ATPL

Principles of Flight
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Chapter 1.

Laws and Definitions

Introduction

Before studying aerodynamics it is essential to have a thorough grounding in basic mechanics and any related units of measurement. In aeronautics all measurements world-wide are based on the International System (SI) of units, but in practice some anomalies exist, for example altitude, which is quoted in terms of feet, and airspeed which is quoted in nautical miles per hour (knots).

SI Units

The fundamental SI units are those of:

- **Mass**: The amount of matter in a body; measured in kilograms (kg).
- **Length**: The distance between two points; measured in metres (m).
- **Time**: The duration of an event; measured in seconds (s).

From these, other standard units can be derived.

Derived Units

The following quantities and their related units of measurement are extensively used in aerodynamics:

- **Area**: A measure of a surface; measured in square metres (m²).
- **Volume**: A measure of the bulk or space occupied by a body; measured in cubic metres (m³).
- **Velocity**: A measure of motion in a specified direction: measured in metres per second (m/s).
- **Acceleration**: A measure of the change of velocity of a body: measured in metres per second per second (m/s²).
- **Momentum**: The product of the mass and the velocity of a body: measured in kilogram metres per second (kgm/s).
- **Force**: An external influence capable of altering the state of rest or motion of a body, and is proportional to the rate of change of momentum of a body.
Force = Mass \times Acceleration

The unit of force is the Newton (N), which is the force required to give a mass of one kilogram an acceleration of one metre per second per second.

Weight

The gravitational force of attraction that the earth exerts on a body of given mass: measured in kilograms (kg).

Weight = Mass \times Acceleration Due to Gravity

Unlike the mass of a body, which remains constant irrespective of its location, the weight of a body varies with distance between the body and the centre of the earth. This is because acceleration due to gravity varies with geographical location and altitude, but under standard conditions this term is assumed to be 9.81 m/s².

Work

The effort needed when a force is applied to a body causes it to be lifted or moved a given distance in the same direction as the force.

Work = Force \times Distance

The unit of work is the Joule (J). One Joule is the work done when a force of one Newton moves a body through a distance of one metre in the direction of the force.

Work is also stated in terms of Newton Metres (Nm), where 1 Joule = 1 Nm.

Power

The rate of doing work: measured in units of work per unit time: measured in Watts (W), where 1 watt = 1 J/s or 1 Nm/s.

Power = Force \times Velocity

Energy

The capacity for doing work, which in mechanics exists in two basic forms:

Potential energy - Due to position.

Kinetic energy - Due to motion.

The unit of energy is the Joule (J), where 1 Joule = 1 Nm.

Pressure

The force per unit area acting on a surface: measured in Newton’s per square metre (N/m²), which is properly called the Pascal (Pa). In aviation the bar is more commonly used to measure pressure where 1 bar = 10⁵ Pa, or 1mb = 1hPa. In aerodynamics three types of pressure exist:

Static Pressure (Pₛ). When air is stationary it exerts pressure equally in all directions. For example a mass of stationary air
in a container, will exert a certain amount of static pressure on the surrounding walls (Fig. 1.1)

**FIG. 1.1**

**Dynamic Pressure** ($P_d$). This occurs when moving air is brought to rest on the surface of a body, i.e. when relative movement exists between the surface and the airflow (Fig.1.2).

**FIG. 1.2**

Dynamic Pressure is expressed as:

$$Q = \frac{1}{2}\rho V^2$$

Rho ($\rho$) is the air density, which decreases with altitude, and $V$ is the speed of the body relative to the airflow.

**Total Pressure** ($P_t$). The sum of both the static and dynamic pressures. This is a very important term in aerodynamic formulae and is used in the calculation of lift, drag and indicated air speeds (these terms will be explained later).
Total pressure = Static Pressure + Dynamic pressure

In aerodynamics this is also referred to as Pitot Pressure.

**Density**
The mass of material per unit volume: measured in kilograms per cubic metre (kg/m$^3$). The density of air is an important property in the study of aerodynamics and varies with changes in pressure, temperature and humidity. Such changes have a significant effect on aircraft performance.

**Temperature (T)**
A measure of the hotness of a body: measured in Degrees Celsius (°C). The unit of thermodynamic temperature is the Kelvin (K) and is the unit normally used in scientific calculations. To convert from the Celsius system to the Kelvin system, 273 must be added to the temperature in °C.

eg. 15°C = 15 + 273 = 288K

**Viscosity**
A measure of the resistance to motion. In aerodynamics it is the resistance to movement of one layer of air over another, and in the case of a fluid, how easily it flows over a surface. For example cold engine oil has high viscosity, and hot engine oil has low viscosity.

**Wing Loading**
The total aircraft weight supported per unit area of the wing: measured in Newton’s per square metre (N/m$^2$).

\[
\text{Wing Loading} = \frac{\text{AUW}}{\text{wing area}}
\]
# Airspeeds

<table>
<thead>
<tr>
<th>Airspeed Type</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Indicated Airspeed (IAS)</strong></td>
<td>The airspeed of an aircraft as shown on its pitot-static airspeed indicator (ASI), that provides vital airspeed information, e.g. stalling and structural limitation airspeeds, to the flight-crew. It is calibrated to reflect standard atmospheric adiabatic compressible flow at sea level, and is uncorrected for airspeed system errors.</td>
</tr>
<tr>
<td><strong>Calibrated Airspeed (CAS)</strong></td>
<td>The Indicated airspeed of an aircraft corrected for position and instrument errors. CAS is equal to the True Airspeed (TAS) in standard atmosphere at sea level only.</td>
</tr>
<tr>
<td><strong>Equivalent Airspeed (EAS)</strong></td>
<td>The calibrated airspeed of an aircraft corrected for compressibility error at a particular altitude. EAS is equal to IAS at airspeeds less than 300 knots, and is equal to TAS in standard atmosphere at sea level only.</td>
</tr>
<tr>
<td><strong>True Airspeed (TAS)</strong></td>
<td>The actual speed of an aircraft through the air relative to the air that is uninfluenced by the aircraft. TAS is important for navigation purposes only. The relationship between EAS and TAS is as follows:</td>
</tr>
<tr>
<td></td>
<td>[ \text{TAS} = \text{EAS} \left( \frac{\rho_o}{\rho} \right)^{1/2} ]</td>
</tr>
<tr>
<td></td>
<td>where ( \rho_o ) = density at sea level</td>
</tr>
<tr>
<td></td>
<td>( \rho ) = density at altitude</td>
</tr>
<tr>
<td><strong>Mach No.</strong></td>
<td>The ratio of the TAS of an aircraft to the speed of sound in the surrounding atmosphere, i.e. the local speed of sound (LSS).</td>
</tr>
<tr>
<td></td>
<td>[ \text{Mach No} = \frac{\text{TAS}}{\text{LSS}} ]</td>
</tr>
</tbody>
</table>

## Newton’s Laws of Motion

<table>
<thead>
<tr>
<th>Law</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Newton’s 1st Law</strong></td>
<td>States that a body will continue in a state of rest, or in uniform motion in a straight line, unless acted on by an external force, i.e. it has inertia.</td>
</tr>
<tr>
<td><strong>Newton’s 2nd Law</strong></td>
<td>States that a body at rest or in uniform motion will when acted on by an external force accelerate in the direction of the force. The magnitude of the acceleration for any given mass is directly proportional to the size of the force applied, i.e. when a force of 1N is applied to a mass of 1kg it will accelerate at 1m/s(^2).</td>
</tr>
<tr>
<td></td>
<td>[ \text{Force} = \text{mass} \times \text{acceleration} ]</td>
</tr>
<tr>
<td><strong>Newton’s 3rd Law</strong></td>
<td>States that for every action there is an equal and opposite reaction.</td>
</tr>
</tbody>
</table>
Chapter 2.

The Flight Environment

Introduction

In order to study the principles of flight it is first necessary to understand the medium in which flight takes place.

The Atmosphere

The atmosphere is a region of air surrounding the earth up to a height of approximately 500 miles (900 Km). Air is a mixture of gases, the principal ones being oxygen 21% and nitrogen 78% by volume. Up to a height of 6 miles (11 Km) water vapour is also found in varying quantities. The actual amount of water vapour in a given mass of air depends on the temperature and whether the air has recently passed over a large area of water. Generally the higher the temperature, the greater the amount of water vapour a given mass of air can hold. Air has weight and is also compressible. Its pressure, density and temperature all decrease with increasing altitude. An aircraft does work on the air to sustain flight, and any change in pressure, density and temperature will effect the amount of energy that the aircraft can extract from the air.

The Universal Gas Law

The relationship between pressure, density and temperature is:

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

where
- $P$ = Pressure (N/m²),
- $T$ = Absolute Temperature (Kelvin),
- $\rho$ = Density or Mass per unit volume (kg/m³)

This equation only applies to a perfect gas when a given mass occupies a given volume, but from this equation it is possible to establish effect of pressure and temperature on density.

The Effect of Pressure on Density

Air is a gas, and can be easily compressed or expanded. When air is compressed the number or mass of molecules in a given volume increases and the density rises. Conversely as air is expanded the original volume will contain fewer molecules and the density will reduce. Density is therefore directly proportional to pressure, i.e. if the pressure is doubled then the density is also doubled. This statement is only true if the temperature remains constant.

Density $\alpha$ Pressure
The Effect of Temperature on Density

As air is heated the molecules tend to speed up and increase the spacing between them. A given volume will contain fewer molecules and air density will decrease. Conversely as air is cooled the degree of molecular movement will decrease and the given volume will now contain a greater number of molecules, so the density will increase. Air density is inversely proportional to temperature, i.e., if the temperature is doubled the density will half. This statement is only true if the pressure remains constant.

\[
\text{Density} \alpha \frac{1}{\text{Temperature}}
\]

The Effect of Altitude on Density

With increasing altitude pressure and temperature both decrease. As stated previously the drop in temperature will cause an increase in density whilst the drop in pressure will cause a decrease in density. These factors act in opposition to each other, but pressure has a dominating influence over density. Consequently, pressure, temperature and density all decrease with increasing altitude.

The Effect of Humidity on Density

It has been assumed that air is perfectly dry. In fact there is always a certain amount of water vapour in the atmosphere, which varies from place to place, and day to day. When water vapour is present in the air it affects its density because it is less dense than dry air, and therefore lighter. The density of water vapour under standard sea level conditions is 0.760 kg/m³, whereas the density of dry air is 1.225 kg/m³. Water vapour therefore weighs 5/8 as much as dry air. This means that, for a given volume, air is least dense when it contains a maximum amount of water vapour and most dense when it is perfectly dry. Humidity will therefore affect aircraft performance.

The International Standard Atmosphere

In order to compare aircraft performance and calibrate aircraft instruments, it is necessary to have an internationally agreed Standard Atmosphere covering temperature, pressure and density for varying altitudes. The International Standard Atmosphere (ISA), part of which is shown below, is laid down by the International Civil Aviation Organisation (ICAO).
<table>
<thead>
<tr>
<th>HEIGHT ABOVE SEA LEVEL FEET</th>
<th>DENSITY KG/M$^3$</th>
<th>PRESSURE MILLIBARS</th>
<th>TEMPERATURE °C</th>
</tr>
</thead>
<tbody>
<tr>
<td>52,496</td>
<td>0.166</td>
<td>104</td>
<td>-56.6</td>
</tr>
<tr>
<td>45,934</td>
<td>0.288</td>
<td>142</td>
<td>-56.5</td>
</tr>
<tr>
<td>39,372</td>
<td>0.312</td>
<td>194</td>
<td>-56.5</td>
</tr>
<tr>
<td>32,810</td>
<td>0.414</td>
<td>265</td>
<td>-50</td>
</tr>
<tr>
<td>26,248</td>
<td>0.526</td>
<td>357</td>
<td>-37</td>
</tr>
<tr>
<td>19,686</td>
<td>0.660</td>
<td>472</td>
<td>-24</td>
</tr>
<tr>
<td>13,124</td>
<td>0.819</td>
<td>612</td>
<td>-11</td>
</tr>
<tr>
<td>6,562</td>
<td>1.007</td>
<td>795</td>
<td>2</td>
</tr>
<tr>
<td>0</td>
<td>1.225</td>
<td>1013.25</td>
<td>15</td>
</tr>
</tbody>
</table>
Chapter 3.

Aircraft Components and Terminology

Wing Position Terminology

Wings are attached to an aircraft’s fuselage in either a low, high or mid position (Fig. 3.1).

![Diagram of LOW WING, HIGH WING, MID WING](image)

The actual wing position is determined by the following design parameters:

- Engine Positioning/Propeller Blade Length
- Undercarriage Positioning
- Short Take-Off and Landing Capability

The wings are inclined above, or below the horizontal. Wing inclination above the horizontal is known as dihedral, and inclination below the horizontal is known as anhedral (Fig. 3.2).
Wing Planform Terminology

The following terminology is associated with wing planform:

- **Gross Wing Area (S)**. The plan view area of the wing including the portion of the wing normally cut out to accommodate the fuselage (Fig. 3.3).

- **Net Wing Area**. The area of the wing excluding the fuselage portion (Fig. 3.4).
- **Wing Span (b)**. The straight-line distance between wing tips (Fig. 3.5).

- **Average Chord \((C_{AV})\)**. The Mean chord (Fig. 3.5). The product of the span and average chord gives the gross wing area (i.e. \(b \times C_{AV} = S\)).

- **Aspect Ratio (AR)**. The ratio of wing span to average chord. Long thin wings are of high aspect ratio, whilst short stubby wings are of low aspect ratio (Fig. 3.6).
Aspect Ratio = \[ \frac{\text{Wing Span}}{\text{Average Chord}} \] or \[ \frac{\text{Gross Wing Area}}{(\text{Average Chord})^2} \] or \[ \frac{(\text{Wing Span})^2}{\text{Gross Wing Area}} \]

- **Taper Ratio (TR).** The ratio of tip chord (Ct) to root chord (Cr) (Fig. 3.7).
➢ **The Angle of Sweepback.** The angle between the line of 25% chord and a perpendicular to the root chord (Fig. 3.8).

![Fig. 3.8](image)

➢ **Mean Aerodynamic Chord (MAC).** The chord drawn through the centroid (centre of area) of the halfspan area. It must be noted that the MAC and $C_{AV}$ are not the same (Fig. 3.9).

![Fig. 3.9](image)

Aspect ratio, taper ratio, and sweepback are some of the main factors that determine the aerodynamic characteristics of a wing.
Wing Section Terminology

For an aircraft to have acceptable aerodynamic characteristics, various wing or aerofoil sections are used. Reference Fig. 3.10 the terminology associated with aerofoil sections is as follows:

![Diagram of Wing Section Terminology](image)

**The Chord line**  
A straight line joining the leading and trailing edges of a wing.

**The Chord**  
The length of the chord line that is used as a reference for all other dimensions relating to a wing.

**The Mean Camber Line**  
The line drawn equidistant between the upper and lower surfaces of an aerofoil.

**Maximum Camber**  
The maximum distance between the mean camber line and the chord line. This is one of the variables that determines the aerodynamic characteristics of a wing.

**Maximum Thickness**  
The maximum distance between the upper and lower surfaces.

**Maximum Thickness Chord Ratio**  
The ratio of maximum thickness to chord expressed as a percentage. For subsonic wings the ratio is normally 12 - 14%.

Aerofoil Cross-sectional Shapes

A thick well-cambered wing will produce high lift at slow speeds, whereas a thin wing with the same camber will produce good high-speed characteristics (Fig. 3.11).

![Aerofoil Cross-sectional Shapes](image)
The above are both examples of asymmetrical aerofoils. If the mean camber line is coincident with the chord line the wing camber is reduced to zero, and this is known as a symmetrical aerofoil (Fig. 3.12).

FIG. 3.12
Chapter 4.

Lift

Introduction

As air flows around an aerofoil the pressure differential set up over the upper and lower surfaces produces a force. This force acts perpendicular to the relative airflow, and is known as lift. In steady level flight lift directly balances the aircraft's weight. For a given airspeed the lower the weight, the lower the lift.

Airflow

To fully understand how the aerodynamic forces of lift and drag act on an aircraft, it is necessary to study the effect of airflow. In principle it does not matter whether an aircraft is moving through the air, or whether air is flowing over a stationary aircraft, since the result will be the same. Airflow can be either streamline or turbulent in nature.

- **Streamline Flow** This exists when succeeding molecules follow a steady path, with the molecules flowing in an orderly pattern along streamlines around an object (Fig. 4.1).

![FIG. 4.1](image)

At any given point in the streamline, the molecules will experience the same velocities and pressures as the preceding molecules, but the values may alter from point to point along the streamline. Widely spaced streamlines indicate a reduction in velocity, whereas a narrow spacing between the streamlines indicates an increase in velocity. If the streamlines flow without mixing, the flow is known as laminar, which is desirable in most phases of flight, and produces the ideal flow pattern around an aircraft (Fig. 4.2).
If there is a sudden change in the direction of the airflow, the streamline flow will break down, and become turbulent flow.

- **Turbulent Flow** This occurs when the succeeding molecules are no longer able to follow a streamlined flow pattern, and instead travel along a path quite different from the preceding molecules (Fig. 4.3).

Turbulent flow is also termed as ‘unsteady’ or ‘eddying’ flow, and results in wasted energy, which is undesirable in most phases of flight (Fig. 4.4).

- **Free Stream Airflow (FSA)** The airflow that is far enough away from an aircraft so as not disturbed by it.
The equation of continuity applies only to streamlined or steady flow. It states that, if a fluid flows through a pipe its mass flow will remain constant, since mass can neither be created or destroyed. If air flows through a pipe of varying cross sectional area (venturi tube), the mass of air entering the pipe in a given time, will equal the mass of air leaving the pipe in the same time (Fig. 4.5).

The mass airflow at any point in the pipe is the product of the density ($\rho$), the cross sectional area ($A$) and the velocity ($V$).

$$\text{Mass Airflow} = \rho AV$$

Mass Airflow is expressed in kg/s where:

- $\rho = \text{Kg/m}^3$
- $A = \text{m}^2$
- $V = \text{m/s}$

This equation applies equally to both subsonic and supersonic airflow, provided that the flow remains steady. At velocities less than 0.4 Mach, air is considered to be incompressible and inviscid (ideal). Density therefore remains constant, and can be deleted from the equation, such that:

$$\text{Mass Airflow} = AV$$

This shows that velocity is inversely proportional to the cross-sectional area, with any reduction in area, resulting in an increase in velocity, and vice versa. This effect can also be illustrated using streamline flow patterns (streamtube), where converging streamlines indicate an increase in velocity, and vice versa (Fig. 4.6).
Bernoulli’s Theorem

Bernoulli’s Theorem uses the principle of Conservation of Energy. It states that when a fluid flows at a steady rate through a pipe, its total energy remains constant, since energy can neither be created nor destroyed. At any point in a pipe the total energy is a combination of:

- **Potential Energy**: Energy due to height or position.
- **Pressure Energy**: Energy due to pressure.
- **Kinetic Energy**: Energy due to movement.

When considering airflow at a given height, changes in potential energy are negligible, and can be essentially ignored. Total energy therefore equals the sum of the pressure energy and kinetic energy.

\[
\text{Pressure Energy} + \text{Kinetic Energy} = \text{Total Energy}
\]

In aerodynamics, it is the mass airflow per unit volume, which is of most interest to us, so the Conservation of Energy equation is better stated in terms of pressure. At any point in a pipe, the total pressure is the sum of the static pressure and dynamic pressure, and is measured in Pascal’s.

\[
P_s + \frac{1}{2} \rho V^2 = P_T
\]

To satisfy Bernoulli’s theorem, this value must remain constant at all points along the pipe, such that any rise in dynamic pressure will be accompanied by a reduction in static pressure, and vice versa (Fig. 4.7).
From the Equation of Continuity, if a steady stream of air flows through the restricted section of a venturi, its velocity will increase, and vice versa. Any rise in velocity will result in an increase in dynamic pressure and a reduction in static pressure, according to Bernoulli’s Theorem, and vice versa (Fig. 4.8).

The airflow around an aerofoil section also resembles the flow through a venturi (Fig. 4.9).

The flow over the upper surface is representative of a convergent section (1), whilst the flow over the lower surface is representative of a divergent section (2). The static pressure likewise varies, and the resulting pressure differential produces lift.

**Angle of Attack**

The angle of attack (α), is the angle between the free stream relative airflow and the chord line of an aerofoil section (Fig. 4.10).
Changes in the angle of attack cause the velocity and pressure of the flow to vary as the air passes over the upper and lower surfaces. This in turn affects the pressure differential that exists, and hence the amount of lift developed.

Do not however confuse ‘Angle of Attack’ with ‘Angle of Incidence’, which by definition, is the angle at which the wing is fixed to the fuselage, relative to the aircraft’s longitudinal axis (Fig. 4.11).

The angle of incidence is fixed, but the angle of attack changes in flight. Also do not confuse the ‘Pitch Angle’ or ‘Pitch Attitude’ of the aircraft with the angle of attack. Notably for any given angle of attack the pitch angle can vary (Fig. 4.12).
Similarly for any given pitch angle the angle of attack can also vary (Fig. 4.13).

Two – Dimensional Airflow About an Aerofoil

As the streamlines approach an aerofoil, a dividing streamline is developed, which separates the flow going over the upper surface, from the flow going over the lower surface (Fig. 4.14).

As the dividing streamline approaches the aerofoil it slows down, and momentarily comes to rest just below the leading edge, forming a stagnation point. A stagnation point also exists at the rear of the aerofoil. At these points the velocity of the airflow reduces to zero, and the static pressure reaches a maximum value (stagnation pressure), which is higher than atmospheric. The forward stagnation point is situated below the leading edge, allowing the airflow passing over the upper surface to initially travel forwards. The pressure differential (negative pressure gradient) associated with the upper surface also imparts acceleration to the flow, and helps draw the air locally upwards, producing upwash (Fig. 4.15).
At the rear of the aerofoil the faster moving airflow over the upper surface relative to the lower surface tends to force the lower streamlines downwards, producing downwash.

**Effect of Angle of Attack on the Airflow About an Aerofoil Section**

If a symmetrical aerofoil section is placed in a steady airstream at zero degrees Angle of Attack a stagnation point will form on the leading edge (Fig. 4.16).

The airflow velocity above and below the aerofoil will increase by equal amounts, as will the static pressures. Consequently no pressure differential will exist, and no net lift will be created.

If the same aerofoil section is placed at a positive angle of attack, the stagnation point will move below the leading edge point (Fig. 4.17).
Upwash will occur in front of the aerofoil section, and the airflow will be accelerated as it passes over the upper surface (venturi effect), resulting in a reduction in the static pressure. Conversely the airflow passing over the lower surface will reduce in velocity, and the static pressure will increase. A pressure differential now exists, and lift is generated.

If an asymmetrical aerofoil section is placed in the same airstream at zero degrees angle of attack, a stagnation point will form below the leading edge, and upwash will occur (Fig. 4.18).

The velocity of the airflow will increase over the more curved upper surface, whilst the static pressure will fall. A pressure differential now exists and lift is generated. With increasing angle of attack the air flowing over the upper surface now needs to travel a greater distance, and must speed up in order to satisfy the Equation of Continuity. Conversely because the air has to travel a shorter distance over the lower surface, it will slow down. This produces a greater pressure differential, and more lift is generated (Fig. 4.19).
The angle of attack in conjunction with the actual shape of an aerofoil section is therefore one of the factors that is instrumental to the production of lift.

**Chordwise Pressure Distributions About an Aerofoil Section**

The distribution of pressure is important in aerodynamics, since it determines:

- The amount of lift generated.
- Any pitching moments.
- The form drag of an aerofoil section.

The actual pressure distribution over the upper and lower surfaces varies with changes in angle of attack, as does the pressure differential and the amount of lift developed. To appreciate these effects it is useful to display the actual pressure distribution diagrammatically (Fig. 4.20).

A series of pressure vectors drawn normal to the aerofoil surface and joined at their extremities produces a pressure envelope. An arrow on each line pointing inwards represents a positive pressure, i.e. above atmospheric pressure, whilst those pointing outwards represents a negative pressure, i.e. below atmospheric pressure.
Fig. 4.21 represents the chordwise pressure distribution that exists about an asymmetrical aerofoil, and shows how it alters with changes in angle of attack. At small negative angles of attack, in this case minus 4°, the pressure changes are approximately equal, and no net lift is developed. This is known as the ‘Zero Lift Angle of Attack’, which is negative for asymmetrical aerofoils, but zero for symmetrical aerofoils. With increasing angle of attack the pressure differential alters, and the reduction in upper surface pressure far exceeds the increase in pressure over the lower surface. The peak of the negative pressure envelope also increases in height, and moves towards the leading edge. Lift similarly increases, and continues to do so up to a certain angle, in this case somewhere between 16° and 20°. For conventional low speed aerofoils this angle is usually about 15° to 16°, and is known as the ‘Stalling Angle of Attack’. Beyond this angle the streamline flow over the upper surface breaks down, causing an increase in static pressure, and the airflow becomes turbulent. The relationship between velocity and static pressure is no longer applicable beyond this point, since Bernoulli’s Theorem only applies to streamline flow.

The Centre of Pressure

In contrast to complicated pressure plots it is possible to show the overall effect of changes in static pressure using a single aerodynamic force (Fig. 4.22).
This force is called the total reaction (TR), and acts through a single point on the chord line, called the Centre of Pressure (CP). At normal cruising airspeeds, and low positive angles of attack, the centre of pressure is positioned on the chord line near the centre of an aerofoil. With increasing angles of attack, the centre of pressure moves forward towards the leading edge, as does the low pressure peak, and the total reaction increases in magnitude. Beyond the stalling angle of attack the low-pressure peak rapidly collapses, causing the magnitude of the total reaction to decrease and the centre of pressure to move rapidly rearwards towards the trailing edge. Similarly any reduction in the angle of attack will cause the centre of pressure to move rearwards.

The total reaction is the resultant of the components of lift and drag. Lift acts perpendicular to the relative airflow, whilst drag acts parallel to, and in the same direction as the relative airflow (Fig. 4.23).

The two components of lift and drag, therefore, vary in size as the magnitude of the total reaction alters. Drag will be covered later in detail.

Pitching Moments

Consider the pressure distribution around an asymmetrical aerofoil at an angle of attack that gives zero lift (Fig. 4.24).
The pressure distribution gives what is basically a couple and, although there is no net lift, a nose-down pitching moment is produced. This is best demonstrated by replacing the pressure envelopes associated with the upper and lower surfaces, with separate individual pressure vectors. The vectors act through the centre of pressure relating to the respective surfaces and represent the actual lift being generated from each. Notice that the lift vectors are both equal in magnitude, but their lines of action do not coincide. The lift developed by the upper surface acts behind the lift developed by the lower surface, and produces a nose down pitching moment.

If the same aerofoil is placed at an angle of attack giving positive lift, the pressure distribution will alter accordingly (Fig. 4.25).

The lift being generated by the upper surface will now be greater than the lift being generated by the lower surface and a greater nose down pitching moment will occur.

Now consider a symmetrical aerofoil at zero degrees angle of attack when it is developing no lift (Fig. 4.26).
Again the lift being developed by the upper and lower surfaces can be represented by individual lift forces. In this case, the lift forces are equal in magnitude and both act through the same centre of pressure, so that there is no overall pitching moment. If the same symmetrical aerofoil is now given a positive angle of attack, it will result in an imbalance in the upper and lower surface lift vectors, as positive lift is developed (Fig. 4.27).

In this case, the two lift vectors continue to act through the same centre of pressure and produce no resultant pitching moment. Unlike asymmetrical aerofoils, symmetrical aerofoils therefore do not produce pitching moments at any angle of attack.

Aerodynamic Centre

For asymmetrical aerofoils, it has been established that the centre of pressure moves along the chord line with changing angles of attack. As the angle of attack increases the centre of pressure moves towards the leading edge and vice versa. A nose down pitching effect is always present, and increases in intensity with increasing angles of attack. A point however, exists on an aerofoil's chord line about which the pitching moment remains constant,
regardless of any change in the angle of attack. This point is called the Aerodynamic Centre, AC (Fig. 4.28).

![Diagram showing lift and pitching moment](image)

Its position in subsonic airflow is normally 25 percent of the chord line back from the leading edge and, unlike the centre of pressure, is fixed in position. For design purposes it is convenient to consider the overall lift force to act at this point, combined with a pitching moment of constant strength, giving a steady nose down pitch effect. However in supersonic flow the AC moves rearwards to 50 percent of the chord line.

**Lift Formula**

The amount of lift generated by a wing depends on the following:

- Wing Shape
- Angle of Attack
- Air Density (\(\rho\))
- Free Stream Air Velocity Squared (\(V^2\))
- Wing Planform Surface Area (\(S\))

The dynamic pressure possessed by a moving fluid equals half the density times the velocity squared

\[ \text{Dynamic Pressure} = \frac{1}{2} \rho V^2 \]

The combination of this pressure and a wing’s planform surface area (\(S\)) produces a force, which is proportional to the area on which it acts. This force is known as Lift.

\[ \text{Lift} = \text{Pressure} \times \text{Area} \]
\[ = \frac{1}{2} \rho V^2 \times S \]
Lift does not directly equal the product of the two terms and therefore the lifting efficiency of the wing also needs to be taken into account. This involves the wing’s shape, and angle of attack. Lift is expressed in Newton’s (N) and the general lift formula is:

\[ \text{Lift} = C_L \frac{1}{2} \rho V^2 S \]

**Variation of Coefficient of Lift with Angle of Attack**

To establish the effect of angle of attack on the lifting ability of a wing, a graph of coefficient of lift against angle of attack can be plotted. This is known as a lift curve and is used to highlight a number of important aerofoil features. The curve shown is for an asymmetrical aerofoil section, because at zero degrees angle of attack a positive \( C_L \) exists, and positive lift is being produced (Fig. 4.29).

![Lift Curve](image)

**Fig. 4.29**

Between \( 0^\circ \) and \( 12^\circ \) angle of attack the graph is a straight line. This shows that the coefficient of lift, and hence lift is directly proportional to angle of attack in this region. Above \( 12^\circ \) angle of attack the rate of increase in lift reduces and the curve eventually forms a peak. This peak represents the maximum coefficient of lift \( (C_{L_{\text{max}}}) \), which for this particular aerofoil occurs at about \( 15^\circ \) angle of attack. This angle however varies with differing aerofoil sections, but for most light aircraft is a typical value. At angles of attack beyond this point, the lift curve drops rapidly downwards indicating a significant drop in the coefficient of lift and hence lift produced by the aerofoil. The aerofoil is now stalled, and can no longer produce sufficient lift to maintain steady straight and level flight. The angle of attack at which \( C_{L_{\text{max}}} \) is reached is known as the stalling angle of attack of the aerofoil. The normal flight range is considered to exist between \( 0^\circ \) angle of attack and the stalling angle of attack. Each aerofoil possesses its own lift curve so it is possible to compare asymmetrical aerofoils against symmetrical aerofoils (Fig. 4.30).
Asymmetrical aerofoils clearly produce more lift at any given angle of attack, but stall at a lower stalling angle of attack than symmetrical aerofoils.

**Three – Dimensional Airflow About an Aerofoil**

So far, the pressure distribution around an aerofoil has been considered purely in the chordwise direction, but to fully understand how lift is developed by an aerofoil or wing, it is necessary to also consider the spanwise pressure distribution. Previously it was established that as air flows around an aerofoil section a pressure differential is set up over the upper and lower surfaces (Fig. 4.31).
The lower surface of a wing is normally at a pressure above atmospheric, whilst the upper surface is at a pressure below atmospheric. Beyond the wing tips the air is nominally at atmospheric pressure. This causes a spanwise flow of air outwards away from the fuselage on the lower surface, and an inward flow towards the fuselage on the upper surface. The air now flows in both the chordwise and spanwise direction. At the trailing edge of the wing, where the two flows meet, a twisting motion is imparted to the air and a series of vortices are formed. These are known as trailing edge vortices (Fig. 4.32).

At the wing tips, a pressure gradient causes the air to flow towards the upper surface and, in conjunction with the forward velocity of the aircraft, large concentrated vortices are formed. These are known as wing tip vortices. These vortices move in a counter-rotating direction, and become progressively weaker towards the centre-line of the aircraft (Fig. 4.33).

The net effect of these vortices is to deflect the relative airflow downward behind the trailing edge, and within the wingspan. This effect is known as induced downwash. In practice the
size of the vortex determines the amount of down wash, and the larger the vortex the greater the down wash (Fig. 4.34).

The down wash deflects the airflow downwards from the horizontal through an angle; known as the **angle of induced downwash** ($\epsilon$). This effect is not only apparent behind the wing, but also influences the airflow approaching the wing by deflecting it upward from the horizontal through the same angle ($\alpha$). The resulting airflow is known as the **effective relative airflow**, and the angle it makes to the horizontal is better known as the **ineffective angle of attack** ($\alpha_i$) (Fig. 4.35).

