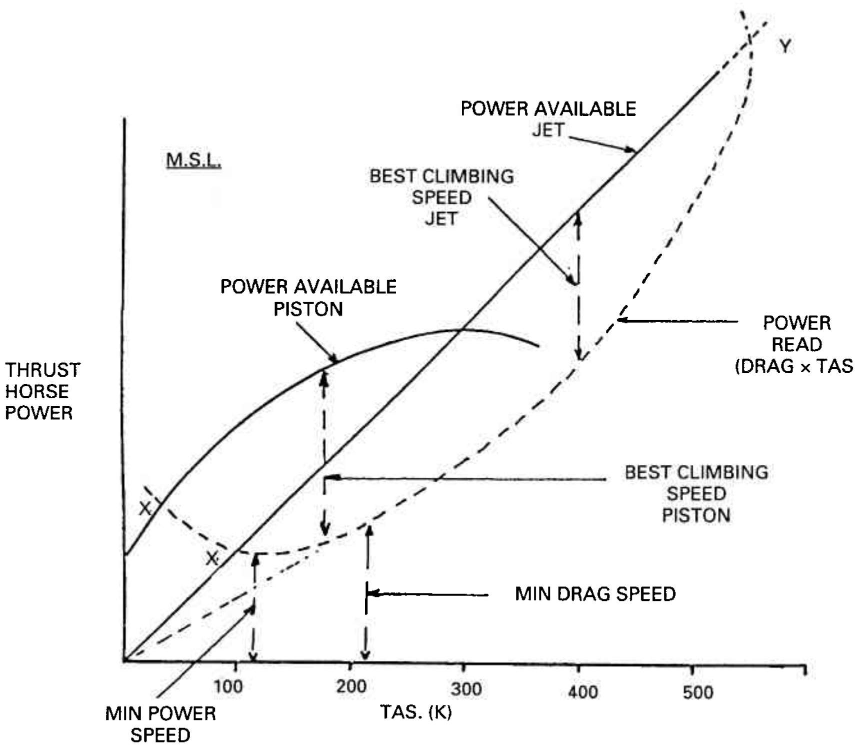


Figure 15-5

Climb Performance

The best climbing speed (highest rate of climb), is that at which the excess power is maximum, so that after some power is used in overcoming drag, the maximum amount of power is available for climbing. The vertical distance between power available and power required represents the power available for climbing at that speed. Note, in Fig 15-5, that this speed for the piston engine is approximately 175K (TAS), and for the jet approximately 400K. In the latter, there appears to be a fairly wide band of speeds which would still give the same excess power; in practice the higher speed is used in the interest of engine efficiency. At points X and Y all available power is being used to overcome drag, therefore these points are the V_{min} and V_{max} for the particular power setting.

Effect of Altitude

The THP of jet and piston engines alike decrease with altitude, due mainly to decreasing air density, so that the power available curves of both types are lowered. Figure 15-6 shows power available and required curves for both engine types, at MSL and 40,000 ft. In Fig 15-4, it is indicated that, at altitude, the power required to fly at minimum drag speed is increased, because though V_{MDrag} remains the same at all heights, in EAS terms, the speed used in calculation of THP is TAS, which increases with altitude for a given EAS. Therefore the THP required to fly at any EAS increases with altitude. Also, from Fig 15-4, speed for best rate of climb also decreases with altitude. The altitude at which rate of climb becomes zero is known as the *absolute ceiling*. *Service ceiling* is the altitude at which the rate of climb has dropped to 100 fpm.

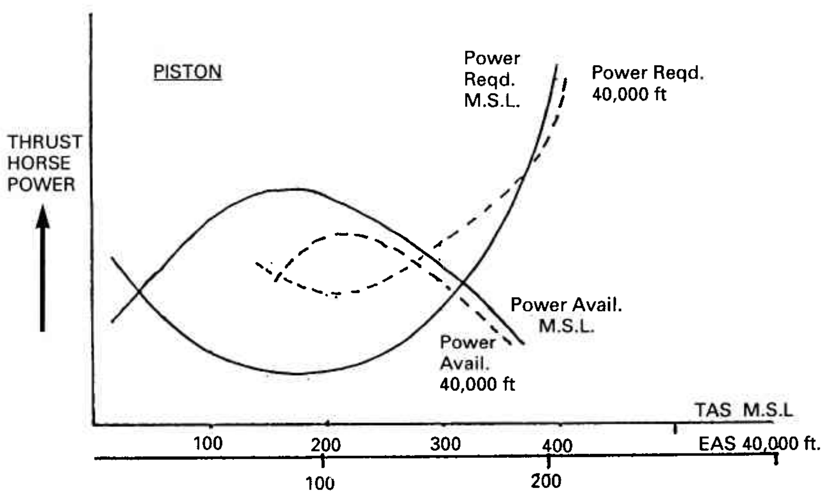


Figure 5-6

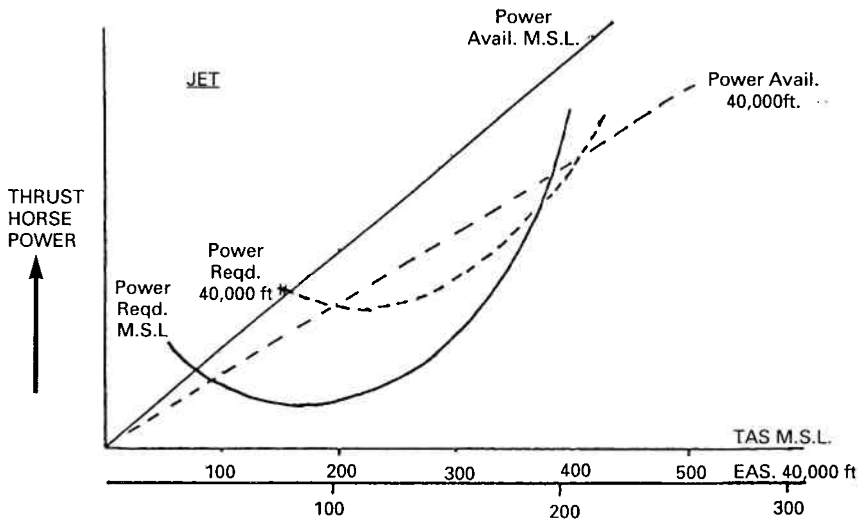


Figure 5-6 (cont'd)

(b) Forces in a Glide

For a steady glide, engine giving no thrust, the Lift, Drag and Weight forces must be in equilibrium (ignoring deceleration effects). Figure 15-7 shows Weight balanced by the resultant of Lift and Drag. The lift vector, acting perpendicular to the flight (glide) path, is now tilted forward, whilst the drag vector still acts parallel to the glide path. To maintain airspeed, energy must be expended to overcome this drag, and the source of this energy is the aircraft's potential energy, ie Attitude.

When the aircraft is placed in a nose down attitude, as in a descent, the component of weight in the direction of the flight path augments the thrust, the aircraft will accelerate, lift and drag will change, so, in order to achieve a balanced condition with a constant airspeed, thrust must be reduced. From the foregoing, it may be seen that the controlling factor of the glide angle is the lift/drag ratio of the aircraft. An increase in weight will not affect glide angle, as all components will expand by the same proportion, but an increase in weight will increase speed along flight path.

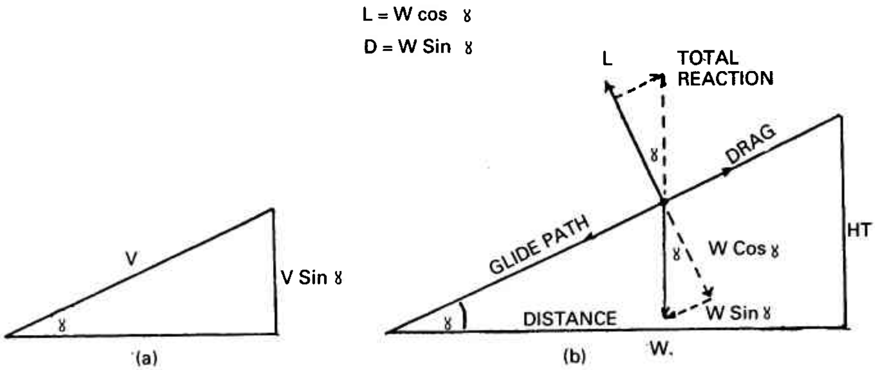


Figure 15-7

Gliding (Descent) for Endurance
It can be seen in Fig 15-7(a) that the minimum rate of descent is achieved by making V Sin γ as small as possible, but, D × V (ie power required) = WV Sin γ (ignoring deceleration effects), thus for a given weight the rate of descent is least, at the speed where the power required (DV) is least.

Gliding (Descent) for Range
If distance is to be maximum, glide angle must be minimum, as evident in Fig 15-7(b).

$$D = \text{Max} = W \text{ Cos } \gamma$$

$$\gamma = \text{Min} = W \text{ Sin } \gamma$$

Therefore

$$\frac{W \text{ Cos } \gamma}{W \text{ Sin } \gamma} = \text{Max}$$

but

$$\frac{W \text{ Cos } \gamma}{W \text{ Sin } \gamma} = \frac{L}{D}$$

The best angle of glide therefore depends on maintaining an angle of attack which gives the best Lift/Drag ratio, or for maximum endurance the aircraft should be flown for minimum drag.

Effect of Wind

Gliding for minimum rate of descent, or for endurance, is unaffected by wind, because the position at the end of glide is unimportant. But when gliding for range, the target is the point of arrival, thus the aim is maximum distance over the ground.

Gliding for range is achieved as explained previously, ie by flying for minimum drag. However, that applies only in still air conditions. The effect of a headwind will be to decrease ground distance by approximately the ratio of $\frac{WS}{TAS}$. An increase of airspeed could reduce the time the wind effect would act, and thus improve ground distance. Similarly, if there were a tailwind, ground distance would be increased, a reduction of speed would improve the distance, since the wind effect time would be increased.

Effect of Weight

Variation in weight will not affect glide angle, provided speed is adjusted to suit the all up weight. A simple method of estimating speed changes, to compensate for weight changes (up to about 20%), is to adjust speed (EAS) by half the percentage change in AUW, eg a weight reduction of 10% would necessitate a decrease in speed of 5%.

Although range is unaffected by weight, glide endurance decreases with weight increase.

Penetration Speed is the optimum gliding speed for any wind speed.

(c) Turning

During a turn weight still acts vertically downwards but a second force, centrifugal force, occurs from the aircraft travelling along a curved path. This centrifugal force has to be opposed by a centripetal force which can only be obtained as a resolved part of the lift force. Because the lift also has to balance the weight in addition to the centripetal force, it is evident that in turn the lift has to be increased to a greater value than weight. This is illustrated in Fig 15-8.

If the aircraft is banked, with the angle of attack constant, the vertical component of lift will be too small to balance the weight, thus the aircraft will descend. Therefore, as angle of bank increases, angle of attack must be increased, the vertical component is then sufficient to maintain level flight, whilst the horizontal component is sufficient to produce the required centripetal force.

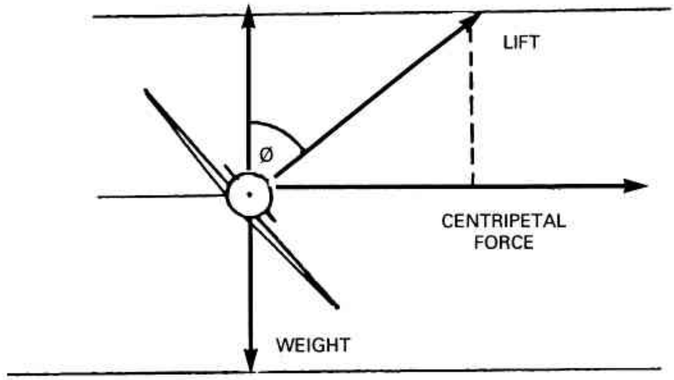


Figure 15-8

Effect of Weight

If the IAS in a turn is maintained at a constant figure the increased lift can only be obtained by an increase in the angle of attack. The increase in lift will, of course, produce more induced drag which will require an increase in thrust. As the angle of attack has been increased the wing is nearer to its stalling angle, therefore the stalling speed will be increased. The increase in the value of lift is, in fact, equivalent to increasing the aircraft's weight. The amount by which this is apparently increased is called the load factor or 'n'. For instance, if the weight is apparently doubled 'n' becomes two and this is called a 2g turn. The increase in stalling speed associated with the load factor may be calculated from the following formula:

$$\text{Stalling speed} = \text{normal stalling speed} \times \sqrt{\text{g load (n)}}$$

For example an aircraft with a normal stalling speed of 100 kt carrying out a 2g turn would have its stalling speed increased by $100 \times \sqrt{2} = 140$ kt approximately.

In a steady level turn, thrust being ignored, lift is providing both a force to balance weight, and a centripetal force to turn the aircraft. If the same TAS and angle of bank can be sustained, the turn radius is basically independent of weight or aircraft type.

Minimum Radius Turn
To achieve a minimum radius turn, it can be shown that:

(a) Wing loading must be as low as possible.

(b) Air must be as dense as possible, ie as at MSL.

(c) The maximum value of product of C_l and angle of bank must be obtained. NOTE: *not* maximum angle of bank, since angle of bank is increased to increase the lift force required for the centripetal force. To do this, at the critical angle of attack, speed must be increased, but an increase in speed may cause a fall in maximum value of C_l.

The Maximum Rate Turn
To achieve a maximum rate turn, it can be shown that:

(a) Wing loading must be as low as possible.

(b) Air must be as dense as possible, as at MSL.

(c) The maximum value of the product of angle of bank, speed and C_l must be obtained. Note, as for the same reasons given in preceding paragraph.

Altitude
With increase in altitude, there is an increase in the minimum radius, mainly due to the EAS/TAS relationship. An additional increase is caused by the reduction in C_{lmax}, because Mach No is higher at altitude for a given TAS.

An increase in altitude will cause the rate of turn to decrease.

Effect of Thrust
Even in level flight, it can be seen that some aircraft have their thrust line inclined to the horizontal, thus producing a component of thrust augmenting lift. In the minimum radius turn, and maximum rate turns discussed, the aircraft is flown for C_{lmax}, which is obtained at the critical angle, the thrust component assists lift, so either less lift is required from the wing, or the turn can be improved. However, the reduction of thrust with increasing altitude will cause a reduction in turning performance, in addition to that caused by the EAS/TAS relationship and the greater C_{lmax} reduction.

Effect of Flap

Lowering of flap produces more lift, also more drag at any given EAS. A smaller radius of turn may thus be achieved with flap, providing the flap limiting speed is not a critical factor, and the available power is sufficient to overcome the extra drag.

(d) Turning and Manoeuvres: Essential Points to Note

Centripetal force:

Consider an object swinging around at the end of a piece of string – the object travels along a curved path produced by the pull of the string acting on the object. Since this radial force is directly towards the centre the acceleration must also be towards the centre. This centre-seeking force is called *centripetal* force, and in accordance with the third law of motion, is opposed by an equal force called the *centrifugal* force. Centripetal force in the case we are considering is also called Horizontal Component of Lift.

Although the object on the string is following a curved path of motion, it is continually trying to obey the first law of motion, ie to continue travelling in a straight line . . . true or false? True – should the string be released, centripetal force is removed and the opposite reaction (centrifugal force) disappears simultaneously. In this instance, the object at once obeys the first law of motion and flies off in a straight line at a tangent to its previous circular path.

It is important to realise that, without centripetal force, no object whether a car or aircraft can be made to turn, and the first law of motion applies.

Centripetal force during a given turn is directly proportional to the mass of the body, the square of its speed and is inversely proportional to the radius of the turn. It is calculated from the formula:

$$\text{Centripetal force} = \frac{W}{g}\frac{V^2}{r} \text{ (in lb)}$$

Or

$$\text{Centripetal force} = \frac{m\,V^2}{r} \text{ (in Newtons)}$$

Where:

W = the weight/or m is the mass

V^2 = the square of the TAS in feet/sec or m/sec

r = the radius in feet or metres

g = the gravitational force of 32.2 ft/sec/sec

To calculate the acceleration towards the centre, the following formula applies:

$$\text{Acceleration} = \frac{V^2}{r}$$

Where V^2 = the velocity in feet/sec or metres/sec

r = the radius in feet or metres

Turning

For an aircraft to turn, centripetal force is required. This centripetal force is derived by resolving the inclined total lift force into two components, namely:

(a) Vertical lift component

(b) Horizontal lift component

Thus, it is the horizontal lift component that provides the centripetal force required to pull the aircraft towards the centre of the turn as it moves along a path of circular motion. (Refer to Fig 15-9). However, during a turn, lift has a double role to play. Not only is it resolved into a horizontal component to provide centripetal force, but also has to provide a lifting force such that the aircraft maintains a constant height during the turn. It will be seen from Fig 15-9 that any inclination of total lift from the

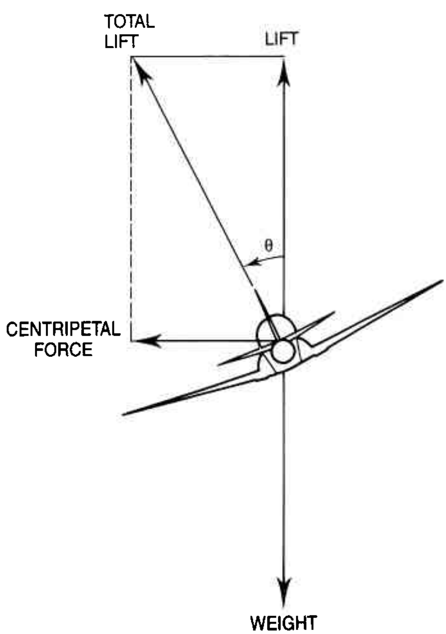


Figure 15-9

vertical results in a smaller vertical component of lift, which would then be too small to balance the weight of the aircraft.

Therefore, to prevent the aircraft from descending, the angle of attack must be progressively increased to maintain a greater total lift. Once this has been accomplished, the vertical component of lift is large enough to maintain level flight, while the horizontal component is large enough to generate the required centripetal force. However, an increase in angle of attack results in an increase in drag, which must be balanced by an increase in power if the speed is to remain constant.

Steep Turns

A steep turn is classified as a turn having an angle of bank in excess of 45 degrees. Larger bank angles require a larger angle of attack to produce the required total lift increment. However, the penalty of large angles of attack is drag. Eventually, the aircraft will reach a speed so low, that any further increase in angle of attack will result in a stall. At this instant, angle of attack and induced drag are so high that full power is usually necessary to keep the speed constant.

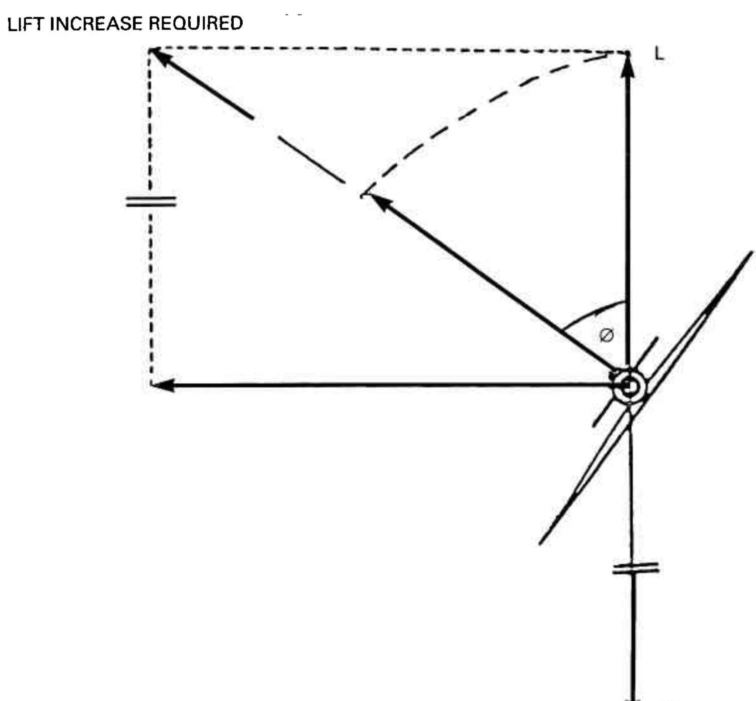


Figure 15-10

Wing Loading

Wing loading is the weight of the aircraft divided by the wing area.

$$\text{Since } L = W \text{ and } L = C_L \tfrac{1}{2}\rho V^2 S \text{, then } W = C_L \tfrac{1}{2}\rho V^2 S$$

Thus wing loading (ie the weight carried by a wing of given area) affects both the maximum and minimum stalling speeds.

However, modern tendency is to increase the wing loading by decreasing the wing area and increasing the speed, and to use flap to reduce landing speeds.