The **resultant lift force**, which acts perpendicular to the local relative airflow, is also deflected rearward through the same angle, ($\alpha_i$). The angle of attack producing this lift force is known as the effective angle of attack ($\alpha_e$), which is the angle between the chord line and the effective relative airflow. Part of the lift now acts horizontally rearwards, and tends to retard the forward motion of the aircraft. This is known as **induced drag**, and will be covered in detail in another chapter. The amount of lift acting vertically upwards is known as the effective lift, and is determined by the size of the effective angle of attack. Hence the smaller the effective angle of attack, the greater the reduction in lift. The amount of lift developed by
the wing therefore decreases in the spanwise direction towards the wing tips, where the vortex, and hence downwash is strongest (Fig. 4.36).

In order to recover the lost lift the angle of attack must be further increased, giving an increased effective angle of attack. This however, causes a corresponding increase in the drag component.

**Wing Shape and its Effect on Lift**

The actual amount of lift generated by a wing depends on both the degree of induced downwash, caused by wing tip vortices, and its chordwise pressure distribution. The actual lift developed by a wing therefore depends, among other things, on its planform shape. Also since the wings on an aircraft are symmetrical about its centre line it is appropriate to consider the lift distribution about only one wing, referred to as a semi-span.

Fig. 4.37 shows how the lift distribution varies over the semi-spans of tapered, elliptical and rectangular wings. The lifting ability of a wing is also affected by its angle of attack. In
practice it is the effective angle of attack of each spanwise section that determines the actual amount of lift developed, and this varies depending on the strength of the wing tip vortices.

![Diagram of effective angle of attack for different wing planforms](image)

**FIG. 4.38**

Fig. 4.38 shows how the effective angle of attack of each section varies with distance from the aircraft’s centre line. Notice that the effective angle of attack of the rectangular planform wing remains fairly constant over the first 50% of the semi-span, but quickly reduces to zero degrees over the latter 50%. By comparison the tapered planform wing shows an increasing effective angle of attack over 70% of the semi-span and a reduction to zero degrees over the final 30%. This shows that a wing's chord length directly affects the size of a wing tip vortex, with the rectangular wing producing a much larger vortex than the tapered section. This is because a wider tip chord allows more air to flow towards the upper surface, so intensifying the size of the vortex.
Chapter 5.

Drag

Introduction

During flight, all of the parts of an aircraft exposed to the airflow produce an aerodynamic force, which opposes the forward motion of the aircraft. This force is known as drag, and is the air resistance experienced by an aircraft as it moves through the air (Fig. 5.1).

FIG. 5.1

Drag acts parallel to and in the same direction as the relative airflow. In steady level flight (SLF), drag is directly balanced by the thrust produced by an engine or propeller. It follows that, for a given airspeed, the lower the drag the less the thrust required to balance it (Fig. 5.2).

FIG. 5.2
Low drag is therefore beneficial since it leads to reduced fuel consumption and lower operating costs.

The total drag acting on an aircraft in flight comprises of:

- Profile drag
- Induced drag
- Interference drag

Profile drag can be further separated into:

- Form drag
- Skin friction drag

Form Drag

Form drag is produced whenever the streamline airflow passing over an aircraft separates from the surface and becomes turbulent. An example of extreme form drag is the effect of a flat plate placed at right angles to the airflow (Fig. 5.3).

The pressure immediately in front of the plate will be above atmospheric, whilst the pressure behind the plate will be below atmospheric. This results in a sucking effect behind the plate and the formation of vortices. The more rapidly the airflow changes direction, the greater the pressure gradient, the earlier the separation and the higher the form drag. It is therefore essential to delay the point at which separation of the airflow takes place and this is achieved by changing the shape or streamline of a given object (Fig. 5.4).
For example on aircraft, a fairing is often fitted around a fixed undercarriage leg to reduce form drag to an acceptable level (Fig. 5.5).

A relationship exists between the length (a) and maximum thickness (b) of a streamlined body, and its resultant form drag (Fig. 5.6).

The ratio of length to maximum thickness is called the fineness ratio:

\[
\text{Fineness Ratio} = \frac{a}{b}
\]

Streamline shapes, which give the least form drag at subsonic speeds normally, have a fineness ratio of 4 to 1, but these values may vary considerably without increasing drag to any great extent.
Additionally as air flows around an object there becomes a point where it is unable to remain on the surface, and instead separates away. This is known as its separation point (Fig. 5.7).

With increasing angles of attack the separation point moves steadily towards the leading edge and the further forward this occurs, the greater the form drag.

**Boundary Layer**

Before covering skin friction drag in detail it is important to firstly consider the layer of air closest to the surface. As air flows over a wing, the roughness of the surface and the viscous property of the air itself slow it down. Much like fluids, the more viscous the air the greater its retardation. At the surface the air particles will adhere to it and their relative velocity will reduce to zero (Fig. 5.8).

With distance from the surface the subsequent layers of air particles are slowed down due to friction effects between the particles, but are not completely brought to rest. The relative velocities of the air particles steadily increases with distance, until a point is reached where the particles do not slow up at all, and instead travel at the free stream velocity.
The layer of air between the surface and the free stream velocity, in which local retardation takes place, is known as the **Boundary Layer**. Like the main airflow, the boundary layer flow can be either laminar or turbulent in nature (Fig. 5.9).

**The Laminar Boundary Layer** is a very thin layer of smooth airflow. It consists of a series of laminations or smooth regular streamlines, in which the air particles do not intermingle.

**The Turbulent Boundary Layer** is a layer of disturbed or turbulent airflow, in which the streamlines break up. The air particles become intermingled and move in a random, irregular pattern. Notably, the turbulent boundary layer creates greater drag than the laminar boundary layer.

The usual tendency is for the boundary layer to start in a laminar condition near the leading edge of an aircraft wing, and then become turbulent. The change from laminar to turbulent flow takes place in the transition region. In fact the transformation from laminar to turbulent flow can be clearly seen in the smoke rising from a cigarette in still air (Fig. 5.10).

The boundary layer also increases in thickness as it moves rearwards over an aircraft wing, with the turbulent boundary layer being proportionally thicker than the laminar boundary layer under the same free stream velocity conditions (Fig. 5.11).
Following the transition from laminar to turbulent boundary layer, the boundary layer will thicken and grow at a more rapid rate. The maximum thickness of the boundary layer is however comparatively small, and in practice is only about 0.01 m in depth. It is also possible to compare the characteristics of laminar and turbulent boundary layer using velocity profiles (Fig. 5.12).

These profiles show the variation in boundary layer velocity with distance above a surface. Notice that the turbulent boundary layer has much higher local velocities immediately adjacent to the surface. The airflow in this region therefore possesses much higher kinetic energy than the laminar boundary layer at the same distance above the surface. The nature of the boundary layer is extremely important in aerodynamics, since it determines the maximum coefficient of lift, and the stalling characteristics of an aerofoil, (these will be explained later).
Skin Friction Drag

The reaction to the retardation of the airflow within the boundary layer is known as skin friction. In practice the amount of skin friction depends on the rate at which the air adjacent to the surface is trying to slide relative to it. The retarded air will try to drag the surface along with the flow, in much the same way as two solid surfaces sliding over each other do. This is known as shear stress, and is directly proportional to the speed of flow. The velocity profiles previously used to compare laminar and turbulent boundary layers also represent the shear stress patterns that exist between layers (Fig. 5.13).

![Skin Friction Drag](image)

FIG. 5.13

The gradual velocity change associated with the laminar boundary layer shows that low shear stresses exist near the surface, resulting in low skin friction drag. Conversely the rapid velocity change associated with the turbulent boundary layer is evidence of high skin friction drag.

If the conditions of flow were such that either a turbulent or laminar boundary layer could exist, laminar skin friction drag would be about one-third of that of the turbulent flow. Laminar boundary layers are therefore desirable, but the natural transition into a turbulent boundary layer prevents this occurring. The point where transition takes place is therefore important when determining the amount of skin friction drag that exists. Fig. 5.14 shows how the boundary layer develops on a typical aerofoil.

![Skin Friction Drag](image)

FIG. 5.14
Factors Affecting Skin Friction Drag

The velocity of the airflow and the surface condition over which it is flowing will ultimately effect the position of the transition point.

Effect of Speed. With increasing forward airspeed, the transition point moves progressively towards the leading edge resulting in a larger turbulent boundary layer. This causes greater skin friction drag.

Effect of Surface Roughness. If the upper surface of a wing has a roughened surface, it may result in a premature transition from a laminar to a turbulent boundary layer; for example the accumulation of ice (Fig. 5.15).

![FIG. 5.15](image)

This will also cause a large increase in skin friction drag. The degree of skin friction drag may be minimised by polishing and de-icing the surface. Since all of the aircraft skin is exposed to the airstream, this type of drag will affect all surfaces.

Interference Drag

When an aircraft is considered as a whole, the total drag acting on it may be greater than the sum of the drags of its individual components. This is a result of the airflow being greatly disturbed where the various components are joined together, principally between the wing and fuselage. The disturbance that is imparted to the airflow produces additional drag, known as Interference Drag (Fig. 5.16).
This type of drag occurs because a large pressure gradient is set up across the junction that causes the boundary layer to prematurely separate from the surface and form a turbulent wake. To minimise this effect suitably shaped fairing or fillet is placed over the intersection to encourage streamline flow, thereby reducing interference drag. Minimising interference drag is particularly important at high airspeeds.

**Note:** The sum of Profile and Interference Drag is sometimes referred to as Parasite Drag and is attributed to those parts of an aircraft that do not contribute to lift.

**Induced Drag**

Whenever a wing is producing lift concentrated vortices are formed at the wing tips. These vortices are strongest at the wing tips and become progressively weaker towards the centre-line of the aircraft (Fig. 5.17).

These vortices induce downwash to the airflow behind the wing, causing the lift vector to tilt rearwards. The horizontal component of lift opposes the forward flight of the aircraft, and is known as induced drag (Fig. 5.18).
The larger the vortex, the greater the induced downwash, and the greater the induced drag.

The formula for induced drag is:

$$\text{Induced Drag } (D_I) = \left( \frac{C_L^2}{\pi A} \right) \frac{1}{2} \rho V^2 S$$

where  
- $C_L$ = Coefficient Of Lift  
- $A$ = Aspect Ratio  
- $\rho$ = Density (kg/m$^3$)  
- $V$ = Velocity m/s  
- $S$ = Planform Surface Area (m$^2$)  

$$\left( \frac{C_L^2}{\pi A} \right)$$ = Coefficient of Induced Drag

**Factors Affecting Induced Drag**

From the induced drag formula the main factors affecting induced drag are wing planform, aspect ratio, speed and weight/lift.

**Effect of Planform.** This is the principal factor affecting induced drag. The size of the wing tip vortex is directly related to the length of the wing tip chord, and also the larger the vortexes, the lower the sectional effective angle of attack (Fig. 5.19).
Clearly a rectangular planform wing produces a much larger vortex than the tapered section. This is because the wider the tip chord, the greater the spillage of air from the lower surface onto the upper surface, and the larger the wing tip vortex. In aerodynamic terms, the elliptical planform wing is the most aerodynamically efficient because the downwash remains constant across the complete wingspan, giving minimal induced drag (Fig. 5.20).

From a practical point of view however, the manufacturing and structural problems associated with an elliptical planform wing preclude its use. For structural reasons, a straight tapered wing provides a good compromise, giving low induced drag (Fig. 5.21).
To preserve the aerodynamic efficiency, the resulting planform is usually tailored, by using wing twist and cross sectional variation, in order to obtain as near as possible the elliptical lift distribution.

**Effect of Aspect Ratio.** Another practical method of minimising induced drag is to make the wingspan as long as possible i.e. increase the aspect ratio. This has the effect of reducing the overall size of the wing tip vortices, and hence induced drag (Fig. 5.22).

![Figure 5.22](image)

Induced drag varies inversely with aspect ratio.

**Effect of Airspeed.** Induced drag is most significant at low airspeeds and high angles of attack i.e. during take-off and landing, when it can account for approximately three-quarters of the total drag. To maintain steady level flight, as the airspeed reduces and the angle of attack increases, the slower passage of air rearwards over the wing will increase the spanwise flow of air around the wing tip. This will result in larger wing tip vortices and greater induced drag (Fig. 5.23).
Induced drag is inversely proportional to airspeed².

**Effect of Lift and Weight.** Induced drag is a by-product of lift, so the greater the lift the greater the induced drag. There is no induced drag produced by an aerofoil at zero lift. Any increase in $C_L$, i.e. during manoeuvres or due to increased weight, will increase the amount of induced drag at a given airspeed. Induced drag varies as $C_L^2$, and therefore weight², at any given airspeed (Fig. 5.24).

**Methods to Reduce Induced Drag**

It is advantageous to reduce induced drag to a minimum, and this is achieved using the following design features to minimise the formation of wing tip vortices.

**Tapering the Wing in Planform.** By tapering the wing in planform towards the tip it reduces the amount of air flowing from the lower surface to the upper surface, thereby reducing the size of the wing tip vortex (Fig. 5.25).
Washout. With washout the wing is twisted so that the inboard wing section is at a higher angle of incidence and hence angle of attack compared to the wing tip (Fig. 5.26).

This ensures that most of the lift is generated on the inner part of the wing, thus minimising the leakage of airflow around the wing tips. This in turn reduces the size of the wing tip vortex and reduces the total amount of induced drag.

Wing Tip Modification. The wing tip may be modified to reduce the leakage of airflow around the wing tip and limit the size of the vortex. Some of the more typical designs are shown in Fig. 5.27.
Drag Formula

Like lift, a drag formula can also be derived. The drag acting on an aircraft depends on the following factors:

- Shape
- Angle of attack
- Air density (\( \rho \))
- Air velocity squared (free stream air velocity) (\( V^2 \))
- Wing planform surface area (S)

Dynamic pressure takes into account the air density and velocity, but when this pressure is combined with the wing planform surface area (S) it produces a force. This force is known as drag. Like lift, drag is not exactly equal to the dynamic pressure times the area, but varies with shape and angle of attack. These factors are represented by the Coefficient of Drag (\( C_D \)).

The general drag formula is thus:

\[
\text{Drag} = C_D \frac{1}{2} \rho V^2 S
\]

A graph of Coefficient of Drag against angle of attack illustrates how drag varies in flight (Fig. 5.28).
Drag Curves

Plotting graphs of profile or parasite drag and induced drag on the same axes shows the relationship between them (Fig. 5.29).

The two drags can be added together at any velocity to produce a total drag curve but it is essential to remember that this curve only applies to an aircraft of constant weight and configuration in level flight at any given altitude. Total drag becomes a minimum when the profile or parasite drag, and induced drag are equal.
This is known as the minimum drag point and occurs at the **minimum drag speed** ($V_{\text{IMD}}$). This is the speed where the required lift is developed with the minimum amount of drag, and is also the most economical speed at which an aircraft flies.

**Effect of Weight on the Drag Curves.** If the weight of an aircraft alters, a corresponding change in the coefficient of lift must occur if level flight is to be maintained at a given airspeed. It follows that since induced drag is proportional to $C_L^2$ (or weight$^2$), any change in weight will also alter the total drag curve (Fig. 5.30).

Any change in weight will move the point at which the induced drag and profile drag curves cross. This will lead to a change in the minimum drag speed; e.g. an increase in weight will increase $V_{\text{IMD}}$ as well as the total drag.

**Effect of High Drag Devices on the Drag Curves.** In flight it is sometimes necessary to decrease $V_{\text{IMD}}$ by deliberately increasing the total drag. This is achieved by using airbrakes or spoilers, which cause an increase in profile drag (Fig. 5.31).
The point at which the profile or parasite drag and induced drag curves cross will similarly alter and the minimum drag speed will reduce.

**Lift/Drag Ratio**

To determine the efficiency of an aircraft, it is necessary to consider the lift and drag curves together (Fig. 5.32).
For maximum efficiency the wings should produce maximum lift with the least possible drag. The lift curve shows that maximum lift is achieved at 15° angle of attack, whereas the drag curve shows minimum drag occurs at -2° angle of attack. Neither of these angles is satisfactory, since the ratio of lift to drag at both extremes is very low. In practice maximum lift at minimum drag, i.e. maximum lift/drag ratio (L/D Ratio), should occur at the same angle of attack. To establish where this occurs it is necessary to examine the Lift/Drag ratio at various angles of attack using the lift and drag formulae’s. The Lift/Drag ratio for an aerofoil at any selected angle of attack can be found using the following formula:

\[
\begin{align*}
\text{Lift} & = C_L \frac{1}{2} \rho V^2 S = C_L \\
\text{Drag} & = C_D \frac{1}{2} \rho V^2 S = C_D 
\end{align*}
\]

Notably the same result is obtained irrespective of whether the lift and drag, or their coefficients are used in the calculations. By plotting lift/drag ratio against angle of attack it is possible to establish where the most efficient angle of attack occurs.

Fig. 5.33 shows that the lift/drag ratio increases rapidly up to about 3 or 4°, at which point the lift is nearly 24 times the drag, but this figure varies depending on the type of aerofoil. For transport aircraft this value is typically 12–20, and for propeller powered trainer aircraft is typically 10-15.

At higher angles of attack the lift/drag ratio steadily reduces because, even though the Coefficient of Lift (C_L) continues to increase, the Coefficient of Drag (C_D) increases at a greater rate. In fact, at the stalling angle, lift may only be 10 to 12 times greater than drag.

The most important point on the lift/drag curve is the angle of attack that gives the best lift/drag ratio, in this case 3 or 4°. This is the most efficient (optimum) angle of attack at which the aerofoil gives its best all round performance, i.e. it produces the required lift for the
minimum cost in drag. At any other angle of attack the same lift will be obtained at a greater cost in drag.

In steady level flight since drag is balanced by thrust, it follows that, by minimising drag, thrust can also be minimised. This allows a smaller engine to be fitted, resulting in better fuel economy, and lower maintenance costs. In practice most aircraft are not fitted with an instrument which indicates angles of attack, so the pilot must rely on the airspeed indicator, since airspeed is related to angle of attack (Fig. 5.34).

![Diagram](image)

The minimum drag speed is therefore related to the angle of attack that gives the best lift/drag ratio, i.e. 3 or 4°. Consequently aircraft are flown at the minimum drag speed to give best all round performance, but remember this is only correct for a given weight, and any change in weight will necessitate a change in airspeed to maintain the best lift/drag ratio. The best lift/drag ratio is also unaffected by changes in altitude.
Chapter 6.

Flying Controls

Introduction

An aircraft can be rotated in flight about any one, or a combination of its three axes. These axes act at right angles to each other, and all pass through the aircraft's centre of gravity (Fig. 6.1).

Movement about the lateral axis is known as pitch, movement about the longitudinal axis is known as roll and movement about the normal axis is known as yaw. This is achieved via a primary flying control system, which in its basic form consists of moveable control surfaces linked by a series of cables and rods to controls in the cockpit (Fig. 6.2).
The primary control surfaces are the elevators, the ailerons and the rudder. These surfaces are hinged at the trailing edges of the main surfaces and are used to manoeuvre the aircraft about its three axes, producing both primary and secondary effects.

**Elevators**

The primary effect of elevators is to provide pitch control about the lateral axis (Fig. 6.3).

Pushing the control column forwards causes the elevator to move downwards. This produces an aerodynamic force acting on the tailplane in an upward direction causing the aircraft to pitch nose-down. Pulling the control column rearwards has the reverse effect and causes the aircraft to pitch nose-up. The elevators produce no real secondary effect on an aircraft, although changes in pitch attitude result in changes in angles of attack, and hence airspeed.

**The Stabilator**

On some aircraft the tailplane and elevator are combined into one surface, known as a stabilator, or an all-moving tailplane (Fig. 6.4).
Forward movement of the control column will cause the leading edge of the stabilator to rise, thereby generating a force that causes the tail to rise, and the aircraft's nose to drop. A rearward movement of the control column will have a reverse effect.

**The Rudder**

The primary effect of the rudder is to provide yaw control about the normal axis (Fig. 6.5).

If the right rudder pedal is moved forwards the rudder will move to the right. In flight this will produce an aerodynamic force on the fin and the aircraft will yaw to the right. If the left rudder pedal is moved forwards the reverse action will take place, and the aircraft will yaw to the left. The secondary effect of rudder is roll in the same direction as yaw (Fig. 6.6).
This occurs because the outer wing travels faster than the inner wing, thereby generating more lift.

Ailerons

The primary effect of ailerons is to provide roll control about the longitudinal axis (Fig. 6.7).

If the control column is moved to the right the ailerons will move in opposite directions by equal amounts; the right aileron will deflect upwards, and the left aileron will deflect downwards.
downward. This will locally alters the shape of the wing where the ailerons are attached. In flight this will produce a downward aerodynamic force on the right wing and an upward aerodynamic force on the left wing, causing the aircraft to roll to the right. If the control column is moved to the left, the reverse effects occur.

In principle a downward movement of the aileron causes an increase in the effective angle of attack and a corresponding increase in lift, whilst an upward movement of the aileron causes a reduction in the effective angle of attack and a decrease in lift. The difference in lift between the two wings produces the necessary rolling moment.

In addition to changes in lift the deflection of the ailerons also causes variations in drag. The down-going aileron will produce a predominance of induced drag, whilst the up-going aileron will produce a predominance of profile drag. At slow airspeeds the increase in drag will be greater on the down-going aileron than the up-going aileron and the aircraft will yaw in the opposite direction to the roll (Fig. 6.8).

This is the secondary effect of ailerons, and is known as adverse aileron yaw.

Adverse Aileron Yaw

In order to compensate for adverse aileron yaw, the drag produced by the ailerons must be equalised, which is achieved by fitting either Differential or Frise type ailerons.

**Differential Ailerons.** These are designed so that the upgoing aileron is deflected through a greater angle than the downgoing aileron (Fig. 6.9).
Frise Ailerons. These are designed so that the leading edge of the upgoing aileron projects beneath the wing (Fig. 6.10).

On some aircraft the two methods are combined together to form Differential/Frise type ailerons (Fig. 6.11).

Combined Primary Control Surfaces

Some types of aircraft have the primary flight control system arranged so that one type of control surface combines its function with that of another; e.g. Elevons, Ruddervators and Flaperons.

Elevons. These are fitted to the trailing edge of the wings on delta winged aircraft such as Concorde, and perform the function of both ailerons and elevators. When the control column is moved backwards or forwards the Elevons move like elevators, and deflect by equal amounts in the same direction. For example if the control column is moved rearwards the Elevons will deflect upwards, and the aircraft will pitch nose-up (Fig. 6.12).
If the control wheel is turned the Elevons on one wing will rise, and on the other wing will lower, as in the case of conventional ailerons. For example when the control wheel is turned to the right the right Elevons will rise and the left Elevons will lower, causing the aircraft to roll to the right (Fig. 6.13).

The control systems are also interconnected so that the surfaces can be deflected simultaneously to produce combined pitching and rolling moments.

**Ruddervators.** Ruddervators are fitted on light aircraft having a Vee or Butterfly tail. They combine the function of the rudder and elevators (Fig. 6.14).
the control column and rudder pedals. When functioning as elevators the Ruddervators move in unison in the same direction by equal amounts. For example pulling the control column rearwards causes the Ruddervators to move up, and the aircraft to pitch nose-up (Fig. 15).

![Diagram of Ruddervators functioning as elevators](image)

**FIG. 6.15**

When functioning as rudders the Ruddervators move by equal amounts in opposite directions. For example when pushing the right rudder pedal forwards the left Ruddervator will move up and the right Ruddervator will move down, causing the aircraft to yaw to the right (Fig. 6.16).

![Diagram of Ruddervators functioning as rudders](image)

**FIG. 6.16**

The control column and rudder pedal systems are also connected to the surfaces through a differential linkage or gearing arrangement, so that combined pitching and yawing moments can be obtained.

**Flaperons.** These are fitted to some light aircraft that are designed to operate from short runways. They combine the operation of ailerons and flaps to create a full span trailing edge flap. When lowered the flaperon is able to move up and down providing roll control whilst still contributing to the wing's overall lift.
Aerodynamic Balance

When the control surfaces are deflected, the product of the aerodynamic force acting through the centre of pressure of the surface and its distance from the hinge-line will produce an opposing moment (Fig. 6.17).

\[ \text{Hinge Moment} = FX \]

FIG. 6.17

This is known as the hinge moment of the control surface and its magnitude determines the amount of effort required by the pilot to maintain its position i.e., stick force. Stick force also depends on the method by which the control column is linked to the control surface. The ratio of stick movement to control surface deflection is known as stick-gearing (Fig. 6.18).

\[ \text{Stick Gearing} = \frac{\text{Stick Movement}}{\text{Control Surface Deflection}} \]

FIG. 6.18

If the stick forces are high some form of assistance will be needed to help move the control surface, and if the stick forces are too light the surface must be artificially loaded to increase the opposing moment. To achieve the necessary stick forces the control surfaces are aerodynamically balanced using one or more of the following methods:

Inset Hinge. With this method the hinge-line is placed inside the control surface. This reduces the length of the moment arm and therefore the size of the hinge moment, thus reducing the overall stick force (Fig. 6.19).
The amount of inset is normally limited to **20 - 25%** of the chord length to ensure that the centre of pressure does not move in front of the hinge-line at high deflection angles.

If the centre of pressure is allowed to move ahead of the hinge line the resulting hinge moment will no longer oppose the movement of the control surface, but will instead assist it (Fig. 6.20).

This is known as **control surface overbalance**, and is detected as a decrease instead of an increase in the progressive stick forces required to move the control surface through a given deflection angle. In this condition the control surface will automatically move towards full deflection and to stop this the control input should be reversed. This is known as **control reversibility**.

**Horn Balance.** This method is used on most control surfaces, but mainly on rudders and elevators (Fig. 6.21).
By design the area of the control surface ahead of the hinge-line is concentrated in one place to form a horn. As the surface is deflected the horn projects into the airflow and assists the movement ahead of the hinge-line. The action of the horn balance is similar to the inset hinge and reduces the overall stick force.

**Internal Balance.** This method is used on ailerons and elevators. It operates in conjunction with tabs to reduce the stick force, and unlike other methods is totally contained within the control surface (Fig. 6.22).

A hinged balance panel divides the area ahead of the control surface into two vented compartments. When the control surface is deflected, e.g. upward, the higher pressure developed in the upper compartment forces down on the balance panel producing a partial balancing moment, thereby reducing the overall stick force.
Tabs

Unlike the previous methods of aerodynamic balance tabs are small hinged surfaces forming part of the primary control surface. In its basic form the pilot does not directly control the tab, but its deflection angle is changed automatically whenever the main control surface is moved. These tabs are used to partially balance the aerodynamic load acting on a control surface, thereby reducing the overall stick force.

**Balance Tabs.** These tabs are sometimes incorporated as part of the elevator on conventional tailplanes. They are mechanically linked to the tailplane by a linkage that causes them to move in the opposite direction to the control surface (Fig. 6.23).

![Fig. 6.23](image-url)

For example if the control column is moved forwards the elevator will move downwards and the balance tab will move upwards. The resultant aerodynamic force acting on the tab will produce a balancing moment, and reduce the overall stick force.

**Anti-Balance Tabs.** These tabs are fitted on aircraft, when it is necessary to increase the stick force. This is because small movements of a control surface can produce large aerodynamic loads, and may lead to over control (Fig. 6.24).

![Fig. 6.24](image-url)
These tabs operate in the same manner as balance tabs except that they move in the same direction as the control surface to increase the stick force, i.e. control surface down, tab down.

**Servo Tab.** In this design the tab is directed controlled by the pilot through a pivot point, and is deflected to supply the hinge moment necessary to move the main control surface (Fig. 6.25).

![Diagram of Servo Tab](Fig. 6.25)

Movement of the tab provides an aerodynamic force that produces a hinge moment about the hinge-line of the control surface. This causes the control surface to move to a new position of equilibrium in a direction of travel opposite to that of the tab i.e. **tab down, control surface up**. The stick forces involved are therefore determined by the hinge moments acting on the tab. In practice the servo tab lacks effectiveness at low airspeeds when large control deflections are required. This is because the amount of airflow passing over the tab is too low to produce the necessary hinge-moment, and hence the required deflection.

**Spring Servo Tab.** This type of tab overcomes the low speed problems associated with a servo tab by including a spring box in the system (Fig. 6.25).

![Diagram of Spring Servo Tab](Fig. 6.25)

The spring tension is such that the tab will not come into operation until the stick force exceeds a predetermined value. At low airspeeds the spring tension will prevent movement of the servo tab and any control input by the pilot will move the control surface and tab as one piece. At higher airspeeds the springs will compress and the
tab will move by way of the pivot point in the opposite direction to the control surface, providing the necessary aerodynamic assistance.

Mass Balance

During flight the main control surfaces may suffer from vibration. This condition is better known as flutter. It is caused by the combined effects of changes in the pressure distribution around the control surface with changing angles of attack (aerodynamic forces), and the forces due to the elastic nature of the aircraft structure (aeroelastic forces) itself. If these forces become coincident, or act in phase with each other the resultant oscillations will quickly increase in amplitude, and if left unchecked may ultimately lead to structural failure.

To help eliminate flutter in flight all manually operated control surfaces are generally mass balanced. This is achieved by attaching weights forward of the hinge-line so that the centre of gravity acts through the hinge-line, thus altering the period of vibration and the liability to flutter. These additional weights are usually stored internally along the leading edge of the control surface, inside the horn balance, or on an arm attached to the surface (Fig. 6.26).

![FIG. 6.26](image)

Powered Flying Controls

Powered flying controls are used on most transport category aircraft to provide assistance in moving the primary and secondary control surfaces against the large aerodynamic loads, which may exceed the physical capabilities of the pilot at high airspeeds.

The primary flying control surfaces are arranged in the same configuration as on light aircraft, with ailerons, elevators and a rudder, although some differ because they are additionally fitted with inboard ailerons (Fig. 6.27).
The control surfaces are hydraulically activated and are powered from the aircraft’s main hydraulic systems. Due to the importance of the flying control systems the surfaces are also normally powered by at least two independent hydraulic systems (Fig. 6.28).

Each system is primarily pressurised in flight by engine driven pumps, or alternatively by electrically driven pumps, but for emergency purposes are normally backed up by either an electrical pump, or by a Ram Air Turbine (RAT) (Fig. 6.29).
Powered Flying Control System Layout and Requirements

A basic powered flying control system comprises of the following components:-

- A control input system
- A power control unit
- An artificial feel system

Irrespective of their design all powered flying control systems are regulated by the Joint Airworthiness Requirements (JARs), and must comply with the following standards:-

**Sense.** The aircraft must move in the direction signified by the control input, e.g. control column back, pitch nose-up.

**Rigidity.** The control system must be strong enough to withstand any operating loads without excessive distortion, e.g. airloads on the control surfaces (irreversibility).

**Stability.** The control surfaces must remain where selected by the pilot and must not be affected by signals which are not self initiated, e.g. vibration and aerodynamic loads.

**Sensitivity.** There must be immediate response at the control surfaces to the pilots input signals.

**Safety.** Passengers, cargo and loose articles must safeguard the control system against jamming, chafing, and interference. Guards must therefore be fitted where appropriate to provide the necessary protection.

**Fail-Safe.** The control system must be duplicated or be capable of manual operation in the event of hydraulic power failure.
Control Input Systems

This is the method by which signals from the flight deck controls are relayed through an aircraft to position a servo control valve, which in turn manually determines the position of a flying control surface via a hydraulic actuator. In operation movements of the control column or rudder pedals are passed to the servo valve by one of the following methods:

**Hydro-Mechanical.** In this system the control signals are relayed through a series of cables and linkages to mechanically position the servo valve (Fig. 6.30).

![FIG. 6.30](image)

**Electro-Hydraulic.** In this system the control signals are measured by electrical transducers, whose output is amplified and then relayed to electrically position the servo valve (Fig. 6.31).

![FIG. 6.31](image)

This is commonly known as a fly-by-wire (FBW) system. In some aircraft the application of this system is limited to the control of only certain flying control surfaces, e.g. spoiler control panels in the case of the Boeing 767 or Airbus 320.
The Power Control Unit (PCU)

The power control unit is the main component in a power operated control system and provides all of the force necessary to move a control surface, with the pilot only having to supply a small force to operate a servo valve (Fig. 6.32).

![Diagram of the Power Control Unit (PCU)](image)

It consists of a jack ram/piston arrangement, which is fixed to the aircraft structure, hydraulic fluid, inlet/outlet ports, and a jack body. These parts form a hydraulic actuator, which is controlled by a servo (control) valve and is connected via a control run to the flight deck controls. When the valve is displaced in either direction from its neutral position it allows hydraulic fluid under pressure to pass to one side of the piston, and opens a return path from the other side. For example a rearward movement of the control column will cause the servo valve to move to the left (Fig. 6.33).
Since the jack is fixed in position the resulting pressure differential across the piston will cause the jack body to move to the left, which in turn will deflect the control surface upwards via a mechanical linkage. The body continues to move until it centralises itself on the servo valve, i.e. returning it to its neutral position (Fig. 6.34).

At this point the hydraulic fluid will be trapped either side of the jack and will form a hydraulic lock. This in turn will maintain the control surface rigidly in its selected position, and it will continue to remain so, irrespective of the aerodynamic loads acting on it, until the servo valve is repositioned by further flight deck control inputs. This is alternatively known as an irreversible control system. Conversely if the control column is moved forward the sequence of operations will be reversed; i.e. if the servo valve moves to the right, the jack body will move to the right, and the control surface will deflect downwards. Some power
control units also operate in response to electrical inputs from the Autopilot and Autostabilisation systems when they are engaged.

**Artificial Feel Systems**

In a manually operated flying control system the aerodynamic loads acting on a control surface are fed directly back through the control runs to provide stick force or feel on the flight deck controls. The loads thus vary depending on control surface deflection and airspeed. In the case of a power operated flying control system there is however no direct linkage between the control surface and the flight deck controls. In fact the only force felt is that associated with the movement of a servo valve, and the effort provided by the pilot therefore bears no direct relationship to the actual loads acting on the control surface.

These loads are alternatively dissipated through the aircraft structure via the body of a dedicated power control unit, thereby relieving the pilot of all control loads. Consequently, to prevent over-controlling and overstressing of an aircraft, some form of artificial feel is incorporated in the control system, so that the forces experienced are representative of a manually controlled aircraft. A suitable feel unit must therefore be capable of producing an opposing force that varies with airspeed and control surface deflection.

On transport category aircraft the requisite feel forces are provided by; spring or Pitot-static Q feel units, or in some cases a combination of both. Artificial feel systems normally also incorporate a self centring mechanism, so that if the flight deck controls are released they will automatically return to their neutral position, and will also centralise the control surface.

**A Simple Spring Feel Unit.** This is the simplest form of artificial feel unit and is normally fitted in the operating linkage between the flight deck controls and the power control unit (Fig. 6.35).

It is designed so that any flight deck control movement is firstly made against spring tension, so the larger the movement, the greater the opposing spring force. For example if the control column is moved rearward the left-hand side of the spring in the feel unit will be compressed in proportion to the control column movement and subsequent deflection of the control surface. When the control column is centralised the spring unit off-loads itself, thereby centralising the linkage and returning the flying control surface to its neutral position. This type of feel unit by itself may be adequate at low airspeeds, but at higher airspeeds greater resistance to flight deck control movement is needed to prevent over-stressing the aircraft. This is because the amount of feel only varies in proportion to control surface deflection, and takes no account of airspeed. On transport category aircraft this type of feel unit is only normally used by itself in aileron control systems.
Q Feel Units. These units like spring feel units are fitted in the operating linkage between the flight deck controls and the power control unit (Fig. 6.36)

A basic Q feel unit consists of a diaphragm with static pressure acting on one side and pitot pressure on the other, with the difference between the two being dynamic pressure. The unit is also arranged so that movement of the flight deck controls in either direction deflects the diaphragm against pitot pressure (Fig. 6.37).
For example if the dynamic pressure increases due to an increase in forward airspeed (IAS), the forces required to move the flight deck controls would similarly increase. Conversely, a reduction in forward airspeed will cause the load on the flight deck controls to decrease. This system therefore ensures that the stick forces vary during flight in proportion to varying loads acting on the control surfaces. These units however tend to be very large, so the sensing pressures are alternatively used to operate a piston subjected to hydraulic pressure, thereby providing hydraulic Q feel (Fig. 6.38).

In this system artificial feel is supplied hydraulically, enabling the unit itself to be much smaller. Like the spring feel unit the Q feel unit also incorporates a self-centring mechanism that operates when the flight deck controls are released. In practice this type of unit is typically used on most transport category aircraft in the rudder and elevator control systems, but is usually operated in conjunction with a spring feel unit (Fig. 6.39).
Two feel unit’s normally act together to resist movement of the flight deck controls from their neutral position.

Trimming Control Systems

The trimming control system is principally designed to reduce the stick forces or control forces to zero. This allows an aircraft to maintain any yaw, pitch or roll attitude set by the pilot without further control input. On small aircraft this system comprises of moveable or fixed auxiliary surfaces. These surfaces are called trim tabs and are normally hinged at the trailing edge of the primary control surfaces (Fig. 6.40). Most light aircraft are fitted with elevator and rudder trim, but aileron trim is normally only fitted on more sophisticated types of aircraft.
Principle of a Trim Tab

In operation a trim tab creates a hinge moment which exactly balances the hinge moment produced by a control surface (Fig. 6.41).

In this condition the control surface remains in its set position without any effort from the pilot, i.e. the control forces are zero.

**Moveable Trim Tabs.** These trim tabs are normally fitted on elevator and rudder control surfaces. In each case the tabs are connected via a cable and gearing system to trim wheels in the cockpit. For example consider the operation of an elevator trim tab (Fig. 6.42).
In this system the trim wheel is mounted to give movement about a lateral axis and is rotated in the natural sense to give the required pitch trim change. A forward movement of the trim wheel will produce a **nose-down trim change**, and vice versa (Fig. 6.43).

Notice that the trim tab moves in the opposite direction to the control surface. In practice if it is necessary to trim the aircraft in a given pitch attitude the elevator should **firstly** be moved to produce the desired pitch. The force necessary to maintain this pitch is then eliminated by rotating the trim wheel in the same direction as the control column until the stick loading reduces to zero.

**Note:** Trim tab deflection reduces the maximum available elevator authority.
Rudder trim is provided in a similar manner to elevator trim, except in this instance the trim wheel is mounted so that it rotates about a normal axis. For example to provide nose right trim the trim wheel is rotated in the clockwise direction and vice versa (Fig. 6.44).

On some light aircraft the trim tabs are moved electrically instead of mechanically. In this case the switch is normally spring-loaded towards the central off position and in order to reduce the stick forces to zero the switch is operated in a natural sense. When the switch is released it is designed to return to the ‘Off’ position.

Note: irrespective of the positioning method, the tab will remain in the same fixed position relative to the control surface until it is necessary to re-trim the aircraft in a new attitude.

Fixed Trim Tabs. These tabs operate completely independently of the pilot and can only be adjusted on the ground (Fig. 6.45).
Their actual setting is determined by flight tests and when they are set to give no resultant stick forces, the trailing position of the control surfaces is governed by the actual deflection of the tab. On light aircraft this type of tab is only normally fitted on ailerons to make wing level flight more easily achievable without having to maintain a constant stick force, i.e., to correct for a wing-low tendency.

**Combined Trim/anti-balance Tabs.** On some aircraft the tab has a **dual function**, and can operate as either a trim tab, or as an anti-balance tab, for example an all-moving tailplane. (Fig. 6.47).

![Diagram of ALL MOVING TAILPLANE ANTI-BALANCE/TRIM TAB PIVOT POINT](image)

**FIG. 6.47**

When used to provide elevator trim the tab is positioned by way of a trim wheel until zero stick force is achieved.

**Trimming of Powered Flying Controls**

On aircraft fitted with powered flying controls the position of the control surfaces is not affected by aerodynamic forces, and is only altered by movement of the appropriate servo valve in response to flight deck control inputs. Thus in order to correct for any out of trim condition a device is fitted which re-positions the neutral setting of the servo valve, and moves the control surface to a new neutral position. The flight deck controls similarly take up a new neutral position in the direction of the required trim. These devices are fitted in the control-input system, and consist of an electrical or mechanical linear actuator, controlled from the flight deck (Fig. 6.48).
When a trim actuator is operated it alters the effective length of the input lever to the servo valve, thereby making a selection, and moving the control surface to a new neutral position. A protection device in the form of a spring strut is also fitted between the trim actuator and the control input linkage, which normally operates as a fixed member, but should the servo valve seize a spring inside the unit compresses or extends to protect the valve from further damage.

**Aileron and Rudder Trim.** On most transport category aircraft aileron and rudder trim is applied through the movement of electrically operated trim switches. These switches are normally positioned on the centre pedestal and are spring loaded to their central ‘off’ position (Fig. 6.49).

To provide aileron trim both switches are simultaneously moved in the same direction to provide system integrity. The amounts of aileron and rudder trim applied are conventionally displayed on dedicated trim indicators. The rudder trim indicator is normally sighted on the centre pedestal, whilst aileron trim indicators are normally sighted on each control column.
Variable Incidence Horizontal Stabiliser. On most transport category aircraft varying the angle of incidence of the horizontal stabiliser provides pitch trim. This has the same effect as that achieved by movement of the elevator, but is aerodynamically more efficient, particularly at high airspeeds, and if the required trim range is considerable. The angle of incidence of the stabiliser is varied by an actuator assembly which is positioned near the leading edge and normally operates in conjunction with the elevator, the combination of the two providing the least trim drag (Fig. 6.50).

![Variable Incidence Horizontal Stabiliser](image)

**FIG. 6.50**

The leading edge of the stabiliser is normally moved up or down in flight by an electrically signalled hydraulic trim motor. This is in response to signals primarily from the electrical trim switches that are situated on the control column, or alternatively in response to electrical signals from the Autopilot pitch channel (Fig. 6.51)

Like the aileron trim system, pitch trim is provided by the simultaneous movement of two switches, located on each control wheel. This provides system integrity and prevents pitch trim runaway. The rate of trim varies and is controlled by trim control modules. The trim rate decreases with increasing IAS.

In the event of a malfunction in any of these methods pitch trim is alternatively provided by manually positioning the hydraulic servo valves via a series of control cables and pulleys. Inputs are normally provided by trim wheels, sited on either side of the centre console on the flight deck, and correspondingly rotate together (Fig. 6.52).
FIG. 6.51
The amount of stabiliser trim is indicated by a pointer, which forms an integral part of the trim wheel. The pointer moves up and down a fixed scale adjacent to it and a green band, which indicates the normal, trim settings for take off. The wheels also move whenever an electrical input is made, by way of feedback through the mechanical linkage.

Some aircraft are alternatively fitted with stabiliser trim levers, eg. Boeing 757. These are sited on the centre console, and like trim wheels pass manual pitch trim control signals to the pitch trim servo motors (Fig. 6.53).
To vary the amount of pitch trim, **both levers must be moved simultaneously together in the same direction.** This increases the overall system integrity, and **prevents** the possibility of **pitch trim runaway.** This system is also designed so that signals from the trim levers override all other trim inputs, and unlike trim wheels, they **do not move in response to electrical trim inputs.** On aircraft fitted with this facility all stabiliser trim indications are alternatively displayed on **two dedicated indicators,** which are normally sited on the centre pedestal (Fig. 6.54).
Spoilers

Spoilers are flap type control surfaces, which are normally, located on the upper surface of the wing, just in front of the trailing edge flaps (Fig. 6.55).

These surfaces are individually hinged at their leading edges and are actuated by hydraulic power supplied by dedicated power control units. As their name implies the main
purpose of the surfaces is to disturb the smooth airflow over the top of the wing, thereby reducing the wings lifting capability, and when fully extended considerably increasing aircraft drag. The spoilers are positioned so that aircraft pitch trim is not adversely affected by their deployment and the surfaces are extended in various sequences depending upon the mode of flight. They are used as:

**Roll Spoilers.** In this mode the spoilers operate asymmetrically in flight whenever the control wheel is rotated to assist the ailerons in providing roll control, particularly at high airspeeds. When this occurs the appropriate spoiler servo valves are signalled by way of the aileron operating system and the requisite surfaces are deflected upwards in proportion to the roll input. During a roll the spoilers on the lowering wing are deflected upwards, so decreasing its overall lifting capability and increasing the aircraft’s roll rate, whilst the spoilers on the opposite wing remain retracted (Fig. 6.56).

The innermost spoiler on the lowering wing also remains retracted to prevent tail buffet, and degraded pitch control.

On aircraft fitted with two sets of ailerons, (inboard and outboard), as the airspeed increases the aerodynamic loads on the ailerons tend to twist the wing at the tips, where it is more flexible. To overcome this tendency some aircraft use the technique of locking the outboard ailerons in the faired or neutral position, and use an inboard aileron/spoiler combination above flap retraction speeds to provide the necessary roll control (Fig. 6.57).
**Flight Spoilers.** In this mode the spoilers are operated symmetrically about the aircraft’s longitudinal axis to slow an aircraft down in flight (speed brakes). Their deflection angle is determined from the flight deck in response to movements of a speedbrake lever, which is typically located on the centre pedestal on transport category aircraft (Fig. 6.58).
With all spoilers fully retracted the lever is held in its down (DETENT) position to prevent inadvertent operation (Fig. 6.59).

Movement of the lever from this position signals the flight spoilers to rise and they reach their maximum attainable in flight deflection angles with the lever in its flight detent position (Fig. 6.60).

Movement of the speed brake lever for in-flight use is normally limited by a solenoid actuated stop.
In this mode the outboard spoilers usually remain retracted to prevent the aircraft pitching nose-up whilst the innermost spoilers are deflected by a lesser amount to prevent tail buffet. There are sometimes occasions in flight when both airbrake and roll commands occur together. On these occasions both inputs are fed into a complex box, containing a mixture of levers, bell cranks and quadrants, called a spoiler mixer unit (Fig. 6.61).

This unit sums both inputs and gives a revised output, which in turn varies the movement of the spoilers during an aileron input depending upon the amount of speedbrake selected (Fig. 6.62).
Spoilers in this role can normally be used at any airspeed but at increasingly higher airspeeds they are forced down (blowback) progressively.

**Ground Spoilers (lift dump).** In this mode the spoiler panels on both wings automatically rise to their full extension after touchdown to increase an aircraft’s rate of retardation (ground spoilers), when certain ground conditions are fulfilled (Fig. 6.63).

These conditions are typically:-

- Speedbrake lever in the **armed** position.
- Aircraft **weight on the undercarriage** (through the air/gound sensing system).
- All thrust levers in their **idle** positions.
- Aircraft **wheels rotating** (provides a time delay and ensures the aircraft is on the ground).

As the spoilers deploy the speedbrake lever automatically moves to the up position in line with their movement (Fig. 6.64).

![Fig. 6.64](image)

The maximum deflection angles are greater in the ground mode than the flight mode. With the spoilers in their fully extended position approximately 80% of the wing/flap lift is destroyed and the aerodynamic drag of the aircraft more than doubles. The subsequent loss of lift causes the aircraft to fully settle on the main undercarriage and increases its potential braking force. The flaps are also left in their landing configuration because of the drag benefits on deceleration. Should any of the **thrust levers be advanced** the speedbrake lever automatically moves to the down position, and the spoilers retract.
Chapter 7.

Lift Augmentation

Basic Lift Augmentation System

The lift augmentation system on light aircraft usually consists of two control surfaces, which are fitted inboard of the ailerons, along the trailing edge of the wing (Fig. 7.1).

These surfaces are known as trailing edge flaps, and are lowered in unison to primarily increase the wings lifting capability at any given angle of attack (Fig. 7.2).
The lift required to support a given weight can therefore be developed at a lower airspeed.

**Trailing Edge Flaps**

Trailing edge flaps are normally extended or deployed during the **take-off and landing** phases of flight when low airspeeds are beneficial. When the flaps are no longer required they are returned to their neutral position and the wing regains its former aerodynamic characteristics. The increase in lift is mainly attributed to the **variation in effective camber** that occurs when the flaps are deflected. This alters the lift curve and also produces a **higher maximum coefficient of lift** (Fig. 7.3).

In some instances the trailing edge flaps may also increase the wings surface area. When the trailing edge flaps are extended it alters the pressure distribution around the wing (Fig. 7.4).
It alters the pressure distribution not only over the rear of the wing, where the flap is situated, but also over the front of the wing. The majority of the additional lift is developed over the rear of the wing, and results in the **centre of pressure moving aft** as the flaps are lowered (Fig. 7.5).

![FIG. 7.5](image)

This also alters the **lift/weight couple** and produces a **nose down pitching moment**, which must be corrected for whenever the flap setting is changed. Any flap deflection increases the effective camber of the wing and affects the coefficient of drag (Fig. 7.6).

![FIG. 7.6](image)

**Types of Trailing Edge Flaps**

Several different types of trailing edge flap are used on light aircraft but the most common types are as follows:

**Plain Flap.** A simple hinged portion of the **trailing edge** of the wing (Fig. 7.7).
The flap increases the wings effective camber and alters the curvature of the wings upper surface. The increase in curvature causes earlier separation of the boundary layer and increases form drag.

**Split Flap.** A plate which is hinged to, and set into the lower surface of the wing trailing edge (Fig. 7.8).

When deflected the wings effective camber is increased, but the curvature of the upper surface remains unchanged. This produces a large turbulent wake at low angles of attack and hence drag, but provides better lift performance than the plain flap at high angles of attack. This is because the less curved upper surface delays the separation of the boundary layer.

**Slotted Flap.** This flap is similar to the plain flap except that when it is deflected a slot forms between the flap and main wing (Fig. 7.9).
This allows high-pressure air below the wing to flow through the slot and re-energise the boundary layer over the upper surface of the flap. The combination of variable geometry and boundary layer control thus increases the wing’s lift performance beyond that of the plain flap at all angles of attack.

**Fowler Flap.** This flap arrangement is similar to the slotted flap, except it firstly moves aft along rollers in a track before being deflected downward (Fig. 7.10).

The rearward movement of the flap increases the wing chord and increases the overall effective wing area. This enhances the wing’s lift capability without any flap deflection, but the resulting reduction in the thickness-chord ratio, causes the wing to stall at a lower angle of attack. The increase in drag is also small compared to the other types of flaps, because of the slot effect, and the wing’s reduced thickness-chord ratio.

**Comparison of Different Types of Trailing Edge Flap**

Trailing edge flaps not only alter the wing’s coefficient of lift \( C_L \), but they also alter its coefficient of drag \( C_D \). It is therefore possible to compare the efficiency of the various types of trailing edge flap by plotting \( C_L \) against \( C_D \) curves (Fig. 7.11).
By drawing tangents to the curves it is also possible to compare the lift/drag ratio, and hence the efficiencies of each type of flap. The maximum lift/drag ratio in each case occurs where the line touches the curve, and this shows that the Fowler flap produces the largest amount of lift for the least amount of drag, i.e. it has the best lift/drag ratio. The gradient of the tangents also shows how efficient each type of flap is, and the steeper the gradient, the more efficient the flap.

**The Effect of Trailing Edge Flaps on the Stalling Angle**

The greater the flap deflection, the lower the stalling angle of attack. This is because the more cambered the wing, the greater the adverse pressure gradient, and the earlier boundary layer separation will occur. By comparison with a clean wing it will also stall at a lower aircraft pitch attitude (Fig. 7.12).
This is because the chord line with the flap deflected alters its position relative to the wing region. This is known as the effective chord line. In practice, however, the stalling angle is always referenced to the chord line of the original clean wing, which acts as a datum line. Thus the greater the flap deflection, the steeper the effective chord line, and the lower the stalling angle of attack. The wing actually stalls when the angle between the effective chord line and the relative airflow, i.e. the effective angle of attack, reaches its normal stalling value of 15° or 16° in the case of light aircraft.

**The Effect of Trailing Edge Flaps on the Stall Speed**

The amount of lift developed at the stall by a wing at any given attitude is dependent on the maximum coefficient of lift, and the indicated airspeed squared. Thus for a given aircraft weight and hence lift, the resulting increase in the maximum coefficient of lift with flaps deflected will produce a lower stalling speed (Fig. 7.13).

![FIG. 7.13](image)

The stalling speed depends on the amount of flap deflection and the greater the deflection, the lower the stalling speed, which is due to the variation in the maximum coefficient of lift (Fig. 7.14).

![FIG. 7.14](image)
Operation of Trailing Edge Flaps

On light aircraft the trailing edge flaps are either electrically or manually operated. In the manually operated flap system the flaps are controlled by a series of cables and a flap lever, which is normally positioned between the pilots seats (Fig. 7.15).

The flaps are raised or lowered by way of the flap lever, which operates in a similar manner to the handbrake on a car (Fig. 7.16).
To extend the flaps the lever is pulled upwards through a series of ratchet settings, each setting relating to a fixed angular deflection, i.e. 10°, 25° and 40°. The first two settings are normally referred to as take-off settings and the latter as the landing setting, which is clearly indicated on the housing. To retract the flaps the button on the end of the lever is pushed in and the lever moved downward. In the case of an electrically operated flap system the flap setting is determined by the position of a flap selector switch, which is normally positioned on the instrument panel. When the switch is moved to the desired flap setting an electric motor operates until the pre-selected flap position is reached, and a micro-switch then cuts off the current supply to the motor. The actual flap position is usually displayed on an indicator positioned beside the switch.

Use of Trailing Edge Flaps for Take-off

For take-off purposes the flaps are normally lowered to a position, which provides the best lift-drag ratio that can be obtained with the flaps in any position other than fully, up. Since the required lift can be obtained at a lower airspeed the take-off run will also be reduced (Fig. 7.17).

Larger amounts of flap will cause a significant increase in drag, which will greatly reduce the acceleration, and increase the take-off run. The reduced stalling angle of attack and increased drag associated with flaps will also reduce the rate and angle of climb.

The Effects of Raising the Flaps in Flight

Shortly after the take-off with the aircraft accelerating and climbing, the action of raising the flaps will cause an immediate reduction in the coefficient of lift. Unless this is counter-acted by increasing the angle of attack the aircraft will momentarily lose altitude or sink until it has accelerated to an airspeed that offsets the effect of the reduction in lift. The more efficient the flaps the greater the reduction in lift and the more corrective action required in order to prevent a loss of altitude. Thus it is recommended that the flaps are raised in stages to enable the coefficient of lift to reduce gradually, and avoid any marked or exaggerated corrections, particularly when the aircraft is heavily loaded.
The Use of Trailing Edge Flaps During the Approach and Landing

When the flaps are initially lowered in flight the subsequent increase in the coefficient of lift produces lift in excess of that required to support the weight of the aircraft. Unless the aircraft’s nose is lowered to decrease the angle of attack, and hence lift, the aircraft will momentarily experience an unpleasant climb tendency known as ballooning (Fig. 7.18).

This effect is only short-lived because the subsequent increase in drag associated with the flap deflection quickly slows the aircraft down, and the excess lift reduces. Ballooning can however be prevented if the aircraft's nose is lowered when the flaps are deflected. Once the aircraft has returned to its former equilibrium state the aircraft will naturally settle in a nose-down pitch attitude due to the rearward movement of the centre of pressure. This ultimately provides improved visibility, which is especially important during the approach and landing phases of flight (Fig. 7.19).

In the landing configuration the flaps are normally fully extended to achieve the greatest increase in the coefficient of lift at any given angle of attack. This results in a significant reduction in the stall speed, and hence landing speed. The landing speed in this configuration must however be at least 1.3 times the stalling speed (1.3 Vs) to provide adequate controllability. It is also important that the flaps are not lowered at an airspeed greater than the maximum flaps extended speed, $V_{FE}$. If the flaps extend at different rates, flap asymmetry will occur, and a rolling moment will be set up.
The increase in drag associated with flap deflection will also **require an increased power setting** in order to maintain a given airspeed and attitude, or maintain a steady rate of descent. The reduction in the lift-drag ratio with flaps lowered also affects an aircraft's glide performance.

**High Lift Devices on Transport Category Aircraft**

With the introduction of larger and heavier aircraft it has become necessary to design a wing which provides good high speed cruise performance for economic purposes, but also possesses good low speed handling characteristics for the landing and take-off phases of flight.

In practice however, no single wing shape satisfies both these requirements; thick well cambered wings needed for low speed flight produce unacceptable profile drag penalties at high speeds, whilst thin wings with little camber lead to high stalling speeds and excessive take-off and landing speeds.

Thus to satisfy both needs high lift devices are fitted to the **leading and trailing edges** of the basic wing section (Fig. 7.20).

![Diagram of FLAPS RETRACTED and FLAPS FULLY EXTENDED](image)

All aircraft are fitted with trailing edge flaps, but most transport category aircraft sweptback planform wings are additionally fitted with leading edge high lift devices to further enhance the wings lifting capability at low airspeeds. The devices most commonly used are flaps, slats and slots, e.g. Boeing 757, which is fitted with **trailing edge flaps and leading edge slats** (Fig. 7.21).
Other types of aircraft are alternatively fitted with leading edge flaps, whilst others employ a combination of leading edge flaps and leading edge slats, but the actual configuration is determined by the manufacturer. Trailing edge flaps are normally fitted at inboard and outboard positions along the wing. These are normally slotted Fowler flaps and are configured during the landing and take-off phases of flight to provide the requisite lift/drag characteristics (Fig. 7.22).

For take-off purposes the trailing edge flaps are normally set to provide the best lift/drag ratio other than that associated with a clean wing, by only increasing the wings surface area. When the flaps are set to the landing configuration they not only increase the maximum coefficient of lift, but also increase drag, and help to retard the aircraft.

**Leading Edge High Lift Devices**

High lift devices applied to the leading edge are primarily designed to delay separation of boundary layer from the upper surface of the wing (stall) to a higher angle of attack. This is done by increasing the energy of the boundary layer, thereby maintaining a smoother flow for longer and reducing the airflow's tendency to become turbulent, or by making the leading edge more rounded. The devices used to achieve this are:
**Leading Edge Slats.** These are movable control surfaces attached to the leading edges of the wings along the complete span (Fig. 7.23).

![Diagram of leading edge slats]

When the slat is closed (retracted) it forms the leading edge of the wing, but in the open position (extended) a slot is created between the slat and the upper surface of the wings leading edge (Fig. 7.24).

![Diagram of slat extended]

This allows air to pass through the slot from the high-pressure region below the wing into the low-pressure region above the wing, thereby accelerating the flow by the venturi effect and re-energising the boundary layer. This delays its separation from the upper surface and **substantially increases the wings overall lifting capability** ($C_L$), by delaying the stall until a higher angle of attack (Fig. 7.25).
The subsequent increase in the maximum coefficient of lift like other high lift devices similarly lowers the aircraft’s stalling speed. The deployment of slats may increase the maximum coefficient of lift by more than 70%, and the stalling angle of attack from 15° to 22°. When operating at high angles of attack the slat has no significant effect on the wings camber, but it affects the pressure distribution over the upper surface of the wing (Fig. 7.26).

This produces a more gradual pressure gradient, and even at moderate angles of attack, enables the boundary layer to penetrate almost the full chord of the wing before separation takes place. This results in a stronger pressure distribution than that obtainable from a wing without slats. The deployment of the slats also affects the airflow around the wing (Fig. 7.27).
The slats are normally arranged in sections along the leading edge, so the combined effect of the airflow through each slot will reduce the overall spanwise flow of the boundary layer and help alleviate the tendency for wing-tip stalling on sweptback planform wings.

The deployment of slats is normally manually controlled from the flight deck in conjunction with the trailing edge flaps, but on some aircraft the slats move from the take-off position to the landing position automatically whenever the stall warning system is activated. The slats then return to their former set position when the warning cancels.

**Leading Edge Flaps.** These are used to improve the wings lifting capability at low airspeeds in a similar manner to that of trailing edge flaps by principally increasing the wings camber. These devices are comparable to slats in that they produce approximately the same increase in the maximum coefficient of lift, although this occurs at a slightly lower stalling angle of attack (Fig. 7.28).
The lift curve differs slightly from that associated with slats due to the additional camber effect. These devices are particularly beneficial on wings of high-speed section (thin with little camber) to improve their otherwise poor low speed handling characteristics. This is because the sharp leading edge associated with this type of wing makes it difficult for the air to negotiate, and stall consequently occurs at moderate angles of attack. The main types which exist are:

**Drooped Leading Edge Flaps (Droop Snoot).** These *normally cover the complete span*, and are drooped at high angles of attack, but retracted at low angles of attack, to give the required leading edge profile (Fig. 7.29).

These are extended via a jackscrew arrangement and pivot about a hinge on the lower wing surface (Fig. 7.30).
Krueger Flaps. These are located on the inboard leading edge section and are similar to drooped leading edge flaps, except when they are retracted they form part of the under surface, but when extended they hinge downwards and forwards (Fig. 7.31).

They are extended by way of a screwjack arrangement to produce a well-rounded leading edge (Fig. 7.32).
The Effect of Leading Edge Flaps on the Stalling Angle

Leading edge flaps tend to increase the wings stalling angle of attack, and also its level flight pitch attitude. This is because compared to a clean wing the effective chord line is deflected downwards. The wing stalls when the angle between this chord line and the relative airflow reaches the stalling angle of attack (Fig. 7.33).

Since the angle of attack by definition is referenced to the original chord line of an aircraft wing when the leading edge flaps are deflected the aircraft will stall with a more nose-up pitch attitude and thus a higher stalling angle of attack.

The Operation of High Lift Devices on Transport Category Aircraft

On most transport category aircraft the high lift devices are normally moved by hydraulic power (actuator or motor) supplied from the aircraft’s main hydraulic systems. The movement of a single flap control lever on the flight deck simultaneously determines the position of these devices (Fig. 7.34).
In operation the flap lever is moved in a flap quadrant in which a series of detents mark the various flap settings. To alter the flap setting the control lever must therefore be physically lifted before it can be moved to its next designated detent. The flap lever then forwards a signal to the trailing edge power drive unit (PDU), which hydro-mechanically alters the position of the flaps (Fig. 7.35).
An alternative electrical system allows the high lift devices to be operated by way of electric motors if the hydraulic system should fail. As the flaps move towards their selected position a signal is also transmitted to a separate power drive unit, which hydro-mechanically, drives the leading edge high lift devices to their selected position. On some aircraft the leading edge high lift devices are pneumatically and not hydraulically driven to their selected position in normal operation.

The leading edge high lift devices normally have only two extended settings (take-off and landing) and extend first and retract last, whereas the trailing edge flaps have various take-off settings, but normally only one landing setting (fully extended). The aircraft can however land with any flap setting, but it is important to remember that the flap position also determines the landing speed and distance. The flap quadrant is additionally fitted with gates (baulks) at specified flap settings (Fig. 7.36).
These gates are designed to prevent inadvertent rearward movement of the flap lever if specified flight and aircraft conditions do not exist. The first gate is normally set at a point which allows the **airspeed to build up sufficiently high enough** before the leading edge high lift devices are fully retracted, whereas the second gate normally marks the flap setting required for go-around with all engines operating.

**Protection of High Lift Devices on Transport Category Aircraft**

The high lift devices are normally protected against asymmetry and excessive flap loads:

**Asymmetry Protection.** This system is required to prevent the angular deflection between the high lift devices on the two wings from significantly differing, and resulting in severe roll. This may prove hazardous if left unchecked, and the high lift devices are therefore prevented from further deflection by asymmetry brakes if the angular difference exceeds set limits.

**Flap Load Relief.** If the flaps are fully lowered at high airspeeds serious structural damage may occur. An automatic system is therefore incorporated in most systems to partially retract the flaps, but if the airspeed subsequently reduces the flaps will automatically return to their former set position, commonly called **blow back**. The maximum flap extension speeds are placarded on the flight deck (Fig. 7.37).
<table>
<thead>
<tr>
<th>FLAPS LIMIT (IAS)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.230K</td>
</tr>
<tr>
<td>2.230K</td>
</tr>
<tr>
<td>5.225K</td>
</tr>
<tr>
<td>10.210K</td>
</tr>
<tr>
<td>210 K ALT FLAP EXT</td>
</tr>
</tbody>
</table>

FIG. 7.37
Chapter 8.

Stalling

Introduction

As the air flows around the aerofoil both the velocity and static pressure vary with distance from the leading edge. The pressure distribution over the upper surface greatly affects the flow characteristics of the boundary layer and will eventually cause it to breakaway or separate from the surface. When the upper surface of an aerofoil is predominantly covered in separated airflow the aerofoil is ‘stalled’. This occurs when the stalling angle of attack is reached. At this point the wing can no longer produce sufficient lift to support the weight of the aircraft, and the separated airflow results in a dramatic rise in form drag. It is desirable for any wing to stall firstly at the root, but this is not always possible, and principally depends on a wing’s cross-section, and planform surface area.

Separated Airflow

The static pressure varies over the upper surface of a typical aerofoil section. Close to the leading edge the airflow comes to rest, and the static pressure reaches a maximum value. This is the stagnation point, and is where the boundary layer first forms (Fig. 8.1).

![Diagram of airflow and pressure distribution](image)

FIG. 8.1

Proceeding rearwards from this point the static pressure decreases, and forms a favourable pressure gradient, i.e. positive to negative, which continues to exist until the air reaches its point of minimum pressure. Beyond this point the pressure begins to increase, forming an unfavourable, or adverse pressure gradient, i.e. negative to positive (Fig. 8.2).
The pressure gradient opposes the flow of the boundary layer and impedes its progress rearwards. The velocity of flow near the surface is also reduced in this region, and this is high-lighted using velocity profiles (Fig. 8.3).