Load Factor

The load factor of a given aircraft in a given condition of flight is defined as the lift divided by the weight. It is denoted by n.

In straight and level flight, $L = W$; therefore $n = 1$. In any manoeuvre where lift is greater or smaller than weight, $L = nW$.

In any manoeuvre, the stalling speed is proportional to the square root of the load factor. (Vn) Limit load factor.

Calculation of centripetal force and loading during a turn:

Consider an aircraft weighing 11,500 lb, flying at 200 knots (338' sec) in a turn having a radius of 2000 feet.

$$\text{Centripetal force} = \frac{W}{g}\frac{V^2}{r}$$

so:

$$= \frac{11500 \times 338^2}{32.2 \times 2000} =$$

$$\text{Centripetal force} = 20{,}400 \text{ lb}$$

Refer to Figure 15-11 for the wing loading calculation. The wing loading which is equal to lift may be calculated by Pythagoras' Theorem, where:

$$TL^2 = F^2 + L^2$$

$$TL = \sqrt{F^2 + L^2}$$

$$TL = \sqrt{20400^2 + 11500^2}$$

$$TL = 23418 \text{ lb}$$

Accelerated or g stalls in a turn:

As already discussed, any increase in bank angle (tightening of the turn) adversely affects the stalling speed. Eventually, the angle of attack reaches the critical angle, resulting in the buffet.

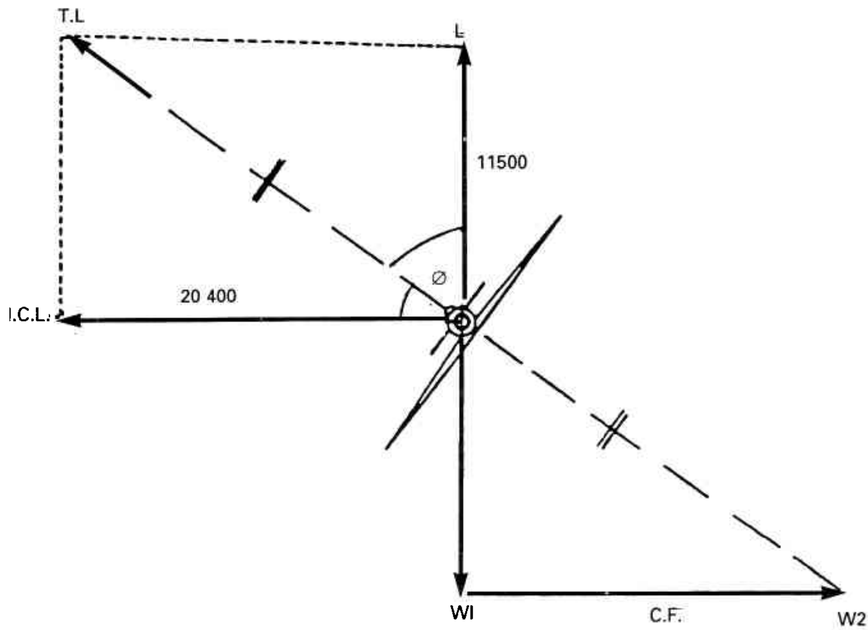


Figure 15-11

However, should one wing stall before the other, the aircraft would tend to roll in or out of the turn, due to unequal lift distribution. This roll may in some cases result in a 'flicking movement'.

Recovery is initiated by moving the control column forward and, in doing so, decreasing the angle of attack, thereby unstalling the aircraft. This occurrence is known as a 'high speed' or 'accelerated' stall.

Minimum Radius and Maximum Rate of Turns

As the angle of bank is increased, the horizontal component (ie centripetal force) consequently increases. The greater this centre-seeking force, the less the radius becomes (ie minimum radius).

Figure 15-12 demonstrates that:

$$\tan Ø = \frac{L}{HCL} = \frac{W}{WV^2/gr} = \frac{V^2}{gr}$$

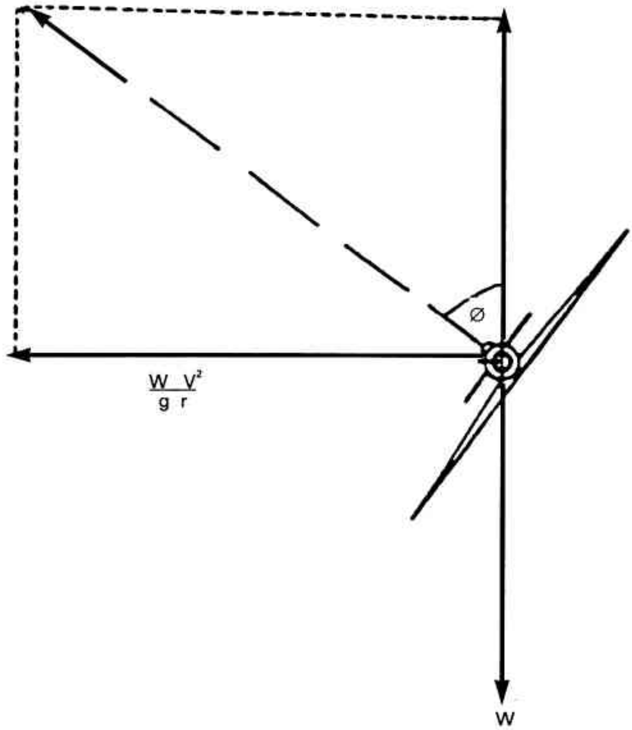


Figure 15-12

A vertical banked turn is impossible because even if Total Lift becomes infinity no vertical component can be obtained. However, even with a vertical bank there is a limit to the radius of the turn because (apart from side slipping), the wings must provide all the horizontal force (ie C_L ½ V^2S), represented by the formula:

$$\text{Centripetal force} = W\ V^2/g\ r \qquad \text{(i)}$$

$$\text{but – Centripetal force} = C_L\ \tfrac{1}{2}\rho V^2 S\ \text{(vertical bank)} \qquad \text{(ii)}$$

$$\text{Therefore :}\quad W\ V^2/g\ r = C_L\ \tfrac{1}{2}\rho V^2 S \qquad \text{(iii)}$$

$$\text{or}\quad r = 2W/(C_L\rho\ S\ g) \qquad \text{(iv)}$$

Straight and level stalling speed is given by the equation:

$$W = L = C_{Lmax}\ \tfrac{1}{2}\rho V^2 S \qquad \text{(v)}$$

Therefore, by substituting this value of W into the equation (iv), we get:

$$r = (2\,C_{Lmax})\ \frac{\frac{1}{2}\rho V^2 S}{C_L \rho S\, g}$$

$$r = \left(\frac{V^2}{g}\right) \times \frac{C_{L\,(max)}}{C_L} \qquad \text{(vi)}$$

Equation (vi) shows that when C_{Lmax} is equal to the C_L, the radius of the turn will be minimum. (Obviously, when C_L is at a maximum value, the angle of attack is the stalling angle and the radius of turn $= \frac{V^2 S}{gr}$

$$\text{Load factor: } \frac{\text{Total weight}}{\text{Aircraft weight}} = \frac{23418}{11500} = 2.0$$

The loading on the aircraft is thus 2.0.

Furthermore, it is true to say that the load factor varies as the secant $\left(\frac{1}{\text{(Cosine)}}\right)$ of the bank angle.

Consider an aircraft in a 60 degree bank turn.

Cosine 60 degrees = 0.5

Secant 60 degrees = 2.0

Thus, the aircraft has a load factor of 2.0 during a 60 degree bank turn.

Determination of the stalling speed during a turn:
Any manoeuvre which requires additional lift consequently increases the load factor and thus raises the stalling speed. This is true of any turn and the stalling speed may be calculated from the formula:

$$\text{New stalling speed} = \text{Old stalling speed x } \sqrt{\text{load factor}}$$

From the foregoing example, let us assume the aircraft had a basic stalling speed of 85 knots at gross weight. The new stalling speed during the turn is therefore:

$$\begin{aligned} \text{New V} &= \text{old } V_s \times \sqrt{n} \\ &= 85 \times \sqrt{2} \\ &= 120 \text{ knots.} \end{aligned}$$

Answer the following questions

1 During a turn having a bank angle of 45 degrees, the stalling speed is 100 knots. Calculate the basic stalling speed.

Answer : 84 knots

2 Assume an aircraft weighing 11,500 pounds pulling out of a dive. If a force of 4 g was registered, what was the centripetal force and the new stalling speed if the basic stalling speed was 78 knots?

Answer : 46,000 pounds ; 156 knots.

Thus, the minimum radius of turn is settled by the stalling speed of that aircraft. However, engine power is the final deciding factor in settling the minimum radius.

To summarize:

1) Fly at any speed provided engine power can maintain it.
2) Fly at the maximum permissible load factor (C_{Lmax}).
3) Air must be as dense as possible (density is a factor in the lift formula).

Maximum Rate Turns
In this turn, the angular velocity of the aircraft during a turn must be as high as possible:

$$\text{ie } \frac{\text{Velocity (ft/sec)}}{\text{radius (ft)}} = \text{Time}$$

Thus minimum time will result if the radius is kept at a minimum value and the aircraft is flown at a minimum speed (ie where C_L is maximum).

To summarize:

1) Fly at the stalling speed.
2) Fly at maximum load factor.
3) Air must be as dense as possible.

Climbing and Descending Turns
During a climbing turn, the outer wing travels a greater distance than the inner wing. This results in the outer wing having a larger angle of attack which leads to an overbank situation.
Bank must therefore be held off during climbing turns. The opposite applies for descending turns, where bank must be held on.

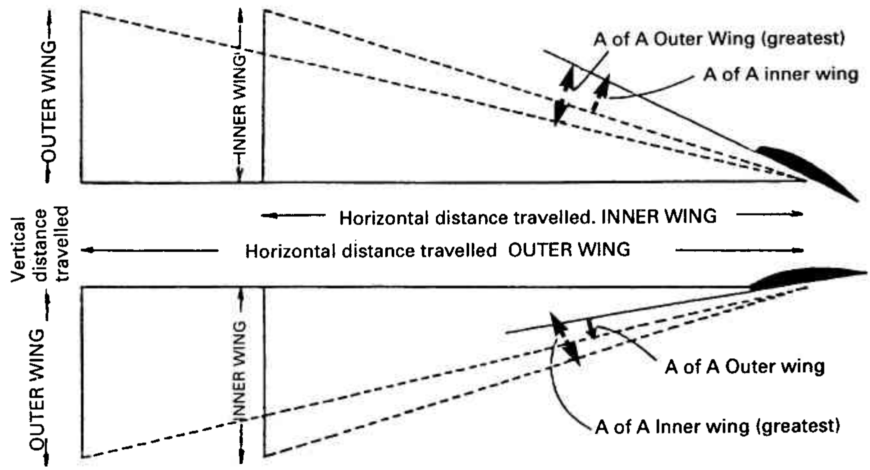


Figure 15-13

Chapter 15 Test Yourself.

1 With increasing altitude the power required from a piston engine:

a) reduces and power available increases.
b) increases and power available increases.
c) increases and power available reduces.
d) increases and power available remains constant.

Ref para 15.3

2 The height at which the rate of climb drops to 100 fpm:

a) is termed the absolute ceiling.
b) is known as the rated ceiling.
c) is the service ceiling.
d) is the critical height.

Ref para 15.3

3 If the weight of an aircraft is increased, its glide range will:

a) be the same.
b) be increased.
c) be reduced.

Ref para 15.3

4 With an increase in aircraft weight:

a) glide endurance will remain the same.
b) glide endurance will increase.
c) glide endurance will reduce.

Ref para 15.3

5 In a turn the centrifugal effect is opposed by:

a) centripetal force
b) thrust only.
c) a component of weight.
d) a component of thrust and weight.

Ref para 15.3

16

High Speed Flight

16.1 Introduction

Low speed aerodynamics is based on the assumption that air is incompressible; the attendant errors are negligible since at low speeds the amount of compression is negligible. At speeds approaching that of sound, however, compression and expansion in the vicinity of the aircraft are sufficiently marked to affect the streamline pattern about the aircraft. At low subsonic speeds a flow pattern is established about the aircraft, but at high subsonic and supersonic speeds the flow around a given wing can be controlled, and its behaviour predicted. In the transonic range where a mixture of subsonic and supersonic flow exists, marked problems of control and stability arise, necessitating special design features to minimise the effects of compressibility.

16.2 Definitions

(a) *Speed of Sound* The speed at which a very small pressure disturbance is propagated in a fluid under certain conditions. Speed of sound is proportional to the absolute temperature (K) and can be calculated from the formula:

$$\text{Local speed of sound (LSS)} = 39 \times \sqrt{\text{absolute temperature}}$$

Therefore, the higher the temperature, the higher the LSS. In fact, at MSL at ISA LSS = 661 kt, and at 30,000 ft LSS = 589 kt.

Derivation of the formula for ISA conditions is as follows:

$$\text{LSS} = C \times \sqrt{288°K} = 661 = C\sqrt{288°K}$$

Therefore

$$C = \frac{661}{\sqrt{288°K}} = 38.95°$$

For practical purposes, the figure of 39 may be used.

(b) *Mach Number (M)* The ratio of True Airspeed (TAS) to the local speed of sound applicable to air temperature. Thus

$$\text{Mach No (M)} = \frac{\text{TAS}}{\text{LSS}}$$ therefore at sea level temperature 15°C

TAS = 529 kt. LSS = 661 kt. $M = \frac{529}{661} = 0.80$

(c) *Free Stream Mach No (M_{FS})* The Mach number of the flow at a point unaffected by the presence of the aircraft.

(d) *Local Mach Number (M_L)* When an aerofoil is placed in a subsonic airflow, the flow is accelerated in some places, and slowed down in others. The local Mach number is the speed at some specified region of flow, and may be greater than, the same as, or lower than M_{FS}.

(e) *Critical Mach Number (M_{crit})* This is the lowest M_{FS} which for a given aerofoil and angle of attack, gives rise to a M_L of 1.0 on the aerofoil. As will be seen, M_{crit} for a wing varies with angle of attack.

(f) *Compressibility Mach Number* The Mach number at which, because of compressibility effects, control of an aircraft becomes difficult, and beyond which loss of control is probable.

(g) *Critical Drag Rise Mach Number* relates the Mach number to an appreciable increase of drag associated with compressibility effects, usually 10-15% higher than M_{crit}.

16.3 Airflow

(a) *Subsonic* flow when free stream Mach numbers are such that local Mach numbers are less than M 1.0 at all points.

(b) *Transonic* flow, the M_{FS} is high enough to produce M_L, some of which are greater than M 1.0.

(c) *Supersonic* flow, M_{FS} is such that at *all* points M_L are greater than M 1.0.

(d) *Hypersonic* flow, M_{FS} is greater than M 5.0.

16.4 Speed of Sound

Anything which moves through the air creates pressure waves and, what may not be generally realised, these waves not only travel out in all directions from the object but they radiate at the speed of sound. If the object is moving at a speed less than the speed of sound these pressure waves will be able to move away from the object. When considering an aircraft moving at very high speed it is possible that the sound waves cannot get away from it, because the aircraft's speed is close to the radiation speed of the waves. It is this which gives rise to the problems of high speed flight.

Figure 16-1(a) illustrates the situation of an aircraft flying at less than the speed of sound. If its starting point is A, then the pressure waves sent out in all directions from the aircraft are moving steadily away and by the time point B is reached they will be well clear of the aircraft. This should be contrasted with the situation illustrated in Fig 16-1(b) where the aircraft is travelling just at the speed of sound. The pressure waves are also travelling at the speed of sound with the result that they pile up ahead of the aircraft and form into a pressure wave, also called a shock wave,

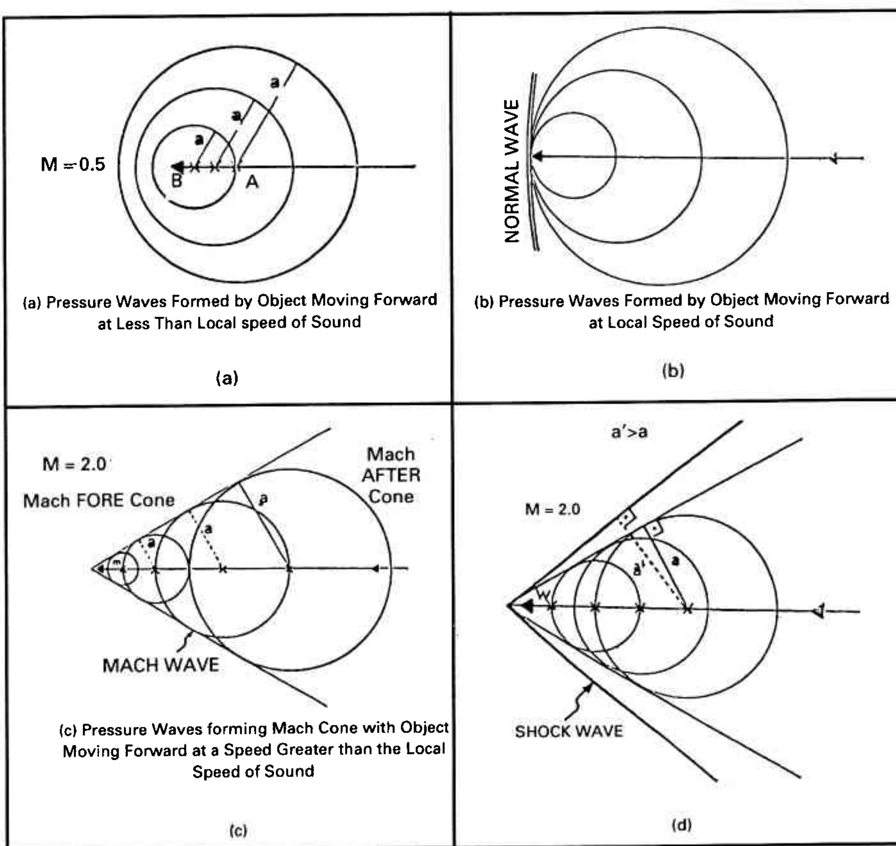


Figure 16-1 (a), (b), (c) & (d)

shown in Fig 16-1(c). An aircraft travelling substantially faster than the speed of sound will leave its own pressure waves behind and form a cone of pressure waves as illustrated in Fig 16-1(d).

16.5 Shock Waves

When a shock wave is formed the pressure distribution over the wings is materially altered, causing considerable alterations in the values of lift and drag and also affecting control operation. It could be argued that few civil passenger transport aircraft are capable of reaching the speed of sound, however, the air over the upper surface of the wing is deliberately accelerated in order to produce lift and even though the aircraft itself may be flying below the speed of sound, some of the air flowing over the wings may be accelerated to Mach 1.0. When the airflow over the upper surfaces of the wing reaches Mach 1.0, the actual speed of the aircraft is called the critical Mach Number or M_{crit}. When this point is reached a shock wave forms over the upper surface of the wing because the pressure waves from the rear of the wing that are trying to move forward are meeting air travelling at exactly the same speed flowing backward. This is similar to trying to move along a moving walkway in the wrong direction at the same speed as the walkway is travelling. The point at which this shock wave usually forms is just aft of the point of maximum camber of the wing where the acceleration of the air is greatest. In front of the shock wave the flow is at or higher than Mach 1 whilst behind the flow it is still subsonic.

At the shock wave, the normal laws of physics seem to break down and as the air passes through the shock wave the pressure *increases* and the temperature increases. If the speed of the aircraft is increased still further the region of supersonic flow on top of the wing also increases and the shock wave will start to move back towards the trailing edge. On the undersurface the curvature of the wing is usually less than on the upper surface and the shock wave will form later. However, once having formed, if the actual speed of the aircraft is further increased, this shock wave will also move rearward and when the actual speed of the aircraft reaches Mach 1 both shock waves will have migrated to the trailing edge of the wing. At the same time another shock wave will form close to the leading edge of the wing, this is called the bow wave. If speed is further increased this bow wave will actually touch the leading edge of the wing and is then termed an 'attached bow wave'. This is illustrated in Fig 16-2 and further speed increases will not change the relative positions of these two shock waves, but will just bend them backwards. The next diagram, Fig 16-3, illustrates the behaviour of the shock waves from a speed below Mach 1 to one well in excess of the speed of sound.