In the presence of a strong adverse gradient the boundary layer will eventually separate from the surface (separation point). The airflow behind this point is turbulent in nature and effectively destroys the lift capability of the aerofoil in this region. This is because the energy possessed by the boundary layer is too low to overcome the adverse pressure gradient (Fig. 8.4).
The absence of the boundary layer behind the separation point allows some air to flow forward towards the leading edge (reversed flow). With increasing angles of attack the adverse pressure gradient increases in magnitude, and the separation point moves closer to the leading edge. This causes a large turbulent wake to form behind the wing, resulting in a reduction in lift, and an increase in drag.

When the separation point occurs so far forward that the majority of the aerofoil is covered in turbulent airflow, the wing is stalled. The lift generated by the aerofoil is drastically reduced, and it is no longer possible to maintain steady level flight (Fig. 8.5).
The Stalling Angle of Attack

The aerofoil stalls when the **stalling angle or critical angle of attack** is reached. This occurs when the **coefficient of lift** reaches a **maximum** value. Beyond this point the breakdown of the airflow results in a marked reduction in the coefficient of lift, and a rapid increase in the coefficient of drag, although lift is not totally lost when this angle of attack is reached (Fig. 8.6).

![Diagram showing the stalling angle and usual angles of flight](image)

**FIG. 8.6**

Most light aircraft tend to stall when the wing reaches an angle of attack of approximately **15 - 16°** in any phase of flight, **regardless of the airspeed**, provided that the aircraft configuration is not altered (Fig. 8.7).
Definition of the JAR/FAR Stalling Speed (Vs)

The calibrated stalling speed, or the minimum steady flight speed, in knots, at which the aircraft is controllable.

The Relationship between Stalling Speed and Lift

A relationship can be shown between lift and the indicated stalling speed by transposing the lift formula:

\[ \text{Lift} = C_L \frac{1}{2} \rho \frac{V^2}{S} \]

where
- \( C_L \) = coefficient of lift
- \( \rho \) = air density (kg/m\(^3\))
- \( V \) = airspeed (m/s)
- \( S \) = wing planform area (m\(^2\))

If the air density and wing planform area remain constant at a given altitude, then the lift formula can be simplified as follows:

Lift is a function of \( C_L \times (\text{IAS})^2 \)
At the stalling angle of attack the coefficient of lift reaches a maximum value so that:

\[ \text{Lift is a function of } C_{\text{L MAX}} \times (\text{IAS}_{\text{STALL}})^2 \]

Since \( C_{\text{L MAX}} \) is a constant value for a given aerofoil section, the amount of lift produced at the stall will be directly proportional to the indicated stalling speed squared, so that:

\[ \text{Lift is proportional to } (\text{IAS}_{\text{STALL}})^2 \]

Thus the stalling speed depends on the amount of lift a wing needs to generate, and it is influenced by the following factors:

**Weight.** To maintain steady level flight sufficient lift must be developed to support the total weight of the aircraft. A heavier aircraft will require greater lift, and will have an increased stalling speed (Fig. 8.8).

![Diagram showing stalling speeds at different IAS for different weights](image)

This relationship is true for any given angle of attack provided that the maximum coefficient of lift is not affected by airspeed. If the aircraft weight reduces by 10% the stalling speed will be affected as follows:

\[
\text{New Stalling Speed} = \text{Original Stalling Speed} \times \sqrt{\frac{\text{New Weight}}{\text{Original Weight}}}
\]

\[
= \text{Original Stalling Speed} \times \sqrt{\frac{0.9 \times \text{Original Weight}}{\text{Original weight}}}
\]

\[
= \text{Original Stalling Speed} \times 0.95 (95\%)
\]
**Aircraft Configuration.** The maximum coefficient of lift not only depends on a wing's angle of attack, but also on its wing shape, i.e. the greater the camber, the higher the maximum coefficient of lift. One common method of altering the camber of a wing is by changing the aircraft's configuration, i.e. extending the trailing edge flaps (Fig. 8.9).

This alters the shape of a wing, and increases its maximum coefficient of lift. This enhances the wing's overall lifting capability at any given angle of attack, and enables it to **support the same weight at a lower airspeed.** Thus the aircraft's **stalling speed is reduced.** Stalling with flaps may also be accompanied by a wing drop, which should be picked up using the rudder, and not ailerons. Notably, trying to raise a dropped wing using opposite aileron may have a reverse effect when operating near to the stall, and the wing will drop quickly.

**Power.** Up until now it has been assumed that the weight of an aircraft is completely supported by its wings. This remains the case when a piston engine is throttled back, but when **power is applied** the resultant slipstream behind the propeller provides **additional kinetic energy to the airflow** (Fig. 8.10).
This delays the separation of the boundary layer from the upper surface of a wing, and results in the aircraft stalling at a lower indicated airspeed. As the stalling angle is approached with ‘power on’, a component of thrust partially supports the weight of the aircraft. The wings will become slightly off-loaded, and will produce less lift.

As the ‘power-on’ stall is approached the airflow increases over the tail section, causing the rudder and elevator to be more effective. The slipstream also generates greater lift from the inner sections of the wing, but the outer sections may stall first. The ailerons will become ineffective, and one wing may stall, causing the wing to drop.

Manoeuvres. For an aircraft to carry out a manoeuvre it is necessary for the wings to generate more lift, causing the stalling speed to increase, e.g. during a turn.

Wing Loading. This is a measure of the total aircraft weight supported per unit area of the wing. If two aircraft are identical, except for their weights, then the heavier aircraft, i.e. higher wing loading, will have an increased stalling speed.

Wing Contamination. Any ice or snow on the wing will cause the total aircraft weight to increase, and thus the stalling speed.

Recognition of the Stall at Low Airspeeds

Most stalls occur at low airspeeds, and high angles of attack. This causes the separated airflow leaving the wing to pass over the tail surfaces, and results in shaking or buffeting of the control surfaces (Fig. 8.11).
This is known as pre-stall buffet, and normally takes place a few degrees before the stall. This buffeting is usually felt through the control column and rudder pedals, and provides adequate warning of an impending stall. The reduction in lift at the stall also results in an aircraft sinking or losing altitude at any given airspeed.

When the stalling angle of attack is reached the pressure envelope over the upper surface of a wing collapses, and the centre of pressure moves rapidly rearwards. This in turn alters the wings pitching moment, and in conjunction with the change in downwash acting on the tailplane most aircraft will experience a nose-down pitching moment at the stall.

**Stall Warning in Light Aircraft**

In addition to pre-stall buffet light aircraft are normally fitted with an audible stall-warning device that operates just before the stall. This device is activated by way of a moving vane (Flapper Switch), which is positioned approximately midway along the wing, just below the leading edge (Fig. 8.12).

At normal operating angles of attack the vane is held down by the airflow, but just before the wing stalls movement of the stagnation point around the leading edge lifts the vane (Fig. 8.13).
This closes a micro-switch and sounds a buzzer in the cockpit, giving warning of an impending stall. On some light aircraft the aural warning is replaced by a flashing red light on the instrument panel. On aircraft cleared to operate in icing conditions the sensing device is also electrically heated. Stall warning normally activates 5 to 10 knots above the stalling speed.

**Recovery from a Normal Stall**

When an aircraft stalls in level flight the reduction in lift normally causes the nose of the aircraft to drop. The speed of the airflow over the wing will increase as the aircraft dives and regains flying speed. This is a safe and desirable characteristic, but the pilot may elect to lower the nose in order to reduce the wing’s angle of attack, and increase the power setting as required.

**The Effect of Wing Section on the Stall**

The shape of the wing section will affect the overall stalling characteristics of the wing. With some sections the stall will occur suddenly without much warning, whilst others will approach the stall more gently and give adequate warning. If the wing stalls too suddenly it may be necessary to utilise an artificial pre-stall warning device.
Fig. 8.14 shows typical lift curves for two different wing sections. A lift curve with a sharp peak and a rapid drop after the stall indicates bad stalling characteristics, whereas a flatter peaked curve depicts a more gentle approach to the stall. By carefully altering the wing section across the complete span a predetermined stall pattern will be achieved. The designs which affect this behaviour are:

- Thickness-chord ratio.
- Camber.
- Chordwise location of maximum thickness.
- Leading edge radius of curvature.

Thus the sharper the leading edge, or the thinner the wing, or the further aft the positions of maximum camber and thickness, the more sudden the stall.

**The Effect of Wing Planform on the Stall**

Stalling does not necessarily occur simultaneously over all sections of a wing due to their variation in the coefficient of lift. In fact each section possesses its own individual coefficient of lift, and the wing stalls progressively as each section reaches its maximum value. To analyse the stalling characteristics of a wing it is necessary to consider the ratio of the section coefficients of lifts ($C_l$), to the wing's overall coefficient of lift ($C_L$). The stall firstly occurs where the ratio of the coefficients is highest, i.e. $(C_l/C_L)_{\text{max}}$.

![Diagram showing the effect of wing planform on the stall](image)

Fig. 8.15 shows how this ratio varies from the root to the tip, and where the stall first commences. On elliptical wings the stall occurs simultaneously over the complete span, on rectangular wings it occurs at the wing root, and on tapered wings it occurs at the wing tips. The stalling characteristics of a wing vary depending on their planform as follows:

**Elliptical Wing.** On an elliptical wing the local coefficients of lift remain constant over the complete semi-span, so that all sections reach the stall at approximately the same angle of attack. The stall will therefore progress uniformly along the span (Fig. 8.16).
An elliptical wing is capable of reaching high coefficients of lift prior to the stall, but there is little advance warning of the complete stall. The ailerons may also lack effectiveness when the wing is operating near the stall, and lead to poor lateral control.

**Rectangular Wing.** On a rectangular wing the stall commences at the wing root, where the highest local coefficient of lift exists, and then spreads progressively outward towards the outboard regions (Fig. 8.17).

This produces a strong root stall tendency and gives adequate stall-warning buffet as the separated air passes over the tail section of the aircraft. The resultant loss of lift associated with the stall is also initially felt near the rolling axis of the aircraft, so even if one wing stalls before the other, which is often the case, there is little tendency for the aircraft to roll.

Aileron effectiveness is maintained up to the stall and the natural in-built tendency of this type of wing automatically places the aircraft in a nose-down pitch attitude as the centre of pressure moves rapidly rearwards. This is the most desirable response to the stall, but the wings structural inefficiency limits its application to low cost, low speed, light aircraft.

**Tapered Wing.** On a highly tapered wing the stall commences near the tips, before spreading inward towards the inboard sections (Fig. 8.18).
This is an extremely **undesirable stalling characteristic** because the loss of lift at one wing tip before the other **may set up a considerable rolling moment**, and may lead to autorotation unless recovery action is taken. Tip stalling will also result in a **loss of lateral control** since the ailerons are located in this region.

**Sweptback Wing.** On a sweptback wing the stalling pattern resembles that of a tapered wing with the maximum section coefficient of lift existing near the wing tips (Fig. 8.19).

This is where the stall commences, and then spreads inward towards the inboard sections (Fig. 8.20).
Like the tapered wing this stalling pattern may also lead to large rolling moments and a loss of lateral control. Since the wing tips are situated well aft of an aircraft’s centre of gravity a loss of lift in these regions will also result in a severe nose-up pitching moment, known as pitch-up, which will result in a further increase in the angle of attack, rather than a reduction.

The normal recovery procedure from the stall is to reduce the angle of attack since the aircraft will continue to move in the wrong direction. This may lead to an extremely dangerous situation, especially if it occurs near the ground, during landing or take-off, when the aircraft is operating at high angles of attack. Even well away from the ground this pitch-up generally results in an overall loss of pitch control, and can prove extremely difficult to recover from, particularly at high airspeeds.

**The Cause of Pitch-up on Sweptback Wings at the Stall**

If the tips stall on a sweptback wing, the overall reduction in lift will cause the centre of pressure to move rapidly forwards, thus reducing the overall nose-down pitch moment (Fig. 8.21).
At the same time the downwash from the inner wing sections becomes concentrated on the tailplane, giving a more severe nose-up pitch effect. The resultant shift in the centre of pressure and increase in downwash acting on the tailplane will produce an overall nose-up pitching moment, ‘pitch-up’ (Fig. 8.22).

Devices to Alleviate Wing Tip Stalling

Pitch-up may be prevented on sweptback wings by incorporating design features, which help to alleviate tip stalling. This is achieved by using one or a combination of the following methods:

**Washout.** With washout the wing is geometrically twisted, so that the angle of attack at the tip is less than that at the root (Fig. 8.23).
This ensures that stalling at the wing tip is delayed until stall has firstly taken place at the wing root. The amount of washout is however limited, because too much, may, when operating at high airspeeds, result in the wing tip angle of attack becoming less than the zero lift angle of attack. This will cause the tip to carry a download, and will reduce the wings overall efficiency.

**Wing Fences (Boundary Layer Fences).** These are chordwise fences, which are fitted on the upper surface of wing to restrict the outwards flow of the boundary layer. This is instrumental in delaying wing tip stall (Fig. 8.24).

**Vortex Generators.** These are small upright aerofoils, which, are normally fitted on the upper surface of the wing in front of the ailerons (Fig. 8.25).
Vortex generators are designed to re-energise the low energy boundary layer, which exists at the wing tips, by making it more turbulent, and thus alleviating tip stall.

**Sawtooth Leading Edge.** This creates a vortex over the wing behind it reducing the magnitude of the vortex over the tip area, thus the magnitude of the tip stall (Fig. 8.26). A device called a Vortillon that is basically a protrusion ahead of the leading edge has a similar effect.

**Stall Sensing in Transport Category Aircraft**

To provide adequate warning of an impending stall, most modern transport category aircraft are fitted with angle of attack measuring probes or vanes. These are normally positioned either side of the aircraft in the nose area, and are mounted so that an undisturbed airflow can pass over them. The sensor that is placed in the airstream is shaped like an
aerofoil surface and its fixing link is attached to the fuselage skin via a pivoted joint (Fig. 8.27).

![Diagram of aerofoil attached to fuselage](image_url)

The sensor measures the angle of attack, as the aerofoil varies its position relative to the airflow, and when it exceeds a predetermined limit an electrical signal is generated to operate the stall warning system. This limit is usually 12° - 14° angle of attack, but this depends on aircraft design. On some aircraft these sensors may even compute the rate of change of angle of attack, which provides much earlier warning of an impending stall. These devices are also heated so that they remain operational throughout the flight.

The Stall Warning System on Transport Category Aircraft

The modern aircraft wing is designed so that airflow separation occurs at increased angles of attack. This gives it greater maximum lifting capability, but in doing the aerodynamic stall warning (pre-stall buffet) is either absent, or occurs too late to provide adequate warning. An artificial stall-warning device in the form of a stick shaker is therefore fitted to provide the necessary warning. These devices are clamped to the base of each control column and consist of a simple electrical motor, to the output shaft of which an eccentric weight is attached (Fig. 8.28).
The devices are designed to activate at greater than Vs (1.05 Vs) to vibrate the control column whenever the motor operates, but since the control columns are joined together the activation of either stick shaker will cause both columns to shake. It is usual however for both stick shakers to operate simultaneously, by way of stall warning computers, whenever the aircraft angle of attack, configuration and airspeed are such that a stall condition is imminent. The system is energised in flight at all times, but is deactivated on the ground via a weight on undercarriage safety sensor.

The Stall Prevention System on Transport Category Aircraft

Some transport category aircraft are additionally fitted with a stick pusher (nudger). This device is installed in the elevator (pitch) control system, and consists of a pneumatic ram that is supplied with high-pressure nitrogen or air. It is operated by the stall warning system and is designed to push the control column firmly forward just before the maximum coefficient of lift is reached. This puts the aircraft into a nose-down pitch configuration, thus reducing the angle of attack, and moving it away from the stalled condition.

Super Stall (deep stall)

Conventional straight winged aircraft with low tailplanes possess ideal stall characteristics. Prior to the stall the separated airflow from the wing causes buffeting of the tail surfaces, and at the stall the aircraft automatically pitches nose-down, thereby reducing its angle of attack. The tailplane also retains its effectiveness at the stall because the disturbed air passes above it (Fig. 8.29).
Jet transport category **aircraft with sweptback wings, a high T-tail configuration, and rear fuselage mounted engines** do not however behave in this manner. They **possess no pre-stall buffet warning** because the separated airflow from the wing does not pass over the tail surface, and the progressive stalling of the wing-tips causes the aircraft to pitch nose-up, thus intensifying the stall. The whole of the **tailplane is covered in disturbed air**, and the pitching capability required for recovery will be lost (Fig. 8.30).

The resulting loss of lift, and rapid increase in drag will also intensify the aircraft's rate of sink. In this condition an aircraft is considered to be **'super-stalled'**. **Recovery is impossible**, and the fitment of a **'stick pusher' is mandatory**.
Accelerated or ‘g’-stall

Another type of stall, which can occur, is the accelerated or ‘g’-stall. This can occur during a manoeuvre, e.g. turning, when the aircraft’s wings are subject to high load factors (g). It will occur when either wing reaches the stalling angle of attack, and like a conventional stall can occur at any airspeed. If the turn is over tightened, i.e. increasing load factor, either wing may stall without prior warning, causing the aircraft to flick in, or out of the turn. On sweptback planform wings this may also be accompanied by pitch-up. To recover from this condition the control column should be moved forwards to decrease the angle of attack.

Spinning

Spinning is a condition of stalled flight in which an aircraft describes a downward spiral path, and is normally the by-product of wing drop when operating near the stall (Fig. 8.31).

The spin manoeuvre can be divided into the following three distinct phases:-

The Incipient Spin (Autorotation). If the wing drops at the stall the resulting rolling action will alter the direction of the relative airflow onto the wing. It will increase the angle of attack of the down-going wing, and reduce the angle of attack of the up-going wing (Fig. 8.32).
This will alter the wings’ coefficients of lift and drag; the down-going wing will become more stalled leading to a reduction in the coefficient of lift, and an increase in the coefficient of drag. Conversely the up-going wing will become less stalled, leading to an increase in the coefficient of lift, and a reduction in the coefficient of drag. The difference in lift between the wings will produce a rolling moment and the aircraft will roll in the direction of the down-going wing. This will be further aided by the yawing moment resulting from the large difference in drags (Fig. 8.33).

These moments will lead to autorotation, the aircraft will continue to roll, a side-slip will develop and the nose will drop. If no corrective action is taken the rate of rotation will steadily increase and a fully developed spin will develop. To recover from the incipient spin:

- Ease the control column forward to unstall the wings.
- Apply rudder to prevent further yaw.
- As the airspeed increases, level the wings using co-ordinated rudder and aileron inputs.
- Apply power and recover any lost altitude.
The Fully Developed Spin. The development and characteristics of a spin depend on the aircraft's design, the distribution of its masses, and the operation of its control surfaces. It is usual for the aircraft to rotate several times around a spin axis before settling into a steady state spin, with the final pitch attitude, depending mainly on the position of the centre of gravity. The spin can be either flat or steep. This condition is reached when the rate of rotation and sink stabilise at constant values. When this occurs the forces and moments acting are considered to be in equilibrium and the relative airflow comes vertically upwards to meet the aircraft (Fig. 8.34).

In a steady stable spin the forces are in equilibrium. Weight acts vertically downwards through the centre of gravity, and is balanced by the aircraft drag, whilst lift acts at 90° to the relative airflow towards the centre of the spin (centripetal force) and is balanced by the centrifugal forces arising from the distribution of the aircraft's masses or inertia's. The moments about the centre of gravity determine the aircraft's state of equilibrium, and the recovery characteristics. The main forces affecting this are the resultant of the aerodynamic forces (lift and drag), and the centrifugal forces resulting from the distribution of masses or inertia's in the nose and tail of the aircraft (Fig. 8.35).
The centrifugal forces produce moments which tend to flatten the spin, whilst the resultant aerodynamic force produces a moment tending to steepen the spin. The position of the centre of gravity consequently determines the final attitude of the aircraft, and its spinning characteristics. The position of the centre of gravity, even if it remains within its permitted safety limits, effects the spin as follows:

**Forward Centre of Gravity.** This results in a steeper spin, and a faster rate of sink. This makes the recovery easier because the spin is less stable. If the centre of gravity is forward of its permitted limits it will significantly reduce the likelihood of a spin occurring, and will instead result in an unusually high spiral descent, during which the indicated airspeed will increase.

**Aft Centre of Gravity.** This results in a flatter spin and a lower rate of sink. This makes the recovery more difficult, because the spin is more stable. If the centre of gravity is aft of its permitted limits it will significantly reduce the likelihood of recovery from a settled spin condition.

When an aircraft is in a steep spin rotation it is primarily in roll, whereas in a flat spin it is primarily in yaw (Fig. 8.36).
The Recovery from a Spin. Spin recovery, like recovery from a simple stall, requires the separated airflow over the wings to be reattached. For a successful recovery from a spin it is necessary to firstly stop the yawing moment, and reduce the rolling moment. The usual recovery technique is to:

- Centralise the control surfaces and reduce the power setting.
- Verify the direction of spin on the turn and balance indicator, and apply full opposite rudder.
- Allow the rudder to become effective, then ease the control column forwards to reduce the angle of attack, and unstall the wings.
- As the rotation stops centralise the rudder, level the wings, and gently pull out of the ensuing dive.
- Apply power and climb the aircraft to regain any lost altitude. Note care must be taken when pulling out of the ensuing dive to prevent an accelerated or ‘g’-stall, and subsequent entry into another spin.
Chapter 9.

Forces Acting On An Aeroplane

Forces in Steady Level Flight

An aircraft is said to be in steady straight and level flight when the forces acting on it are in an equilibrium, or trimmed condition, i.e. there is no resultant force to accelerate or decelerate the aircraft. The main forces acting on an aircraft are shown in Fig. 9.1.

- Lift acts through the centre of pressure and Weight acts through the centre of gravity.
- Thrust and drag act in opposite senses, parallel to the direction of flight, through points, which vary with aircraft attitude and design.
- In steady level flight:

  \[
  \text{Lift} = \text{Weight} \quad \text{and} \quad \text{Thrust} = \text{Drag}
  \]
Lift/Weight and Thrust/Drag Couples

It would be convenient if all four forces acted through a single point, i.e. the centre of gravity (Fig. 9.2).

Unfortunately during flight the forces alter their points of action, and are normally arranged so that the lift/weight and thrust/drag forces are as follows:

**Lift/Weight Couple.** Lift acting behind weight causes a nose-down pitch moment and lift acting in front of weight causes a nose-up pitch moment (Fig. 9.3).

**Thrust/Drag Couple.** Thrust acting below drag causes a nose-up pitch moment, and thrust acting above drag causes a nose-down pitch moment (Fig. 9.4).
To prevent the aircraft rotating the forces can be distributed as shown in Fig. 9.5.

Most aircraft have the forces arranged, so that if the thrust is removed, i.e. in the event of engine failure, the remaining lift/weight couple will pitch the aircraft nose-down (without any action by the pilot), so that it assumes a gliding attitude. Alternatively, when power is added, thrust will increase and the nose will tend to pitch nose-up towards a level flight attitude (Fig. 9.6)

The forces are, normally arranged so that lift acts behind weight and thrust acts below drag. There is usually also a considerable difference in magnitude between the two pairs of forces, with lift and weight being greatest. In an effort to balance the pitching moments the spacing between the thrust and drag forces is normally greater than the spacing between the lift and weight forces. Ideally, the pitching moments should cancel each other out, but in practice this is not always possible and a secondary method of balancing must be used. This is normally provided by the tailplane (Fig. 9.7).
On some aircraft Canards or Foreplanes provide the secondary balancing required. Canard type aircraft do not use a conventional tailplane, in this design the horizontal tailplane is replaced with horizontal surfaces mounted in front of the main wing. The rearward location of the main wing allows more of the fuselage to be used as cabin space than where the wing is mounted at the mid fuselage position (Fig 9.8).

Unlike the conventional tailplane both the canard surface and the wing of the aircraft create an upwards lifting force under all normal conditions of flight, thus the weight of the aircraft is shared over both surfaces allowing a lighter wing loading to be obtained and therefore a lighter structure to be used.

One of the effects of mounting a small wing forward of the main wing is that the lift from the canard counters the negative pitching moment of the wing.

Canards are mounted with a slightly larger angle of incidence than that of the main wing; with the result that canards have a greater angle of attack than the main wings (Fig. 9.8). Creation of a positive pitching moment by the wing will lift the nose of the aircraft further increasing the angle of attack of the canard, if the pitching continues the canard will stall before the wing stalls.
The Contribution of the Tailplane

The tailplane is able to supply the force necessary to balance any residual pitching moments because it is positioned some distance from the aircraft’s centre of gravity and has a large moment arm (Fig. 9.9).

For this reason the area of the tailplane and subsequent lift force required need only be small, compared to the lift force produced by the main-plane. If the overall pitching moment is normally nose-down, it will provide a downward aerodynamic force (down-load) and vice versa.

During some phases of flight, the correcting moment provided by the tailplane may be insufficient to counteract the out of balance moments, which exist. In this case the lifting capability of the tailplane needs to be increased and this is achieved by altering the position of the elevator to provide an upward or downward force. On other aircraft, the actual tailplane position is adjustable to maintain level flight. In producing this force at any given airspeed an aircraft will experience an increase in drag, known as Trim Drag (Fig. 9.10).
Straight Steady Climb

An aircraft possesses a steady climb capability by converting propulsive energy in excess of that required to maintain steady level flight into potential energy. An aircraft can either be climbed steeply at a low airspeed, or be climbed at a higher airspeed at a shallower angle (Fig. 9.11).

If the airspeed is too low or too high, all of the power or thrust available will be needed to overcome the drag, thus reducing an aircraft's climb capability to zero. In a steady climb at a constant airspeed in a given period of time an aircraft can be climbed at: -

The maximum angle of climb ($V_x$). This is achieved when an aircraft gains the most altitude in the shortest horizontal distance covered, i.e. best gradient. This occurs when it is flown at a relatively low airspeed, and gives good ground obstacle clearance.

The maximum rate of climb ($V_y$). This is achieved when an aircraft gains the most altitude in the shortest time. This occurs when it is flown at a small angle of climb and a high airspeed.

Forces in a Straight Steady Climb

When an aircraft climbs at a constant airspeed the forces acting on it are in equilibrium, and act as shown in Fig. 9.12.
The angle between the flight path and the horizontal is known as the **angle of climb** ($\gamma$). The weight is resolved into two components; one opposing the lift and the other acting in the same direction as drag.

The following relationships therefore exist:

\[
\begin{align*}
\text{Thrust} &= \text{Drag} + \text{component of weight opposing flight} \text{ or } T = D + W \sin \gamma \\
\text{Lift} &= \text{component of weight acting perpendicular to the flight path} \text{, or } L = W \cos \gamma.
\end{align*}
\]

With increasing angles of climb the amount of lift required steadily decreases, whilst the thrust requirement increases (Fig. 9.13).

**In a steady climb thrust is always greater than drag, and lift is always less than weight.**
Straight Steady Descent (Dive)

If the aircraft is placed in a nose-down pitch attitude, and the thrust remains constant, the forces acting on the aircraft will change. This new attitude will cause a corresponding decrease in the angle of attack, and lift will momentarily become less than weight, causing the aircraft to begin the descent. A component of weight will act forward along the flight path, and together with the component of thrust, will cause the aircraft to accelerate. The engine is now doing less work compared with climbing and level flight, and to maintain a constant airspeed the thrust will need to be reduced, until the two components acting along the flight path oppose the drag (Fig. 9.14).

The forces can be resolved as follows:
Thrust = Drag – component of weight acting along flight path, or
\[ T = D - W \sin \gamma \]

Lift = component of weight acting perpendicular to the flight path, or
\[ L = W \cos \gamma \]

This shows that in a straight steady descent lift is less than weight and thrust is less than drag.

**Steady Straight Glide**

If the amount of power available is reduced to zero, i.e. on engine failure, the component of thrust will reduce to zero, and the drag force will act to decelerate the aircraft. This will lead to an overall reduction in lift, thus unbalancing the lift/weight couple and placing the aircraft in a nose-down pitch attitude, i.e. a glide. A component of weight will act forward along the flight path, and will oppose the drag (Fig. 9.15).

**FIG. 9.15**
Forces in a Steady Straight Glide

In a steady straight glide the aircraft will be moving at a constant indicated airspeed, with the engine producing no thrust, and the remaining aerodynamic forces, i.e. lift, drag and weight, being in equilibrium (Fig. 9.16).

![Diagram of forces in a steady straight glide]

The angle between the flight path and the horizontal is known as the aircraft's angle of glide (γ). The aircraft's weight is balanced by the total reaction (resultant of lift and drag), and is resolved into two components. One component acts perpendicular to the flight path, and balances the lift, whilst the other acts along the flight path, and balances the drag.

The forward component also determines the aircraft's forward airspeed. For an aircraft of given weight, any reduction in the angle of glide will result in a smaller component of weight acting forward along the flight path (Fig. 9.17).
This will reduce the amount of drag required to maintain a steady glide, and the lift/drag ratio will increase. The shallowest glide is obtained when the drag is least for the required lift, i.e. best lift/drag ratio. The lift/drag ratio is therefore a measure of the aircraft's gliding efficiency or performance. The aircraft will glide furthest through the air, i.e. best glide performance, when it is flown at an angle of attack and airspeed that gives the best lift/drag ratio ($V_{IMD}$) as illustrated in the polar diagram (Fig. 9.18).

The Effect of the Lift/Drag Ratio on Glide Performance

Most aircraft are not fitted with an angle of attack indicator, so the airspeed is normally adjusted to correspond to that relating to the best lift/drag ratio, i.e. minimum drag speed ($V_{IMD}$). This is possible because in a glide a similar, although not exactly the same, relationship exists between indicated airspeed and the angle of attack as that in level flight.
This speed is found in the flight-operating manual and is based on an aircraft's all up weight (AUW).

Furthermore since the **minimum drag speed produces the best glide performance**, flight at any other speed will **reduce the lift/drag ratio**, and consequently increase the angle of glide. This will **reduce the aircraft's glide performance**, and **reduce the overall glide distance** (Fig. 9.19).

![Diagram showing glide performance at different speeds](image)

**FIG. 9.19**

The reduction in the lift/drag ratio at airspeeds above and below the minimum drag speed is due to; **high induced drag at slow airspeeds**, and **high profile drag at high airspeeds**. If the aircraft is gliding at the recommended airspeed for maximum glide distance, and it looks like it will not reach its designated landing point, **the nose should not be raised**, since the higher nose attitude will decrease the glide distance.

**The Effect of a Steady Wind on Glide Performance**

A steady wind alters an aircraft's actual flight path and its effective gradient over the ground, thus altering its angle of glide (Fig. 9.20).
A tailwind will increase an aircraft’s gliding distance over the ground, i.e. reduce the angle of glide, whilst a headwind will reduce the distance over the ground, and i.e. increase the angle of glide. The time taken to reach the ground from a given start altitude in either case will remain the same, i.e. glide endurance is unaffected by a steady wind.

The Effect of Weight on Glide Performance

Any change in aircraft weight will require a similar change in lift and drag to maintain the best glide distance, i.e. range in a straight steady glide (Fig. 9.21).
The best lift/drag ratio will therefore remain unchanged, as will the angle of glide, provided that the airspeed is adjusted to maintain the optimum angle of attack. The glide distance is therefore unaffected by changes in aircraft weight, but the glide endurance will decrease with increasing weight, due to the higher indicated airspeed, and vice versa.

**Steady Co-ordinated Turn**

To understand turning it is important to be familiar with Newton's three laws of motion. When an aircraft is in steady level flight it is in a state of equilibrium (Newton's First Law), but to turn, it is necessary to apply an external force to change the direction and/or airspeed, i.e. it involves an acceleration (Newton's Second Law). In a steady co-ordinated turn at a constant altitude the aircraft will also be subject to Newton's Third Law of motion.

**Forces Acting on an Aircraft During a Steady Co-ordinated Turn**

During a steady co-ordinated turn, at a constant altitude, a force must continually act towards the centre of the turn (centripetal force). If it is not present the aircraft will be unable to maintain a curved path, but instead will fly off at a tangent. When the aircraft is banked the total lift force is tilted; the horizontal component of lift will provide the centripetal force, and the vertical component of lift will support the weight of the aircraft (Fig. 9.22).

![Diagram of forces during a steady co-ordinated turn]

**Calculation of the Centripetal Force**

Newton's second law states that when an external force acts a body it will accelerate in the direction of the force. The centripetal force acting on an aircraft will give it acceleration towards the centre of the turn (radial acceleration). In a steady co-ordinated turn an aircraft of weight \( W \) Newton's, travelling at a velocity (TAS) of \( V \) metres per second, around the
circumference of a circle of radius \( r \) metres, will have an acceleration towards the centre of the circle of \( \frac{V^2}{r} \) metres per second per second, so that:

\[
\text{Centripetal Force} = \text{Mass} \times \text{acceleration or } F = \text{mass} \times \frac{V^2}{r} = \frac{W \times V^2}{g \times r}
\]

where \( W \) = Aircraft Weight (kgf), \( g \) = Acceleration due to Gravity (9.81 m/s\(^2\))

**Turning an Aircraft**

To perform a steady co-ordinated turn at a constant altitude the ailerons are used to maintain the desired angle of bank, whilst the elevator is used to increase the wings angle of attack, and the rudder is used to balance the turn. The airflow acting on the parts of the aircraft behind the centre of gravity will cause the aircraft to yaw towards the dropped wing, but the natural inbuilt stability will try to resist the turn. The rudder is used to counter adverse yaw due to the ailerons, and also balances the turn. If the total lift force remained the same as in steady straight and level flight, the actual amount of lift supporting the weight of the aircraft will effectively reduce (Fig. 9.23).

Unless the lost lift is recovered, by primarily increasing the aircraft's angle of attack, it will lead to a loss of altitude. Thus to maintain a steady balanced turn at a constant altitude, the greater the angle of bank, the greater the centripetal force, and the greater the total lift requirement (Fig. 9.24).
Factors Affecting an Aircraft’s Radius of Turn

In order to establish the factors, which affect an aircraft’s radius of turn it is appropriate to consider the forces acting on it (Fig. 9.25)

The resulting horizontal and vertical forces, which exist during a steady level turn, are:
$$L \sin \phi = \frac{W \cdot \frac{V^2}{g}}{r} \quad \text{and} \quad L \cos \phi = W$$

So that:

$$\tan \phi = \frac{\frac{V^2}{g}}{r}$$

The aircraft's angle of bank is therefore **totally independent of its weight**, and **depends only on its airspeed (TAS)**, and its **radius of turn**. By transposing the above formula it is possible to establish the effect of airspeed and angle of bank, on the radius of turn:

$$r = \frac{\frac{V^2}{g \times \tan \phi}}{1}$$

For a **given angle of bank** the aircraft's radius of turn is solely determined by its **airspeed**, and for a **given radius of turn**, the **steeper the angle of bank**, the **higher the airspeed**.

Other factors that determine an aircraft's radius of turn are:-

**Wing Loading.** This is a **fixed value** for a given **aircraft type and weight**, and does not vary during a turn, even though the wings are producing greater lift per unit area. For a given wing area, the **greater the weight**, the **greater the wing loading**. To establish the effect of wing loading on the radius of turn it is appropriate to consider two similar types of aircraft having the same wing areas, but differing gross weights, i.e. different wing loadings (Fig. 9.26).

![Diagram](image.png)

**FIG. 9.26**
If both aircraft are turning with their wings at the highest usable angle of attack, and developing the same total lift, the lighter aircraft will be able to achieve a lower minimum radius of turn. This is because the lighter aircraft has a lower vertical lift requirement than the heavier aircraft, thereby providing a greater centripetal force for acceleration purposes, thus the higher the wing loading, the greater the minimum radius of turn.

Flaps. The lowering of flaps to the take-off setting in flight is advantageous, particularly when manoeuvring an aircraft at low indicated airspeeds, or in poor visibility. This is because the subsequent rise in the coefficient of lift associated with flap deflection increases an aircraft’s overall lift capability at any given indicated airspeed. It follows that since the weight of the aircraft remains unchanged, then the additional lift produced can provide a larger centripetal force, thus reducing an aircraft's minimum radius of turn (Fig. 9.27).

This can only be achieved if the flap limiting speed is not exceeded, and also if sufficient thrust is available to overcome the additional drag associated with flap deflection.

Altitude. The effect of increasing altitude is to increase an aircraft's minimum radius of turn. This is due to the IAS/TAS relationship, and the reduction in thrust horsepower.

IAS/TAS Relationship. For a fixed indicated airspeed the true airspeed steadily increases with altitude, so that for any given angle of bank the minimum radius of turn must likewise increase.

Reduction of Thrust Horsepower. With increasing altitude the reduction in thrust horsepower steadily increases, thereby limiting the maximum attainable indicated airspeed, and reducing the margin above the stalling speed. It follows that insufficient thrust horsepower may be available at altitude to overcome the higher drag associated with the increase in angle of attack needed to provide enough lift to
sustain a steady level turn. The minimum radius of turn therefore increases with increasing altitude.

Balancing the Turn

If the turn is unbalanced an aircraft will either slip into, or skid out of the turn, thereby reducing the aerodynamic efficiency of the aircraft. To help correct for these unwanted conditions the balance part of the turn and balance indicator is used (Fig. 9.28).

When the ball is in the centre it indicates that the turn is balanced, whilst any displacement of the ball indicates that the turn is unbalanced (Fig. 9.29).

If the ball is only partially displaced from its centre position the turn may be balanced using only the rudder, i.e. ball to the right, apply right rudder. If the ball is at the extremities of the indicator the use of rudder alone will produce a highly inefficient turn, and therefore the ailerons are used initially to primarily balance the turn.
Slipping turn. A slipping turn will occur if the angle of bank is too large for a given rate of turn, i.e. the aircraft is over-banked. This is indicated by the ball moving towards the lower side of the balance indicator, and you leaning in towards the centre of the turn (Fig. 9.30).

Skidding turn. A skidding turn will occur if the angle of bank is too small for the rate of turn, i.e. the aircraft is under-banked. This is indicated by the ball moving towards the upper side of the balance indicator, and you leaning towards the outside of the turn (Fig. 9.31).

Balanced turn. During a balanced turn the ball remains in the centre of the balance indicator, and you remain upright in your seat relative to the aircraft (Fig. 9.32).
Any deviation from a balanced turn is corrected by applying rudder according to the position of the ball in an attempt to maintain it in the centre, thereby maintaining a balanced turn.

Rate of Turn

This is measurement of how long it takes for an aircraft to turn, and is measured in degrees per second. This is particularly important when instrument flying where rate-1 turns are usually carried out at a rate of 3° per second. This means that the aircraft will turn through 180° in 1 minute, or 360° in 2 minutes. A steeper angle of bank will be required to carry out a rate-1 turn at higher airspeeds.

Load Factor

The additional lift required to maintain a steady co-ordinated turn at a constant altitude will place an increased structural loading on the aircraft’s wings. This is because the wings need to produce lift in excess of the aircraft weight to provide the necessary centripetal force. A ratio exists between the total lift produced by the wings, and the weight of the aircraft. This gives a measure of the structural loading which will occur during a balanced turn. This ratio is termed the load factor.

\[
\text{Load factor} = \frac{\text{Total Lift}}{\text{AUW}}
\]

In straight and steady level flight the load factor is one, but during a turn the increased lift requirement will produce a value greater than one. A relationship exists between angle of bank and load factor, such that:

\[
\text{Load Factor} = \frac{\text{Total Lift}}{\text{AUW}} = \frac{1}{\cos \phi}
\]
The total amount of lift required in a turn can be directly measured in terms of weight. For example in a 60° banked turn:

$$\text{Total Lift} = \text{AUW} \times \frac{1}{\cos 60°} = 2 \times \text{AUW}$$

The wings will have to produce lift equivalent to twice the weight of the aircraft; i.e. the load factor is 2. This is more popularly referred to as the g loading, which is sensed as an apparent increase in weight, e.g. at an angle of bank of 60°, 2g will be experienced. The steeper the angle of bank the greater the load factor, but this is limited by structural considerations (Fig. 9.33).

**Fig. 9.33**

The Effect of Turning on Stalling Speed

For any given aerofoil section the indicated stalling speed is related to lift as follows:-

Lift is proportional to (IAS$_{\text{STALL}}$)$^2$

The additional lift required to carry out a steady co-ordinated turn at a constant altitude thus results in an increase in the stalling speed. Since the angle of bank also determines the amount of extra lift required to support an aircraft in a turn, it is also related to the stalling speed (Fig. 9.34).
Fig. 9.34 shows the percentage increase in stalling speed compared to that in steady level flight with increasing angles of bank. At angles of bank less than 30° the percentage increase in stalling speed is minimal, but at greater angles it becomes more marked. For example if an aircraft normally stalls at 50 Knots in steady level flight, in a 60° banked turn it will stall at:

\[ 141\% \text{ of } 50 \text{ Knots} = 71 \text{ Knots} \]

Like lift, the stalling speed is also related to the load factor, so that:

\[ \text{Stalling Speed in Turn} = \text{Normal Stalling Speed} \times \sqrt{\text{Load Factor}} \]

For example if an aircraft has a normal stalling speed of 60 knots, and a load factor of 2 (60° angle of bank) the stalling speed in the turn will be:

\[ 60 \times \sqrt{2} = 85 \text{ Knots} \]

In this example the stalling speed in the turn has increased by 25 knots, which effectively reduces the safety margin, i.e. the margin above the stalling speed (Fig. 9.35).
This margin is further reduced by the reduction in airspeed resulting from the increase in induced drag, thus steep turns must be avoided when operating at low airspeeds.

**Aircraft Response During a Level Banked Turn**

Once an aircraft begins a level banked turn the *outer wing* will start to *travel faster* than the inner wing, thus *producing greater lift* (Fig. 9.36).
The aircraft's **angle of bank will steadily increase**, or the aircraft will have a tendency to **overbank**. It is therefore important to apply aileron as appropriate in order to counteract this effect and avoid any associated problems, i.e. increased load factor. The rudder is also applied to maintain a curved flight path, since the **faster moving outer wing** also develops increased profile drag, and will tend to yaw the aircraft.

**Aircraft Response During Climbing and Descending Turns**

Previously it was established that during a steady level banked turn an aircraft has a tendency to overbank because the faster moving outer wing produces greater lift. In **climbing and descending turns** this effect is however further complicated by the fact that an aircraft is **subject to additional rolling moments** resulting from the **variation in each wings angle of attack**. This is because although the whole aircraft covers the same altitude during a complete turn, the wings follow differing spiral paths due to the variation in turn radii.

**Climbing Turns.** During climbing turns an aircraft describes an upward spiral path. The **airflow** therefore **comes downwards to meet the wings**, thereby **reducing** their **angles of attack**, and hence their coefficients of lift (Fig. 9.37).
The faster moving outer wing however results in a smaller reduction in angle of attack, so that the net coefficient of lift is higher than on the inner wing, thereby producing greater lift. Its increased velocity further enhances the lifting capability of the outer wing. An aircraft in a climbing turn will therefore tend to overbank more than in a steady level turn, and if necessary the desired angle of bank should be maintained using the ailerons.

**Descending Turns.** During descending turns an aircraft describes a downward spiral path. The airflow comes upwards to meet the wings, thus increasing their angles of attack, and hence their coefficients of lift (Fig. 9.38).
The increase in angle of attack of the outer wing is however less than that of the inner wing, as is its lifting capability. The faster outer wing may compensate for the variation in lift due to the difference in angle of attack, although the aircraft may still tend to underbank. If this occurs the desired angle of bank should again be maintained by using the ailerons.
Chapter 10.

Stability

Introduction to Stability

Stability is the natural tendency of an aircraft to return to its former equilibrium or trimmed position, i.e. straight and level flight, following a disturbance without any pilot assistance. The stability of an aircraft is static and dynamic in nature. The actual stability characteristics of an aircraft are not only governed by its design, but are also dependent on crew workload. Thus a close relationship exists between stability and controllability.

Controllability

This is the ability of the pilot to alter the position, or attitude of an aircraft using the flying control surfaces. Adequate controllability does not necessarily exist with adequate stability. In fact high stability will make an aircraft resistant to change, and will reduce its controllability, i.e. good stability makes it harder for the pilot to control and manoeuvre an aircraft. Thus the upper limits of stability are determined by the lower limits of controllability. No aircraft is completely stable, but all must possess desirable stability and handling characteristics. Stability naturally occurs whenever an aircraft is rotated about any one, or a combination of its axes (Fig.10.1)

These axes act at right angles to each other, and all pass through the aircraft's centre of gravity. Stability about the lateral axis (pitch) is known as longitudinal stability. Stability about the longitudinal axis (roll) is known as lateral stability. Stability about the normal
axis (yaw) is known as directional stability. Lateral and directional stability are however not entirely independent of each other, and tend to act together to produce certain undesirable motions. In fact an aircraft can be unstable about two of its axes, but stable about the third or vice versa.

The degree of stability also differs, between types of aircraft, with transport category aircraft being generally more stable than light aircraft. Equilibrium of an aircraft in flight is more usually referred to as its trimmed condition and occurs when no net moments act to displace it from this condition, i.e. its moments in pitch, roll and yaw are zero. Trimming of an aircraft is normally attributed to trimming devices such as tabs, but in terms of stability it simply means that no net moments exist. Stability falls into two main categories; static stability and dynamic stability. Furthermore, whenever an aircraft is displaced from its normal trimmed position the air loads acting on it will oppose and damp out the subsequent motion. This is known as aerodynamic damping, and it greatly affects an aircraft’s degree of dynamic stability.

Static Stability

Static stability is the initial tendency an aircraft displays after it has been displaced from a given equilibrium position. If an aircraft tends to return to its former position it is said to be statically stable, and if it continues to move in the direction of the displacement it is said to be statically unstable. Finally if an aircraft tends to remain in the disturbed position it has neutral static stability. This type of stability can be demonstrated using ball bearings and a curved container (Fig. 10.2).

The Degree of Stability

The different degrees of stability are categorised by how quickly an aircraft tends to return to its trimmed position following a disturbance. To analyse this it is appropriate to again consider the analogy of a ball in a curved container (Fig. 10.3).
In this case the steeper the container the greater the static stability, but as stability increases so controllability reduces. The upper limits of stability are therefore set by the lower limits of controllability.

**Dynamic Stability**

Dynamic stability is the movement with respect to time of an aircraft in response to its static stability following a displacement from a given equilibrium position. For example consider a statically stable aircraft, which following a disturbance overshoots its equilibrium position. Its inbuilt stability will attempt to correct for this, and an oscillatory motion will occur (Fig. 10.4).
The time taken for the motion to subside is a measure of the aircraft’s dynamic stability. If the oscillations damp out with time the aircraft is dynamically stable, and if the oscillations increase in magnitude the aircraft is dynamically unstable. Finally if the oscillations persist without either increasing or decreasing in magnitude the aircraft has neutral dynamic stability. It is overall desirable for an aircraft to be both statically and dynamically stable.

**Static Longitudinal Stability**

This is the aircraft’s natural or inbuilt tendency when disturbed in pitch, to return to its former trimmed angle of attack without pilot input, and is desirable throughout the aircraft’s complete speed range. Conversely if the aircraft continues to diverge away from its trimmed angle of attack following a disturbance it is said to be statically longitudinally unstable, and if it remains at whatever angle of attack the disturbance causes, it is longitudinally neutrally statically stable. This type of stability is mainly provided by the tailplane. For example consider the effect of a gust, which causes an aircraft to pitch nose-up. The aircraft will due to its inertia momentarily continue to follow its original flight path and present itself to the relative airflow at an increased angle of attack. The subsequent increase in angle of attack of the tailplane will produce a small aerodynamic force. This force multiplied by the distance from the centre of gravity will produce a strong restoring pitching moment and will pitch the aircraft back to its former equilibrium position (Fig. 10.5). The pitching moment is defined in a coefficient form \( (C_m) \):

\[
C_m = \frac{M}{qS(MAC)}
\]

Where
- \( C_m \) = Pitching Moment Coefficient, +ve nose up
- \( N \) = Pitching Moment about the CG, +ve nose up
- \( q \) = Dynamic Pressure
- \( S \) = Wing Area
- \( MAC \) = Mean Aerodynamic Chord

**FIG. 10.5**
Mathematical Representation of Static Longitudinal Stability

An aircraft is trimmed longitudinally when any residual out-of-balance couples between the four main forces are balanced by a download ($L_T$) acting on the tailplane (Fig. 10.6).

For this to occur the angle of incidence of the tailplane is usually less than that of the mainplane (Fig. 10.7).

The angle between the chord line of the tailplane and the chord line of the mainplane is known as the **longitudinal dihedral angle**, and is a practical aspect in most types of aircraft. The actual degree of longitudinal stability is determined by the interaction between an aircraft's centre of gravity, its centre of pressure, and the position of its tailplane. For example consider an aircraft in steady level flight where the angle of attack of the mainplane and tailplane are $+6^\circ$ and $-4^\circ$ respectively (Fig. 10.8).
In this case wing lift (LW) acts through the centre pressure (C.P) equivalent to 8 units, whilst a download (LT) of 4 units acts on the tailplane and their points of action are 4 and 8 units respectively from the aircraft’s centre of gravity. The aircraft will be in a trimmed condition when no out of balance forces exist. This occurs when:

\[ LW \times 4 = LT \times 8 \]

i.e. \[ 8 \times 4 = 4 \times 8 = 32 \text{ units} \]

If an aircraft is suddenly subjected to an upgust its nose will rise, but at the same time due to its inertia it will momentarily continue to travel along its original flight path and present itself to the airflow at an increased angle of attack (Fig. 10.9).

This results in an increase in the amount of wing lift to 10 units, and a reduction in the download acting on the tailplane to 2 units), so that the aircraft is no longer in a trimmed condition. For ease of explanation it has also been assumed that even though the angle of
attack increases the points of action do not significantly move with respect to the aircraft's centre of gravity, and the moment arms therefore remain unaltered, so that:

\[
\begin{align*}
L_W \times 4 &= 10 \times 4 = 40 \text{ units} \\
L_T \times 8 &= 2 \times 8 = 16 \text{ units}
\end{align*}
\]

These figures show that the pitching moment due to the wing increases by 20%, whilst the pitching moment due to the tailplane reduces by 50%, so the aircraft is no longer in a trimmed condition. The combined effect of these changes in pitch moment is to rotate the aircraft back to its former trimmed position, with the tailplane having a greater overall effect than the wing.

**Factors Affecting Static Longitudinal Stability**

The degree of longitudinal static stability normally varies depending on the:

**Position of the Centre of Gravity.** Variations in the position of the centre of gravity greatly affect the static longitudinal stability of an aircraft, and generally the **further forward the centre of gravity the greater the stability** (Fig. 10.10).

Its forward position is limited by the fact that high stability results in poor controllability. This is because stability tends to resist movement away from the aircraft's trimmed attitude, which in turn is reflected in the amount of stick force necessary to displace an aircraft from this position. It follows that the further forward the centre of gravity, the greater the stick force, and the greater the effort required to manoeuvre the aircraft. If the centre of gravity is however positioned too far forward the stick forces will become excessive and will make it extremely tiring to fly the aircraft.

The forward position of the centre of gravity is also limited because if it is too far forward, the aircraft will become uncontrollably nose heavy at low airspeeds. This is particularly important in the landing phase when elevator deflection may be insufficient to allow the pilot to flare the aircraft on landing, unless the airspeed is increased to give greater elevator authority.
Conversely as the centre of gravity is moved progressively aft the degree of stability steadily decreases, as do the stick forces, and the aircraft returns less quickly to trimmed flight. Eventually a position is reached where the aircraft has no tendency to return to a trimmed condition following a disturbance and instead remains in its disturbed position. This is the aircraft’s neutral point, and is the centre of gravity position giving statically neutral stability (Fig. 10.11).

![Fig. 10.11](image)

Any movement aft of this point will make an aircraft statically longitudinally unstable. Most aircraft are designed to be statically longitudinally stable, so the centre of gravity is normally positioned ahead of the neutral point. The distance between the centre of gravity and the neutral point is called the static margin (Fig. 10.12).

![Fig. 10.12](image)

**Position of the Wing’s Centre of Pressure.** The position of the centre of pressure is a function of it's angle of attack, and moves towards the wing's leading edge with increasing angles of attack, and vice versa. In general the amount of movement of the centre of pressure varies depending on the aerofoil section used, and the greater
the camber, the greater its range of movement. If the centre of pressure is positioned behind the centre of gravity it will have a stabilising effect on the aircraft (Fig. 10.13).

Conversely if the centre of pressure moves ahead of the centre of gravity, a nose-up movement will be applied to an aircraft in response to a pitch-up disturbance, and will have a destabilising effect (Fig. 10.14).

Design of the tailplane. The overall function of the tailplane is to provide a force to counteract any residual out of balance couples existing between the four main forces. The degree of longitudinal stability is determined by the interaction between the aircraft's centre of gravity, area and the position of its tailplane. Its position relative to the centre of gravity is of most importance, since it has the greatest stabilising effect on the aircraft. This is because the greater the moment arm the greater the stability. If downwash from the wing acts on the tailplane it will also affect the aircraft's degree of stability, by affecting it's angle of attack. Furthermore the tailplane is usually of symmetrical section, and the position of its centre of pressure does not vary much in flight.

Wing Down-wash. Any disturbance in pitch will alter the wing’s angle of attack, and thus the amount of downwash from the wing. This will also alter the angle of
attack of the tailplane, e.g. If the aircraft pitches nose-up the downwash angle will be increased and the effective angle of attack of the tailplane will be reduced. The aerodynamic force produced by the tailplane will thus reduce, as will the restoring moment. This is compensated for by moving the C of G forward to increase the moment arm.

Graphical Representation of Static Longitudinal Stability

This is achieved by plotting a graph of pitching moment against angle of attack (or coefficient of lift) with the elevator fixed in its neutral position (Fig. 10.15).

The graph shows that if the angle of attack increases, e.g. due to a disturbance, a nose-down (-) pitching moment will be created and the aircraft will rotate back to its original trimmed position. Conversely if the angle of attack decreases a nose-up (+) pitching moment will be created. Thus for an aircraft to be **statically longitudinally stable**, the pitching moment must decrease with increasing angle of attack, i.e. have a negative slope. It is the steepness of the slope, which actually determines the aircraft's degree of stability (Fig. 10.16).
Fig. 10.16 shows the static longitudinal stability characteristics of four different aircraft. Aircraft A and B both have negative slopes and are thus longitudinally stable, although aircraft A is most stable because it has a more negative slope. Conversely aircraft C is longitudinally unstable because the pitching moment increases with increasing angle of attack and has a positive slope. Aircraft D on the other hand is different to the other aircraft because the pitching moment remains constant regardless of changes in angle of attack and the aircraft has no tendency to return its former trimmed position following a disturbance. Aircraft D therefore exhibits static neutral stability, and alternatively takes up a new trimmed position. Any aft movement of the C of G will reduce the degree of static longitudinal stability, and will produce a less negative slope. The slope of the graph will also be influenced by the following conditions:

**Stick Fixed Static Longitudinal Stability.** This involves the response of an aircraft to a disturbance in pitch if the flying control surfaces are held in set position. When the disturbance takes place the aircraft will have a natural tendency to return its former equilibrium or trimmed position. The amount of control deflection required to maintain any new equilibrium position is a measure of the aircraft’s stick fixed static longitudinal stability.

**Stick Free Static Longitudinal Stability.** This involves the response of an aircraft to a disturbance in pitch when the control surfaces are free to find their own position depending on the aerodynamic forces acting on them, i.e. with manual flying controls the stick forces have been reduced to zero by way of the trim tab system prior to the disturbance. This only applies to manual flying controls because in power operated flying control systems the surfaces are not free to float, and there is no difference between stick fixed and stick free static longitudinal stability.

**The Effect of Elevator Deflection on Pitching Moments**

To maintain a different flight attitude an equal and opposite moment from the elevators must be applied, e.g. to maintain a nose-up pitch the elevators must be raised. If the position of the C of G is fixed then the degree of static longitudinal stability will remain constant at any
deflection angle (i.e. constant slope), but the change in pitching moment will alter the coefficient of lift at which equilibrium will occur (Fig. 10.17).

This is because the angle of attack of the mainplane has been increased and the tailplane will produce a greater nose-up moment due to the change in effective camber. If the aircraft is trimmed to maintain the new pitch attitude (i.e. zero stick forces), and the elevators are allowed to float free, any change in the aircraft’s angle of attack will cause the control surfaces to move away from their trimmed position in the direction of the relative airflow. For example an increase in angle of attack will result in the elevators floating upwards, thus reducing the lift force (upload) acting on the tailplane, and reducing the aircraft’s static longitudinal stability compared to the stick fixed condition (Fig. 10.18).
Control Force Stability

The coefficient of lift ($C_L$) corresponds to a particular airspeed (dynamic pressure) in steady straight and level flight, and thus any variation in airspeed will correspond to a different value of $C_L$ in the aircraft’s trimmed or equilibrium position (Fig. 10.19).

An aircraft, which demonstrates **stick position stability**, will require the **control column to be moved forwards to reduce the angle of attack** and trim at a higher airspeed, and vice versa, i.e. with increasing forward airspeed an increasing forward stick force will have to be applied to maintain steady straight and level flight. Conversely an aircraft, which exhibits stick position instability, will require the control column to be moved aft to trim at a higher airspeed and vice versa.

In a manually controlled aircraft the control stick forces are dependent on:

- Basic stick force stability, where the force is independent of airspeed.
- The trim tab position, which varies with airspeed.

With **increasing EAS less and less nose-up tab is required**, and if the aircraft is correctly trimmed, i.e. if positive stick force stability exists, a push force will be required to maintain a new attitude with increasing airspeed and vice-versa (Fig. 10.20).
If the position of the C of G is varied whilst maintaining the same trim airspeed its actual position will have an effect on stick force stability. For example an aft movement of the C of G will reduce the negative slope of the graph, and thus the degree of stick force stability as illustrated in fig 10.21.
This also means that smaller stick forces will be required to displace the aircraft from its original trimmed airspeed. In accordance with JAR 25.173 a minimum gradient for stick force is required for an aircraft to be certified, with the following rules being applicable:-

- A pull force must be present to obtain and maintain airspeeds below the specified trim speed, and a push force must be present to obtain and maintain airspeeds above the specified trim speed.
- The airspeed must return to within 10% of the original trim speed during the climb, approach and landing conditions, and must return to within 7.5% of the original trim speed during the cruise.
- The average gradient of the stable slope of the stick force versus speed curve may not be less than 1lb for each 6 knots.

The degree of static longitudinal Stability must also be such that a stable slope exists between 85% and 115% of the airspeed at which the aircraft is trimmed with:

- The flaps retracted.
- The undercarriage retracted.
- At its maximum take-off weight.
- At 75% of maximum continuous power (piston), or maximum power or thrust (jet).

**Manoeuvring Stability**

Whenever an aircraft is manoeuvring it is acted on by acceleration forces, e.g. If the aircraft is pulling out of a dive, its flight path will be curved, and the resultant pitching velocity will provide aerodynamic damping in pitch due to the downward movement of the tailplane. This will act with the inbuilt static longitudinal stability of the aircraft and will tend to resist this motion. The tailplane provides the largest contribution towards damping in pitch, although other aircraft components such as the wings do assist. A graph of stick force versus load factor illustrates the manoeuvring stability of an aircraft (Fig. 10.22).
The gradient of the graph should be positive, i.e. with increasing load factor the stick force must also increase. This gradient must not be excessively high or the aircraft would be difficult and tiring to manoeuvre, or conversely it should not be too low or the stick forces would be too light and the aircraft could be over-stressed.

The manoeuvring stick force gradient or stick force per g for a transport category aircraft will be approximately 9 lbs / g. Aircraft with high static longitudinal stability will possess high manoeuvre stability, i.e. low controllability, and will also be associated with a high stick force gradient. Any aft movement of the C of G will reduce the stick force gradient and will also reduce the longitudinal static stability of the aircraft. With increasing altitude the manoeuvre stick force stability reduces. This is because as the density of the air reduces, the TAS increases, and the amount of pitch damping reduces (10.23)
Tailoring The Control Forces

Many devices can be added to the control system to modify or tailor the stick force stability to desired levels:

**Down-spring.** This device is a preloaded spring that tends to rotate the elevators down and increases the airspeed stick force stability without changing the aircraft static longitudinal stability (Fig. 10.24).

![Down-spring diagram]

This contributes to an increment of pull force that is independent of airspeed or control deflection. When the aircraft is retrimmed for its original airspeed then the airspeed stick force gradient increases resulting in a stronger feel for airspeed. Because the force increment due to the down-spring is not affected by stick position or normal acceleration then manoeuvring stick force stability is unchanged.

**Bob-weight.** This device is designed to improve the stick force stability. It consists of an eccentric mass, which is attached to the flying control system, and in unaccelerated flight acts like the down-spring. In accelerated flight during a manoeuvre the bob-weight will be subjected to the same forces as the aircraft, and will provide an increment of stick force in direct proportion to the magnitude of the manoeuvring acceleration, thus increasing the manoeuvring stick force stability (Fig. 10.25).
Dynamic Longitudinal Stability

Two types exist with one form of oscillation being more serious than the other, but for each type consider an aircraft initially in steady straight and level flight travelling at an airspeed, $V$:

**Long Period Oscillation (Phugoid).** This involves very long periods of oscillation (20 – 100 secs) with noticeable variations in pitch attitude, altitude and airspeed, whilst the angle of attack remains nearly constant, i.e. if an aircraft is subjected to a horizontal gust its airspeed will momentarily change, but its angle of attack will remain virtually constant. Any change in airspeed will be accompanied by a change in drag, (Fig. 10.26)

If the aircraft is statically stable and is operating at an airspeed in excess of $V_{IMD}$ any increase in airspeed will not only increase the drag, but will also increase the lift, and the aircraft will momentarily gain height. Some of the aircraft’s kinetic energy will subsequently be converted into potential energy, and the aircraft will slow down.
As the airspeed drops below its original value, the aircraft momentum will reduce, and the aircraft will descend. An oscillatory motion will take place as the aircraft successively gains and loses altitude. If this motion is damped out the aircraft will be dynamically longitudinally stable, although in some instances the aircraft may be slightly unstable. In either case some form of corrective action will need to be taken by the pilot, but since the period of oscillation is usually long, any necessary action is easily applied.

**Short Period Oscillation.** This involves very short periods of oscillation, typically 1-2 seconds, when an aircraft is subjected to a vertical gust. The disturbance will cause the aircraft to rotate about its lateral axis, and will vary its angle of attack, whilst the airspeed remains virtually constant. The change in angle of attack will also vary the lift, and a pitching moment will result. If the aircraft is statically longitudinally stable any disturbance in pitch will set up an oscillatory motion about the aircraft’s lateral axis, where oscillation will be dynamically stable or unstable.

The frequency of this oscillation is normally high and thus cannot be corrected for by the pilot, as in the case of Phugoidal oscillatory motion. It is therefore essential that this form of oscillation be quickly damped out by way of an automatic stabiliser, which must be included in the aircraft’s flying control system. It follows that an aircraft operating at airspeeds less than the minimum drag speed will show speed instability. If a jet transport category aircraft is flying at an airspeed less than the minimum drag speed, for instance on landing, speed instability can prove extremely serious.

**Pilot Induced Oscillations**

Oscillatory longitudinal motion of an aircraft can occur due to inadvertent movement of the flying controls by the pilot. Short period longitudinal motion of the aircraft can have the most damaging effect where any delay in the pilots control system response (response lag) can quickly produce an unstable oscillation. This can produce damaging flight loads and even lead to a loss of control of the aircraft. When normal response lag and control system lag are added to the actual aircraft motion, any inadvertent control inputs by the pilot may have a negative effect on the oscillatory motion leading to dynamic instability. Since short period motion is of relatively high frequency, the amplitude of the pitching oscillation can quickly reach a dangerous level in a very short time. If pilot induced oscillation is encountered, the most effective solution is to immediately release the controls, since any attempt to forcibly damp out the oscillations will just worsen the situation, and increase the amplitude of motion. Releasing the controls removes the unstable excitation and allows recovery to naturally occur through the dynamic stability characteristics of the aircraft.

**Static Directional Stability**

The directional static stability of an aircraft is its natural, or inbuilt tendency to recover from a disturbance in yaw, which is mainly provided by the fin. For example consider a gust of wind which causes the aircraft to yaw to the left.

The displacement of the aircraft centre-line from some reference azimuth is called the yaw angle (ψ). Positive yaw angle occurs when the aircraft is displaced to the right of the azimuth direction. The aircraft will due to its inertia, momentarily continue along its original flight path, and will begin to sideslip.
The displacement aircraft centre-line from the relative airflow, rather than some reference azimuth is called the *sideslip angle* \( (\beta) \). By convention it is positive when the relative airflow is displaced to the right of the aircraft centre-line. This action gives the symmetrical fin an angle of attack, which is the directional angle of attack of the aircraft. This produces a small aerodynamic force, which when multiplied by the distance from the centre of gravity produces a strong restoring moment, and yaws the aircraft back to its original equilibrium position (Fig. 10.27).

This type of stability is also referred to as *weather-cock stability*.