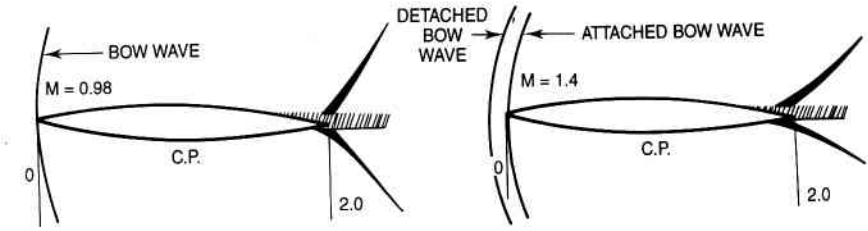


Figure 16-2

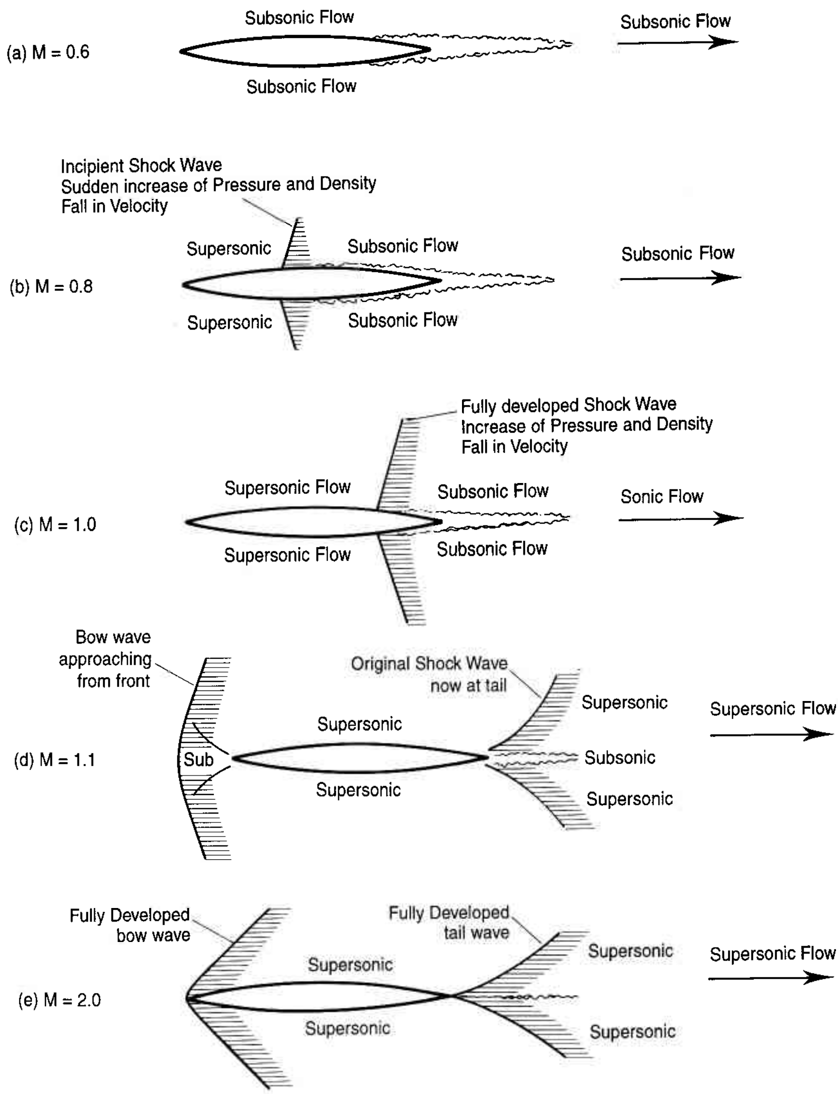


Figure 16-3

16.6 Wave Drag

For aircraft not designed for transonic and supersonic flight, the formation of these shock waves will have a marked effect on lift, drag and also on the general stability of the aircraft and its control. The basic cause of the problems is the separation of the airflow behind the shock wave due to the rise in pressure. This causes the boundary layer to separate, reducing the amount of lift produced by the wing and an increase in drag. This increase in drag is very marked at M_{crit} and produces 'wave drag' – as previously mentioned in the chapter on total drag. If speed can be increased further against this drag force the shock waves will move towards the trailing edge, thus reducing the amount of separated air and, in fact, the lift will start to increase again and the drag decrease. However, this will only occur in aircraft which are designed for transonic flight but the total developed lift at higher speeds is less than at subsonic speeds for the same angle of attack. This is because C_L is less for any given wing section and angle of attack at supersonic speeds. The change in both lift and drag are illustrated in the following graphs, Fig 16-4 and Fig 16-5. It should be noted that this loss of lift which occurs as a result of the shock wave is not dissimilar in effect to that produced by a low speed stall. For this reason it is sometimes called a 'high speed stall' and gives rise to the first of the control difficulties encountered in flying an aircraft at or above M_{crit}.

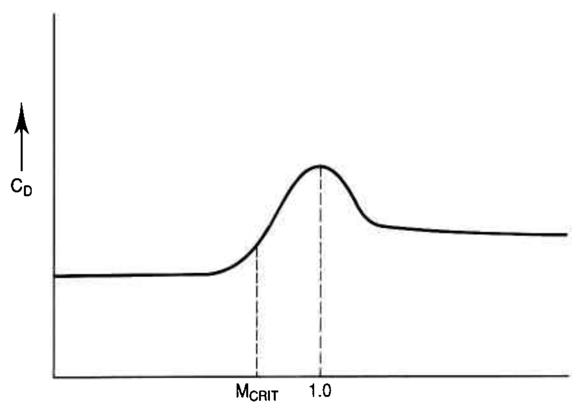


Figure 16-4 Variation of C_D with Mach No at Constant Angle of Attack

Wave drag arises from two sources, viz: energy drag and boundary layer separation.

Energy drag stems from the nature of changes occurring as a flow crosses a shock wave. Energy lost due to temperature rise across the shock wave becomes drag on the aerofoil. The more oblique the shock waves, the less energy they absorb, but since they become more extensive laterally and affect more air, energy drag rises as M_{FS} increases.

Boundary layer separation; at certain stages of shock wave movement there is considerable flow separation (Fig 16-3). The turbulence represents lost energy and contributes to the drag. As M_{FS} increases through the transonic range the shock waves move to the trailing edge and separation decreases, thus drag decreases.

The total effect on drag is shown in Fig 16-4 (C_D broken line), the hump in the curve being caused by drag associated with the trailing edge shocks arising from energy loss, separation of the boundary layer and the formation of the bow shock wave above M 1.0.

16.7 Reduction of Wave Drag

To reduce the effect of wave drag, shock waves 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 so producing a weak bow shock wave. The thin wing will have a reduced camber, thus the adverse pressure gradient across the wing shock waves will be smaller, and the strength of the shock waves will be reduced. Fuselages may be treated in a similar manner, for a given minimum cross section, an increase in length (within reason) will reduce wave drag.

16.8 Effects of Compressibility on Lift

To consider this aspect it is necessary to start at a speed where compressibility effects become significant and see how they vary with increasing Mach No.

(a) *Subsonic Rise in C_L* An increase in velocity is always accompanied by a decrease in pressure, and since the velocity increase in a compressible flow is greater than that in an incompressible flow for the same wing, the pressure will be lower, thus lift is greater for a wing in a compressible flow. At low speed, where air can be considered incompressible, lift is proportional to V^2, ie C_L can be assumed constant for the same angle of attack. At moderately high speeds density changes become significant, lift increases at a rate higher than indicated by V^2, ie C_L increases for the same angle of attack.

Another factor affecting C_L is the amount of warning the air gets of the wing's approach. As speed increases compressibility effects increase and the reduced upstream warning causes flow displacement to start closer to the wing. This effectively increases the angle of attack, so increasing C_L. There is a slight loss of lift due to movement of the stagnation point forward, but overall there is an increase in C_L.

(b) *Transonic Variations in C_L* In considering this aspect, five significant speeds are selected, A, B, C, D and E in Fig 16-5, and are used in Fig 16-6.

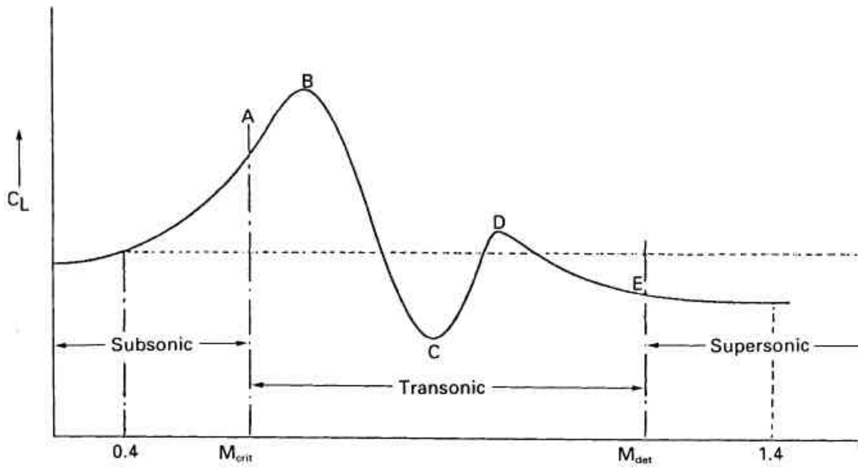


Figure 16-5

At A $M_{FS} = 0.75$, the flow accelerates rapidly from the stagnation point along both upper and lower surfaces, giving a sharp drop in pressure, and the wing is above M_{crit}. Over the top surface of the wing, as yet there is no shock wave, and C_L has risen by 60% of its low speed value for the same angle of attack. Over the bottom surface flow is still subsonic.

At B $M_{FS} = 0.81$. With the acceleration to this speed, the shock wave has formed and is strengthened, and will be approximately 60% chord (Fig 16-3); there is no shock wave on the undersurface. Behind the shock wave on the rear part of the wing there is no real change in pressure differential between upper and lower surfaces; ahead of it and behind the 40% (approximately) chord, pressure differential has increased considerably due to supersonic acceleration up to the shock wave. This effectively increases the C_L to roughly double its incompressible value. It also causes the CP to move rearward to approximately 30% chord. Flow under the bottom surface becomes sonic.

At C $M_{FS} = 0.89$. A shock wave has formed on the undersurface and moved to the trailing edge, while the upper shock wave has remained virtually stationary. The reason for the differing behaviour is the effect each shock wave has on the boundary layer. Such an arrangement of shock waves leads to a pressure distribution such that the wing behind the upper shock wave is producing negative lift, which has to be subtracted from the positive lift producing area. Lift coefficient has dropped to approximately 30% below its incompressible value and centre of pressure moves forward to approximately 30% chord. The reason for the slope between B and C is the relative movement between upper and lower shock waves.

At D $M_{FS} = 0.98$. The top surface shock wave is forced to the trailing edge, the area of negative lift is replaced by the orthodox pressure differential. The C_L is approximately 10% above basic value, and the CP has moved rearward to approximately 45% chord; this movement of the CP is experienced by all aircraft going through the transonic range.

At E $M_{FS} = 1.4$. Above M 1.0 the bow shock wave forms, and at M 1.4 is almost attached to the leading edge. The whole of the wing is producing lift, and the CP is at approximately mid-chord position. The C_L is reduced to a value of 30% less than its incompressible value due to the stagnation point moving to the most forward point on the leading edge, and to the loss of pressure energy through the bow shock wave.

The shock wave positions for each station considered above are shown in Fig 16-6.

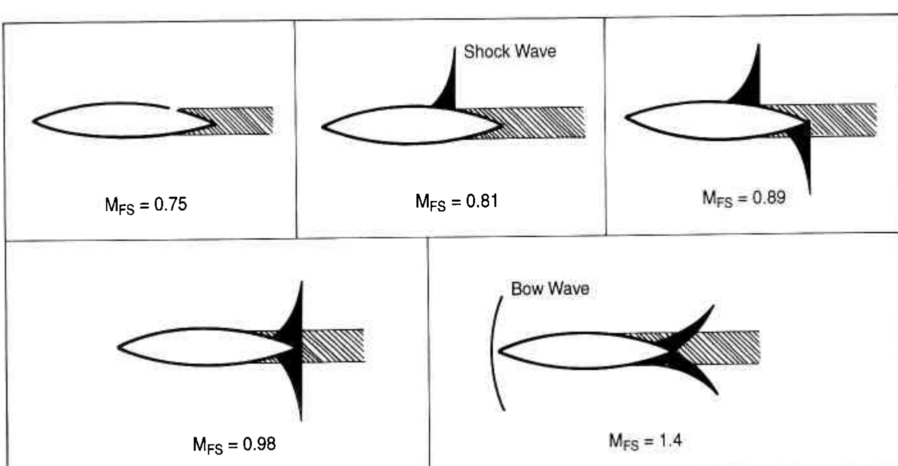


Figure 16-6

16.9 Supersonic Fall in C_L

The full explanation of this aspect is beyond the scope of these notes, but suffice it here to point out that in practice, any decrease in lift between the upper limit of the transonic range and M_{FS} 1.4 would be masked by trim changes resulting from passing through the transonic range. The practical result is that the lift curve slope becomes progressively more gentle with an increase in M_{FS} in the supersonic range. The variations in C_L at supersonic speeds depend mainly on attendant compressibility problems with increase of speed, and adverse pressure gradient increases with speed increase.

16.10 Effects of Increasing Mach No on Stability

Transonic Longitudinal Stability
Most aircraft operating in the transonic range experience a nose down pitch with speed increase, mainly due to two causes:

(a) Rearward movement of CP which increases longitudinal stability.

(b) Modification of airflow over the tailplane. The effect of mainplane shock waves is to modify the flow over the tailplane which will tend to pitch the aircraft nose down.

The effects on an aircraft's handling characteristics of nose down pitch are two-fold.

(i) At some Mach No an aircraft will become unstable with respect to speed, necessitating a rearward movement of the control column. This particular problem is dealt with more fully in Mach Trim.

(ii) The requirement for a large up deflection of elevator/tailplane reduces the amount of available control deflection for manoeuvres.

Supersonic Longitudinal Stability
The rearward movement of the CP in the transonic range continues as the aircraft accelerates into full supersonic flight. Thus all aircraft experience a marked increase in longitudinal stability.

Transonic Lateral Stability
Disturbances in the rolling plane are often experienced in transonic flight,

on some aircraft one wing starts to drop when M_{crit} is exceeded, due mainly to the difference in lift on the two wings because shock waves do not form at identical Mach numbers and positions on each wing.

Supersonic Lateral Stability

Lateral stability depends, after sideslip, on the lower wing developing lift. Since C_L decreases in supersonic flight the correcting force is thus reduced and dihedral and sweepback are consequently less effective. Another adverse effect is the lift/drag ratio decreasing due to surface friction drag, the decrease in lift/drag ratio being due to pressure differences between upper and lower surfaces combined with the pressures at the wing tips and their associated Mach cones.

Directional Stability

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. Subsonically, the fuselage side force in a sideslip acts in front of the CG and the vertical fin surfaces are able to overcome the destabilising condition. In supersonic flight the fuselage side force moves forward. As long as the aircraft is in balanced flight no problem arises, but if the relative airflow is off the longitudinal axis a destabilising force at the nose results. This is caused by asymmetry in the strength of the two shock waves producing a pressure gradient across the nose.

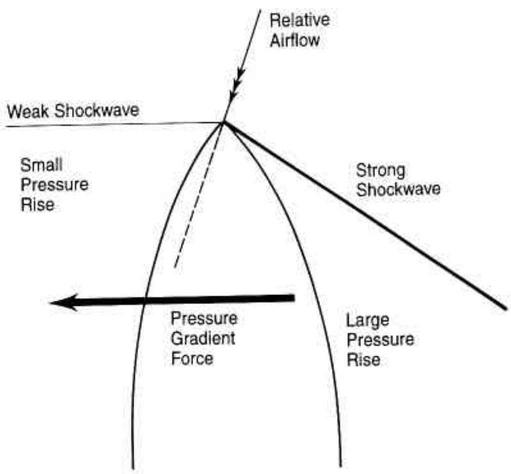


Figure 16-7

The nose force illustrated in Fig 16-7 is tending to prevent the nose being turned into the relative airflow and is therefore destabilising. The force increases with speed and has a longer arm than the fin and rudder. The point of application of the force is difficult to define, but is located at that part of the fuselage where the cross-sectional area is increasing.

One answer to this problem is to fit longer fins and increase their numbers, but there is a limit if only for wave drag considerations. A better method is the fitting of yaw dampers, which have already been dealt with.

Mach Trim

The device which corrects or compensates for longitudinal instability at high Mach numbers is the Mach Trimmer. As stated previously, at some Mach number an aircraft will become unstable with respect to speed; this is potentially dangerous since any inattention on the part of the pilot in allowing a small increase in Mach No will produce a nose down pitch, which will give further increase in Mach No, in turn leading to even greater nose down pitch. However, the Mach Trimmer will in fact correct or compensate for the initial increase in speed.

The Mach Trimmer is sensitive to Mach number and is programmed to feed into the elevator/stabiliser a signal which is proportional to Mach number so that stability remains positive. The signal fed into the elevator/stabiliser simply causes their deflection in a direction to compensate for the trim change.

Mach trim operation in normal conditions will not be shown up by the behaviour of the aircraft, but will usually be indicated by activation of the trim wheel and/or illumination of a monitor light.

Mach trim operation should be checked against Mach number for any significant change in flight condition.

Some Key Points So Far

1 Fixed Trim tabs are used to correct permanent out-of-trim faults and can only be adjusted on the ground.

2 Fixed Trim tabs should only be adjusted by an engineer.

3 A Balance tab is fitted to assist the pilot in moving the controls in flight.

4 A Servo tab is activated by movement of the control column which directly moves the tab which then aerodynamically moves the control surface.

5 On some supersonic aircraft longitudinal trim is achieved by moving fuel from one fuel tank to another. The tanks are positioned fore and

aft under the floor and by pumping fuel from one to the other the C of G is moved.

6 A high aspect ratio wing gives improved lift and reduced drag.

7 Employment of swept wings, or wings with swept leading edges, will delay M_{crit}.

8 For a given wing area at a given angle of attack a swept wing will produce less lift.

9 A swept wing is more prone to tip stall.

10 Spanwise movement of airflow over a swept wing may be reduced by:

Wing Fences.
Leading edge Notches.
Saw, or Dogtooth Leading edges.
Vortex Generators.
Wing Blowing.

Chapter 16: Test Yourself.

1 Most aircraft operating in the transonic speed range experience:

a) no pitch change.
b) a nose up pitch change.
c) a nose down pitch change.
d) none of the above.

Ref para 16.10

2 During the transonic speed range the:

a) C of P moves forward.
b) C of P does not move.
c) C of G moves aft.
d) C of P moves aft.

Ref para 16.10

3 Auto Mach Trim will primarily function:

a) at all speeds.
b) only at high subsonic speeds.
c) only at supersonic speeds.
d) within the transonic speed range.

Ref para 16.10

4 As an aircraft accelerates through transonic to supersonic flight:

a) longitudinal stability increases.
b) longitudinal and lateral stability increases.
c) longitudinal stability reduces.
d) longitudinal and lateral stability reduces.

Ref para 16.10

5 Wave drag arises from two sources:

a) interference drag and boundary layer separation.
b) energy drag and boundary layer separation.
c) energy and induced drag.
d) boundary layer separation only.

Ref para 16.6

17

Fundamental Manoeuvres and Their Affects, Trim and Engine Failure

17.1 Introduction

This chapter is intended to bring together those aspects of the principles of flight that are involved in some of the basic manoeuvres of an aircraft.

17.2 Lift

Example: Increased weight whilst maintaining level flight

An increase of weight will require an increase of lift to maintain level flight, which will normally be initiated by aft movement of the control column to produce an up deflection of the elevators.

The movement of the elevators UP will produce a down load on the tailplane, resulting in the longitudinal axis rotating about the lateral axis to increase the angle of attack.

The increase in angle of attack will result in the following:

(a) *Centre of Pressure*
Will move forward.

(b) *Transition Point*
Will move forward.

(c) *Boundary Layer*
Will become thicker.

(d) *Separation Point*
Will move forward

(e) *Stagnation Point*
Will move down and aft towards the underside of the wing.

(f) *Induced Drag*
Will increase as the angle of attack and the resultant lift increases.

(g) *Upwash and Downwash*
Will increase as the angle of attack is increased.

(h) *Power*
In order to maintain level flight with increased angle of attack and maintaining constant airspeed, then power must be increased to balance the increase in drag.

(i) *Power Required*
It may also be said Power Available reduces and Power Required increases.

(j) *Stalling*
Whilst the stalling angle will remain the same, due to the weight increase the stalling speed increases.

17.3 Lift Related to Camber

Whilst different cambers are used for wing sections to satisfy individual type requirements, a number of basic principles must be appreciated.

Example: High Camber wing at zero angle of attack:

(a) Will produce some lift and some drag

(b) High Camber wing sections will produce some lift and some drag even when at small negative angles of attack.

Example: Symmetrical wing sections:

(a) Will produce no lift and some drag at zero angle of attack.

(b) Must have a positive angle of attack to produce lift.

17.4 Yaw to Port (Conventional Fin and Keel Surface)

The following principles are applied to an aircraft when it is yawed to port.

(a) Left rudder pedal pushed forward causing rudder trailing edge to move to port.

(b) Some part of the leading edge of the rudder will move to starboard.

(c) This action will cause the aircraft to yaw about the normal or vertical axis to port.

(d) The airflow over the starboard wing is now at a greater velocity than that over the port wing and so more lift is being generated by the starboard wing than the port wing, the result being a roll to port.

(e) So a yaw to port will also result in a roll to port.

17.5 Yaw to Port (Large Fin and Keel Surface).

(a) Left rudder pedal pushed forward causing rudder trailing edge to move to port and rudder leading edge to starboard.

(b) Aircraft yaws to port about the normal or vertical axis.

(c) Action of rudder on a large fin causing a lift force of high magnitude to cause the fin to move about the longitudinal axis towards the right in a clockwise rotational movement when viewed from the rear, hence a roll to starboard.

(d) So it can be said on an aircraft with a large fin and keel surface when the aircraft is yawed to port it will tend to roll to starboard.

Note: It can therefore be said that an aircraft with a normal or conventional sized fin and keel surface is spirally stable in that when yawed to port it will readily roll to port and allow a spiral to be executed in a stable manner.

If, however, the aircraft has a large fin and keel surface, when yawed to port it will tend to roll to starboard and will resist a spiral to port by rolling out of it and so can be said to be spirally unstable.

17.6 Increase of Speed Whilst Maintaining Level Flight at a constant altitude

An increase of speed for a given angle of attack will result in an increase of lift and so in order to maintain a constant altitude:

(a) The angle of attack must be reduced by pushing the control column forward.

(b) The reduction in angle of attack will result in the Centre of Pressure moving aft.

(c) The induced drag reducing.

(d) The Transition Point and Separation Point moving aft.

(e) The Stagnation Point moving forward and up towards the leading edge.

(f) The Boundary Layer becoming thinner.

Note: Induced drag will reduce as the square of the speed.

Profile drag will increase as the square of the speed.

17.7 Stalling Angle

It must be noted that for a given wing shape the stalling angle will remain the same regardless of speed, weight, altitude or any other factor and can generally be regarded as being in the order of 14° to 15°.

17.8 Stalling Speed

Unlike the stalling angle the stalling speed is a variable quantity.

(a) The stalling speed will be increased if the C of P is forward of the neutral point.

(b) The stalling speed will be reduced if the C of P is aft of the neutral point.

(c) The stalling speed will be reduced if a power-on approach is made with a propeller-driven aircraft due to an increased thrust component from the propeller and the airflow tending to re-energise the wing boundary layer.

17.9 Multi-Engined Aircraft

Before describing the various procedures involved in multi-engined aircraft we are going to look at the aerodynamics of engine failure and asymmetric flight.

Under normal conditions of flight, thrust is provided in equal proportions to provide Total thrust which is opposed to Total drag, the two forces acting through the aircraft centreline. (Fig 17-1)
Consider that the right-hand engine fails. Immediately, Total thrust moves from the aircraft centreline to the thrust line of the left engine. Furthermore, the right hand propeller not only ceases to produce thrust but generates a considerable amount of drag until the propeller is feathered! With Total thrust moving to the left and Total drag moving to the

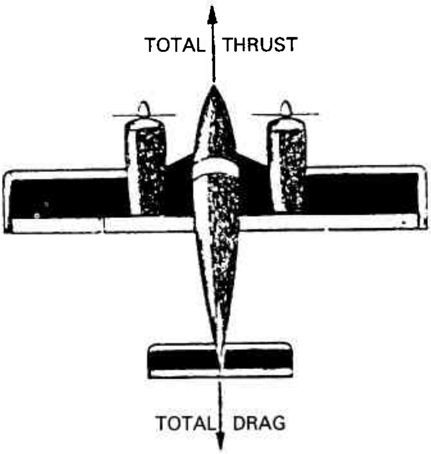


Figure 17-1 Total Thrust and Total Drag in Normal Flight

right, the opposing forces cause a yaw towards the failed engine. (Fig 17-2)

The events that follow if no corrective action is taken are as follows: the yaw produces a roll in the same direction (like further effects of rudder), and the aircraft nose will follow the down-going right wing tip into a spiral dive. It has a similar effect to putting a bootful of right rudder in and then leaving the aircraft to sort itself out without any help from other controls.

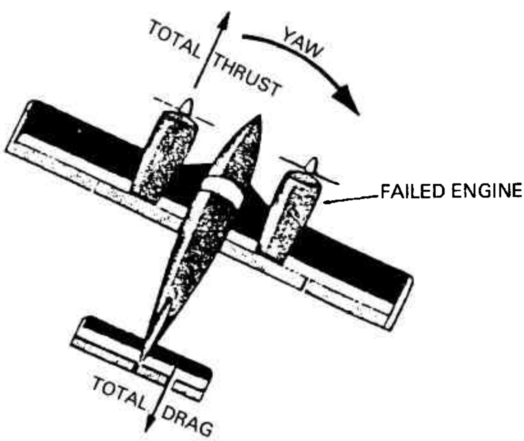


Figure 17-2 Total Thrust and Drag, Right Engine Failed

17.10 Minimum Control Speed

In a multi-engined aircraft the aerodynamic consequences of engine failure are dealt with by applying rudder to oppose yaw to prevent the yaw/roll/spiral dive sequence.

The rudder, however, like any other flying control, is only as effective as the airflow over it and herein lies a problem. If you let the speed drop too low the rudder will lose its effectiveness and will be incapable of combatting the yawing force of the live engine, aided and abetted by drag from the failed engine.

The minimum speed at which it is possible to maintain direction on one engine (known as minimum control speed), cannot be quoted as a single figure for any particular aircraft as it varies according to circumstances. The following are the primary factors that affect it:

1 *Altitude:*
Since more power means more asymmetric thrust (and therefore yawing action) it follows that minimum control speed will be at its highest at full throttle altitude where maximum power can be developed.

2 *Load:*
A fully loaded aircraft must, speed for speed, fly at a higher angle of attack than when nearly empty. A higher angle of attack means more drag and that in turn demands more power. So, back to square one; more power, more yaw, more yaw and, in consequence, a higher minimum control speed.

3 *Drag*:
This takes us back to Point 2. More drag means more power means more yaw, etc. Drag is mentioned here as a separate item to draw attention to the fact that flying with cooling flaps open and the landing gear extended will demand more power from the live engine – and will therefore mean an increase in minimum control speed.

4 *Flaps*:
Use of flap spoils the lift/drag ratio even though some flaps give very little drag increase until after the first 10 to 15 degrees of depression. As a guideline it is probably best to regard the flaps as coming under the heading of 'drag' and leave them up, unless the aircraft manual specifically advises otherwise.

5 *Windmilling*:
While some of the early light twins had fixed-pitch propellers, these days constant speed/feathering types are universal.

The drag from a windmilling propeller is very considerable however, and since it will provide a great deal of 'anti-thrust' (ie drag), minimum control speed will be appreciably higher than usual until the propeller is feathered. Remember that windmilling drag is asymmetric drag – and that is Poison.

6 *Pilot limitations:*
Modern aircraft are equipped with adequate trim, so the pilot should not be hampered by the physical limitation of being unable to apply sufficient rudder. Adequate experience and training in asymmetric procedures will ensure the ability to operate at low minimum control speeds.

7 *Critical Engine:*
When both propellers rotate in the same direction, slipstream and torque effects have a natural tendency to create yaw. In the case of modern piston-engine aircraft, where the propellers turn clockwise when seen from the rear, the yaw tendency is to the left.

Failure of an engine means loss of power and that, in turn, induces a drop in speed. To maintain height the angle of attack must be increased so that the aircraft is flying along in a tail-down/nose-up attitude.

In the tail-down attitude the propshafts are inclined upwards and the tops of the propeller discs are therefore tilted backwards. If you think about it, that means the down-going propeller blade (ie the one on the right of the disc when seen from astern during clockwise rotation) will have a bigger angle than the up-going blade on the other side. It is a curse well known to pilots taking off in a tailwheel aircraft, where, until the tail is raised, the propeller shaft is effectively tilted.

As illustrated in Fig 17-3, and keeping in mind that we are maintaining height on reduced power in a tail-down flight attitude: because the down-going blade has an increased angle, more thrust is being generated by the right half of the propeller disc than by the left. In effect, the centre of thrust for the port engine is moved towards the aircraft centreline whilst that for the starboard engine is moved away.

The amount of yawing force that can be generated by an engine depends on the amount of thrust and the moment arm, through which it acts. Since moment arm B is longer than moment arm A, the starboard engine will clearly exert more yawing force during engine-out flight than could the port engine.

Consequently, the greater yaw (and therefore the higher minimum control speed) would in this instance result from the loss of the port engine. In other words, when the propellers rotate clockwise, the critical engine is on the left.

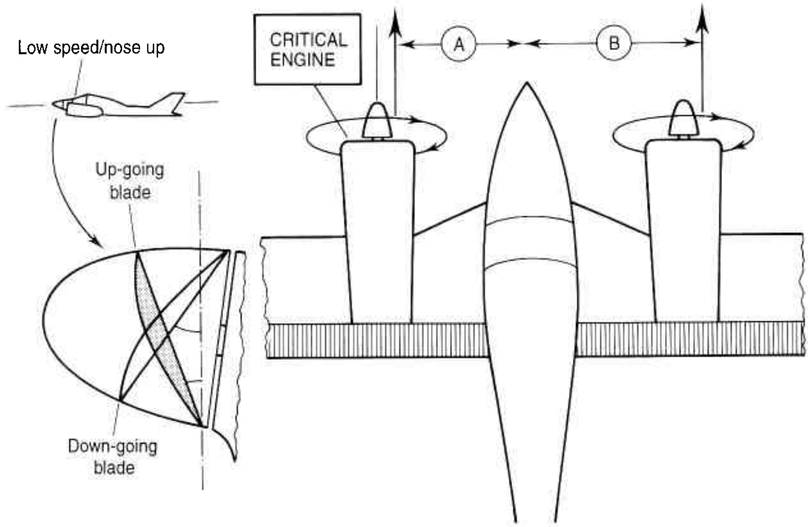


Figure 17-3

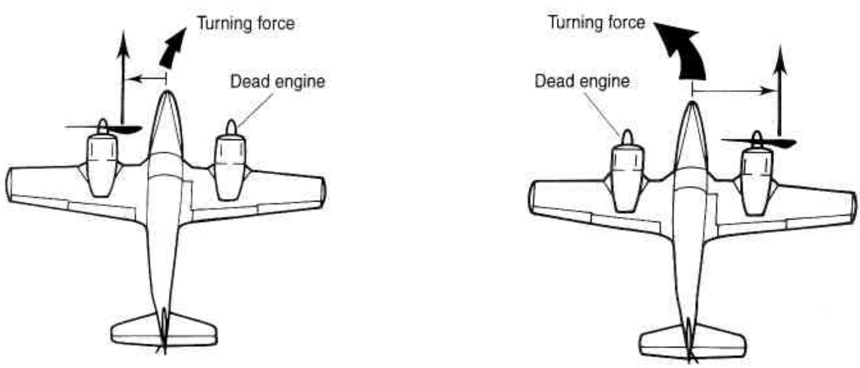


Figure 17-4 An exaggerated comparison of forces which shows that from a control standpoint the port engine is the worse one to lose

It is not always easy to demonstrate a meaningful difference in minimum control speed between the port and starboard engines, but much ado has been made of the subject. In any case, some popular light twins now have handed propellers, the left one turning clockwise and the right one anticlockwise, thus the minimum control speed is the same for both engines. The following is a list of 'V' codes relating to multi-engine aircraft handling:

V_1 *Decision speed during take-off*:
Up to that speed there should be enough runway for the aircraft to stop if, for any reason, you should decide to abandon the take-off. Beyond V_1 you are committed to press on and take off on one engine.

V_r *Rotate speed*:
At this stage the nose should be lifted to attain the take-off attitude.

V_2 *Take-off safety speed*:
This is, in fact, minimum control speed with an added safety margin to cater for the following factors which could apply if an engine fails during or immediately after take-off:
(a) Element of surprise
(b) Failure of the critical engine (ie the port)
(c) Landing gear down, flaps in take-off position, propeller windmilling
(d) Pilot of average strength and ability

Provided the aircraft has attained V_2, it should be possible to maintain direction and height while things are being sorted out.

V_{mcg} *Minimum Control Speed – Ground*:
Should an engine fail during the take-off run, this is the minimum speed at which direction can be maintained.

Some aircraft with good nosewheel steering can handle the situation at any speed provided the nosewheel is in contact with the ground.

V_{mca} *Minimum Control Speed – Air*:
This is the minimum speed at which it is possible to maintain direction after failure of the critical engine. No safety allowance is made for any of the items in V_2 so it is of little practical value other than for demonstration purposes while training multi-engine pilots.

V_{mcl} *Minimum Control Speed – Landing*:
This is the lowest speed at which it is possible to maintain direction when full power is applied following failure of the critical engine while in the landing configuration. This speed is important since it relates to the asymmetric overshoot.

V_{ne} *Never exceed speed*:
The ASI should be marked with a red radial line at this speed.

V_{no} *Normal Operating Speed*:
Also called 'maximum structural cruising speed', this is the top of the green arc marked on the ASI. Beyond this speed we enter the yellow or cautionary area which must be avoided while flying in turbulence.

V_{yse} *Speed for best engine-out rate of climb*:
This should be marked on the ASI as a blue radial line and is often referred to as the 'blue line' speed.

V_3 *The all engines screen speed*:
The speed at which the aeroplane is assumed to pass through the screen height with all engines operating on take-off.

V_4 *The all engines steady initial climb speed*:
The speed assumed for the first segment noise abatement take-off procedure.

V_{AT} *The target threshold speed*:
The scheduled speed at the threshold for landing in relatively favourable conditions.

V_{Tmax} *The maximum threshold speed*:
The speed above which there is an unacceptable risk of overrunning; normally assumed to be V_{AT} + 15 knots.

V_{MU} *The minimum demonstrated lift-off speed*:
The minimum speed at which it is possible to leave the ground (all engines) and climb out without undue hazard.

Additional 'V' codes relating to general aircraft handling:

V_S *Stall speed*:
The speed at which the aircraft exhibits those qualities accepted as defining the stall.

V_{MS} *The minimum speed in the stall*:
The minimum speed achieved in the stall manoeuvre.

V_{SO} *Stall speed in landing configuration*:

V_{MO} *Maximum operating speed*:
The maximum permitted speed for all operations.
M_{MO} – maximum Mach operating speed for all operations.

V_{DF} *Maximum demonstrated flight diving speed*:
The highest speed demonstrated during certification.
M_{DF} – highest Mach speed demonstrated during certification.

V_{RA} *The rough-air speed*:
The recommended speed for flight in turbulence.
M_{RA} – recommended Mach Number for flight in turbulence.

V_B *Design speed for maximum gust intensity*:
One of the parameters used in establishing V_{FA}

V_C *Design Cruising speed*:
One of the speeds used in establishing the strength of the aircraft.

V_D *Design Diving speed*:
Another of the speeds used in establishing the strength of the aircraft.

V_F *Flap Limiting speed*:
The maximum speed for flight with the flaps extended.

V_{IMD} *Speed for minimum drag*:

V_{IMP} *Speed for minimum power*:

V_{fe} *Maximum speed for flight with flaps extended*:

V_{fo} *Maximum speed for operating flaps*:

V_{le} *Maximum speed for flight with gear extended*:

V_{lo} *Maximum speed for operating gear*:

V_A *Design Manoeuvring speed*:

V_A *Maximum speed for full deflection of controls*:

V_x *Best angle of climb speed*:

V_y *Best rate of climb speed*:

Screen speed (and screen height):
The speed assumed at 35 feet above the runway after take-off and at 30 feet above the runway on approaching to land, which is used in establishing the field performance of the aeroplane.

Zero rate of climb speed:
The speed at which, for a given thrust from the operating engines, the drag of the aircraft reduces the climb gradient to zero.

Chapter 17: Test Yourself.

1 On a twin-engined aircraft, with clockwise rotating propellers (Right Handed), with reduced power and a tail down attitude the critical engine will be:

a) either port or starboard.
b) the port engine.
c) the starboard engine.

Ref para 17.9

2 V_{MO}

a) is the maximum operating speed.
b) is the flap limiting speed.
c) is the minimum power speed.
d) is the design manoeuvring speed.

Ref para 17.10

3 V_x is the:

a) zero rate of climb speed.
b) best rate of climb speed.
c) best angle of climb.
d) maximum speed for full deflection of controls.

Ref para 17.10

4 At a constant height an increase of aircraft weight requires:

a) an increase of power.
b) an increase of power available.
c) a reduction in angle of attack.
d) an increased stalling angle.

Ref para 17.2

5 With an increase in angle of attack of an aerofoil:

a) the C of P moves aft.
b) the C of G moves aft.
c) the separation point moves aft.
d) the stagnation point moves aft.

Ref para 17.2

18

Duplicate Inspections of Controls

18.1 Pilot Responsibility

A pilot is authorised to carry out the second part of a duplicate inspection of an aircraft's control systems if:

(a) he is licensed on that type of aircraft.

(b) the control system has only had a *minor adjustment* made to it.

(c) there is no licensed engineer available.

This is intended to cover minor adjustments to control systems made on light aircraft away from base.

British Civil Airworthiness Requirements Chapter A5 – 3 refers to this authorisation.

The following paragraphs are based on BCAR Chapter A5 – 3 and Civil Aircraft Inspection Procedures.

18.2 Control System Definition

A control system is defined as a system by which the flight attitude or the propulsive force of an aircraft is changed. A duplicate inspection is therefore required for the following:

(a) Flying control systems which include primary flying controls (elevator, rudder and ailerons), together with tabs, flaps, airbrakes and the mechanisms used by the pilot to operate them.

(b) Propulsive system controls, which include primary engine controls and related systems (eg throttle controls, fuel cock controls, oil cooler controls) and the mechanisms used by the crew to operate them.

18.3 Duplicate Inspection of Control Systems

1 A duplicate inspection of a control system is defined as an inspection which is first made and certified by one qualified person and subsequently made and certified by a second qualified person. Components or systems subject to duplicate inspection must not be disturbed or readjusted between the first and second parts of the inspection, and the second part of the inspection must, as near as possible, follow immediately after the first part.

2 In some circumstances, due to peculiarities of assembly or accessibility, it may be necessary for both parts of the inspection to be made simultaneously.

3 A duplicate inspection of the control system in the aircraft shall be made:

(a) before the first flight of all aircraft after initial assembly.

(b) before the first flight after the overhaul, replacement, repair, adjustment or modification of the system.

The two parts of the duplicate inspection shall be the final operations, and as the purpose of the inspection is to establish the integrity of the system, all work should have been completed. If, after the duplicate inspection has been completed, the control system is disturbed in any way before the first flight, that part of the system which has been disturbed shall be inspected in duplicate before the aircraft flies.

4 The correct functioning of control systems is at all times of vital importance to airworthiness, and it is essential that suitable licensed aircraft engineers and members of approved inspection organisations responsible for the inspection or duplicate inspection should be thoroughly conversant with the systems concerned. The inspection must be carried out systematically to ensure that each and every part of the system is correctly assembled, and is able to operate freely over the specified range of movement without risk of fouling. Also that it is correctly and adequately locked, clean and correctly lubricated, and is working in the correct sense in relation to the movement of the control by the crew.

18.4 Persons Authorised to Certify Duplicate Inspections

Personnel authorised to make the first and second parts of the duplicate inspection of control systems in accordance with Chapter A5 – 3 of BCAR are as follows:

(a) Aircraft engineers appropriately licensed in Categories A, B, C and D.

(b) Members of an appropriately approved Inspection Organisation who are considered by the Chief Inspector competent to make such inspections.

(c) For minor adjustments to control systems when the aircraft is away from base, the second part of the duplicate inspection may be performed by a pilot or flight engineer licensed for the type of aircraft concerned.