**Graphical Representation of Static Directional Stability**

This is achieved by plotting a graph of yawing moment coefficient \( (C_n) \) versus the sideslip angle (Fig. 10.28). Where the yawing moment coefficient is:

\[
C_n = \frac{N}{qSb}
\]

Where \( C_n \) = Yawing Moment Coefficient, +ve to the right
\( N \) = Yawing Moment, +ve to the right
\( q \) = Dynamic Pressure
\( S \) = Wing Area
\( b \) = Wing Span
The slope of the graph is a measure of the aircraft’s static directional stability. If the aircraft is subject to a positive sideslip angle and a positive yawing moment coefficient exists static directional stability will be present. For example if the relative airflow comes from the right (+\(\beta\)) a yawing moment to the right (+Cn) will be created and will tend to weathercock the aircraft into wind. A positive slope shows that the aircraft is directionally stable, and the steeper the slope the greater the degree of stability. Conversely if the slope is negative it shows that the aircraft is directionally unstable and that it will tend to diverge or move away from the direction of the airflow.

The Factors Affecting Static Directional Stability

The vertical fin is the primary source of static directional stability and is highly stabilising up to the stall. By incorporating fin sweepback directional stability can be improved by reducing the aspect ratio and increasing the stalling angle. Also the centre of pressure moves rearwards thereby increasing the tail yawing moment.

The fitment of dorsal fins, that are a forward extension to the fin, helps to delay the stall by increasing the surface area that is located aft of the CG, and reducing the fins effective aspect ratio therefore increasing the stalling angle of attack. (Fig.10.29)

Ventral fins are located on the underside of the tail, unlike dorsal fins that are an extension of the fin, have no effect on static longitudinal stability, but have a negative effect on static lateral stability and a positive effect on static directional stability. (Fig.10.29)
At high angles of attack the fuselage may cause an overall decrease in static directional stability. This is due to an increase in the fuselage boundary layer at the vertical tail location and is most significant for low aspect ratio aircraft with sweepback. The fitting of strakes improves directional stability by re-energising fuselage boundary layer and stopping cross flow around the fuselage at high angles of attack that may stall the fin due to the resulting disturbed airflow. (Fig.10.30)
The degree of stability provided by the fin thus depends on the:

- Position of the aircraft’s C of G.
- Position of the fins C of P.
- Area and angle of attack of the fin.

The contribution of the wing, particularly wing sweepback, has a small effect on the degree of directional stability compared to other components. The fuselage tends to have a destabilising effect, but at high sideslip angles the degree of instability reduces. The contributions of the aircraft components to static directional stability are shown in Fig. 10.31.

**Static Lateral Stability**

The lateral static stability of an aircraft is its natural or inbuilt tendency to recover from a disturbance in roll. For example if a disturbance causes an aircraft to roll, one wing will rise and the other will drop. The motion in roll will be naturally damped out by the wings, and the aircraft will assume a banked attitude as shown in Fig. 10.32.
In this attitude the lift force is tilted so that it no longer directly opposes weight and the resultant of these two forces will cause the aircraft to sideslip in the direction of the dropped wing. Due to the inertia the aircraft will also continue in a forward direction. As a result of sideslip the aircraft will be subjected to a sideways component of relative airflow, as in the case of directional stability, and with its inbuilt design features will produce a rolling moment. This moment will restore the aircraft to its original wings-level attitude and is defined as the rolling coefficient ($C_L$) in the following formula:

$$C_L = \frac{L}{qSb}$$

Where

- $C_L$ = Rolling Moment Coefficient, +ve to the right
- $L$ = Rolling Moment, +ve to the right
- $q$ = Dynamic Pressure
- $S$ = Wing Area
- $b$ = Wing Span

To provide the necessary stability characteristics one or a combination of the following design features are utilised:

**Wing Dihedral.** As the aircraft sideslips the lower wing due to its dihedral will have an increased angle of attack, whilst the upper wing will have a reduced angle of attack (Fig. 10.33).
The lower wing will produce greater lift than the upper wing, and the increased lift produced by the lower wing will thus produce a rolling moment, which will return the aircraft to its former equilibrium position. The upper wing may also be partially shielded by the fuselage, further reducing the amount of lift it develops.

**Wing Sweepback.** As the aircraft sideslips the lower wing will present more of its span, known as effective span, to the airflow than the upper wing as shown in Fig. 10.34.
The effective chord of the lower wing will also reduce whilst that of the upper wing will increase. The aspect ratio of the lower wing will thus be greater than that of the upper wing, and will produce greater lift. The increased lift will produce a rolling moment, and the aircraft will roll back to its former equilibrium position.

**High Wing and Low Centre of Gravity.** As an aircraft sideslips the higher wing acts in a similar manner to the dihedral wing, with the lower wing producing greater lift than the upper wing. In this case however, the displacement of the overall lift force outwards towards the wing-tip on the lower wing providing the necessary restoring moment (Fig. 10.35).

![Diagram of lift and weight](image)

**FIG. 10.35**

The position of the lift force will produce a rolling moment about the aircraft's centre of gravity and will roll the aircraft back to its former wings-level condition. Thus the lower the centre of gravity, the greater the lateral stability characteristics. On some high winged aircraft the amount of stability is so large that low dihedral, or even anhedral wings are fitted, i.e. to de-stabilise the aircraft. This form of recovery is also known as the 'pendulous effect'.

**High Keel and Low Centre of Gravity.** As an aircraft sideslips its side surfaces, i.e. the fuselage and fin, will assume a position perpendicular to the relative airflow. It follows that those surfaces, which are above the aircraft's centre of gravity, will produce a restoring moment and the aircraft will roll back to its former wings-level condition (Fig. 10.36).
Notably the lower the centre of gravity, the greater the degree of lateral stability.

**Graphical Representation of Static Lateral Stability**

This is achieved by plotting a graph of rolling moment coefficient ($C_l$) against sideslip angle ($\beta$) (Fig. 10.37)

If the aircraft is subject to a **positive sideslip angle**, it will be **laterally stable** if a **negative rolling moment** is applied. For example if the relative airflow comes momentarily from the right ($+\beta$) a negative rolling moment ($-C_l$) will be created and the aircraft will roll to the left, returning it to its former equilibrium position. Static lateral stability thus only exists if a negative gradient exists.
Factors Affecting Static Lateral Stability

High lift devices and power have a destabilising effect on static lateral stability by reducing the dihedral effect. Flap deflection causes the inboard section of the wing to become more effective and the C of P to move inboard closer to the aircraft’s longitudinal axis thus reduces the rolling moment.

Interaction Between Lateral and Directional Static Stability

Up until now lateral and directional stability have been considered as completely separate items. The effects of lateral and directional stability are however so closely interlinked that it is impossible to separate them. Thus a disturbance which initially only involves lateral stability will when the aircraft reacts, also involve directional stability at the same time. The main combinations of these types of stability are spiral instability and oscillatory instability:-

**Spiral Instability.** As previously established the lateral stability of an aircraft depends on the forces that tend to return it to a wing level condition following a wing drop. If an aircraft is fitted with a large fin, as the wing drops a side-slip component acts on the keel surface (including the fin) and tends to yaw the aircraft in the direction of the lower wing. If the aircraft's *directional stability is greater than its roll stability* the aircraft will directionally try and straighten itself up, i.e. line itself up with the side-slip component acting on the wing.

This will cause the higher outer wing to accelerate and produce more lift than the lower inner wing. This will set up a rolling moment towards the inner wing and the aircraft's angle of bank will increase, resulting in further side-slip. When this occurs the aircraft exhibits spiral instability, and if left unchecked will lead to a steep spiral dive. This tendency may be minimised by reducing the total fin area, which effectively reduces the aircraft's directional stability, and its tendency to yaw in the direction of the dropped wing. Lateral stability is consequently more effective than directional stability, and the aircraft becomes more spirally stable.

Another method of minimising spiral instability is to increase the dihedral effect, although this leads to greater oscillatory instability. This form of instability is normally controllable so most aircraft are designed to be spirally unstable. If this characteristic exists an aircraft when yawed, either by the prolonged application of rudder or asymmetric power, a rolling moment in the direction of yaw will rapidly occur, and the aircraft will quickly enter a spiral dive. This can occur when flying at low airspeeds under asymmetric power conditions, when excessive yaw due to too much power, and insufficient rudder control to balance it may quickly place an aircraft in a dangerous attitude.

**Oscillatory Instability.** This is more serious than spiral instability and is commonly found to a varying degree in combinations of high wing loading and sweepback, particularly at low indicated airspeeds, and high altitude. It is characterised by a combined rolling and yawing movement, or wallowing motion, where an aircraft continually yaws and rolls from side to side until corrective action or natural damping takes place. The main forms of oscillatory instability are:- Dutch Roll - where roll predominates yaw, and Snaking - where yaw predominates roll.
When an aircraft is disturbed in roll depending on its design, either of these conditions may result, from which the aircraft will be unable to settle without some form of assistance. The resulting motion can be simply unpleasant, but in some cases may lead to the total loss of the aircraft, particularly when flying under instrument conditions. The main factors which determine the degree of oscillatory instability are the:

- amount of dihedral
- amount of sweepback
- keel surface area (including the fin and rudder)

In transport category aircraft the most common form of oscillatory instability that must be catered for is Dutch Roll.

**Dutch Roll**

Consider an aircraft with sweepback when its **directional stability is less than its lateral stability**. If the aircraft is yawed say to the right the left wing will advance (side-slip) and generate more lift, whilst the right wing will slow down and produce less lift. The result of the imbalance in lift is to roll the aircraft in the direction of yaw. The lift generated by the left wing will be further increased by becoming less sweptback, and will also offer a greater span to the airflow, whereas the right wing will become more sweptback, and will decrease the span to the airflow. This effect is similar to that of dihedral. The advancing wing will also produce greater drag due to the larger areas exposed to the airflow, which will cause the aircraft to yaw in the opposite direction, i.e. to the left. This will result in the right wing producing more lift than the left wing, and the direction of the roll will reverse. The final result is an undulating or corkscrew motion where the rolling and yawing oscillations have the same frequency, but are out of phase with each other (Fig. 10.38).
This unstable motion will continue until corrective action is taken, or alternatively the motion will be naturally damped out. This motion is primarily due to excessive lateral stability; so one method of curing this problem is to reduce the amount of wing dihedral, or even by setting the wings at a slight anhedral. If anhedral wings are fitted the angle of attack of the advancing wing will reduce, whilst that of the retreating wing will increase. This will effectively reduce the aircraft's lateral stability, and thus its tendency to Dutch roll, but it does tend to increase an aircraft's spiral instability. The Dutch roll tendency may also be reduced by increasing the size of the fin/rudder, but this will adversely effect its handling characteristics. This is because it becomes necessary to firstly fight the 'weather cocking' tendency of the fin before the aircraft can be turned, i.e. it increases an aircraft's spiral instability. Conversely if the fin/rudder were too small the aircraft would become oscillatory unstable, i.e. lateral stability would exceed directional stability, and the amplitude of the oscillatory motions in Dutch roll would quickly increase. An aircraft is therefore usually designed with a small degree of spiral instability, in order to help alleviate the less pleasant Dutch roll tendency.

Aircraft with straight wings are less susceptible to Dutch roll because any movements in yaw are quickly damped out, but on aircraft with sweptback wings Dutch roll is more of a problem. This is because sweepback tends to worsen the aircraft's roll and sideslip tendencies. All transport category aircraft are generally prone to Dutch roll and require artificial damping in the form of a yaw damper system. This is because the magnitude of the oscillatory motion is normally comparatively small and is therefore extremely difficult for a pilot to be able to coordinate his reactions in phase with the Dutch roll of the aircraft. Any manual input may result in over correction, and will intensify the resulting oscillatory motion.
Yaw Damper Systems

Yaw damper systems are designed to detect infinitesimal yaw variations from the desired flight path. This is sensed by way of rate gyros and the system corrects for it by mechanically deflecting the rudder by an amount proportional to the rate of yaw, i.e. maximum rudder deflection will be applied when the rate of yaw is maximum (Fig. 10.39).

![Diagram of yaw damper system]

By using this method it is possible to stop the Dutch roll before the effects are felt. Most transport category aircraft are normally fitted with at least two yaw damper systems, which operate continually, and in their basic operation act independently of the autopilot system. On some aircraft however the yaw dampers are additionally used to co-ordinate turns made by the pilot or autopilot from information sensed in the aileron control circuit as shown in Fig. 10.40.
Each system has its own yaw damper controller, which provides signals to operate a yaw damper actuator, which in turn generates rudder control inputs. The inputs normally operate in series with the pilot input and do not result in rudder pedal movement.

**Speed Stability**

It was previously established that aircraft drag varies with speed (Fig. 10.41).
To sustain steady level flight it was also established that the thrust required from aircraft engines must equal the total drag at any given airspeed. The thrust required curve is therefore identical to the drag curve, but it must be remembered that this only applies to an aircraft of constant weight, configuration and altitude. Minimum drag also occurs at one particular airspeed, known as the minimum drag speed ($V_{IMD}$). Any change in airspeed either side of this point will result in an increase in aircraft drag/thrust required. As an example consider two points A and B, where the drag curve relates to an aircraft flying at an airspeed where thrust equals drag, i.e. thrust available equals thrust required (Fig. 10.42).

In each case consider the thrust setting to be constant regardless of any variations in airspeed, i.e. due to a gust. Firstly consider variations in airspeed with respect to point B. If the airspeed increases drag will similarly increase, so that the drag will now exceed the amount of thrust available. The aircraft will therefore decelerate and will return to point B without any pilot input. Conversely if the airspeed reduces below point B the drag will reduce, so that the thrust available will now exceed the drag. The aircraft will therefore accelerate and will similarly return to point B. Thus an aircraft operating at airspeeds greater than the minimum drag speed ($V_{IMD}$) will exhibit speed stability (Fig. 10.43).
Now consider variations in airspeed with respect to point A. If the airspeed increases the drag will reduce, so that the amount of thrust available will now be greater than drag, and the aircraft will accelerate, i.e. it will move away from point A. Conversely if the airspeed reduces the drag will increase, so that the amount of thrust available will now be less than drag, and the aircraft will decelerate further, until it reaches its stalling speed (Fig. 10.44).

Thus an aircraft operating at **airspeeds less than the minimum drag speed** will exhibit **speed instability**. If a jet transport category aircraft is flying at an airspeed less than the minimum drag speed, e.g. on landing, speed instability can prove extremely serious. This is because:

- the response to throttle movement is much slower than for a piston engined aircraft (Fig 10.45).
- of a lack of slipstream effect.
of the high momentum associated with large aircraft which requires a long time, and a lot of thrust to increase speed. It similarly takes time to reduce speed.

Care must thus be taken during the approach, since a situation may arise where the maximum thrust can not be obtained before the minimum airspeed is reached, i.e. the time to reach the minimum airspeed from the minimum drag speed may be less than the time taken for the engines to accelerate to maximum thrust as shown in Fig. 10.46.
By comparison, piston engined aircraft are less susceptible to speed instability because the:

- response to throttle movement is almost immediate.
- propeller slipstream produces additional lift without any change in airspeed.

The deployment of flaps and the lowering of the undercarriage (landing configuration) helps to delay the onset of speed instability by increasing profile drag, and thus reducing the minimum drag speed (Fig. 10.47).
FIG. 10.47
Chapter 11.  
Ground Effect

Introduction

When an aircraft flies close to a surface, e.g. ground or water, the lift, drag and stability characteristics are significantly affected. The changes that occur are collectively known as ground effect, and are noticeable whenever an aircraft is one wingspan or less above the surface. The closer to the surface the more pronounced the effect.

The Characteristics of Ground Effect

Ground effect takes place because the surface interferes with, and alters the airflow pattern around the wings. Primarily the surface restricts the formation of the wingtip vortices (Fig. 11.1).

![FIG. 11.1](image)

This results in a reduction in the amount of induced downwash behind the wing, and increases the wings effective angle of attack (Fig. 11.2).

![FIG. 11.2](image)
It also alters the pressure distribution around the wing, and the amount of lift developed. This occurs because the change in effective angle of attack increases the wing’s coefficient of lift ($C_L$), and thus its lifting capability at any given angle of attack. The aircraft will also stall at a lower angle of attack when flying in ground effect. (Fig.11.3).

The magnitude of the wing tip vortices, and thus downwash, also determine the amount of induced drag produced by the wing. Thus the closer the aircraft is to the surface, the greater the reduction in induced drag, at any given angle of attack (Fig. 11.4).

The reduction in induced drag is approximately 1% at a height of one wingspan above a surface, and approximately 50% at a height of one tenth of the wingspan. This effect is therefore significant during the take-off and landing phases of flight when induced drag may
account for over 80% of the total drag. The overall reduction in drag also reduces the amount of thrust required to maintain a given airspeed (Fig. 11.5).

![Diagram of thrust required vs velocity]

**FIG. 11.5**

The second characteristic associated with ground effect is the change in aircraft trim and stability. This occurs because the downwash acting behind the wing is deflected by the surface, thus altering the angle at which the airflow meets the tailplane (Fig. 11.6).

![Diagram of downwash path]

**FIG. 11.6**

This places the tailplane at a less negative or increased angle of attack, and reduces the download, acting on the tailplane. In this condition the aircraft will experience a nose-down pitching moment, although high ‘T’ tailed aircraft do not respond in this manner, since the tailplane is outside the influence of the downwash.
The Influence of Ground Effect on Landing

When an aircraft enters ground effect during the landing phase of flight the sudden increase in lift and reduction in drag will cause it to experience a floating tendency. The aircraft will also tend to pitch nose-down due to the reduction in download on the tailplane.

The Influence of Ground Effect on Take-Off

During the take-off phase of flight as the aircraft leaves ground effect the wing tip vortices will rapidly grow in magnitude, producing an increase in downwash behind the wing. This will cause the lift produced at a given angle of attack to suddenly reduce, and the associated rapid increase in induced drag may prevent a successful take-off. The increased downwash acting on the tailplane will increase the download, and a nose-up pitching moment will result. It is therefore important that the take-off is not attempted at speeds less than the designated take-off speed, because even though the aircraft may be able to fly in ground effect, it will be unable to climb out of it. The aircraft may even sink as it flies out of ground effect, and will resettle on the ground.

The Influence of Ground Effect on Trailing Edge Flaps

When flying in ground effect with the trailing edge flaps deflected the surface will increase the adverse pressure gradient over the flap, and will also induce earlier separation of the airflow (Fig. 11.7).

![FLOW SEPARATION AT FLAP DUE TO HIGH PRESSURE ON GROUND](Fig. 11.7)

This will significantly reduce the maximum lift capability of the wing, compared to that obtained outside ground effect. This is one of the main reasons why STOL aircraft are manufactured with a high wing configuration.
Chapter 12.

Propellers

Introduction

The purpose of a propeller is to convert the power output from a piston engine into thrust. The power developed (BHP) is transmitted to the propeller shaft as engine torque and rotates the propeller. As the propeller rotates it accelerates a large mass of air rearwards at a relatively low velocity and the reaction to this is a force acting in a forward direction known as thrust, which propels an aircraft along its flight path. Propellers are classified as either left-handed or right-handed, i.e. the direction in which they rotate. If viewed from the cockpit a left-handed propeller rotates anti-clockwise whereas a right-handed propeller rotates clockwise. The propellers on most modern single engine aircraft are right-handed (Fig. 12.1).

Propeller Terminology

Fixed pitch propellers are usually manufactured in one piece, and are normally only fitted to low power single piston engined aircraft. They usually consist of two or more blades which are attached to a central hub, and are mostly attached directly to the end of the engine crankshaft (Fig. 12.2)
The propellers are mainly manufactured from aluminium alloy forgings, and are anodised or painted to provide the necessary protection. The blades are of a aerofoil section, and like a wing have leading edges, trailing edges and tips (Fig. 12.3).

The part of the blade nearest the hub is called the shank and is of greater cross-section than the rest of the blade, because this is where the greatest stresses occur in normal operation. The blades are also twisted along their length, and have a decreasing chord and depth of section from root to the tip (Fig. 12.4).
The blades are more twisted at the root than the tip to maintain a constant angle of attack along the complete blade. The angle at which the chord line of each section is inclined to the plane of rotation, i.e. the plane in which the propeller rotates 90° to the crankshaft centre-line, is called the **blade angle** (Fig. 12.5).

![Diagram of propeller blade showing blade angle, plane of rotation, and other components.]

Where **fine pitch** is a setting that produces less resistance to rotation and is a **low blade angle**. **Coarse pitch** is a setting that increases the resistance to rotation and is a **high blade angle**.

The flat side of a propeller blade is called the **blade face**, **pressure face** or **thrust face** (the side facing the pilot) and the curved face is called the **blade back**. The propeller hub is also streamlined by way of an aerodynamically shaped structural cover, called a spinner.

**Factors Affecting the Blade Angle of Attack**

Propeller blades are of aerofoil section and when they are rotated through the air they behave like an aircraft wing. Each blade section produces a total reaction force whose magnitude is determined by the speed and direction at which the relative airflow meets it. This depends on the rotational velocity of each blade section (RPM), and the aircraft’s forward airspeed (TAS). The two components interact to produce an overall resultant velocity of each blade section through the air (Fig. 12.6).
For any given blade section the forward velocity remains constant but the rotational velocity increases with distance from the blade hub, i.e. the closer to the tip the greater the rotational velocity (Fig. 12.7).

As the blade rotates the air will oppose its movement, and each blade section will experience a different relative airflow (Fig. 12.8).
The angle between the relative airflow and the blade section chord line is the **angle of attack**. This angle consequently varies if either the rotational velocity, or the forward airspeed changes (Fig. 12.9).
The propeller blades are thus twisted from the hub to the tip, i.e. the blade angle reduces towards the tip, so that all blade sections along the entire length operate at the same angle of attack (Fig. 12.10).
Factors Affecting the Blade Thrust Distribution

When the propeller is rotating aerodynamic losses occur near the hub and also at the blade tips. The airflow is disrupted near the hub where the propeller blade sections are thicker, to give them adequate structural strength, and also near the engine.

At the propeller blade tips vortices form in a similar manner to that associated with a wing, resulting in induced drag and a reduction in the actual amount of thrust being developed in this region. In practice only a small part of the propeller blade is effective in producing thrust at normal operating angles of attack. This is generally between 60% and 90% of the tip radius out from the blade hub (Fig. 12.11).

At positive angles of attack the greatest useful thrust is produced at 75% of the tip radius, and is consequently where the blade angle of a propeller is referenced to.

Forces Acting on a Blade Section

At normal operating angles of attack the total reaction attributed to each blade section may be resolved into components with respect to the aircraft's direction of flight (Fig. 12.12).
**Thrust** acts **parallel to the direction of flight**, whilst propeller **torque** (the resistance to the motion in the plane of rotation) acts **perpendicular to the direction of flight**. The propeller torque must thus be overcome or balanced by engine torque for a propeller section to provide thrust. As with a wing the relative size of the components depends on the angle of attack, with the greatest ratio of thrust to propeller torque occurring at an angle of attack of **3 or 4 degrees**. As the angle of attack reduces, so the thrust similarly reduces.

There are five operational forces acting on a propeller, these are:-

- Centrifugal Turning Moment (CTM)
- Aerodynamic Turning Moment (ATM)
- Centrifugal Forces
- Thrust Bending Forces
- Torque Bending Forces

**Centrifugal Turning Moment (CTM)**

This is the tendency for the blades to **turn towards fine pitch**. This is a result of the distribution of the masses within a blade that due to the centrifugal forces tend to rotate the blade around its torsional or pitch axis towards its plane of rotation in an attempt to align the masses with the torsional axis. (Fig. 12.13)
Aerodynamic Turning Moment (ATM)

This is the tendency for the blades to turn towards coarse pitch that is due to the centre of pressure of the blade acting ahead of the torsional axis. The higher the blade angle then the greater the affect of ATM. (Fig. 12.14)

Centrifugal Forces

Due to the centrifugal action high stress is felt on the propeller as the blades try to pull out of the hub. (Fig. 12.15)
Thrust Bending Forces

Due to thrust loads the blades tend to **bend forwards** towards the tips where the blades are thinnest. (Fig. 12.16)

Torque Bending Forces

This is the tendency for the propeller blades to **bend back** on themselves against the direction of rotation. (Fig. 12.17)
Propeller Efficiency

Propeller efficiency is defined as the ratio of thrust horsepower delivered, to the engine power required to drive a propeller at a given rpm (brake horsepower), so that:

\[
\text{Propeller Efficiency} = \frac{\text{Thrust Horsepower}}{\text{Brake Horsepower}}
\]

It is alternatively defined as the ratio of useful work done by the propeller in moving an aircraft, to the work supplied by the engine. The work done by the propeller is the product of the thrust and forward airspeed (TAS), whilst the work supplied by the engine is the torque required to turn the propeller at a given rotational velocity (RPM), so that:

\[
\text{Propeller Efficiency} = \frac{\text{Thrust} \times \text{TAS}}{\text{Propeller Torque} \times \text{RPM}}
\]

The efficiency is therefore zero when either the thrust or forward airspeed is zero. If the airspeed is zero no work is being done in moving the aircraft, and so none of the power being delivered by the propeller is being used.

Conversely if the thrust is zero again no work is being done in moving the aircraft. Furthermore since the brake horsepower delivered by the engine is proportional to the propeller torque the efficiency of the propeller will depend on the ratio of the thrust force to propeller torque, which depends on the blade angle of attack.
Fig. 12.18 shows how propeller efficiency varies with blade angle of attack, which like a wing is most efficient when the blade section angle of attack is 3 to 4 degrees. For a fixed pitch propeller the angle of attack at any given RPM is also directly dependent on the aircraft's forward airspeed, so this will additionally determine a propeller's efficiency (Fig. 12.19).

Maximum efficiency thus occurs at one airspeed, and one particular blade angle of attack. The highest efficiency obtained by a propeller is 85% to 88%. The blade angle is usually set so that the speed for maximum efficiency is close to the cruising speed. At any other airspeed the efficiency will be relatively low and only a small proportion of the power being delivered by the engine will be used to propel the aircraft. Consider a fixed pitch propeller travelling at different forward airspeeds at a constant RPM (Fig. 12.20).
At low airspeeds the thrust will increase as the angle of attack is increased, but the speed is low, so propeller efficiency is low, for example no useful work is being done when the aircraft is held against the brakes, with the angle of attack being the same as the blade angle. At high airspeeds the angle of attack will be minimal and the propeller efficiency will again be low.

**Forces Acting on a Windmilling Blade Section**

If the rotational velocity (RPM) is reduced but the airspeed maintained, the blade angle of attack will eventually become **negative** (Fig. 12.21).
When this occurs the total reaction will act in a rearwards direction, and its components will also alter their orientation (Fig. 12.22).
The components of the total reaction are drag and propeller torque. The torque force no longer opposes the blade rotation, but instead acts in the direction of rotation, and assists its rotation, thus driving the engine. At the same time the airflow impinging on the blade back produces a drag force, which opposes the forward flight of the aircraft. The drag force caused by a windmilling propeller can be extremely high, and has a decelerating effect on the aircraft.

**Propeller Pitch**

Due to the interaction between forward velocity and rotational velocity each propeller blade section follows a corkscrew or helical path through the air. For example consider the helix traced by the blade tip in one revolution (Fig. 12.23)

The theoretical distance moved forward in each complete revolution is known as the geometric pitch, although this does not take into account any losses due to inefficiency. The actual distance moved forward in each revolution is known as the effective pitch, and the difference between the two is known as slip, so that:

\[ \text{Slip} = \text{Geometric Pitch} - \text{Effective Pitch} \]

The effective pitch is however not a fixed quantity and varies with forward airspeed, as does the amount of slip for a given rpm.
Fig. 12.24 shows that slip is directly related to angle of attack, whilst the effective pitch is governed by the **helix angle (angle of advance)**. These two angles together constitute the blade angle, so that:

\[
\text{Blade Angle} = \text{Angle of Attack} + \text{Helix Angle}
\]

The effective pitch varies with changes in angle of attack.

**Disadvantages of Fixed Pitch Propellers**

Because the blade angle is fixed during all phases of flight fixed pitch propellers have the following disadvantages:

- They are only efficient at one particular combination of airspeed and rotational velocity (RPM).
- During take-off the angle of attack is large because the airspeed is low and the rotational velocity is high (Fig. 12.25).
During cruise conditions the angle of attack is small, and so forward airspeeds are limited to prevent engine overspeed (Fig. 20.26).

If good take-off performance is required the cruise performance is reduced and vice versa. More complex single engined light aircraft, and multi-engined aircraft, use variable pitch or constant speed propellers, where the blade angle is varied in flight.

The Variable and Constant Speed Propeller

A variable pitch propeller is of the type that the blade angle can be varied in flight so that engine power may be fully utilised. The variable pitch propeller, which is non-governed, was originally produced with two blade settings: a fine or low pitch for take off and climb, and high or coarse pitch to enable full engine speed to be used for cruising.
The introduction of an engine driven governor enables the blade angle to be altered automatically, and is a constant speed propeller that is defined as one, the pitch setting of which varies automatically to maintain a pre-selected constant rotational speed. Therefore the engine and the propeller can work at their maximum efficiency, regardless of whether the aircraft is at, Take Off, Climb, Cruise or Maximum Speed. The blade pitch is varied by the operation of a propeller governor that controls oil flow in and out of a propeller pitch change mechanism to move a piston, which is connected to the propeller blade, thereby changing the pitch angle. The governor is called either a Constant Speed Unit (CSU) in the case of a piston engine, or a Propeller Control Unit (PCU) for a turboprop. The term variable pitch is often used when describing a constant speeding propeller.

In order for the propeller to be efficient over the whole operating range then the blade angle needs to vary to maintain the optimum angle of attack of the blade, which is approximately 2 to 4°. As forward speed is increased the blade angle needs to increase to maintain the same angle of attack as illustrated in fig. 12.27 below. Therefore for take off condition the blade is set to fine at the maximum RPM and as the aeroplane moves forward the governor progressively moves the blades towards coarse maintaining the optimum angle of attack and preventing the RPM from being exceeded. With a constant speed installation then engine power and RPM can be separately controlled.

Power Absorption

For maximum efficiency a propeller must be capable of fully absorbing an engine's maximum power output during its normal operating range as it accelerates air rearwards. If the engine power exceeds the propeller torque, the propeller will overspeed, causing both the engine and propeller to become inefficient.
A propeller capacity for absorbing power depends on the following design features:

- Blade size, e.g. diameter, chord, area.
- Blade section, e.g. camber, thickness.
- Blade angle-pitch
- Number of blades.

If an engine of greater power output is fitted to the aircraft any of these quantities can be increased, although each has its own limitations, and a compromise is normally necessary in the final propeller design. The load on the engine created by the propeller also limits the engine speed.

The blade diameter is an important factor as the greater the diameter then the greater the tip speeds at a lower RPM thus the tips reach sonic velocity earlier. At sonic velocity then the compressibility effects reduce thrust and increase drag, therefore reducing the propeller efficiency. This effect will also result in high noise levels that are unacceptable. One solution to lower propeller noise is to have rather low tip velocity and increase the number of blades.

**Propeller Solidity**

Solidity is the usual method of increasing the power absorption capability of the propeller and is a ratio between that part of the propeller disc that is solid and the circumference at a specified radius. Normally 70% tip radius is used, as this is the most efficient region of a propeller blade.

Solidity can be increased by two ways, increasing the number of blades or the blade chord and can be expressed by the following formula:

\[
\text{Solidity} = \frac{\text{Number of blades} \times \text{chord at 70\% tip radius}}{\text{Circumference at 70\% tip radius}}
\]

However there is obviously a limit to the size and the amount of blades that can be fitted, but modern day methods are constantly being developed like the swept sabre sword shape that increases the solidity whilst safeguarding tip speed.

**Propeller Effects on Take-off**

On take-off rotation of a single engined propeller causes an aircraft to swing to one side, particularly those fitted with a tail-dragger undercarriage. The main causes of this are:

- Slipstream effect.
- Torque reaction.
- Gyroscopic effect.
- Asymmetric blade effect.
To analyse these effects it is appropriate to consider a right-handed propeller, which rotates in a clockwise direction when viewed from the cockpit.

**Slipstream Effect.** As the propeller rotates in a clockwise direction it imparts a rotational flow to the air and produces a slipstream, which passes around the fuselage, and *strikes the left side of the fin* (Fig. 12.28).

![Diagram of propeller and slipstream](image)

**FIG. 12.28**

This causes the aircraft to *yaw to the left* and the actual amount of rotation imparted to the air is dependent on the power setting. To counteract this effect right rudder needs to be applied, but on some aircraft the fin is alternatively offset (Fig. 12.29).

![Diagram of offset fin](image)

**FIG. 12.29**

**Torque Reaction.** This is caused by the air trying to *resist the motion of the propeller*. In doing this it tends to *twist* the engine and airframe in the opposite direction to the propeller, i.e. *anti-clockwise* (Fig. 12.30).
This will cause the aircraft to roll to the left and whilst on the ground will place more weight on the left wheel than the right wheel. This will effectively increase the rolling resistance, i.e. drag, of the left wheel, thus slowing the aircraft down, and causing it to yaw to the left.