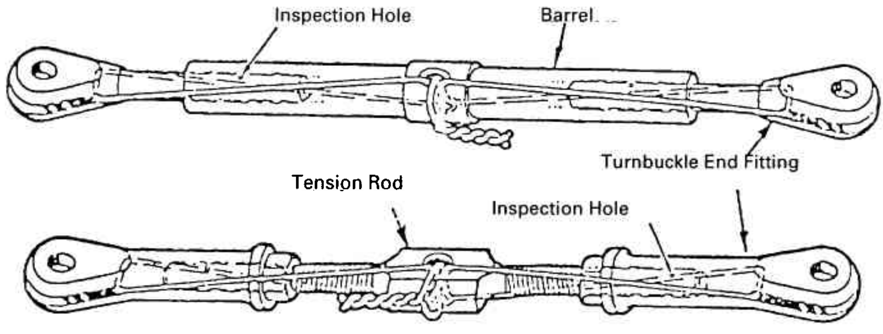


Figure 18-1

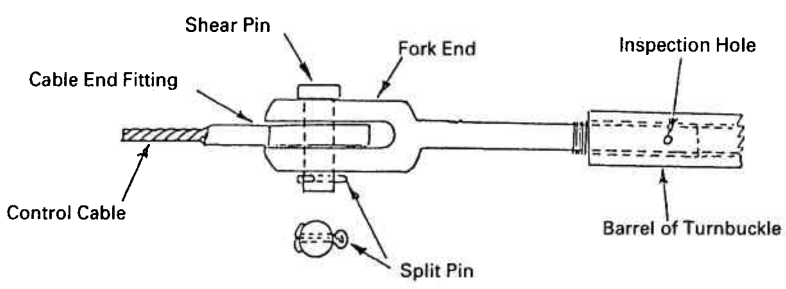


Figure 18-2 (a)

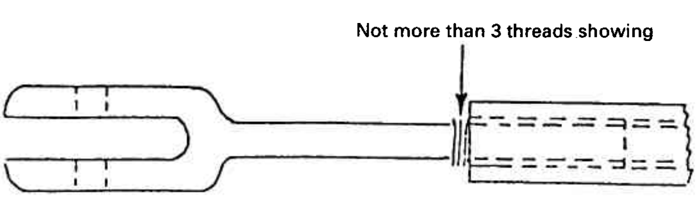


Figure 18-2 (b)

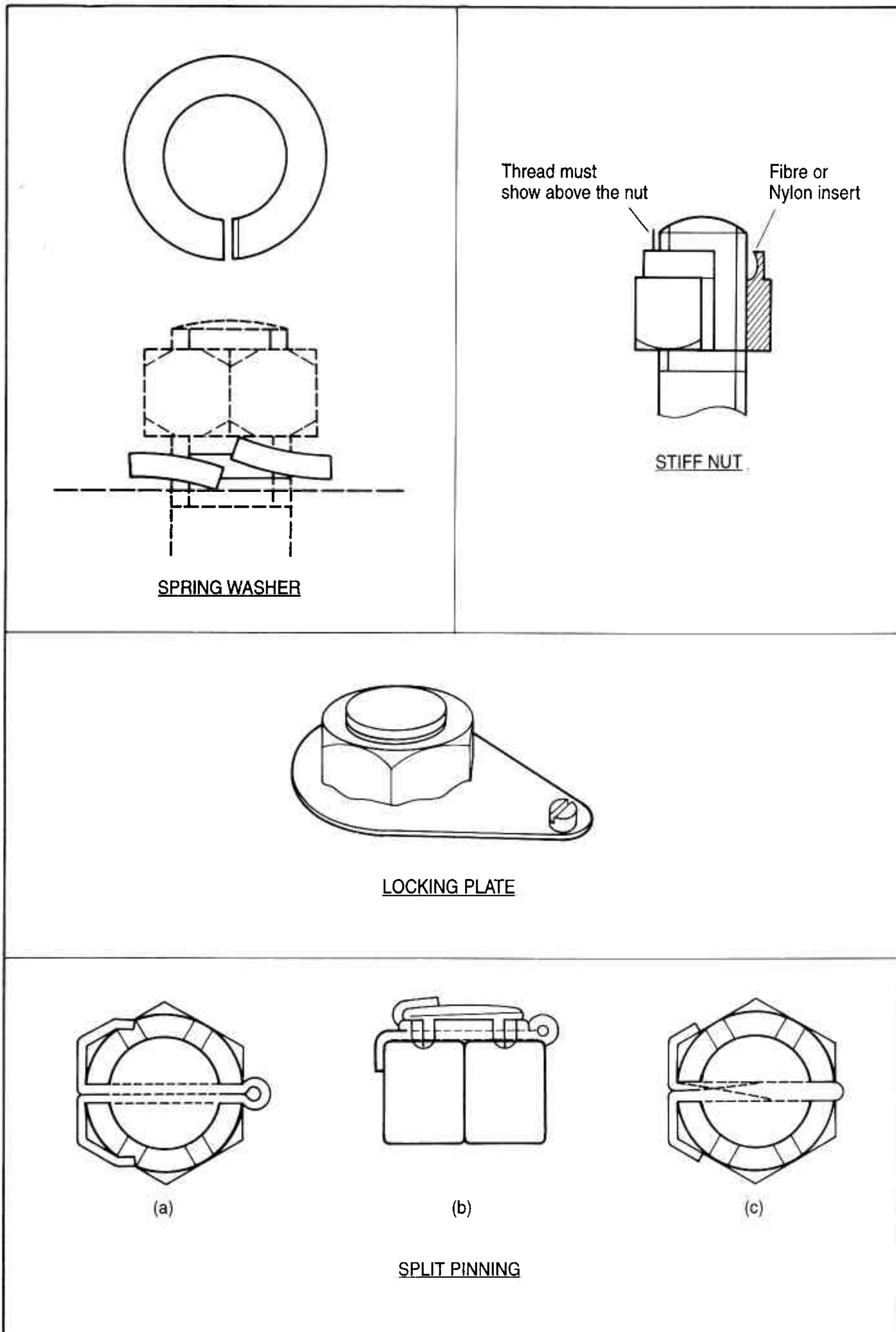

Thread must show above the nut
Fibre or Nylon insert
STIFF NUT
SPRING WASHER
LOCKING PLATE
(a)
(b)
(c)
SPLIT PINNING

Figure 18-3

18.5 Flying Control Systems

General Points
Movement of the pilot's flying controls can be transmitted to the aircraft control surfaces by a system of flexible steel cables. Although one of the oldest methods of control, it is still used extensively today. An alternative type of control system is formed from light alloy tubes. These tubes form a rigid link system that also gives positive control under all flight conditions. Both methods will be considered in some detail in this chapter. Where it is necessary to change the direction of a control run, whilst maintaining a strong, flexible and positive connection, a sprocket and chain system may be fitted at appropriate points in the control run. Since all flying control systems start in the cockpit, we shall begin this discussion by looking at controls which the pilot operates in order to fly the aircraft.

Control Column
The control column is the most important single control that the pilot has to operate. Its movement controls both the ailerons and the elevators. The dual controls illustrated in Fig 18-4 are interconnected so that movement of one has exactly the same effect as movement of the other. We shall, therefore, consider the movement of only one of them. As illustrated in Fig 18-4, the control column is pivoted at a mid-position to allow sideways movement about that point. The bottom of the unit is attached to the aileron control system so that sideways movement of the control column will move the ailerons. The control column is pivoted on the angled crank of a torque tube which also carries a lever arm to which the elevator control system is attached. Moving the control column fore and aft rotates the torque tube, thus moving the elevators.

Rudder Bar and Pedals
The rudder is controlled from the cockpit by moving the rudder pedals. These pedals may be separate units or are attached to a rudder bar. Moving the rudder pedals operates a mechanical system to which the rudder is attached. Putting the left foot forward swings the rudder to port; conversely, putting the right foot forward swings the rudder to starboard. The rudder bar (or individual rudder pedals) can be adjusted to suit the leg reach of the pilot.

Trimming Tab Controls
We have seen that the primary control surfaces (ailerons, elevators, rudders) are moved by operating the pilot's controls. This may be as a direct result of physical effort on the part of the pilot, or it may be as a result of a signal fed from the pilot's controls to a powered flying control

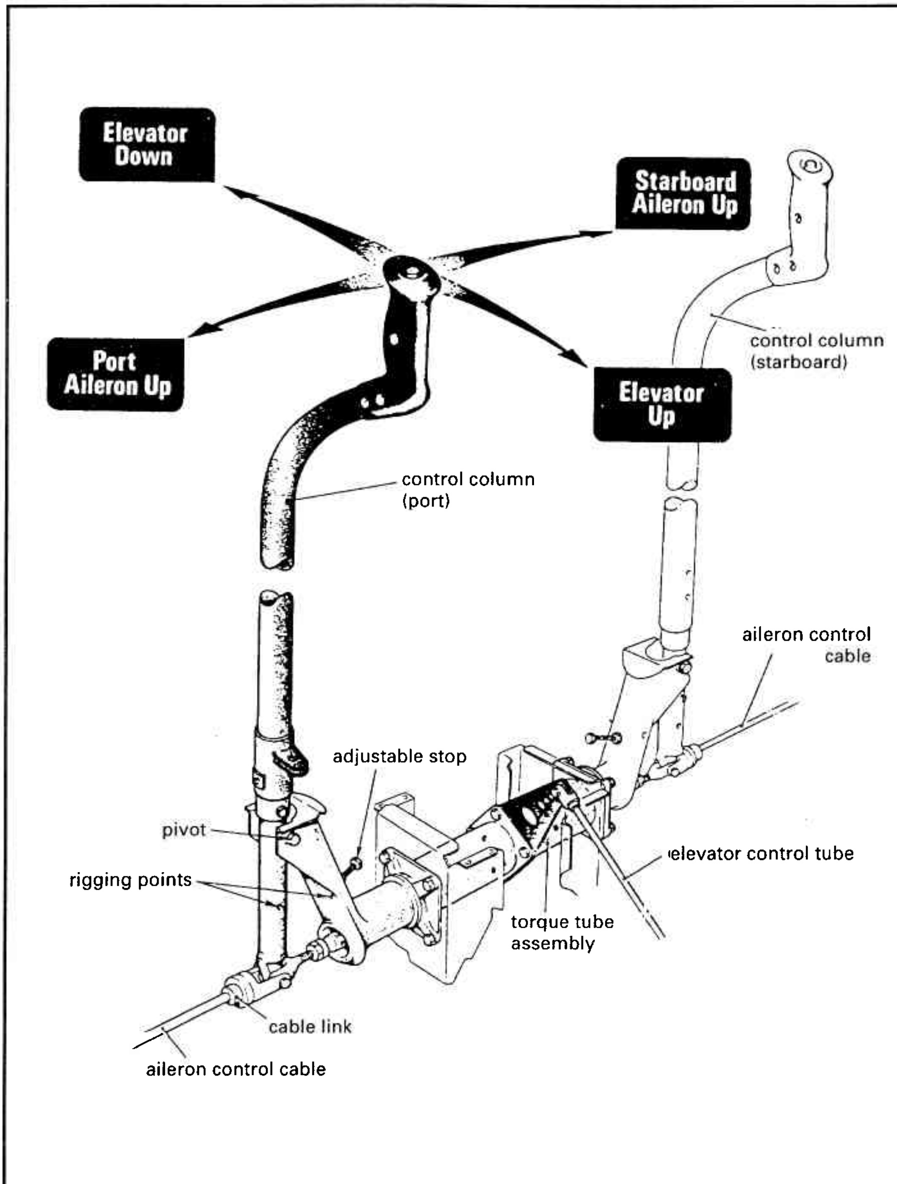


Figure 18-4 Dual Control Columns

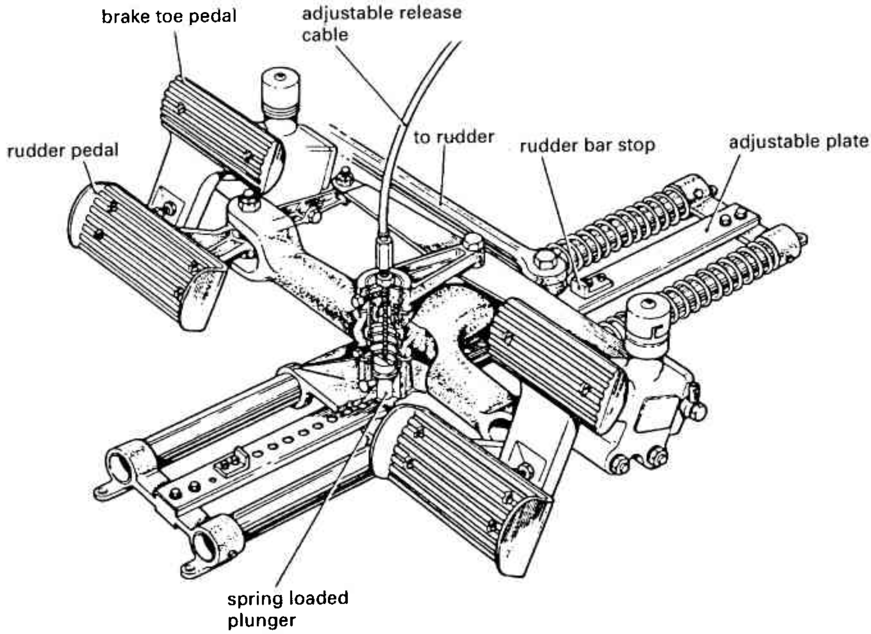


Figure 18-5 Rudder Bar and Pedals

unit. In the first instance – ie with no power assistance – the control surfaces normally have small trimming tabs fitted to them, as described in previous chapters. These trimming tabs are controlled from the cockpit, as illustrated in Figure 18-6.

An aircraft is said to be trimmed when there is no load on the control column or rudder bar/pedals, and the aircraft is flying steadily without any change in direction or altitude. The controllable trimming tabs ensure that, for any unwanted variation from the trimmed condition, the appropriate control surface(s) may be re-trimmed to remove the loading on the pilot's control. Trimming tabs, where fitted, are adjusted frequently during each flight.

Methods of Operating Control Systems
Each control system in an aircraft is constructed for the particular job it has to do. Consequently, there are considerable differences in the design of control systems – not only between those in different aircraft, but between different systems in the same aircraft. We cannot deal with all the variations in a book of this type. We can however, deal with the common areas in control systems.

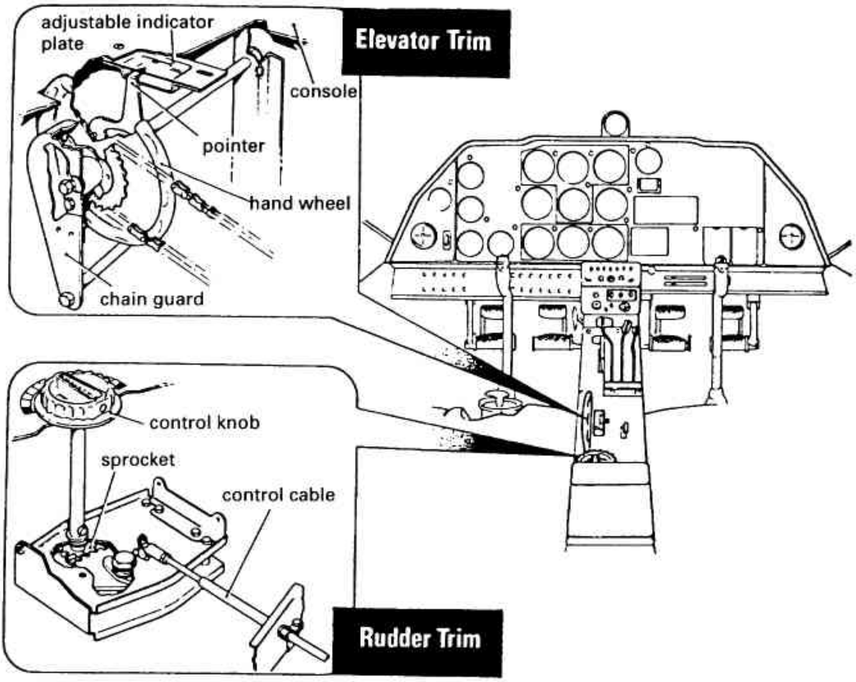


Figure 18-6 Trimming Tab Controls

As already stated, flying controls are normally operated by cables or by control tubes, and each method will now be considered.

18.6 Control Cables

General points

Cables provide a strong, light and flexible method of control and are used extensively in aircraft control systems. Cables operate in tension and can, therefore, only be used to pull the control. However, two cables can be arranged in the form of a continuous loop to provide a pull in both directions (Fig 18-7).

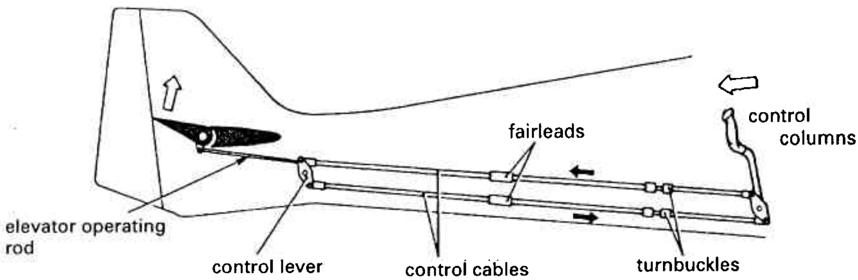


Figure 18-7 Cable System

Construction of Cables

Flying control cables are normally preformed; that is, the strands in the cable are formed into the shape they will assume in the complete cable. The cables, which are made of galvanised or corrosion-resistant steel, are impregnated with an anti-friction lubricant during manufacture.

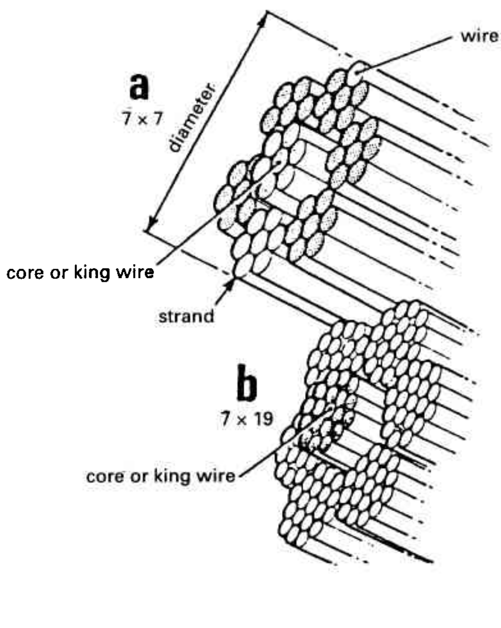


Figure 18-8 Construction of Cables

A cable is made up of steel wires which, in turn, are formed into strands, as illustrated in the two examples of Fig 18-8. Each strand consists of several wires (7 or 19) which are wound helically in one or more layers, the centre wire being known as the core wire or king wire. Each cable is made up of several strands (usually 7), wound helically around the centre or core strand. The cable is described by the number of strands it contains and by the number of individual wires in each strand. Figure 18-8a shows that a 7×7 cable consists of 7 strands, each having 7 wires; Figure 18-8b shows a 7×19 cable – 7 strands, each having 19 wires. The number of wires in each strand, the number of strands, and the overall diameter of the cable determine the breaking load of the cable. For example, a 7×19 cable of 6.4mm (¼in) overall diameter has a minimum breaking load of 7000lbf. Cables are classified either by the minimum breaking load, which may be quoted in cwtf, lbf or kN, or by the nominal diameter in inches.

It is often necessary to coil a cable when handling it for assembly into an aircraft. The coil should be of large diameter; never less than 50 diameters of the cable involved and with a minimum diameter of 150mm. To avoid kinking the cable, and thus making it unserviceable, uncoiling should be done by rotating the coil so that the cable is paid out in a straight line.

Pulleys

Pulleys are used to change the direction of operation of flying control cables, and to give support on long straight runs. A cable guide (or retainer) is fitted to the pulley to ensure that the cable remains on the pulley. A typical pulley, with its retainer, is illustrated in Fig 18-9. When adjusting a control, it is important to ensure that the cable end fittings do not foul the pulley, otherwise the cable movement will be restricted. Also look for possible misalignment between the cable and pulley: this must not exceed 2° (Fig 18-9b).

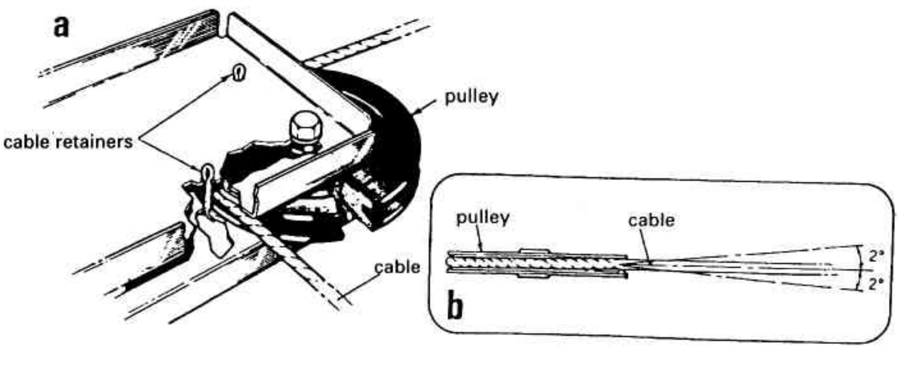


Figure 18-9 Pulley

Screwjack
A cable-operated trimming tab control system usually operates a screwjack at the output end of the system. The screwjack (Fig 18-10) is attached by means of an adjustable rod to the trimming tab. The cable movement rotates the sprocket of the screwjack to reposition the trimming tab. This unit acts as a lock, retaining the trimming tab in the desired position until the cockpit control is next moved.

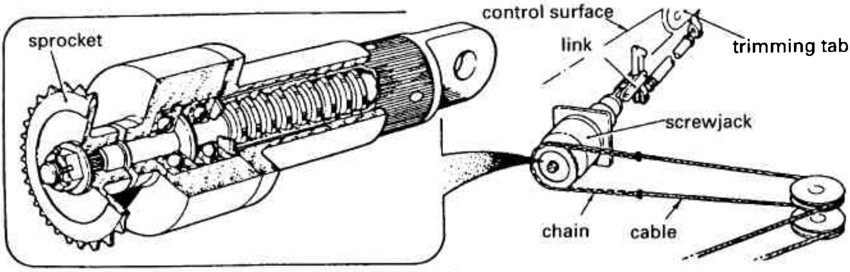


Figure 18-10 Screwjack

18.7 Cable Tensioning

Need for tension. For a wire cable control system to operate effectively, the cable tension must be correct. It should be just sufficient to operate the control – neither too taut nor too slack; excessive tension imposes an unnecessary load on the control system, whilst a slack cable results in ineffective response. We shall see later that cable systems are tensioned to a pre-determined value, in accordance with the servicing instructions for the particular system. The value chosen is such that sufficient tension is maintained over a range of operating temperatures. The range of temperature over which the tension remains satisfactory depends upon whether or not a cable tension regulator is fitted in the system (see later).

Temperature change, cable stretch, and general wear of supporting parts affect the tension which must, therefore, be checked and adjusted as necessary at specified intervals. Some cable systems have compensating devices fitted which ensure effective operation over a much wider range of temperatures than would otherwise be possible.

Turnbuckles
It is normal to use turnbuckles to adjust the tension of cables in flying control systems. There are two types of turnbuckles in common use (Fig 18-11) and the type fitted will match the end fittings on the cables.

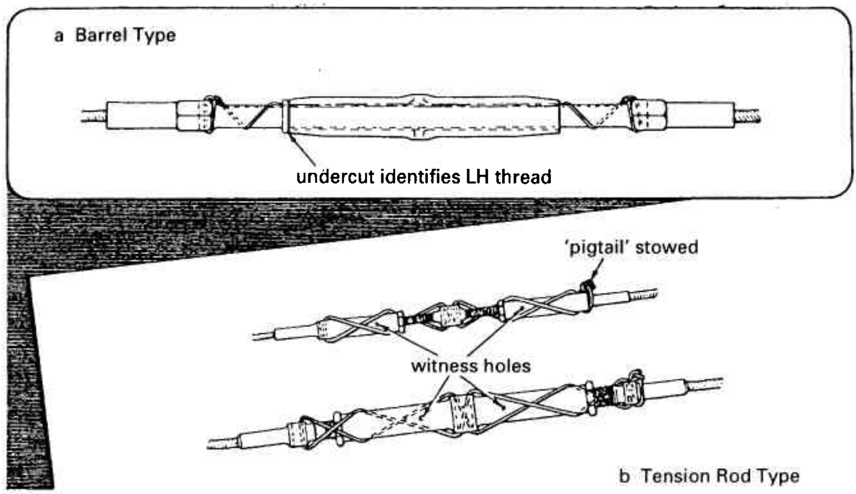


Figure 18-11

When connecting cables together using a turnbuckle the threads must be evenly engaged at each end. It is important to ensure that sufficient threads are engaged, otherwise the load on the cable could strip the threads.

With the American type of turnbuckle (Fig 18-11a) not more than three threads should be visible.

Cable end fittings that engage with the British tension type of turnbuckle (Fig 18-11b) have small 'witness' holes drilled in their shanks. The turnbuckle thread must at least reach these holes for the connections to be 'in safety'.

All turnbuckles are locked in the approved manner using locking wires or clips, as shown in Fig 18-1 and 2. The British type is also locked with locking nuts.

Adjusting the tension in a cable system

There are many different types of metal in an aircraft, each of which expands at a different rate with increasing temperature. The effect of this in a cable system is that the tension tends to decrease with an increase in altitude. Thus, to retain sufficient tension at altitude, the pre-determined load must be high. This requires a strong structure, with a resulting increase in weight. Furthermore, compared with a tension regulated system (see later), stress and static friction are also higher.

While tensioning is being carried out to the correct value of pre-determined load by evenly adjusting all the turnbuckles in the system, the correct relative positions of the pilot's control and the relevant control surface must be maintained. The cable tension is checked frequently using a tensiometer as the adjustments are made.

Cable Tension Regulator
A cable tension regulator is a mechanical device which, when fitted in a cable system, allows the cables under all conditions of temperature change and structural deflections to take up and let out equally on each side of the circuit, thus maintaining uniform tension. The compensating unit of a tension regulator may be manufactured with either one or two springs; a double spring unit is described below and illustrated in Fig 18-12.

This type of regulator consists of a pair of spring-loaded quadrants, with a pointer and scale for recording the change in length of the cables. The cables are inserted through slots in the recessed end of the grooved quadrants and the cable ends are secured at the anchorage points. The basic purpose of the regulator is to keep the effective length of the cable constant even when the actual length has been increased or decreased either by change of temperature or structural flexing. The graph is used, in conjunction with the regulator scale reading, when assessing the cable tension (see later). Let us see how the regulator functions.

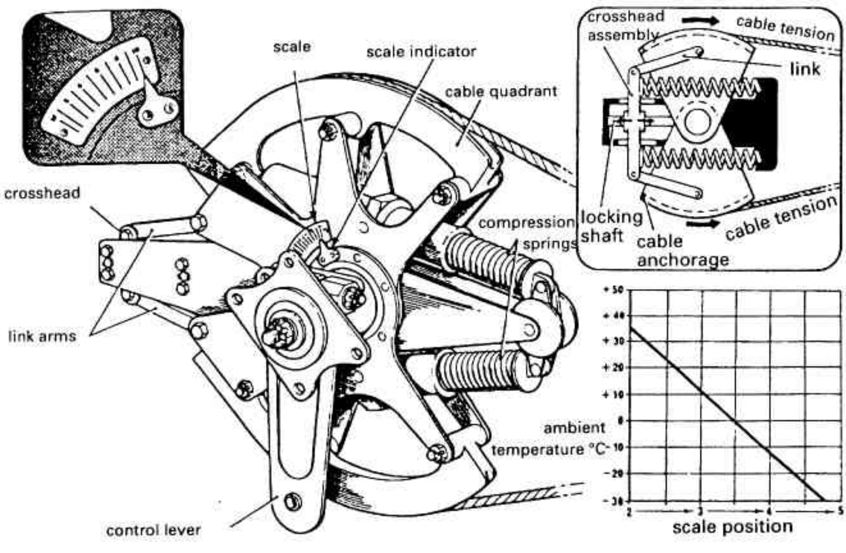


Figure 18-12 Cable Tension Regulator

Figure 18-13a shows that any extension of the cables attached to the quadrant tends to give equal slackening of the cables. The regulator springs then impart a rotary displacement to the cable quadrants, and this movement is transmitted by the link arms to the crosshead, causing it to move freely along the locking shaft. This movement is controlled by the pressure of compression springs to govern the cables at their correct pre-set tension.

Figure 18-13b shows that any shortening of the cables will have the reverse effect, tending to give equal tensioning of the cables. This gives a rotary displacement of the quadrants and moves the crosshead inwards along the locking shaft by the action of the link arms.

Figure 18-13c shows that when the pilot operates a 'regulated' control, the crosshead tilts on its locking shaft, causing it to lock on to the shaft. Both quadrants are now locked together and operate as a lever to give the pilot positive control of the system.

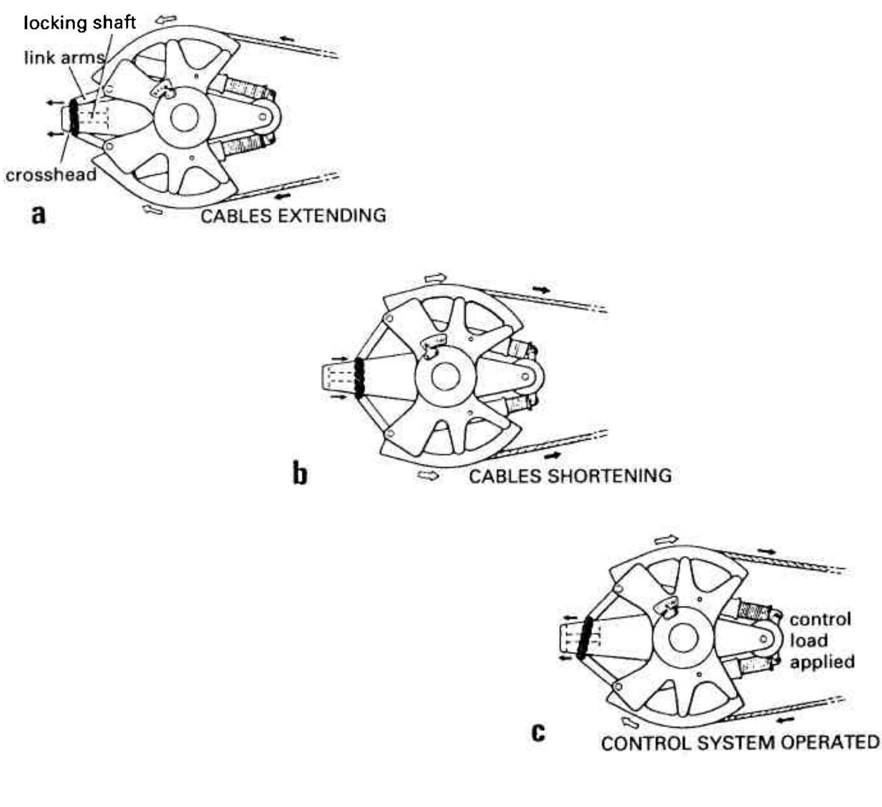


Figure 18-13 Cable Tension Regulator

A tensiometer is not used to check the tension in a regulated system. The cable tension is adjusted on both sides of the circuit (usually by means of turnbuckles) until the correct reading is obtained on the regulator scale. The reading obtained depends upon the prevailing temperature, and the correct reading for that temperature is obtained from a special graph printed in the relevant manual (Fig 18-12).

After the cables have been set to the correct tension, regulator compensation may be checked by grasping both cables near their point of entry to the regulator and forcing both cables in towards each other. The resulting movement of the quadrants should be smooth and even. If the regulator fails to move, or the movement is jumpy, it may indicate that the cables have been wrongly rigged so that the tension is uneven, causing the crosshead to tilt and 'lock' the system.

Compared with a cable system, a regulated cable control system ensures a relatively constant tension. Because of this, the level of tension can be reduced; this in turn provides:

*Lower static friction *Less structural weight *Improved response

18.8 Mechanical Stops

The next check is to ensure that the control surface moves to its designed maximum travel position, in both directions, when moved by the cockpit control.

The maximum travel of a primary control surface is limited in each direction by mechanical (limit) stops. These stops (Fig 18-14) are fitted to limit the control surface movement in either direction and thus avoid damage due to excessive travel. In a manual system, the stops are usually located near the control surface, and a second pair of stops, known as the override stops or secondary stops, are fitted to limit the pilot's control movement should the main stop fail. Secondary stops are adjusted to a specified clearance under normal operating conditions. In powered

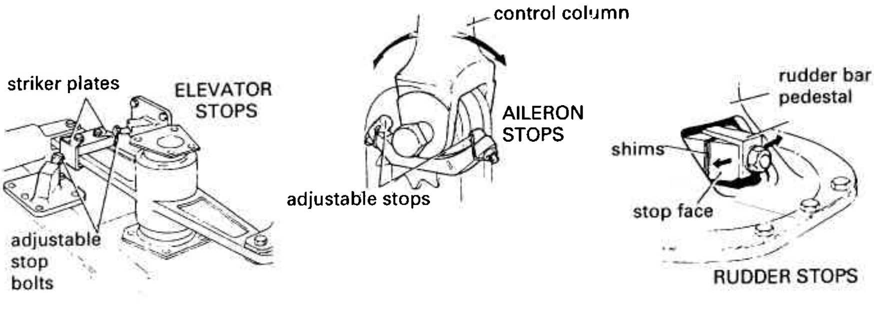


Figure 18-14 Mechanical Stops

control systems, the mechanical stops are located on the input side of the powered flying control unit (PFCU); usually they are located next to the pilot's control in the cockpit, thus limiting the control system movement from that position. During the rigging procedure, the main mechanical (primary) stops may need to be re-set to ensure that the control surface reaches, but does not exceed, its maximum travel position.

Chapter 18: Test Yourself.

1 An American type turnbuckle is in safety when:

a) it is wire locked.
b) not more than three threads are showing.
c) the inspection holes are obscured.
d) the lock nuts are tight.

Ref para 18.7

2 Primary control stops are located:

a) at the control surface.
b) at the control column.
c) at any convenient position in the control run.
d) at the mixer unit.

Ref para 18.8

3 Automatic cable tension is provided by:

a) turnbuckles.
b) control stops.
c) tension regulators.
d) pulleys.

Ref para 18.7

4 If a control system cable tension is too high:

a) control surface range of movement will be reduced.
b) controls will be easier to move.
c) excessive wear will take place on cables and pulleys.
d) flutter is more likely to occur.

Ref para 18.7

5 When a pilot carries out a duplicate inspection:

a) he or she must sign the first signature block.
b) he or she must be an ATPL holder.
c) he or she must also be a type rated engineer.
d) no other engineer must be available.

Ref para 18.4

19

Aircraft Construction

19.1 Airframe Structural Design

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.

19.2 Definitions

To avoid misconceptions of the engineering terms used in this chapter a list of definitions is given below:

Stress:	The force exerted between two contacting bodies or parts of a body. It is measured as the load per unit area.
Strain:	The deformation caused by stress. It is recorded as the change of size over the original size.
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.
Stiffness or *Rigidity*:	The ratio of stress over strain.
Ultimate Strength:	That point beyond which, if stress is increased, the material will fail.

19.3 Designing a new Aircraft

The Specification

The initial step in the production of a new aircraft is the preparation of a specification. This will state the required performance in terms of speed, range, ceiling and payload etc. The general conditions in which the aircraft is required to operate are also stated, such as type of runway surface, temperature conditions and altitude. The strength and length of available runways must also be taken into account so that new aircraft may use existing runways, and in most cases may operate worldwide. The specification will also give the strength requirements and will stipulate a factor of safety to allow for unforseen contingencies or for accidentally exceeding basic design limitations. If the strength requirements are too severe, the aircraft will be penalized by excessive structural weight; if not severe enough, there is a risk of failure of structural components. In choosing these strength conditions, the aim must be to ensure that the aircraft will be able to carry out all the normal manoeuvres appropriate to its role. It is desirable to make the airframe as strong as possible, but the extra structural weight needed to ensure adequate strength for certain manoeuvres must not be out of proportion to the advantage gained. Manoeuvres for which the aircraft is not stressed are called prohibited manoeuvres.

19.4 The Design

When the specification has been produced, the design team will decide what they consider to be the best form and size of aircraft to meet the requirements. At this stage it must be emphasized that any aircraft design is inevitably a compromise between the conflicting demands of payload requirements, performance, economy, reliability, cost, ease of maintenance and so on. The designer's own preference will influence the ultimate layout of the airframe; this explains why there are so many different shapes and layouts for aircraft, although each is more or less correct for its task. The requirements for speed and range are usually so dominant that only an aerodynamically clean monoplane design can be used, and design effort is concentrated on achieving the minimum drag by careful positioning of wings, fuselage, tail unit, and engines, and by the cleanest possible stowage of radar aerials, etc. Still further reductions in drag are possible by eliminating the tail unit, and even the fuselage, by using a flying wing design; but the problems of longitudinal control and stability are then difficult to solve satisfactorily. In every heavy or high speed design, much care must be taken to provide controls which require the least practicable force to operate them and which are effective throughout the speed range.

The next stage in the development of a new aircraft is usually wind-tunnel testing. Work starts in the wind-tunnel on models of the selected design to check the exact outline of the aircraft, the lift and drag, to work out maximum air loads that will be exerted on all surfaces under all possible flight conditions, together with the performance, stability and control of the aircraft, so that errors can be detected at an early stage and the design modified as necessary.

During the wind-tunnel test stage a full-size wooden mock-up is often made so that positions may be found for models of all the equipment to be carried in it. The mock-up is also useful to check the field of view, to ensure sufficient space and headroom for flightcrew and passengers, and to check the position of controls in relation to the instrument panel, etc. In general, the mock-up works as a rough 3-D check on all the dimensions made by the design staff and draughtsmen.

A detailed design of the airframe structure is then begun. Probably the most important part is the calculation of the strength of the aircraft. The airframe has to be sufficiently strong to withstand aerodynamic, landing and handling loads. The aerodynamic loads are calculated from wind-tunnel experiments for the acceleration and speeds in the specification and multiplied by the factor of safety. Loads imposed by manhandling on the ground, especially on light aircraft, are often many times greater than the aerodynamic loads and must be allowed for if the aircraft is not to be covered with 'Do not push here' and 'No step' signs.

Other features the detailed design must include are:

(a) A smooth skin of the required aerodynamic form.

(b) Sufficient stiffness to retain its correct shape under aerodynamic loads.

(c) Mounting points for the engines.

(d) Protection for flightcrew and radio gear, often in a pressurized compartment; heating and/or refrigeration for crew, passengers and equipment.

(e) Suitable breakdown points, to enable the aircraft to be dismantled for transport, or repair by replacement of components.

(f) The minimum number of points requiring servicing and examination, and easy access to them.

The overall design must lend itself to easy and cheap production methods and repairs. With aircraft speeds constantly rising, necessitating a complex structure to provide the strength, this requirement is becoming more difficult to meet.

When the general arrangement of the aircraft is settled, the structural design may proceed. In this the designer has complete freedom of choice, but he is usually influenced by past practice and experience.

One of the main problems that a designer has to overcome is that of excessive weight. An increase of 1% in the weight of the structure can mean as much as 5% to 10% increase in the gross weight of the aircraft. Briefly, this is due to the 'snowball' effect where increased lift creates more drag, necessitating larger engines, and therefore more fuel for the same performance. Conversely, a saving of 1% in the weight of the structure can result in 5% to 10% reduction in total weight of the completed aircraft.

As the various components are manufactured some are set aside for fatigue testing. This, for example, can involve placing whole wing structures in devices which can vibrate and flex the wing at various frequencies, thereby simulating in a relatively short time many thousands of flying hours. These tests are usually continued until a unit fails, in which event the unit can either be modified or strengthened and the safe-life time of the wing can be calculated.

The fuselages of most pressurized civil transport are often subjected to a 'tank test'. The fuselage is completely immersed in a large tank of water, and the pressure inside it is raised until the differential between the inside and outside of the cabin is similar to that during flight at cruising altitudes. The pressure can then be raised and lowered, simulating climbs and descents. This is normally continued until fatigue failure occurs, which may indicate that strengthening is required or it can serve to give an indication of the safe life of the fuselage.

19.5 Structural Rigidity

In the early days of aircraft design an aircraft was considered to be acceptable if it was made strong enough to withstand the direct air loads acting upon it. As aircraft speeds increased it was found that vibration could occur in the wing and tail units and it often appeared to be associated with the control surfaces. In some instances the vibration was sufficiently severe to cause complete disintegration of the airframe.

After several years of research an explanation was evolved for a phenomenon now known as flutter. Design features to overcome flutter are nowadays incorporated as a matter of course in aircraft design. The following paragraphs present a simple non-mathematical explanation of a very complex subject.

Vibration may occur in three ways and can be caused by the wing bending or flexing, by wing twisting, or by control surface movement. The vibration due to wing flexing and twisting can be controlled by structural rigidity, whilst control surface movement is governed by the elasticity in the control cables or rods. Figure 19-1 illustrates the way in which a wing may twist in torsion. The torsional axis can be taken as the line about which the wing will twist if a force is applied to the wing, other than on the line of the axis itself.

A wing will not twist if a force is applied to the torsional axis. The wing may, however, bend or flex under this force, as illustrated in Fig 19-2. It can be seen that the torsional axis is an important feature of the wing structure and can be taken as the point or line about which the wing will either twist in torsion, or bend in flexure.