**Gyroscopic Effect.** With a tail-dragger type aircraft the tail is lifted off the ground as soon as possible during the take-off run in order to minimise drag and place the aircraft in a flying attitude (Fig. 12.31).

As the tailwheel leaves the ground a forward force will be applied to the top of the rotating propeller disc, tending to alter its plane of rotation in the nose-down sense. Since the propeller disc constitutes a large rotating mass it behaves like a basic gyroscope and tends to resist any attempt to change its plane of rotation. The propeller disc is thus subject to gyroscopic precession and a similar force is will be applied 90° later in the direction of the propeller rotation (Fig. 12.32).
FORCE EXERTED HERE

PROPELLER DISC REACTS AS IF FORCE HAS BEEN APPLIED HERE

AIRCRAFT YAWS TO THE LEFT

FIG. 12.32

This will cause a forward force to act on the right hand side of the propeller disc, and the aircraft will **yaw to the left**. Conversely if an aircraft is **purposely yawed** to the right in flight it will experience a **nose down pitching moment** due to the gyroscopic effect of the rotating propeller. The effects on an aircraft will however be reversed if the propeller rotates in an anti-clockwise direction when viewed from the cockpit, ie. a left handed propeller will yaw the aircraft to the right on take-off.

**Asymmetric Blade Effect.** This effect occurs when the axis of rotation of the propeller is **inclined to the direction of flight**. For example when the tail wheel on a tail-dragger type aircraft is in contact with the ground it will be inclined above the horizontal (Fig. 12.33).

FIG. 12.33

This will cause the **down-going blade** to have a **greater effective angle of attack** than the up-going blade, and will thus develop greater thrust (Fig. 12.34).
The thrust asymmetry between the two blades will cause the aircraft to **yaw to the left** on take-off. Conversely if an aircraft is flying yawed then the asymmetry of the thrust will alternatively cause a pitching moment.

These effects all act together on take-off and yaw the aircraft in the same direction. Some aircraft however compensate for some of these effects, eg undercarriage design and biased trim. For example an aircraft fitted with a tricycle type undercarriage is virtually unaffected by asymmetric blade effect and gyroscope effect because it remains in a level flight attitude during the complete take-off run.

**Propeller Icing**

Ice contamination on the propeller has the same effect as ice on the wings in the way that both **reduce the efficiency** of the aerofoils. The leading edge ice on the propeller will reduce the generated thrust in the same way the lift of the wing is reduced by **creating turbulent air flow**.

In the same way, as the drag of an ice contaminated aeroplane increases, the resisting force on the propeller will increase for a given RPM and blade angle. Ice separating from the propeller blades may **cause damage** to the aeroplane structure and **vibration** due to propeller imbalance.

Since ice on the propeller blades reduces the available thrust then **extra power** may be required from the engine to **overcome the increased drag**.
Chapter 13.
Asymmetric Flight

Introduction

If one engine on a conventional twin engined aircraft fails in flight it will adversely affect its performance and controllability (Fig. 13.1).

This is because the subsequent reduction in thrust will drastically reduce the aircraft’s overall climb capability, and if the airspeed is too low, the resulting yawing moments, due to the failed engine, may even make the aircraft uncontrollable.

Single Engine Performance

Consider a twin piston engined aircraft in normal operation at sea level, with each engine supplying 150 Thrust Horsepower (THP), giving a total available power output of 300 THP with both engines operating (Fig. 13.2).
The THP available in this example is well in excess of that required to maintain level flight and the aircraft exhibits a good rate of climb. If one engine however fails the total power available will be immediately reduced by 50% to 150 THP (Fig. 13.3)

Even with the propeller of the inoperative engine feathered, and the aircraft in a clean configuration, additional drag will exist. This will lead to a significant reduction in the amount of excess power available, which in some aircraft can be as high as 80% or more of its original value. The aircraft’s rate of climb is thus substantially reduced during asymmetric
power conditions. In some cases depending on aircraft type, gross weight, configuration and air temperature, a situation may occur where it is impossible even to maintain level flight. For example, consider an aircraft in its take-off configuration, i.e. with its undercarriage and flaps lowered, where any associated increase in drag will result in a power requirement which exceeds the amount of power available, thereby preventing an aircraft from being able to maintain a given altitude (Fig. 13.4).

![Drag Curve Diagram](image)

**FIG. 13.4**

**Yawing Moments**

![Yawing Moments Diagram](image)

**FIG. 13.5**
With both engines operating on a twin engined aircraft at the same power setting, the amount of thrust produced by each engine will be identical and their lines of action will be **symmetrically displaced** about the aircraft's normal axis (Fig. 13.5).

If one engine fails in flight, the **remaining thrust forces will be asymmetrically displaced**, and the aircraft will yaw in the direction of the failed engine (Fig. 13.6).

![Diagram showing asymmetric displacement of thrust forces](image)

**FIG. 13.6**

The resulting yawing moment is a product of the thrust force and its perpendicular distance from the aircraft's centre of gravity. Thus at any given airspeed the moment will be greatest when the operating engine is producing maximum thrust. The strength of the yawing moment will be also determined by how far the engine is positioned out from the aircraft's centre line, i.e. **the further away the engine is, the greater the yawing moment will be**.

Aircraft engines are therefore normally located as close to the fuselage as possible to minimise the yawing tendency if an engine fails. On propeller driven aircraft the yawing moment is further intensified by the significant rise in drag resulting from the windmilling propeller on the failed engine (Fig. 13.7).
The thrust and drag forces produce a couple which acts about the aircraft's centre of gravity, and will further increase the yawing tendency towards the failed engine. Using the rudder can counteract this yawing tendency, e.g. if the left engine fails the rudder should be deflected to the right, i.e. in the same direction as the operating engine (Fig. 13.8).

If the yawing moment produced by the thrust \ drag couple is balanced by the rudder force multiplied by its distance from the centre of gravity the aircraft will continue along its original
flight path. The further aft the centre of gravity (X) the greater the rudder force needed for a given set of conditions (Fig. 13.9).

![Diagram showing thrust and moment arms, long and short moment arms, forward and aft C.G., rudder force, increased rudder force, and the relationship between thrust and rudder force.]

FIG. 13.9

The rudder must therefore be sufficiently effective to be able to overcome the yawing moment produced by the thrust/drag couple. The amount of force being applied by the rudder depends on the aircraft's airspeed (IAS), so the lower the airspeed the greater the rudder deflection required to produce the same force.

**Asymmetric Blade Effect**

If engine failure occurs on a twin-engined propeller aircraft the resultant loss of power will lead to a reduction in airspeed, and a loss of lift, which may be recovered by increasing the aircraft's angle of attack. This will incline the propeller shaft to the relative airflow and will place the propeller blades of the operating engine at differing effective angles of attack. The descending blade will have a higher angle of attack than the ascending blade and will produce greater thrust (Fig. 13.10).

![Diagram showing the direction of relative airflow, incline of propeller shaft, ascending blade, descending blade, lower angle of attack, higher angle of attack, and line of flight.]

FIG. 13.10
The amount of thrust developed by the descending blade is further augmented by the fact that it moves a greater distance forward than the ascending blade in a given time, and therefore travels faster relative to the air (Fig. 13.11).

These two effects combine to create a thrust line, which is slightly offset from the engines centre line. For example if the propeller rotates in a clockwise direction (viewed from the cockpit), the thrust line will be slightly offset to the right, thus intensifying the yawing moment towards the failed engine (Fig. 13.12).

Conversely if both propellers rotate in the same direction, e.g. Clock-wise, failure of the right engine will result in a lower yawing moment, because the thrust line of the left engine is positioned closer to the aircraft's centre line (Fig. 13.13).
The left engine in this case is considered to be the critical engine, i.e. the engine whose failure would most adversely affect the performance or handling characteristics of the aircraft. If the propellers were left-handed the right engine would be the critical engine. To overcome this problem some aircraft have the engines arranged so that the propeller on the left engine rotates clock-wise, whilst the propeller on the right engine rotates anti-clockwise, e.g. Piper Seneca (Fig. 13.14).
This ensures that the thrust lines act the same distance from the aircraft’s centre of gravity, so that no critical engine exists, and the strength of the yawing moment towards the failed engine is identical if either engine should fail.

The Effect of Bank

If the aircraft is **banked in the direction of the operating engine** it will induce sideslip, and the rudder will become more effective at any given airspeed (Fig. 13.15).

This is because the fin and rudder are presented to the airflow at an increased angle of attack. The **angle of bank should however be strictly limited**, since excessive bank will result in a large reduction in the lift force directly opposing the aircraft’s weight (Fig. 13.16).
In order to recover the lost lift it is necessary to either increase the angle of attack or airspeed. These actions result in increased drag, and consequently more thrust is needed to maintain a given altitude, thus worsening the asymmetric effect. The angle of bank available to counter the effect of engine failure is therefore limited to $5^\circ$.

**The Effect of Weight**

Any increase in weight will ultimately reduce the aircraft’s overall performance, but in a banked condition it will induce greater sideslip, and the rudder will become more effective for a given airspeed (Fig. 13.17).
Thus the greater the weight, the greater the induced sideslip, and the greater the rudder effectiveness. This benefit is however insignificant compared to the penalties associated with any additional weight.

**Rolling Moments**

A *secondary effect* of engine failure is a *rolling moment towards the failed engine*, which is mainly attributed to the variation in the wing lift distributions (Fig. 13.18).

This is primarily due to the *absence of propeller slipstream* behind the failed engine and the disturbance of the airflow behind the windmilling propeller. The aircraft will initially continue to travel along its original flight path due to its inertia and as it yaws towards the failed engine the outer wing will travel faster than the inner wing, and will produce more lift. This will cause the roll towards the failed engine to be intensified. Due to sideslip, the fuselage will also shield part of the wing on the side of the failed engine, thus weakening its lift distribution, and *intensifying the rolling moment* towards the failed engine (Fig. 13.19).
Minimum Airspeeds During Asymmetric Flight

If the airspeed is too low, the amount of rudder deflection necessary to overcome the yawing moment may be insufficient to maintain directional control. The speed at which this occurs is known as the Minimum Control Speed (\(V_{MCA}\)), and is the minimum airspeed at which the aircraft can be safely controlled in the air. \(V_{MCA}\) is defined as the limiting airspeed at which it is still possible to maintain directional control if the critical engine suddenly fails during the take-off and climb phases of flight, with an angle of bank of not more than 5°. \(V_{MCA}\) should not exceed 1.2 \(V_{s}\) with the:-

- Operating engine at its maximum available take-off power / thrust setting.
- Aircraft in its take-off configuration.
- Propeller on the failed engine windmilling.
- Centre of gravity as far aft as possible, and the aircraft at its maximum take-off weight.
- Aircraft trimmed for take-off.

If one engine fails when the aircraft is operating close to \(V_{MCA}\), it is vital to reduce the drag as soon as possible, by feathering the propeller on the failed engine.

If the critical engine fails during the take-off phase the airspeed must be maintained above \(V_{MCG}\), which is the minimum control speed with the wheels still on the ground. \(V_{MCG}\) is defined as the minimum control speed on the ground during the take-off run when it is still possible to maintain directional control using the rudder only. \(V_{MCG}\) must be established with the:
Aircraft in its most critical take-off configuration.
- Operating engine at its maximum available power / thrust setting.
- Centre of gravity as far aft as possible, and the aircraft at its maximum take-off weight.
- Aircraft trimmed for take-off.

If the critical engine fails during the approach and landing phase of flight the airspeed must be maintained above $V_{MCL}$, which is the minimum control speed during the approach and landing. $V_{MCL}$ is defined as the minimum control speed at which it is possible to maintain directional control in the landing configuration with an angle of bank of not more than 5°. $V_{MCL}$ must be established with the:

- Aircraft in its most critical landing configuration and all engines operating.
- Centre of gravity as far aft as possible and the aircraft at its maximum landing weight.
- Propeller on the failed engine (propeller aircraft only) in the position it achieves without pilot action, whilst maintaining a 3° glide slope.
- Go-around power / thrust setting on the operating engine.

$V_{MCL}$ is a fixed value for a given aircraft, but $V_{MCA}$ and $V_{MCG}$ both reduce with increasing altitude.

**Turning Flight**

During turning flight the main factors, which affect it under asymmetric power conditions, are airspeed and the direction of the turn relative to the failed engine. For example consider an aircraft where the left engine has failed and the yawing and rolling moments have been stabilised using right rudder. To initiate and hold a balanced left turn the amount of right rudder needed to counteract the yaw will have to be reduced, whilst a balanced right turn will require additional right rudder under the same conditions. If the airspeed is too low, turning towards the operating engine may reduce the control forces to a critical level. At low indicated airspeeds it is therefore necessary to limit turns to only small angles of bank, since rudder deflection may become insufficient to maintain a balanced turn.

**Recognition of a Failed Engine**

In steady level flight an engine failure on a twin engined aircraft will cause roll and yaw in the direction of the failed engine unless corrective action is taken. Similarly if an engine fails during a climbing or level turn the aircraft will tend to yaw towards the failed engine. Any engine failure in these conditions is therefore easily recognisable and the aircraft will continue to remain controllable provided the airspeed remains above $V_{MCA}$. Under low power conditions and relatively high airspeeds, e.g. during the descent, the yaw and roll forces resulting from an engine failure will be relatively small. An engine failure under these conditions will not be easily recognisable.
Chapter 14.

High Speed Flight

Introduction

During high-speed flight range significant changes occur in the flow and pressure distributions around the aircraft, which result in a loss of lift and an increase in drag. This is caused by the formation of shock waves, which adversely effect the stability and control characteristics of the aircraft. Aircraft designed to fly at high Mach numbers therefore incorporate features, which are designed to minimise these effects.

The Speed of Sound

The Speed of Sound is defined as the rate at which small pressure disturbances are propagated through the air, which is solely a function of air temperature (Fig. 14.1).

The speed of sound is therefore solely dependent on the ambient air temperature and varies with altitude as illustrated in the following table.
<table>
<thead>
<tr>
<th>ALTITUDE (ft)</th>
<th>TEMPERATURE (°C)</th>
<th>SPEED OF SOUND (Knots)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sea Level</td>
<td>15.0</td>
<td>661.7</td>
</tr>
<tr>
<td>5000</td>
<td>5.1</td>
<td>650.3</td>
</tr>
<tr>
<td>10000</td>
<td>-4.8</td>
<td>638.6</td>
</tr>
<tr>
<td>15000</td>
<td>-14.7</td>
<td>626.7</td>
</tr>
<tr>
<td>20000</td>
<td>-24.6</td>
<td>614.6</td>
</tr>
<tr>
<td>25000</td>
<td>-34.5</td>
<td>602.2</td>
</tr>
<tr>
<td>30000</td>
<td>-44.4</td>
<td>598.6</td>
</tr>
<tr>
<td>35000</td>
<td>-54.3</td>
<td>576.6</td>
</tr>
<tr>
<td>40000</td>
<td>-56.5</td>
<td>573.8</td>
</tr>
<tr>
<td>50000</td>
<td>-56.5</td>
<td>573.8</td>
</tr>
<tr>
<td>60000</td>
<td>-56.5</td>
<td>573.8</td>
</tr>
</tbody>
</table>

The speed of sound at sea level is approximately 660 Knots and steadily reduces up to the base of the tropopause, where after it remains constant.

Fully subsonic aircraft can be heard approaching because they send out pressure disturbances, or waves in all directions, which travel at the Speed of Sound. This enables an approaching aircraft to be heard, and more importantly for the aircraft, the air to be warned of its approach. The sound is transmitted to ones ears by way of a series of molecular collisions. Conversely aircraft travelling supersonically can not be heard because the air ahead gets no warning of their approach, and no molecular collisions take place.

**Pressure Waves From a Moving Source**

If the object is moving at a speed less than the local speed of sound, sound waves will move out from the source in all directions, but will tend to be closer together ahead of the source than behind it (Fig. 14.2)
The waves maintain their separation and have no tendency to bunch up. As long as the pressure waves continue to travel faster than the source a disturbance will occur ahead of the object as a normal wave under subsonic flow conditions. If the source is moving at the speed of sound, the waves will no longer move ahead of the source and will bunch up to form a normal Mach wave, which acts at right angles to the direction of movement (Fig. 14.3).

If the source travels faster than the wave progression, i.e. at a speed greater than the speed of sound, a supersonic flow condition will exist and the waves will pile up on each other to form a boundary beyond which no wave passes. This boundary is called the ‘Oblique Mach Wave’, and the angle it makes with the flight path is called the ‘Mach Angle’ (Fig. 14.4).
The Mach Wave is inclined backwards to form a Mach Cone and as the speed of the object increases the angle of the cone or 'Mach Angle' becomes more acute. The Mach Cone described later under supersonic flight.

**Nature of Compressibility**

Air is termed incompressible whenever it undergoes changes in pressure without apparent changes in density, i.e. air is analogous to the flow of water, hydraulic fluid or any other incompressible fluid. If air was fully incompressible, the speed at which pressure disturbances travelled would be infinite. The disturbance created by the aircraft would thus be felt everywhere instantaneously, regardless of the aircraft speed. Air is however compressible a change in density and temperature accompanies a change in pressure. The speed of propagation of the pressure waves therefore has a finite value, which is directly related to the Speed of Sound. The principle difference between low subsonic and supersonic airflow, is that supersonic airflow is compressible. Any change in the velocity or pressure of the supersonic airflow will result in a change in density. The following table identifies the main differences between subsonic and supersonic airflow in a stream-tube:

<table>
<thead>
<tr>
<th>SUBSONIC FLOW</th>
<th>SUPersonic FLOW</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>CONVERGING CHANNEL</strong></td>
<td>increasing velocity,</td>
</tr>
<tr>
<td></td>
<td>decreasing static pressure,</td>
</tr>
<tr>
<td></td>
<td>constant density</td>
</tr>
<tr>
<td></td>
<td>increasing static pressure,</td>
</tr>
<tr>
<td></td>
<td>increasing density</td>
</tr>
<tr>
<td></td>
<td>decreasing static pressure,</td>
</tr>
<tr>
<td></td>
<td>reducing density</td>
</tr>
</tbody>
</table>

**Mach Number**

Mach number is named after Ernst Mach, an Austrian physicist, and is the ratio of the actual speed of a body or flow, to the speed of sound in the surrounding atmosphere, so that:
Mach Number \( (M) = \frac{\text{The Speed of a body or flow}}{\text{The Speed of Sound in the same region}} = \frac{V}{a} \)

For example, if an aircraft is travelling at half the speed of sound the Mach number will be 0.5. From the basic Mach number the following definitions can be derived:

**Free Stream Mach Number \( (M_{FS}) \).** This is the Mach number of the airflow sufficiently remote from the aircraft so as not to be affected by it, so that:

\[
M_{FS} = \frac{\text{Aircrafts True Airspeed}}{\text{Local speed of Sound}} = \frac{TAS}{LSS}
\]

**Local Mach Number \( (M_{L}) \).** This is the actual speed of the flow over a surface. For example the airflow accelerates as it passes over a wings surface, so that the local Mach number on top of the wing is always greater than the free stream Mach number (Fig. 14.5).

![FIG. 14.5](image)

For a given aerofoil section and free stream Mach number the local Mach number also varies directly with changes in the angle of attack.

**Critical Mach Number \( (M_{CRIT}) \).** This is the value of the free stream Mach number when the local Mach number first becomes sonic anywhere on an aircraft, which normally initially occurs on the upper surface of an aircraft’s wing near to the point of maximum thickness (Fig. 14.6).

![FIG. 14.6](image)
Flight Speed Classifications

In flight the airspeed of an aircraft determines whether the airflow is travelling subsonically or supersonically around the aircraft:

**Subsonic.** Aircraft speeds approximately **Mach 0.75 or less**, where the total airflow around an aircraft is travelling at a speed less than the speed of sound.

**Transonic.** Aircraft speeds between **Mach 0.75 and Mach 1.2**, where the airflow around an aircraft is partly subsonic, and partly supersonic.

**Supersonic.** Aircraft speeds between **Mach 1.2 and Mach 5.0**, where the total airflow around an aircraft is travelling at a speed greater than the speed of sound.

Comparison of Subsonic and Supersonic Flow Patterns

As an object moves through the air, velocity and pressure changes occur, which create pressure disturbances in the airflow surrounding the object. In subsonic flow, a pressure wave is set up ahead of the object, which results in upwash and causes the flow to change direction well ahead of the leading edge (Fig. 14.7)

![Flow Changes Direction Well Ahead of Leading Edge](image)

**FIG. 14.7**

In supersonic flow, where the object is travelling at speeds in excess of the speed of sound, the pressures acting upon it will not influence the flow ahead of the object. It will only be influenced when the air particles are suddenly forced out of the way by a concentrated pressure wave set up by the object (Fig. 14.8).
NO CHANGE OF FLOW DIRECTION
APPARENT AHEAD OF LEADING EDGE
FIG. 14.8

The Development of Shock Waves

When aircraft fly through the air every part of them sets up tiny pressure disturbances, which radiate outwards at the speed of sound in all directions. This is similar to the ripples produced when a stone is dropped into stagnant water, where the resulting pressure waves travel outwards from the source in expanding spheres, each travelling at the speed of sound (Fig. 14.9).

At low subsonic airspeeds, eg. Mach 0.6, the disturbances travel forward faster than the oncoming flow, thus warning the air ahead of the aircraft's approach. As the air flows over the wing it accelerates, but remains fully subsonic, and no shock wave will form (Fig. 14.10).
With increasing Mach number the flow over the wing continues to accelerate and eventually reaches a sonic value at a particular point on the wing, normally the point of maximum thickness (Fig. 14.11).

The aircraft is now travelling at its critical Mach number with the airflow either side of this point remaining subsonic. With increasing Mach number this point grows into an area of supersonic flow, so the air moving over the upper surface will now be moving rearwards faster than the pressure disturbances can move forwards. These disturbances consequently pile up on each other and form a shock wave (Fig. 14.12).

The shock wave acts perpendicular or normal to the surface, and is more commonly referred to as normal shock wave. Notably this occurs where the flow changes from supersonic back to subsonic. With increasing Mach number the shock wave grows in intensity and moves rearwards with a greater portion of the upper surface being covered by supersonic flow. The overall direction of the airflow remains the same, but as it passes through the shock wave the following changes take place:

- The flow is rapidly decelerated to a subsonic Mach number.
- The static pressure is substantially increased.
- The density of the air is suddenly increased.
- The kinetic energy of the boundary layer is substantially reduced.

At Mach 0.82 a weak shock wave also forms on the lower surface of the wing (Fig. 14.13).
As the free stream Mach number approaches the speed of sound the areas of supersonic flow continue to grow as both shock waves increase in intensity and move rearwards (Fig. 14.14).

At speeds just above the speed of sound (M 1.05) another shock wave appears ahead of the wing, which is known as a bow wave, and the original shock waves move to the trailing edge (Fig. 14.15).
Behind this shock wave a small region of subsonic flow exists, but everywhere else the flow is supersonic. Finally when the flow is fully supersonic ($M \geq 2.0$), fully developed bow and tail waves firmly attach themselves at the leading and trailing edges respectively (Fig. 14.16).

The Mach number when this occurs is called the ‘Shock Attachment Mach number’ ($M_{SA}$).

**Shock Stall**

At airspeeds above the critical Mach number the formation of a shock wave and its associated pressure gradient results in a significant increase in drag and a reduction in lift (Fig. 14.17).
This results from the **sudden increase in pressure** across the shock wave, which causes **localised heating** of the air, and the eventual **separation** of the boundary layer **behind the shock wave**. At airspeeds just above the critical Mach number the increase in drag is mainly due to the loss of kinetic energy used in heating the air, which needs to be continuously supplied by the engines.

With **increasing Mach number** the **strength of the shock wave steadily increases**, as does the size of the adverse pressure gradient, and this determines the point at which the boundary layer separates from the surface. Both upper and lower surface shock waves can cause separation of the airflow, and as in the case of a conventional low speed stall, the larger the adverse pressure gradient the larger is its associated turbulent wake (Fig. 14.18).

When separation firstly occurs the coefficient of lift begins to fall and the coefficient of drag begins to rise rapidly (Fig. 14.19).
This is called the 'shock stall'. It differs from a conventional low speed stall because it normally occurs at low angles of attack, although with increasing angles of attack the stall will occur at a lower Mach number. The combined effect of the energy loss across, and the turbulent wake behind the shock wave is called 'wave drag' (Fig. 14.20).

The drag varies significantly from the standard drag curve at the 'drag divergence speed', and the associated increase in drag is known as the 'transonic drag rise'. This is similar to a conventional low airspeed stall since the separation of the boundary layer during the shock stall also results in buffeting of the aircraft, and a reduction in control effectiveness.

As the upper layer surface shock wave moves rearwards with increasing Mach number, the region of shock induced separation reduces, and once the lower surface shock is established at the trailing edge some measure of recovery may occur. A common characteristic of shock wave induced separation is the increasing severity of buffet intensity with increasing Mach number. In fact, it is possible that the maximum angle of attack may not
be achievable due to this severe buffet intensity. The aircraft’s manoeuvring capability (load factor) will also be reduced.

The Effect of Altitude on the Shock Stall

Apart from the engine limitations, which affect an aircraft’s maximum attainable altitude, it is also determined by the height at which an aircraft can fly without stalling (Fig. 14.21).

![Subsonic Speed Range of Flight - True Airspeeds](image)

Stalling can occur conventionally at low airspeeds and high angles of attack, or at high airspeeds due to shock stall. A specified range of flight speeds is attainable between the two limits at a given altitude.

The margin between the two types of stall however narrows with increasing altitude. Notably the true airspeed of the low speed stall will increase with increasing altitude for a given indicated airspeed, whilst the true airspeed of the shock stall will reduce up to the base of the tropopause, and there after will remain constant. During manoeuvres the two stalls will occur at considerably lower altitudes, because the high angle of attack stalling speed will increase, whilst the shock stalling speed will decrease.

The point at which the two stalls coincide is often referred to as Coffin Corner. The altitude that that is eventually reached at which an aircraft can fly at one airspeed in a 1g manoeuvre is called the Aerodynamic Ceiling. Since this condition has no safety margin then aircraft must be operated within a buffet margin of 0.3g therefore it is normal to draw a buffet onset boundary chart for 1.3g.
Buffet Onset Boundary Chart

This chart is specific to a particular type of aircraft and shows the relationship of altitude, load factor, cruise data, weight, Mach number and IAS (Fig. 14.22). The chart can be used to find (1) the manoeuvre margin in terms of load factor and bank angle and (2) the low and high speed buffet for 1g flight. For example, given the following data: Airspeed of M 0.72, a flight level of 350, a CG of 10% MAC and a gross mass of 50000 kg.

1. Entering the chart at point A with the airspeed of M 0.72, a line is drawn vertically up until it intersects the flight level curve of 35000 ft at point B. Now a horizontal line is drawn to the right until it intersects the CG reference line at point C. From point C a line is drawn back following the slope to point D that is the 10% MAC point. Now a line is drawn horizontally to the right until it intersects the gross weight curve at point E. A line is then drawn vertically downwards to intersect the load factor and bank angle scale to determine the values for the initial buffet, in this case 1.47g and 47° respectively.

2. Entering the chart at the 1g position a line is drawn vertically upwards until it intersects the gross weight curve at point F. From point F a horizontal line is drawn to the left to the 10% CG position and back to the CG reference line and then to the left to intersect the flight level curve of 35000 ft at point G. From this point a vertical line is drawn down until it intersects the speed scale at M0.814, which is the 1g high speed buffet boundary. To find the 1g low speed buffet boundary the horizontal line from point G is continued to point H on the flight level curve. A vertical line is drawn down until it intersects the speed scale, giving a speed of M 0.54 or 178 kts.

If the aircraft is assigned a different flight level but maintains its IAS the Mach number will vary, e.g. if the altitude reduces the Mach number will reduce. This is particularly important if the aircraft is assigned a higher altitude than expected by Air Traffic Control, and the pilot needs to be aware of the aircraft’s margin to buffet.
Methods of Reducing or Delaying the Transonic Drag Rise

Due to the fact that undesirable effects are associated with shock waves various design features are used to either overcome, or reduce these effects. In subsonic jet aircraft this is achieved by using designs, which increase the value of $M_{CRIT}$, so that the aircraft enters the transonic range at a higher airspeed. Other aircraft incorporate features, which are designed not to delay the onset of transonic flow, but instead to minimise the adverse effects. Various design features that affect the wing and tailplane of the aircraft are used to accomplish this:

**Wing Section.** The following design features of a wing section are used to increase $M_{CRIT}$:

- Low thickness/chord ratio.
- Maximum thickness well aft.
- Small leading edge radius of curvature.

**Wing Planform.** This design has the most significant effect on $M_{CRIT}$. It not only delays the shock stall, but also significantly reduces the severity when it occurs. If a wing has sweepback the effective chord (parallel to the aircraft’s longitudinal axis) will be lengthened, but the wing’s thickness will remain unchanged (Fig. 14.23).

![FIG. 14.23](image)

This will reduce the thickness/chord ratio of the wing. This will result in a higher value of $M_{CRIT}$, and will delay the transonic drag rise (Fig. 14.24).
Thus the greater the sweepback the higher the value of $M_{CRIT}$, and the greater the reduction in drag under all transonic speeds. Although sweepback is a great asset in increasing the critical Mach number it does have a number of disadvantages, which are:

- A reduction in the coefficient of lift, which increases stall speed.
- Wing tip stalling leading to pitch-up.
- Low aspect ratio leading to increased induced drag at high angles of attack, which is particularly dangerous during take-off and landing.

The Tailplane. The tailplane behaves similarly to the wing, where shock associated drag is reduced by utilising thin sections and sweepback. The tailplane is also designed to have a higher critical Mach number than the wing, so that shock stall can be avoided, and full elevator efficiency maintained.

Transonic Area Rule

Regardless of the aircraft’s configuration there is always additional drag due to interference between the various components. Interference drag can reach extremely large values at transonic airspeeds, and thus to minimise it the cross-sectional area along its complete length must follow a smooth pattern, with the area gradually increasing to a maximum, and then decreasing again giving the optimum area distribution (Fig. 14.25).
Supercritical Wings

To reduce the severity of the shock stall, and allow aircraft to travel faster some modern jet transport category aircraft have supercritical wings. The point of maximum thickness is positioned close to the trailing edge, and the upper surface has a very slight curvature. This ensures that the localised Mach number remains just above the critical Mach number, and results in a flattish pressure distribution over the majority of the upper surface (Fig. 14.26).

This ensures that the flow will gradually decelerate near the trailing edge, to a subsonic speed, to discourage the formation of shock waves. The wings are thicker at the root than conventional wings and more fuel can be stored in them. The increased thickness at the root also allows the wings to be of lighter construction. These wings also have less sweepback, giving them a higher aspect ratio, and thus better lift characteristics at a given angle of attack.
Control Problems in Transonic Flight

When shock waves form on the aircraft's wing the shock induced separation behind them, can lead to vibration and control surface ineffectiveness (Fig. 14.27).

The disturbed airflow over the control surfaces may cause uncommanded erratic movements, although this will not directly affect the air ahead of the shock wave, because the resulting pressure disturbances are prevented from travelling forwards. The pressure distribution over the front of the wing is however altered; which will vary the position of the wing's centre of pressure, and it's overall pitching moment. This will alter the wings angle of attack, and will result in rapid backwards and forwards movements of the shock waves. A kind of instability will be set up, and the rapid changes in the pressure distribution will result in vibration of the whole aircraft. This is primarily due to the distributed airflow behind the shock wave hitting the tailplane. If shock waves form on the control surfaces it will also affect the stick forces by altering their hinge moments (Fig. 14.28).

Since the hinge moment, which opposes the movement of the control surfaces is the product of the force acting through its centre of pressure times its distance from the hinge line, it will fluctuate, as will the stick force in phase with any shock wave movement. Thus any rearward
**movement of the shock wave** acting on the control surface will result in **increasing stick forces**. This will reach a maximum value when the shock wave is at the trailing edge (Fig. 14.29).

![Diagram of shock wave and control surface](image)

**FIG. 14.29**

If the centre of pressure moves ahead of the hinge-line transitory overbalance will occur and **control surface reversibility** will take place (Fig. 14.30).

![Diagram of shock wave and control surface](image)

**FIG. 14.30**

Since the shock waves move quickly with changes in control surface deflection the effect will be felt on the flight deck as **snatching or buffeting**, depending on the position of the control surface. The disturbed air resulting from shock induced separation also precludes the use of aerodynamic balance methods, in particular tabs, so power operated controls are normally used in preference to manually operated controls. Other methods used to overcome these control problems are:

- All moving surfaces.
- Surfaces of low thickness/chord ratio.
- Larger control surfaces.
- The position of the tailplane.
- The fitment of vortex generators.
Vortex Generators

These are small wing like surfaces, which are fitted in front of the control surface, and project vertically upwards into the airstream (Fig. 14.31). They operate by forcing high-energy air into the boundary layer, thus enabling it to overcome the adverse pressure gradient caused by the shock wave, and thus delaying its separation.