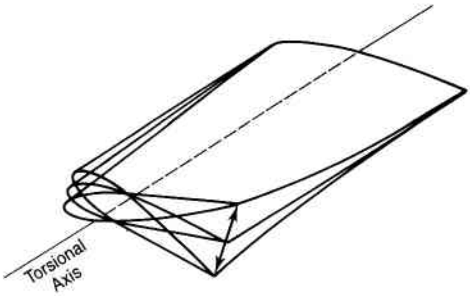


Figure 19-1 Wing Twist or Torsion

The third form of vibration is caused by the control surface itself vibrating in the airstream because of incorrect balancing or slackness in the control runs. The first two forms of vibration are, in themselves, harmless and can quickly be damped out by the rigidity of the airframe. However, when brought about by an external force, for example the airstream, further reactions will occur which may eventually lead to structural failure.

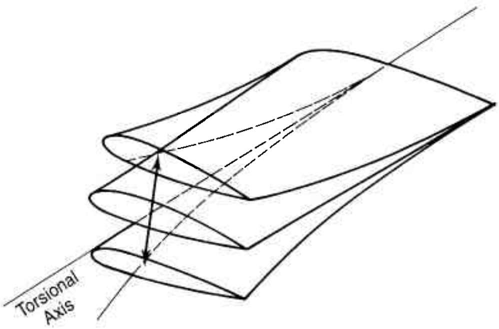


Figure 19-2 Wing Bending or Flexure

19.6 Flutter

Flutter is a possible cause of structural failure. It is a violent vibration of the aerofoil surfaces caused by interaction of their mass and aerodynamic loads. Three forms of flutter affect the wing:

(a) Torsional flexural flutter.

(b) Torsional aileron flutter.

(c) Flexural aileron flutter.

Torsional Flexural Flutter
This occurs as a result of the wing flexing and twisting under the influence of aerodynamic loads. The sequence of events is as follows:

(a) The wing is taken to be in stable horizontal flight with the torsional axis ahead of the CG of the wing. The lift (L) is balanced by the reaction (R) caused by the bending of the wing due to the aircraft weight.

(b) A disturbance causes the incidence of the wing to be momentarily increased, resulting in an increase in lift; L is now greater than R and the wing flexes upwards. Because of inertia, the CG will lag behind the torsional axis and thereby further increase the angle of incidence, and so increase lift even more.

(c) Stiffness of the wing brings the torsional axis to rest, but inertia causes the CG to travel farther, decreasing the angle of incidence. L is then less than R, and the wing starts to descend.

(d) Stiffness of the wing brings the torsional axis to rest, but inertia causes the CG to travel farther, increasing the incidence. L is again greater than R and the flutter cycle begins again.

Torsional flexural flutter can be prevented in the design, either by ensuring that the wing is sufficiently stiff so that the critical flutter speed is far in excess of the permissible maximum speed, or by ensuring that the CG of the wing is on, or ahead of, the torsional axis.

Torsional Aileron Flutter
This is caused by the wing twisting under loads imposed on it by the movement of the aileron. Figure 19-3 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.

(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, together with the 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 with fully powered controls and no manual reversion do not require mass-balancing; all other aircraft have their control surfaces mass-balanced.

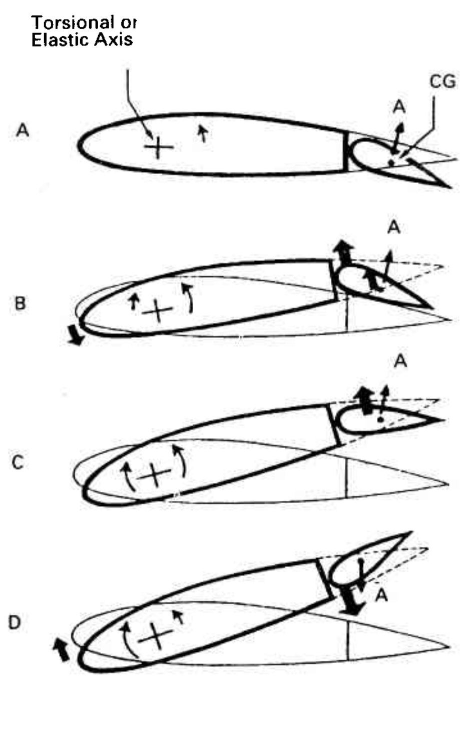


Figure 19-3 Torsional Aileron Flutter

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 wing as it flexes, therefore 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, thereby increasing the stiffness of the aileron and preventing a concentrated weight starting torsional vibrations in the aileron itself.

So far only wing flutter has been discussed, but a few moments consideration will show that mass-balancing must also 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 consequently fail.

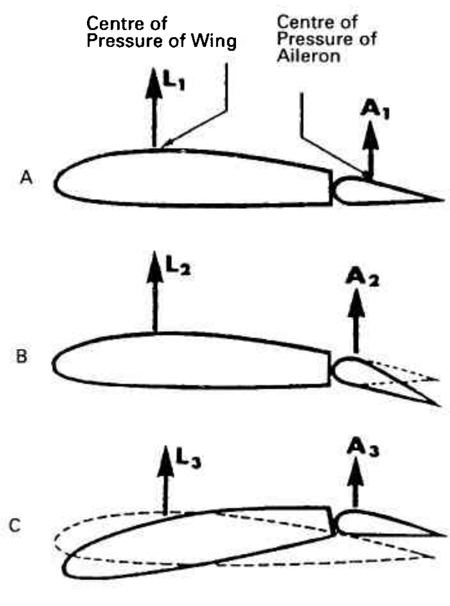


Figure 19-4 Aileron Reversal

To raise one wing, the aileron attached to that wing is lowered (Fig 19-4). This increases the lift of the aileron (A_2), exerting an upward force on the hinge. If the wing has insufficient stiffness it will twist about its torsional axis, raising the trailing edge, relative to the leading edge, thereby reducing the incidence of the wing. This in turn decreases the lift of the wing (L_3), and in particularly bad cases may exceed the lifting effect of the aileron (ie $L_3 + A_3$ becomes less than $L_1 + A_1$). As a result the wing goes down – the opposite effect to that intended. This is known as aileron reversal.

Divergence
In extreme cases, lack of torsional rigidity in the wing causes divergence. If the incidence of a wing is momentarily increased, the lift of the wing will also increase, and the centre of pressure will move forward. Should the torsional axis of the wing be behind the centre of pressure, both the increase of lift and its forward movement magnify the couple which is twisting the wing in the direction of increased incidence. Conversely, should the initial disturbance decrease the angle of incidence, the decreased lift and the aft movement of the centre of pressure behind the torsional axis tend further to reduce the incidence.

In both cases this twisting action is opposed by the torsional reaction of the wing; but since the lift force increases with the square of the speed, there is a critical speed (known as the divergent speed), beyond which the aerodynamic couple will build up more rapidly with change of incidence than the torsional reaction of the wing, and consequently the wing will continue to twist until it breaks off. This is avoided in either of two ways: by making the wing sufficiently stiff in torsion (but not necessarily in flexure) so that the divergent speed is well beyond the maximum permissible speed for the aircraft; or by designing the wing so that its torsional axis is in front of the aerodynamic axis, in which case divergence cannot occur at any speed.

19.7 The Structure

The basic forces acting on an aircraft in flight, ie lift, weight, thrust, drag, are all primary criteria in the design of the aircraft's structure. The designer has to ensure that the strength of the airframe exceeds the normal maximum operating loads imposed on it, by the required safety margin. These forces will vary considerably throughout the flight envelope, and are dependent on such things as loading (g), airspeed, turbulence, movement of control surfaces, changes in configuration (lowering of landing gear, etc) and landing.

On older types of aircraft, a biplane configuration was almost standard and the use of external wires and bracing struts enabled wing structures to be made which were extremely rigid compared with the weight and strength of the component parts. The wings and fuselages were built up from a light framework of wooden ribs, spars and formers covered with a skin of fabric tightened by doping. The wires and struts bracing the mainplanes formed what was, in effect, a large lattice girder; wires were also used to brace the tailplane and fin.

As the top speed of aircraft increased, so the shape and layout of the aircraft, and the materials used in its construction changed. The monoplane layout became universal, bringing with it the more sophisticated problems of designing a thin unbraced wing that was strong enough to resist the tension, compressive and twisting loads imposed upon it. Metal was used for formers, ribs and as an outer skin in place of the plywood and fabric of the earlier aircraft. The conflicting requirements of light weight and strength usually resulted in a compromise, and aluminium alloys are used extensively in medium speed, subsonic aircraft. For supersonic aircraft, the kinetic heating effect of prolonged supersonic flight could cause the conventional light alloys to lose some of their strength and specially formulated light alloys have to be used; other materials, such as stainless steel, which is heavier, stronger and more expensive, have to be used in the construction of aircraft designed for continuous supersonic flight at the higher Mach numbers. Some examples of airframe construction are shown in Fig 19-5 a, b, c and d.

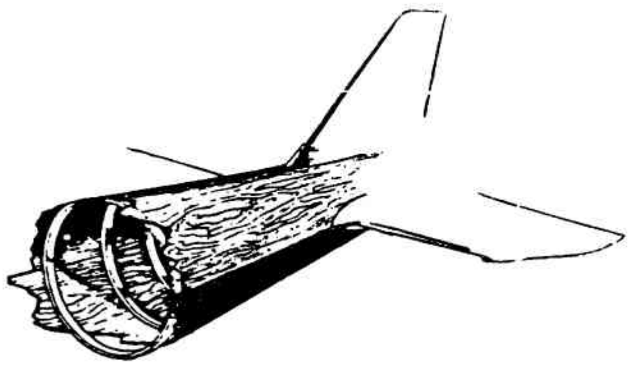


Figure 19-5a Stressed Skin or Monocoque Construction

Stressed skin is a type of construction in which the skin of the aircraft takes a considerable proportion of the load on the aircraft.

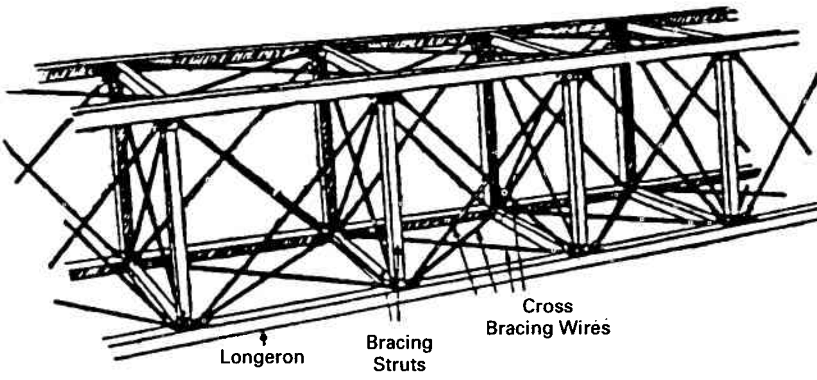


Figure 19-5b Warren Girder

Used extensively on older light aircraft designs in which the skeleton frame takes most of the load and the skin very little.

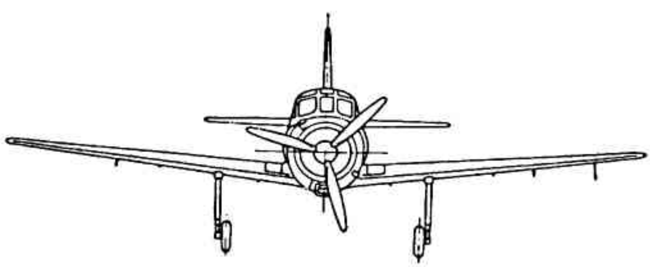


Figure 19-5c Cantilever Wing

Figure 19-5d Braced Wing

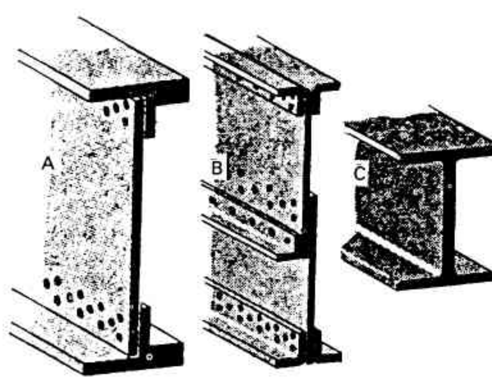
19.8 Wing Construction

Spars

In order to resist the bending forces imposed on it, an ideal spar is given a certain depth. An example of this is an ordinary ruler, which will flex easily when loaded on the upper or lower flat surfaces, but is very stiff when a load is applied to the edge. Unfortunately, the modern wing is thin in cross-section, precluding the use of a deep spar. Two, three, or more spars are used in the wing to give the necessary strength. A spar usually consists of solid booms at the top and bottom, connected by a thin plate web. Normally these are manufactured as separate items and riveted together, but some spars are made in one piece from monobloc forgings, machined to perfect shape. Figure 19-6 illustrates three typical spar sections.

Stressed-skin

Although some light aircraft still have parts of the airframe covered in fabric, most aircraft today are metal clad. In subsonic aircraft, the wing skeleton of spars and ribs is covered with a light alloy skin. This is riveted to the framework and is designed to stiffen the wing by taking some of the loads. This type of construction if known as 'stressed skin' and produces a relatively strong wing without too large a weight penalty. The wing can withstand twisting or torsion loads, and is usually strengthened by the addition of span-wise stringers to withstand the bending or flexure loadings.



a. Simple plate web and extruded booms.
b 'Fail-safe' spar in which no crack can propagate.
c Spar machined from single forging.

Figure 19-6 Typical spar sections

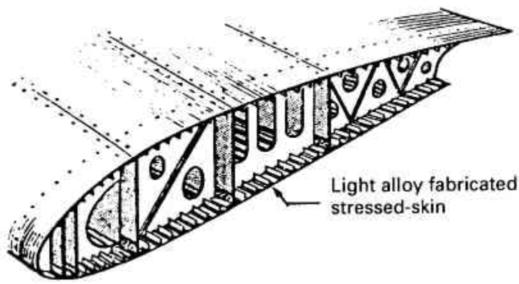


Figure 19-7 Stressed-skin wing construction

Machined Skin

The faster an aircraft flies, the greater the rigidity required of the structure. To achieve this the stressed-skin of the slower aircraft is replaced by a machined skin manufactured from a solid billet of metal. The metal is milled away by high precision machines so that in its final form the contour of the wing is very accurately reproduced, together with the necessary strengthening buttresses and ribs. Altogether up to 90% of the original metal will be cut away, leaving a structure that is not only extremely strong and precisely shaped, but also light in weight. The panels so produced are joined together to form a rigid, strong wing.

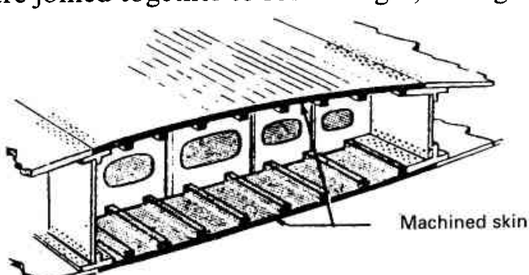


Figure 19-8 Machined skin wing construction

Torsion-Boxes

In this form of construction the skins of the upper and lower surfaces of the wing join the front and rear spars rigidly together to form a box. To the front spar is attached the leading edge and to the rear spar the trailing edge, aileron and flaps. To increase the load-carrying capacity of the skin between the spars, it is common to corrugate it and then cover the corrugations with thin sheet. This form of construction is much used and a variation of it, which has a number of spars, one behind the other, forming a series of boxes, appears particularly suited to aircraft with low aspect ratios.

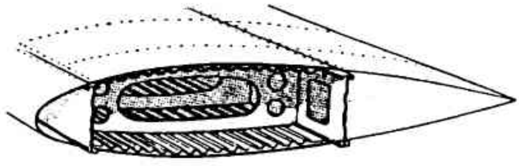


Figure 19-9 Torsion Box Construction

D-Spar Construction
The front spar, which takes most of the bending load, is placed as near as possible to the point of maximum thickness of the wing, and the skin of the leading edge is rigidly attached to it to form a D-shaped tube, which takes nearly all the torsional stresses of the wing.

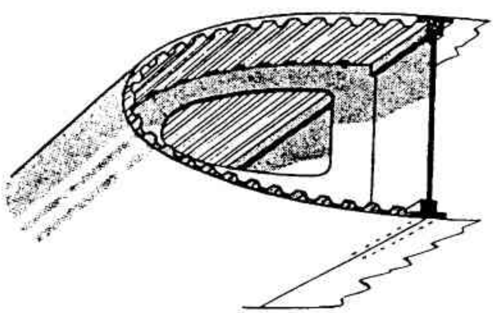


Figure 19-10 D-Spar Construction

Control Surface
For speeds up to 300–350 kt fabric-covered ailerons built up on a spar and ribs are usually satisfactory. Higher speeds demand a rigidity that can only be obtained by a stressed-skin covering built up in much the same way as a D-spar wing. Additional stiffness can be obtained by employing longitudinal fluting of the skin (ie spaced corrugations); in this design most of the ribs can be eliminated.

Braced Wings
This design feature is used almost exclusively in small high wing aircraft. The bracing struts, running from the fuselage to a point about half-way along the wing, relieve the spars of much of their vertical load and anchor them in tension. The designer can therefore save weight in the wing, but because of the additional drag, this form of construction is limited to aircraft with a low top speed.

Fuselage Construction

Fuselages present a basically simpler structure problem than do wings. A fuselage is usually built up from a skeleton of frames or transverse members joined by longitudinal girder members or 'stringers', the whole framework being covered by stressed skin. The shape of the cross-section of the fuselage will vary with the job that the aircraft has to perform. Pressurized transport aircraft have circular cross-sections; this has been found to be the most suitable shape to resist the differential pressures. Light aircraft often have a rectangular section fuselage; this being an easy and strong shape to construct.

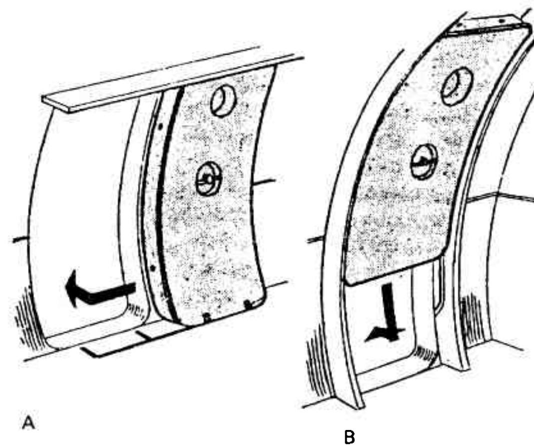


Figure 19-11 Typical pressure cabin doors

Pressurization

The ideal shape for a pressure vessel is a sphere; passengers and freight are best carried in a box shape. In pressurized transport aircraft the designer combines these two shapes as much as possible and the pressure cabin is usually in the form of a circular tube with hemispherical ends. This structure is easy to construct from light alloys and the stresses induced by pressurization are not difficult to calculate. The problems of providing openings for doors, windows, etc are more difficult. Where cut-outs are made in the stress-carrying skin, additional strengthening is needed around the edges to provide a stress path around the aperture; strong rims alone are not sufficient, the loads must be gradually absorbed by the surrounding structure to prevent any sudden stress concentration that could lead to fatigue. The ideal shape for any opening in a pressure cylinder should be an ellipse which is of course, why many aircraft have their windows this shape. Elliptical door shapes are not so practicable from a loading aspect and the more common shape is a rectangular door with rounded corners. (Fig 19-11).

Sealing Problems

Ideally, a pressurized cabin should be airtight; in practice, leaks are kept to a minimum. Sealing must be effective under all conditions, including the structural flexing that occurs during flight, and the expansion and contraction caused by temperature variation. For doors, the sealing medium normally used is an inflatable tube, fitted between the door edge and the aircraft structure and inflated to form an airtight seal (Fig 19-12). Control rods or cables passing out of the cabin must be adequately sealed against leakage, whilst allowing movement and self alignment with a minimum of friction. The seal shown in Fig 19-13 is a typical example that relies on grease in conjunction with packing rings to provide an airtight seal.

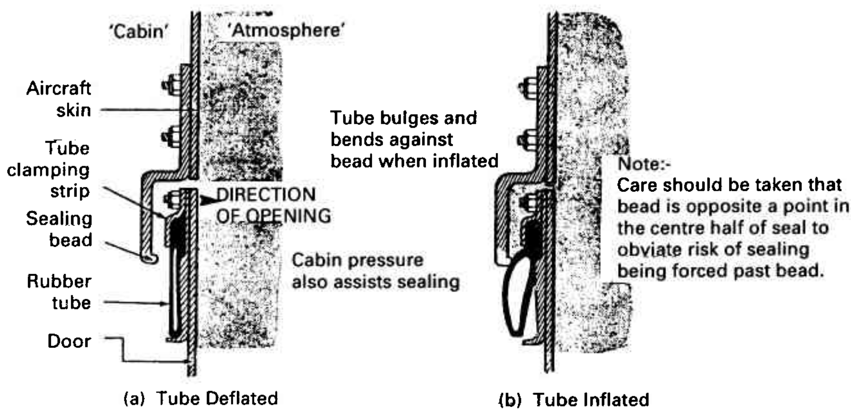


Figure 19-12 Method of Sealing Door

Supersonic Aircraft Structures

To achieve stronger airframes the machined skin type of construction originally devised for wings can also be applied to most parts of the fuselage. However, for aircraft designed for prolonged flight at high supersonic speeds even stronger materials have to be used. Because of the kinetic heating effect at high Mach Number, parts of the skin can be raised to over 120°C at M2.0. At this temperature aluminium alloys lose 40% of their strength. Therefore, in order to retain acceptable strength and rigidity, large panels are made from a stainless steel honeycomb sandwich, (Fig 19-14). This consists of a core, built from thin strips of stainless steel in the form of a honeycomb, and brazed together. The finished core is then machined to the shape of the required panel and placed between ready-shaped inner and outer skins, also of stainless steel. It is then heated in an inert atmosphere until all the joints have been brazed together. This results in an extremely rigid and relatively light structure which will retain its strength at temperatures of around 260°C.

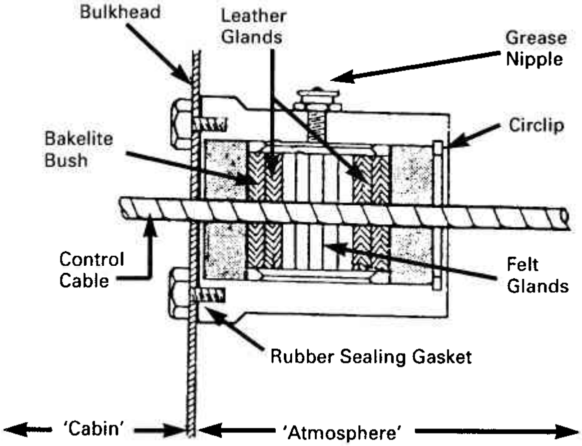


Figure 19-13 Method of sealing control cable

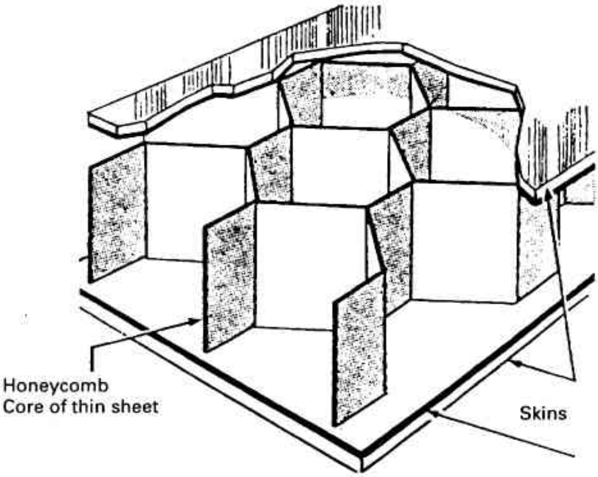
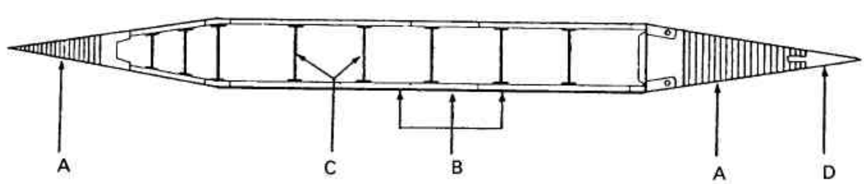


Figure 19-14 Section of stainless steel honeycomb sandwich panel

Complete airframes can be built up from honeycomb sandwich panelling, pre-shaped as described in the previous paragraph, with extruded stainless steel boundary members, transverse struts and attachment points incorporated. Areas subjected to large stresses have the density of the core increased, and the skin thickened. Figure 19-15 illustrates the section of the wing of an aircraft designed for continuous flight at speeds of around Mach 3. The leading and trailing edges have a solid honeycomb core and the skin is of a honeycomb sandwich supported by welded stainless steel spars. At this point mention must be made of the structural concept of 'safe-life' and 'fail-safe'. A structure designed for a given safe life is one in which actual testing of similar structures has enabled the designer to calculate the minimum flying hours before which structural failure will occur. This figure is then the 'safe-life' for that particular structure. A 'fail-safe' structure is one in which, by duplicating primary structures, an alternative path is available for a load. Therefore, if one member fails, the remaining structure can carry the load for a limited time. In some cases this will involve an extra weight penalty, but often the standby part can justify its existence by performing some separate task. An example of this is the window of a pressure cabin, which consists of two layers of glass with a sandwich of dry air between. Normally, the pressure differential is supported by the inner layer, but should this fail then the outer layer can be made to take the load.



A Solid honeycomb leading and trailing edges

B Honeycomb sheeting

C Stainless steel welded spars

D Machined extruded sharp trailing edge

Figure 19-15 Wing section designed for Mach 3

Airframe Limitations

Except during landing, or manoeuvring on the ground, all loads on an aircraft structure are imposed aerodynamically in two ways, either as the result of a manoeuvre or because of atmospheric disturbance, (eg gusts). Limitations, such as indicated speeds, Mach number, accelerations, weights and CG positions, are imposed for reasons of safety. These usually depend on factors not related to the skill of the pilot. All airframe limitations are quoted in the Flightcrew Manual for the type, and must not be exceeded intentionally.

Limitations take into consideration the aircraft's role, structure, and controllability, and are imposed only when they are essential. Disregard of limitations leads to damage and weakens the aircraft structure so that it may fail immediately or on a subsequent flight.

IAS Limitations

The airloads acting on the airframe depend principally upon dynamic pressure (the $\frac{1}{2}\rho V^2$ effect) and vary roughly as the square of the IAS. Figure 17-16 shows how the dynamic pressure, which is 35 lb per square foot at 100 knots, increases to no less than 875 lb per square foot at 500 knots. Therefore at a certain speed the total load on some parts of the airframe, usually the wings or tail structure, increases up to the safety limit. The strength of the tail structure is often the limiting factor because a considerable down load, produced by the elevators or tailplane, is required to keep the wings at the angle of attack necessary to produce the large amount of lift when manoeuvring at high g.

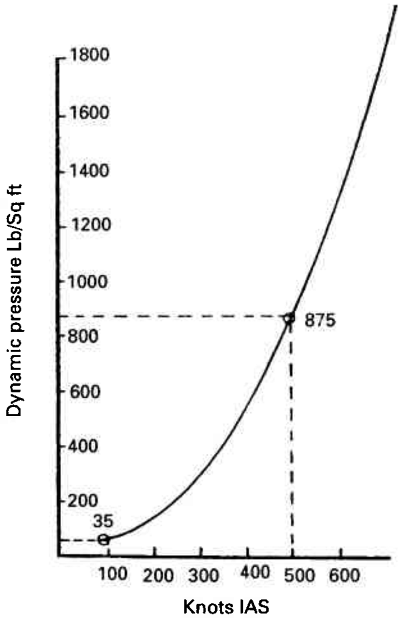


Figure 19-16 Effect of IAS on the dynamic pressure experienced by an aircraft

A further consideration is that at high IAS the loads on the airframe may be great enough to cause aeroelastic distortion which could so alter the stability characteristics of the aircraft as to make its behaviour unpredictable.

The maximum permissible IAS given as the service limitation in the Flightcrew Manual is slightly lower than the design maximum IAS, which is the highest figure for which the aircraft is stressed. The difference between the two gives the pilot a small safety margin. If the design maximum IAS were permitted, even the slightest inadvertant exceeding of it would almost certainly cause damage to the aircraft.

Mach Number Limitations

A Mach number limitation is usually imposed when violent compressibility buffetting may lead to structural failure, or when loss of control due to compressibility effects may cause the aircraft to exceed the structural limitation before control can be regained. Alternatively it may be necessary to impose a Mach number limitation in the early stages of an aircraft's service life because trials have not been completed to allow clearance to a higher Mach number. When a Mach number limitation is imposed it may be quoted as a definite figure, such as 0.88M, or as a specific condition of flight, eg when a nose-up trim change occurs.

On some aircraft Mach number limitations are imposed at low altitudes, because even temporary or partial loss of control at the high accompanying IAS could quickly result in a dangerous situation; the larger aerodynamic and g loads set up by violent behaviour, added to the already large loads imposed by the high IAS, might well be more than the airframe could absorb.

Flight in Turbulence

Turbulent air imposes g loads on the airframe, the effect of which is proportional to the IAS. If turbulent air is encountered when flying at high IAS, the air speed should be reduced to that recommended in the Flightcrew Manual for safe flight in these conditions. Speeds higher than the recommended figure may result in damage to the airframe, whereas lower speeds may lead to difficulty in control.

Prohibited Manoeuvres

The flying controls enable the pilot to manoeuvre the aircraft into any attitude. Some of these attitudes may lead to dangerously high loadings and air speeds which the aircraft has not been designed to withstand. To protect the pilot and the aircraft certain manoeuvres are prohibited.

Undercarriage and Flap Limiting Speeds

The speed limitations for the raising and lowering of the flaps and undercarriage arise either from the limited strength of the components to withstand the air loads, or from the power of the operating mechanism. The limiting speed still applies with the service in the extended position unless the Flightcrew Manual states a higher speed. Further, should the undercarriage or flaps be lowered at higher speeds the trim and stability of the aircraft may be markedly affected and the airframe overstressed. Unless the Flightcrew Manual for the type states that the flaps are designed to assist manoeuvres, they should not be used under conditions of loading appreciably greater than those of steady level flight. It should be noted that the figures quoted are limitations and are *not* recommended as the best speeds at which to perform these operations.

Weight Limitations
Weight limitations are imposed on all aircraft, the determining factors being the strength of the undercarriage, particularly for the landing case, and the loads that can be absorbed by the wings when manoeuvring at the maximum permissible g. On twin and multi-engined aircraft the performance on asymmetric power is sometimes critical, and exceeding the weight limitations may result in a serious drop in performance.

Flightcrew Manuals often give more than one weight limitation, for example:

a) Maximum weight for take-off and gentle manoeuvres only, and a lower limit—

b) Maximum weight for all other permitted forms of flying, and a still lower limit—

c) Maximum weight for landing.

This means that at the highest weight the aircraft must be handled gently, moderate turns should be made and only small amounts of g imposed. Also the IAS and Mach number should be kept well within the limitations until the weight falls to the limit at which all forms of flying are permitted. The limits imposed for landing should be exceeded only when an emergency landing must be made and excess load cannot be jettisoned. In this case every care must be taken to avoid large shock loads and the aircraft landed as gently as possible.

CG Limitations
Flying limitations include the most forward and most rearward permissible positions of the CG. The aircraft should be flown at standard loadings at which the CG is within safe limits. Allowance should always be made for any shift of the CG as fuel is used. Non-observance of CG limits can lead to instability at all speeds and to uncontrollable nose or tail-heaviness at low speeds, the latter because of the elevators reaching the limit of their movement.

Chapter 19: Test Yourself.

1 In an aircraft structure STRAIN is:

a) measured as the load per unit area.
b) the change of size over the original size.
c) the original size over the change of size.
d) measured as the total force acting on a given structural section.

Ref para 19.2

2 Rigidity is the:

a) change of size over the original size.
b) ratio of strain over stress.
c) ratio of stress over strain.
d) measure of load per unit area.

Ref para 19.2

3 A monocoque structure is also known as:

a) rigid construction.
b) stressed skin.
c) warren girder.
d) quasi construction.

Ref para 19.7

4 Warren girder construction employs the principle of:

a) the aircraft skin taking most of the load.
b) the aircraft skin taking minimal load.
c) all metal construction.
d) all wood construction.

Ref para 19.7

5 Torsional aileron flutter may be caused by:

a) wing flexure.
b) mass balance forward of the aileron hinge line.
c) control surface C of G on the hinge line.
d) fitting of hydraulic servos to the aileron control system.

Ref para 19.6

Some More Key Points

1 In level flight Lift, Weight, Thrust and Drag are said to act through the centre of gravity.

2 Induced drag is proportional to lift.

3 Induced drag is greatest at the wing tip.

4 Induced drag is inversely proportional to speed.

5 Profile drag is proportional to speed.

6 The stalling angle is the angle above which a wing will stall.

7 At zero angle of attack a cambered wing will produce some lift and some drag.

8 At zero angle of attack a symmetrical wing will produce NO LIFT but some drag.

9 Induced drag reduces with increased aspect ratio.

10 A high aspect ratio wing has a long span and a short chord.

11 An increase of aspect ratio results in a reduction in stalling angle.

12 With increasing altitude the stalling angle of wing remains the same.

13 With increasing speed the stalling angle remains the same.

14 With increased aircraft weight the stalling speed increases.

15 With the aircraft CG on its forward limit the stalling speed is increased.

16 3° to 4° angle of attack is known as the optimum angle of attack.

17 From zero degrees angle of attack up to the optimum angle of attack the Lift/Drag ratio increases.

18 Above the optimum angle of attack the Lift/Drag ratio reduces.

19 The optimum angle of attack is the best angle of attack in the cruise.

20 Aileron flutter is primarily caused by wing flexure.

21 Aileron flutter is most likely to occur on a flexible wing with rigid ailerons at high speed.

22 Aileron flutter may be reduced with mass balance of the control surfaces.

23 The objective of mass balance is to bring the control surface CG to the surface hingeline.

24 Mass balancing is achieved by fitting weights to the control surface which act forward of the hingeline.

25 When the angle of attack is increased in flight the C of P will reach its farthest forward point just below the stalling angle.

26 In flight an increase in angle of attack will result in the transition point moving forward.

27 As the angle of attack is increased in flight the Boundary Layer will thicken.

28 In level flight the stagnation point is that position where air is brought to rest just in front of the aerofoil leading edge.

29 The stagnation point is static pressure plus dynamic pressure.

30 In flight, with an increase in angle of attack, the stagnation point will move down and aft.

31 When trailing edge flaps are lowered in flight the wing centre of pressure moves aft.

32 When trailing edge flaps are lowered in flight the wing stalling angle is reduced.

33 Slats are normally fitted in front of the ailerons at the wing leading edge to increase the stalling angle.

34 Swept wings are employed to delay M_{crit}.

35 For a given wing area and angle of attack a swept wing will produce less lift than a straight wing.

36 A swept wing tends to increase lateral stability.

37 A swept wing is more prone to tip stall.

38 Spanwise movement of airflow over a swept wing may be reduced by Wing Fences, Leading Edge Notches, Extended or Saw Tooth Leading Edges, or Vortex Generators.

39 The purpose of a vortex generator is to re-energise the boundary layer.

40 Vortex generators are normally fitted on the upper wing surface towards the leading edge in front of the control surfaces. Some aircraft may have them across the complete span.

41 Balance tabs are fitted to control surfaces to assist the pilot in moving the controls by reducing control column loads.

42 A Spring tab is fitted to reduce control column loads at high speed.

43 An American Type Turnbuckle is in safety when not more than three threads are showing.

44 An American Type Turnbuckle is locked by wire locking.

45 As an aircraft accelerates through the transonic speed range the wing C of P will move aft producing a nose down pitching moment.

46 The nose down pitching moment generated as an aircraft accelerates through the transonic speed range is adjusted or trimmed out by the Auto-Mach Trim System.

47 Washout of a wing is the reduction in angle of incidence from root to tip.

48 Induced drag equals profile drag at V_{md}.

49 An increase of aircraft weight will have no affect on glide range but will reduce glide endurance.

50 Generally a V or Butterfly Tail, will aid spin recovery.

FINAL TEST.

1 The angle of attack of an aerofoil is the angle between:

a) chord and the longitudinal axis.
b) wing and the lateral axis.
c) wing leading edge and trailing edge.
d) chord and the relative airflow.

2 Directional control of an aircraft is achieved by use of the:

a) rudder.
b) elevators.
c) fin.
d) ailerons.

3 The wing span is the distance from:

a) leading edge to trailing edge.
b) wing tip to wing tip.
c) wing tip to fuselage centre line.
d) wing tip to wing tip minus the width of the fuselage.

4 The three axes of an aircraft are said to act through the:

a) centre of pressure.
b) wing leading edge centre section.
c) centre of gravity.
d) transition point.

5 The chord line is a:

a) line tangential to the wing surface at the leading edge.
b) line equidistant from upper and lower surfaces.
c) line equidistant between leading and trailing edges, from root to tip.
d) straight line from leading edge to trailing edge.

6 Yawing is a rotation about the:

a) longitudinal axis.
b) lateral axis.
c) transition point.
d) normal axis.

7 A high aspect ratio wing has a:

a) long span and long chord.
b) long chord and short span.
c) long span and short chord.
d) short span and high chord.

8 Rolling is a rotation of the aircraft about the:

a) longitudinal and normal axis.
b) lateral and normal axis.
c) longitudinal axis.
d) lateral axis.

9 Lateral control is achieved with the use of:

a) rudder.
b) tailplane.
c) elevators.
d) ailerons.

10 Pitching is the movement of the aircraft about:

a) the normal axis.
b) the lateral axis.
c) the longitudinal axis.
d) all three primary axes.

11 Adverse yaw when rolling about the longitudinal axis may be prevented by use of:

a) a smaller fin.
b) equal deflection lateral control surfaces.
c) differential ailerons.
d) increased dihedral.

12 In a Frise Aileron control system:

a) the up-going aileron moves through a greater angle than the down going aileron.
b) the down-going aileron leading edge protrudes into the airflow.
c) the up-going aileron produces increased drag.
d) the down-going aileron allows air to spill from below the wing to the upper surface of the aileron.

13 When an aircraft fitted with spoilers is rolled to port the:

a) port spoiler is deflected up.
b) stbd spoiler is deflected down.
c) port spoiler is deflected up and the stbd down.
d) port upper spoiler up and port lower spoiler down.

14 The primary control stops:

a) will be engaged at the control column when the surface is fully deflected.
b) when engaged will leave a small clearance at the control column secondary stops.
c) when engaged will leave a small clearance at the control surface secondary stops.
d) are duplicated, one at the control column the other at the control surface.

15 An American type control Turnbuckle is in safety:

a) when at least three threads are showing.
b) when the inspection hole is covered by thread.
c) when the inspection hole is clear.
d) when not more than three threads are showing.

16 A stiff nut is in safety when:

a) it cannot be tightened with the fingers.
b) the thread of the bolt is level with the face of the nut.
c) the threaded portion of the bolt is above the level of the nut.
d) the threaded portion of the nut is above the level of the end of the bolt.

17 Inset hinges provide:

a) mass balance to assist control movement.
b) aerodynamic balance to prevent flutter.
c) aerodynamic balance to prevent control snatch.
d) aerodynamic balance to assist in control movement.

18 The lowering of leading edge flaps will cause the C of P to:

a) move aft.
b) move aft and towards the wing root.
c) move forward.
d) remain in the same position.

19 A servo tab is normally employed on:

a) transonic aircraft.
b) large subsonic aircraft.
c) light aircraft only.
d) control surfaces subjected to occasional heavy loads.

20 To limit the range of movement of control surfaces in flight:

a) cables are tensioned to a set value.
b) primary and secondary internal control stops are provided.
c) primary and secondary external control stops are provided.
d) control tension regulators are provided.

21 For an aircraft without cable tension regulators fitted in the flying control systems, an increase in temperature will cause cable tension to:

a) decrease.
b) increase only at high altitude.
c) increase only at low altitude.
d) increase.

22 Anti-balance Tabs:

a) move in the same direction as the control surface.
b) move in the opposite direction to the control surface.
c) have a fixed value and do not move in relation to the control surface.
d) are directly connected to the control column.

23 The purpose of a spring tab is to:

a) provide feel feed back in a control system.
b) provide a reduction in the pilot's effort to move the controls against high air loads.
c) provide a constant static friction for the controls.
d) provide a constant load resistance to surface deflection at all speeds.

24 As a trailing edge plain flap is lowered to the max lift position the C of P will:

a) move forward.
b) move forward and towards the wing root.
c) move aft and towards the wing tip.
d) move aft and towards the wing root.

25 A Fowler Flap will increase:

a) wing area.
b) wing area and camber.
c) wing area and aspect ratio.
d) wing area and fineness ratio.

Final Test Answers

1. d	6. d	11. c	16. c	21. d
2. a	7. c	12. c	17. d	22. a
3. b	8. c	13. a	18. c	23. b
4. c	9. d	14. b	19. b	24. d
5. d	10. b	15. d	20. b	25. b