![Diagram of Vortex Generators](image)

FIG. 14.31

The Effect of Transonic Flight on Aircraft Trim and Stability

In transonic flight it is the nature of the airflow behind a shock wave, which determines the stability characteristics of the aircraft. The common effects are:-

**Longitudinal Stability.** This is the most commonly affected form of stability. It is determined by movement of the centre of pressure, coupled with changes in the coefficient of lift, and the amount of downwash acting on the tailplane. In straight and level flight at low subsonic airspeeds the centre of pressure is typically located aft of the centre of gravity (Fig. 14.32).
This will result in a **nose-down pitching moment**, which must be counteracted by placing a small download on the tailplane. If the shock stall occurs any lift aft of the shock wave will be destroyed, and the tailplane will become covered in disturbed airflow (Fig. 14.33).

The **downwash acting on the tailplane** will consequently be **eliminated** causing a smooth and radical change in its angle of attack. The angle of attack will become more positive and the download acting on the tailplane will become an upload, which in conjunction with the wing pitching moment will cause a **violent nose-down pitching moment**, known as ‘tuck-under’. The exact nature and strength of the changes in trim and stability will however be dependent on the design of the aircraft.

**Lateral Stability.** Disturbances about the longitudinal axis are often encountered in transonic flight and are characterised as a **wing heavy tendency** as the critical Mach
number is exceeded. This occurs because shock waves do not always form simultaneously, nor at identical places on opposite wings. The design features, which normally provide lateral stability, may consequently reverse the effect, and will aggravate a dropping wing. This occurs because the downgoing wing sideslips and causes the airflow to accelerate. This will intensify the shock wave and will cause the wing to drop further.

**Directional Stability.** This form of stability is affected by the variation in wing shock wave formations, which results in different drag characteristics. For example if a shock wave first forms on the left wing, the associated increase in drag will cause the aircraft to yaw in the same direction. Whilst yawing to the left the airflow will then accelerate over the right wing, so intensifying the shock wave and increasing the drag. This process is thus self-perpetuating and will result in snaking or Dutch roll, depending on the lateral and directional characteristics of the aircraft.

**Mach Trim**

To guard against nose tuck under frequent pitch trim changes are required. This is carried out by a variable incidence tailplane, which is automatically positioned by way of a Mach Trim System. This system is designed to aid aircraft longitudinal stability, and ensures that the forward stick forces increase proportionally with increasing Mach number. It is operational at high Mach numbers in the Transonic speed range.

**Supersonic Flight**

The supersonic flight range starts at about Mach 1.2 to 1.3, depending on the individual aircraft design. The airflow about a surface varies immensely from that in transonic flight and forms a series of oblique shock waves and expansion waves.

**Oblique Shock Wave**

An oblique shock wave is a compression wave and is similar to a normal shock wave, except that the airflow changes direction into a corner and its velocity decreases to a lower supersonic value (Fig. 14.34).
The wave angle depends on the Mach number of the approaching flow, and the angle of the wedge. This type of shock wave is weaker than the normal shock wave, but the energy loss still has to be overcome by the aircraft engines. As the air passes through an oblique shock wave its pressure, temperature and density all increase.

Mach Cone

Only the region behind the oblique shock wave will be affected by disturbances and is sometimes referred to as the zone of action. The region ahead of the oblique shock wave is not affected by the disturbances, and is called the zone of silence. In three dimensions the disturbances emanating from the moving body expands outwards as spheres and not circles. When the speed is above Mach 1 these spheres are enclosed within a cone, the Mach Cone and it is within the Mach Cone that disturbances are felt.

If the source of the disturbance is a wing then the Mach Lines generate two oblique plane waves forming a wedge.

Expansion Wave

 Expansion waves are the opposite of shock waves (compression waves), and form where the airstream turns around a convex corner (Fig.14.35).
Pressure, temperature and density all decrease as the air flows through an expansion wave. The velocity of the air also increases to a higher supersonic value as it passes through the wave, but no energy is lost, and lift is produced as the static pressure decreases. This is the main reason for using double wedge aerofoils for supersonic flight, although the subsonic characteristics will be very poor. To avoid these subsonic problems circular arc or bi-convex aerofoil sections are used, which uses two arcs of circles to define their shape.
Summary of Supersonic Wave Characteristics

Below is a table summarising the changes that occur as the airflow passes through various forms of supersonic waves.

<table>
<thead>
<tr>
<th>Change of Direction of Flow</th>
<th>Normal Shock Wave (Compression)</th>
<th>Oblique Shock Wave (Compression)</th>
<th>Expansion Wave (No Shock)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>No Change</td>
<td>“Into a Corner” where it turns into the preceding airflow</td>
<td>“Around a Corner” where it turns away from the preceding airflow</td>
</tr>
<tr>
<td>Effect on Velocity and Mach No</td>
<td>Decreases to Subsonic Speed</td>
<td>Decreases but remains Supersonic</td>
<td>Increases to a Higher Supersonic Speed</td>
</tr>
<tr>
<td>Effect on Static Pressure</td>
<td>Large Increase</td>
<td>Increases</td>
<td>Decreases</td>
</tr>
<tr>
<td>Effect on Density</td>
<td>Large Increase</td>
<td>Increases</td>
<td>Decreases</td>
</tr>
<tr>
<td>Effect on Temperature</td>
<td>Large Increase</td>
<td>Increases</td>
<td>Decreases</td>
</tr>
<tr>
<td>Effect on Total Pressure</td>
<td>Large Decrease</td>
<td>Decreases</td>
<td>No Change since there is no Shock</td>
</tr>
</tbody>
</table>
Chapter 15.

Flight in Adverse Weather Conditions

Introduction

When considering flight in adverse weather conditions it is vital to know how windshear and any accumulation of ice or frost on the aircraft will affect its flight performance. Either condition will seriously affect the aircraft's climb capability, and may even prove fatal if ignored.

The Effect of Ice, Frost and Snow on the Aircraft's Performance

Any accumulation of ice, frost or snow on the main surfaces of the aircraft will have a detrimental effect on its overall performance and handling characteristics. The most critical areas of contamination are the wings and tail surfaces for frost, snow or ice, and the engines and pitot-static systems for ice (Fig. 15.1).

![WING & TAIL SURFACES](image1)

![ENGINES](image2)

**FIG. 15.1**

The most important areas are the wings and tailplane, where the lift capability depends on the section shape and camber. For example a clean modern wing will produce approximately twice the amount of lift developed by a flat plate at the same angle of attack. If the aerofoil is contaminated with frost, snow or ice its maximum lift capability will steadily deteriorate and under severe contamination may reduce to that achievable from a flat plate (Fig. 15.2).
Any surface contamination will reduce the aircraft's stalling angle of attack and its overall climb performance. Pitch and roll pre-stall flight characteristics may also occur before the stick shaker activates during a normal take-off.

The Effects of Contamination on Maximum Wing Lift Capability

The formation of ice along a wing's leading edge will greatly alter its aerodynamic contour and will also lead to premature separation of the boundary layer. This will reduce the wing's stalling angle of attack, and also its maximum coefficient of lift by up to 50% (Fig. 15.3).
The formation of ice will additionally increase the aircraft's gross weight, and may even increase the stalling speed by up to 30%. Ice on the surface’s will also cause a large increase in drag, and will require additional thrust for the aircraft to be able to maintain steady level flight (Fig. 15.4).

![Diagram showing increase in stalling speed and drag](image)

**FIG. 15.4**

By comparison a coating of hard frost will not significantly alter a wings aerodynamic contour, but it will produce a surface of considerable roughness (Fig. 15.5).

![Diagram showing lift and roughness](image)

**FIG. 15.5**

The roughness of frost is similar to that of sandpaper and produces a proportionately large increase in skin friction, which results in a substantial reduction in boundary layer energy. A wing contaminated with frost will thus stall at a lower angle of attack compared to a clean wing, and will also have a reduced maximum coefficient of lift (Fig. 15.6).
The overall reduction in these values is however not normally as great as those associated with an ice contaminated wing.

The Effects of Contamination on Flaps and Slats

Most transport category aircraft are fitted with a combination of slotted trailing edge flaps and leading edge slats, to improve the low speed handling characteristics, and any contamination will have a detrimental effect on their overall efficiency. Firstly consider a slotted flap, which operates by venting high energy air through a slot, to re-energise the low energy upper surface boundary layer, and delay the separation of the airflow over the flap (Fig. 15.7).
Air passing through the slot from the lower surface will increase the flap lift capability, reduce the thickness of the separation wake behind the flap, and give an overall reduction in profile drag. Any upper surface contamination will reduce the energy possessed by the boundary layer, and will result in earlier separation of the airflow. This will lead to a reduction in the flap lift capability, and will increase the form drag (Fig. 15.8).

A coating of frost or frozen slush on the lower surface of the wing will also act to decelerate the flow, and will further reduce the energy possessed by the upper surface boundary layer. This will result in earlier separation of the airflow, and if ice forms in the slot the condition will be worsened. Slats operate in the same way as slotted flaps by venting high velocity air into the upper surface boundary layer (Fig. 15.9).
ACCELERATED FLOW
STAGNATION POINT

FIG. 15.9

This will produce a 20-50% increase in the maximum coefficient of lift, but if ice forms on the leading edge of the slats this may reduce to only 5-10%. Contamination of the slats will also create boundary layer disturbances, which will tend to downgrade the efficiency of the slotted trailing edge flaps, and will reduce the wing’s stalling angle of attack (Fig. 15.10).

FIG. 15.10

The Effect of Contamination on Take-off Performance

Any wing contamination will greatly affect the take-off performance of the aircraft, with the associated increase in drag not only extending the take-off run, but also reducing the aircraft’s climb performance (Fig. 15.11).

FIG. 15.11
Any increase in the stall speed may also dangerously reduce the margin to stall on takeoff, and may even reach a critical level, especially when operating in turbulent conditions (Fig. 15.12).

![Diagram showing increased stall speed and reduced margin to stall](image)

**FIG. 15.12**

The resulting increase in aircraft drag will also reduce the amount of excess thrust available at a given climb speed, thus decreasing the aircraft's angle of climb. The aircraft's rate of climb is similarly reduced. With severely contaminated wings, the pitch and roll pre-stall buffet characteristics compared to a clean wing, may occur before the stick shaker activates (Fig. 15.13).

![Diagram showing stick shaker activation and wing angle of attack](image)

**FIG. 15.13**

This occurs because the stick shaker normally activates at a specific angle of attack, and a contaminated wing normally stalls before this angle is reached. The stick shaker may therefore not provide adequate stall warning, although pre-stall warnings in the form of buffet
should normally be sufficient to warn of an impending stall. A contaminated aircraft may even stall at an angle of attack lower than that associated with normal rotation. Any wing and tailplane contamination can also upset the trim characteristics of an aircraft, and may lead to a nose-up out of trim condition during the rotation (Fig. 15.14).

Surface contamination may also cause a decrease in stick force, and in conjunction with the out of trim condition will result in higher rotation rates for the same stick-force input, and will require a push force to counter these effects. On twin engined aircraft with both engines operating any wing contamination will result in a reduction in the climb capability, but with one engine inoperative the reduction in climb capability will be much more pronounced, and may even produce a rate of sink. It is therefore important to ensure that all the lifting surfaces are free of ice, frost and snow before a take-off is commenced.

The Effect of Contamination on Aircraft Landing Performance

Wing and tail contamination affects an aircraft's landing performance, because it increases the aircraft's stall speed by up to 20-30%, and also reduces the margin to stall. To compensate for this it is necessary to increase the landing speed to provide a safe margin above the stall speed. This will substantially increase the aircraft's landing distance, and will also reduce its braking efficiency (Fig. 15.15).
If the margin to stall is not increased then high sink rates will result due to increased drag, and may even cause an uncontrollable loss of altitude, particularly during windshear conditions. If the landing is subsequently aborted it must be remembered that the reduction in excess thrust due to contamination will reduce the aircraft's overall climb capability, particularly at large flap angles, to a marginal level. The formation of ice along the leading edges of the wing and tailplane during flight must thus be prevented. This is achieved using aircraft anti-icing systems; such as hot engine bleed air or pneumatic boots.

**Tail Icing**

As the trailing edge flaps are extended on an aircraft the wing centre of pressure moves **steadily rearwards**, producing a large **nose-down moment**, tending to rotate the aircraft nose-down (Fig. 15.16).
To compensate for this the **down load** acting on the tailplane will **naturally increase** due to the **change in downwash** and direction of the airflow behind the wing. With increasing forward airspeed the nose-down moment will steadily increase and will require a further increase in the download acting on the tailplane. On aircraft with highly efficient flaps the download on the tailplane will become excessively large, and the tailplane may even stall if the flaps are extended at too high a speed. Any accumulation of ice along the leading edge of the tailplane will seriously affect its maximum lift capability, and it may even stall at the normal approach airspeed (Fig. 15.17).

If the tailplane stalls, the **download will be suddenly removed**, the aircraft will **pitch nose-down**, and will go into an **uncontrollable dive**. Care should therefore be taken when increasing an aircraft's forward airspeed to compensate for ice on the wings, since every knot of airspeed added to prevent wing stall, will be a knot closer to tailplane stall. It is therefore vital that the tailplane, like the wing, is free from contamination. This type of stall does not usually occur, but if it does, **recovery is normally impossible**.

**Windshear**

Windshear is one of the leading causes of weather related aviation accidents and can occur at any altitude, but is **particularly serious below 1500 feet**. This is **Low Level Windshear (LLWS)** and occurs, whenever, the aircraft is configured for the take-off, approach, and
landing phases of flight. Windshear by definition is a variation in wind velocity and/or direction over a short period of time or distance. It alters the direction of the relative airflow, thereby greatly affecting the aerodynamic forces and moments acting on the aircraft. This alters the response of the aircraft to control inputs, and may require substantial control action in order to effect a recovery.

The following types of windshear exist:

**Vertical Windshear.** This is caused by wind-speed and direction changes with altitude.

**Horizontal Windshear.** This is caused by changes in wind-speed and direction along a given horizontal distance.

**Updraught/Downdraught Shear.** This is caused by changes in the vertical component of wind with horizontal distance.

The most potent forms of windshear are normally associated with thunderstorms or cumulonimbus clouds, although severe windshear can also be experienced due to topography or buildings, particularly in the presence of strong winds. By comparison with swept-wing aircraft, light high winged piston engined aircraft tend to react totally differently when entering a windshear of given strength.

**Vertical Gusts**

Vertical gusts acting on the aircraft principally alter its angle of attack. An up-gust will increase the angle of attack, whilst a down-gust will reduce the angle of attack. The variation in angle of attack is determined by the interaction between the vertical gust velocity and the aircraft's forward air velocity (Fig. 15.18).

![FIG. 15.18](image)

Any change in angle of attack will alter the total amount of lift developed by the wing, and if a strong up-gust occurs at high airspeeds it may even cause structural damage to the aircraft. Conversely, if vertical gusts are encountered at low airspeeds, e.g. during the approach, landing or take-off phases of flight, the changes in angle of attack will lead to
incipient stalling or sinking, rather than overstress. The wing shape that is least affected by turbulence is the Swept Wing, since it is a low aspect ratio, low lift wing.

**Horizontal Gusts**

Horizontal gusts differ from vertical gusts because they result in a change in airspeed, rather than a change in angle of attack, initially without any change in pitch attitude. For example consider an aircraft trimmed for straight and level flight whose airspeed reduces by 20% to 80% of its original value when acted on by a sharp horizontal gust. This will result in the aerodynamic forces of lift and drag at the same angle of attack falling to 64% of their original values. Due to the inertia of the aircraft it will momentarily continue to fly along the same flight path, but the subsequent reduction in airspeed and lift, will cause it to sink, and lose altitude until a new equilibrium condition is reached (Fig. 15.19).

![Figure 15.19](image-url)

Conversely if a sharp windshear is encountered, which increases the airspeed, the aircraft will tend to float, and gain altitude before equilibrium conditions are again reached (Fig. 15.20).

![Figure 15.20](image-url)
Downdraughts and Updraughts

If the aircraft enters a vertical up-draught or down-draught from a horizontal airflow its momentum will temporarily maintain its original flight path relative to the new direction of the airflow. In either case the airspeed will reduce and the aircraft's angle of attack will either increase or decrease in magnitude. For example consider the effect of a downdraught acting on the aircraft (Fig. 15.21).

The resulting reduction in airspeed will initially cause an energy loss and a subsequent reduction in aircraft performance. The reduction in the angle of attack will lead to a reduction in lift, and will cause the aircraft to pitch nose-down. Conversely if the aircraft is subject to an updraught it will cause the angle of attack to increase, thus increasing the aircraft's lift capability. Any small increase in the angle of attack will pose no significant problems in the controllability of the aircraft, but if operating at high angles of attack, which are normally associated with the approach and landing phases of flight, the wing may stall. Neither condition is desirable, especially when operating close to the ground.

Indications of a Windshear Encounter

The occurrence of one or more of the following changes will indicate a potentially severe windshear at altitudes below 1500 feet:-

- Any uncommanded change in IAS of ±15 kts
- Any uncommanded change in vertical speed of ± 500 feet per minute
- Any uncommanded change in pitch angle ± 5°
These indications will be displayed on the **airspeed indicator**, **vertical speed indicator** and **attitude indicator** respectively.

**General Recovery from a Windshear Encounter**

To counter the effects of a windshear at low altitude the following actions should be taken:

- Briskly increase power (Full thrust if necessary)
- Increase the pitch attitude to check the descent
- Co-ordinate power and pitch
- If on the approach, be prepared to carry out a missed approach rather than risk landing from a destabilised condition.

**Recovery from a Windshear Encounter During the Take-off, Approach and Landing**

If preventative action following a windshear encounter is unsuccessful, or flight path control becomes marginal below 500 feet during the take-off or landing phases of flight the following actions should be carried out simultaneously.

- Disengage the auto-throttle and aggressively advance the thrust levers to ensure maximum rated thrust is attained.
- Disengage the auto-pilot and smoothly increase the aircraft pitch attitude using the stick shaker as an upper limit if necessary in order to check the descent.

In general flight director guidance should be ignored and the **attitude director indicator** used as the primary reference for pitch attitude during the recovery from a windshear encounter. The **aircraft configuration**, e.g. undercarriage and flaps, should also not be altered until the **vertical flight path is under control**. Any attempt to regain lost airspeed should also be disregarded until ground contact is no longer a factor.

**Note:** During the recovery refer to the **vertical speed indicator** and altimeter when co-ordinating power and pitch attitude **until the rate of sink reduces to zero, or a positive rate of climb is achieved**. Conversely if a windshear encounter occurs near the **normal point of rotation**, indicated by a sudden rise in IAS, and quickly followed by a **decrease in airspeed**, the subsequent loss of lift may totally preclude a successful take-off. If insufficient runway remains left to stop in at this stage, then either increasing the airspeed and/or increasing the pitch attitude can alternatively obtain the required lift. Additional thrust will also help to accelerate the aircraft, but if the remaining runway is insufficient to reach the normal take-off speed, even at maximum thrust, the pitch attitude should be increased to make use of the available airspeed in order to generate enough lift, thus trading airspeed for altitude.
Microbursts

These are the most lethal forms of windshear. They normally occur within the vicinity of thunderstorms, and are mainly associated with cumulonimbus clouds (Fig. 15.22).

As the thunderstorm progresses intense downdraughts of air (microbursts) are generated, which travel towards the ground at speeds of up to 60 knots. The columns of air then splay out in all directions as they impact with the ground, and reduce in velocity. These are typically 5 km in diameter, and last between one and five minutes.

Microbursts can however also occur below altocumulus clouds where evaporative cooling enhances the strength of the downdraught. If an aircraft enters a microburst it will firstly encounter increasing headwinds, which will be followed by a downdraught, and finally increasing tailwinds. This sequence of events can prove disastrous, especially if they are encountered during the approach, or during the take-off phases of flight.

The Effect of a Microburst Encounter During the Approach

Fig. 15.23 depicts how an aircraft typically reacts when subject to a microburst during the approach. The sequence of events which occur are:
Increasing headwinds, causing the IAS to rise (Temporary Energy Gain), and the aircraft to climb above the glidepath. A reduction in thrust to increase the rate of descent, and/or a change in pitch attitude may counter this.

A downdraught that increases the rate of descent and reduces the IAS, thereby causing the aircraft to drop further below the glidepath. This is further worsened if the nose is still high and the thrust setting low. To counter this power is re-applied, but with transport category aircraft the thrust does not increase instantaneously because the engines take time to speed up, and the IAS continues to fall.

On leaving the downdraught increasing tail winds will cause a further reduction in IAS (Temporary Energy Loss), and the rate of descent may lessen due to increased thrust availability. To counter this full power must be achieved whilst maintaining pitch attitude, to check the descent, and the approach abandoned.

![Diagram showing the effect of microbursts on flight](image)

**FIG. 15.23**

A successful recovery from a microburst encounter will depend on the altitude, thrust, and speed reserves available. In addition to these effects severe wind turbulence, heavy rain and blinding flashes of lightning often accompany microbursts.

**The Effect of a Microburst on Take-off**

If a microburst is encountered on take-off the increasing headwind component will initially be beneficial because of the temporary increase in performance, but the rapid transition to a downdraught will quickly kill any rise in airspeed, and may even result in a drop in airspeed. The rate of climb will lessen, or may even show a rate of sink, which will be worsened by the shift to an increasing tailwind, and will cause a further reduction in airspeed. This is because the amount of excess power available on take-off to provide a good climb capability may only be enough to balance the drag associated with the higher
pitch attitudes. The aircraft may or may not have enough excess power to enable it to successfully traverse a microburst on take-off, particularly when operating at marginal airspeeds. If the airspeed falls below the trim airspeed, unusual stick forces may also be required to minimise any reduction in the pitch attitude. Attention must be particularly paid to the stick shaker, and if activated the pitch must be reduced just enough to silence the stick shaker. **Flight with intermittent stick shaker may also be required to keep the aircraft climbing.**

**Airborne Windshear Detection Systems**

Because low level windshear (LLWS) is mainly encountered during the take-off and landing phases of flight, between the ground and 1500 feet, it is important to provide as much warning as possible of windshear activity, since any encounter may rob an aircraft of lift.

To provide the necessary warnings, airborne detection systems are fitted to most modern aircraft. Flight-deck warnings are normally displayed as red annunciators for windshear, and a single amber one for turbulence. This is additionally accompanied by an Aural Alert (Master Warning Sound) on the flight-deck speaker, followed by a voice annunciation of "WINDSHEAR, WINDSHEAR, WINDSHEAR" (Fig. 15.24)

Some aircraft are additionally fitted with visual cues, e.g. the word ‘Windshear’ appears in red across the lower portion of the electronic attitude director indicator (EADI).
The Effect of Heavy Rain on Aircraft Performance

Windshear is often accompanied by heavy rain, which greatly affects the aircraft's overall performance. This is because it increases the gross weight of the aircraft due to the weight of the rainwater film and degrades the aerodynamic properties of a wing when the water film is roughened by the impact of raindrops. Rain induced surface roughness has the greatest effect on aircraft performance because it:

- increases the coefficient of drag (Skin Friction).
- reduces the maximum coefficient of lift (premature separation of the boundary layer).
- reduces the wing's stalling angle of attack.
Chapter 16.

Operating Limitations

Introduction

Every aircraft structural component comprises of an array of members, which are manufactured from various materials, and are, designed to safely distribute the forces or stresses acting on the aircraft. However some structures do not carry any structural stresses, and are purely designed to provide a streamlined shape, eg. engine cowlings and wing fairings. The various structures must also be capable of dissipating additional stresses, which exist during the manoeuvring phases of flight, e.g. banking and turning, when an aircraft is subject to acceleration forces (g forces). All aircraft are therefore built to safely accommodate the highest stress level anticipated during normal operations, i.e. the maximum load its structure must withstand without causing permanent damage. All aircraft also have a specific range of g forces and speeds at which they can be safely operated. During the normal working life of the aircraft its structure is also constantly subject to varying stresses, which occur due to:

- Flight manoeuvres.
- Atmospheric turbulence.
- Ground loads.
- Cabin pressurisation and depressurisation.
- Thermal effects.
- Vibrations.

The Flight Operating Envelope

Each type of aircraft has its own specific flight envelope, within which it can be safely operated in accordance with the Joint Airworthiness Requirements (JAR's). This is represented on a velocity against load factor, or V-n diagram (Fig. 16.1).
The above graph illustrates the limit load factors and limit speeds, which if exceeded, may result in permanent structural damage. The limit loads vary dependent on aircraft type, but for transport category aircraft these are generally +2.5g and -1g.

To provide a safety margin all aircraft are built to be able to withstand loads up to 1.5% limit load without failure occurring. This produces ultimate load limit factors of +3.75g and -1.5g. Some structural damage may however occur if the structure is loaded between the load limit and the ultimate load limit. It is thus not always safe to assume that the load factor may be increased above the limiting value just because a safety factor exists.

The high speed limit is the never exceed speed (V_{NE}) and is a design reference point for the aircraft beyond which structural damage or failure may occur. Conversely the stall governs the low speed limit, and this limit increases with increasing load factor.

At low airspeeds the stall may occur before the limit load factor is reached, but at high airspeeds the limit load factor may be reached before the stall. Gusts also affect the load factor, and the additional acceleration forces (g forces) associated with turbulence, e.g. due to an upgust at high airspeeds, can result in structural damage. In these conditions it is therefore advisable to limit the airspeed to the maximum normal operating speed (V_{No}) to avoid excessive flight loads. Other important reference speeds the pilot should be aware of are:-

- $V_S$ - Unaccelerated stall speed in the clean configuration
- $V_A$ - Maximum design manoeuvring speed and is the highest speed where the aircraft will stall before it exceeds the maximum load factor.
- $V_B$ - Maximum design speed for maximum gust intensity.
- $V_C$ - Maximum design cruising speed.
**V\textsubscript{NO}** - Maximum normal operating speed normally associated with light aircraft.

**V\textsubscript{NE}** - Never exceed speed.

**V\textsubscript{MO} / M\textsubscript{MO}** - Maximum operating speed limit. \( V\textsubscript{MO} \) is normally expressed as IAS. Where \( M\textsubscript{MO} \) is the same as \( V\textsubscript{MO} \) but is stated as a Mach Number and is the high altitude limiting speed since it is achieved before limiting IAS in the less dense air. In the climb and maintaining a constant IAS the Mach Number will increase, therefore \( M\textsubscript{MO} \) may be exceeded. Similarly when descending at \( M\textsubscript{MO} \) then care should be taken not to exceed \( V\textsubscript{MO} \).

**V\textsubscript{D} / M\textsubscript{D}** - Maximum design diving speed.

**V\textsubscript{LE}** - Maximum landing gear extended speed.

**V\textsubscript{LO}** - Maximum landing gear operating speed and may be the same as \( V\textsubscript{LE} \) depending on aircraft design.

**V\textsubscript{RA}** - Maximum speed in rough Air. This speed must be less than \( V\textsubscript{MO} \).

**V\textsubscript{FE}** - Maximum Flap Extension Speed.

\( V\textsubscript{D} \) is airspeed, which is set sufficiently above \( V\textsubscript{C} \) to allow a safe margin for the effects of a defined upset.

Fig 16.2 illustrates the effect on the **manoeuvre envelope with the flaps down**. It can be seen that \( V\textsubscript{F} \), \( V\textsubscript{C} \) and \( V\textsubscript{D} \) have also been added.
Manoeuvre and Gust Loads

The loads imposed on an aircraft in flight are the result of manoeuvres and gusts. Using the basic lift equation, a relationship between airspeed and aircraft mass can be established:

\[ \text{Lift} = C_L \frac{1}{2} \rho V^2 S = \text{Mass} \]

**Mass is directly proportional to** \( V^2 \), **so if the aircraft mass is reduced** then the **new value of** \( V_A \) **will be as follows:**

\[ \text{New } V_A = \frac{\text{New Mass}}{\text{Original Mass}} \]

For example if the **aircraft mass is reduced by 19%** then new value of \( V_A \) will be as follows:

\[ \text{New } V_A = \frac{0.81M}{M} = 0.9 \times \text{Original } V_A \]

\( V_A \) **will be reduced by 10% of its original value**, because the **lighter aircraft** will **respond more quickly** to any given input, thus preventing overstress. In addition to maximum manoeuvring speeds vertical gusts may have an adverse effect on the aircraft, since they directly affect its angle of attack (Fig. 16.3).

**FIG. 16.3**

At **high airspeeds** a sudden resultant change in angle of attack will momentarily alter the load factor, and may lead to **possible overstress**. The increment change in load factor due to a vertical gust can be determined from the following equation:

\[ <n = 0.115 \sqrt{\frac{m\sigma}{W/S}} V_e (KU) \]

where \( <n = \) **change in load factor due to gust**
m = slope of the lift curve, unit of $C_L$ per degree of angle of attack

$\sigma$ = altitude density ratio

$W/S$ = wing loading ($N/m^2$)

$V_e$ = equivalent airspeed (knots)

$KU$ = equivalent gust velocity (m/s)

For example, consider an aircraft flying at sea level at 200 knots (103m/s), which encounters a gust of 9m/s (30 ft/s). If the slope of the lift curve is 0.08 and the wing loading is 15 $N/m^2$, the gust would produce a load factor of increment of 0.57. This increment when added to the flight load factor of the aircraft prior to the gust, gives a new load factor of $(1.0 + 0.57) = 1.57$. Gusts in the region of 21.34 m/s (70 ft/s) are considered to be extreme, and flight in these regions should be avoided.

**Aeroelastic Distortion (Aileron Reversal)**

During **high-speed flight** any **aileron deflection** may cause the **wing to twist** about its torsional axis (Fig. 16.4).

This is because the **wing is flexible** and the ailerons are fitted near the wingtip, where the wing is less rigid. The actual torsional rigidity of a wing depends on its structure, but will normally be strong enough to prevent any distortion at low airspeeds. Aileron power however increases as the **square of forward airspeed**, whereas the torsional stiffness of a wing remains constant with speed. At high airspeeds a twisting moment due to aileron deflection will eventually overcome the torsional rigidity of the wing and alter its angle of attack, thereby reducing the rate of roll. The rising wing will twist nose-down, so reducing its **effective angle of attack**, and thus its coefficient of lift (Fig. 16.5).
Conversely the lowering wing will twist nose-up, so increasing its effective angle of attack, and its coefficient of lift (Fig. 16.6).

Eventually an airspeed will be reached where the incremental change in the coefficients of lift due to aileron deflection will be completely nullified by the wing twisting in the opposite sense. At this speed the lift produced by each wing will be the same irrespective of aileron deflection, i.e. the ailerons will become totally ineffective. Above this speed a downward deflection of the aileron will result in a reduction in lift, whilst an upwards deflection of the aileron will result in an increase in lift, and the aircraft will roll in the opposite direction to that applied by the control input. This is known as ‘aileron reversal’, and the speed at which it occurs is known as the ‘aileron reversal speed’. This airspeed normally occurs outside the aircraft's flight envelope, but at any airspeed below this point there will be an apparent reduction in roll rate for a given aileron deflection.
Emergency Descents

An emergency descent made if a sudden and complete failure of the cabin pressurisation system occurs. If the aircraft is operating at a **typical maximum operating altitude of 43,000 feet** the average person will become **unconscious within 15 seconds**, but will **reduce the crew's capability sooner**. If this occurs the emergency descent procedure should be initiated immediately, although the **structural integrity** of the aircraft must not be compromised. The procedure varies between aircraft types, but all involve **retarding the throttles to their flight idle position, operating the speedbrakes, and placing the aircraft in a steep descent** (Fig. 16.7).

![FIG. 16.7](image)

This should be initially carried out at a target speed of \(M_{\text{MO}}\) where the aircraft is **Mach limited**, and \(V_{\text{MO}}\) where the aircraft is **speed limited at lower altitudes**. To assist the pilot some aircraft are fitted never exceed speed needles on the airspeed indicator, which is datumed to \(M_{\text{MO}}\) at high altitude, and \(V_{\text{MO}}\) at lower altitudes. Whether the aircraft is cruising at \(M_{\text{MO}}\), or not the effect of reducing the thrust setting, and operating the speedbrakes will initially reduce the airspeed, and will require a fairly steep descent to return to \(M_{\text{MO}}\). Care should be taken not to overshoot \(M_{\text{MO}}\) during the descent and when the aircraft reaches the altitude where \(M_{\text{MO}} = V_{\text{MO}}\) at approximately **24 500 feet**, the **dive angle must be reduced to prevent** the aircraft exceeding \(V_{\text{MO}}\). At about 15 000 feet the cabin pressure is acceptable and the aircraft should be slowly returned to level flight. If the aircraft is pulled out of the dive too quickly it may result in overstress.