Theory of Flight

(According to the Syllabus Prescribed by Director General of Civil Aviation, Govt. of India)

FIRST EDITION

THEORY OF FLIGHT

Prepared by

L.N.V.M. Society Group o f Institutes
* School of Aeronautics
(Approved by Director General of Civil Aviation, Govt. of India)

* School of Engineering & Technology
(Approved by Director General of Civil Aviation, Govt. of India)

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Dedicated To

Shri. Laxmi Narain Verma

[Who Lived An Honest Life]

Preface

From the Gliders & Hot air balloons to the flying giants of the sky, the science of the flight has always fascinated human minds. In an attempt to explain the fundamentals of flight & to provide more light on its mechanism, School of Aeronautics, publication division is adding one more feather in its cap by bringing out the book on 'Theory of Flight'.

This book contains the elaborate explanation on the various topics like Aircraft performance, Stability & Control, Transonic & supersonic flight, propellers, Aerodynamic effects etc.

In short this book is an attempt to explain the flight of an aeroplane is simple & interesting way.

This is an endeavour to provide valuable knowledge for the aspirants of Aircraft Maintenance Engineer. It covers the entire syllabus of 'Theory of Flight' for paper II of D.G.C.A. licence examination which will help the students in their examination and also will empower them in their professional life. In present form this issue is the result of continuous development through experiences as a teacher and feed back received from our students as well as people related to the Aviation Industry.

My heartful thanks to all who have helped me with their valuable suggestions and able guidance.

I am very much thankful to our esteemed Director Mr. C.C. Ashoka for his sincere guidance and encouragement for preparing this book.

I would very much appreciate all forms of suggestions and constructive criticisms. I shall gratefully acknowledge, if any error, is brought to my notice.

Mr. Bipin Dwivedi

(Senior Instructor) ***Dated : August, 2007***

CONTENTS

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SYLLABUS COVERED IN THIS BOOK FOR BAMEL, PAPER-II

Knowledge of the terms lift, drag, angle of attack, stall etc.

CHAPTER: 1
THE STANDARD ATMOSPHERE

TROPOSPHERE, STRATOSPHERE AND TROPOPAUSE

The portion of the atmosphere below the height at which the change occurs is called the **troposphere,** and the portion above, the **stratosphere**. The interface between the two is called the **tropopause**. The lapse rate and the height of the tropopause vary with latitude. In Arctic regions, the rate of temperature change is lower, and the stratosphere does not start until around 15 500 m. The temperature in the stratosphere varies between about - 30°C at the equator to - 95°C in the Arctic. In temperate regions such as Europe the temperature in the stratosphere is around - 56.5°C.

INTERNATIONAL STANDARD ATMOSPHERE

For aircraft performance calculations, it is normal practice to use a standard set of conditions called the **International Standard Atmosphere (ISA)**. This defines precise values of lapse rate, height of the tropopause, and sea-level values of temperature, pressure and density. For temperate regions the ISA value of the lapse rate is - 6.5°C per 1000m, the tropopause is at 11 km, and the sea-level values of pressure and temperature are 101.325 k N/m^2, and 15°C respectively.

Modern long- and medium-range airliners cruise in or very close to the stratosphere, and supersonic airliners such as Concorde fly in the stratosphere well above the tropopause. When piston-engined aircraft first started to fly in the stratosphere, conditions were very uncomfortable for the crew. The low density and pressure meant that oxygen masks had to be worn, and at temperatures of - 56°C, even the heavy fur-lined clothing was barely adequate. Nowadays, the cabins of high-flying airliners are pressurised, and the air is heated, so that the passengers are unaware of the external conditions. Nevertheless, above every seat there is an emergency oxygen mask to be used in the event of a sudden failure of the pressurisation system.

Despite the low external air temperature in the stratosphere, supersonic aircraft have the problem that surface friction heats the aircraft up during flight, so means have to be provided to keep the cabin cool enough.

The upper atmosphere

The atmosphere with which we have been concerned in the flight of aeroplanes - i.e. the troposphere and the stratosphere - is sometimes called **the lower atmosphere**; the remainder is called **the upper atmosphere** (Fig. 1.1).

In the lower atmosphere the temperature had dropped from an average of + 15°C (288 K) at sea-level to - 57°C (217 K) at the base of the stratosphere, and had then remained more or less constant. The pressure and density of the air had both dropped to a mere fraction of their values at sea-level, about 1 per cent in fact. One might almost be tempted to think that not much more could happen, **but such an assumption would be very far from the truth**.

MESOSPHERE

There is a lot of atmosphere above 20 km - several hundred kilometres of it, we don't exactly know, it merges so gradually into space that there is really no exact limit to it - but a great deal happens in these hundreds of kilometres. The temperature, for instance, behaves in a very strange way; it may have been fairly easy to explain its drop in the troposphere, not quite so easy to explain why it should then remain constant in the stratosphere, but what about its next move? For from 217 K it proceeds to rise again - in what is called the **mesosphere** - to a new maximum which is nearly as high as at sea-level, perhaps 271 K; then, after a pause, **down it goes again to another minimum** at the top of the mesosphere. Estimates vary of just how cold it is at this height (only 80 kilometres, by the way, only the distance from London to Brighton), but all agree that it is lower than in the stratosphere, lower, that is to say, than anywhere on earth, perhaps 181 K (-92°C). But its strange behaviour doesn't stop at that and, once more after a pause at this level, as the name of the next region, the **thermosphere**, suggests, **it proceeds to rise again**, and this time it really excels itself **rising steadily, inexorably** to over 1200 K at 200 km, nearly 1500 K at 400 km, and still upward in the exosphere until it reaches **over 1500 K at the outer fringes of the atmosphere.**

An interesting point about these temperature changes in the upper atmosphere is their effect upon the speed of sound which, rises with the temperature, being proportional to the square root of the absolute temperature. The interest is not so much in the effects of this on shock waves, or on the flight of rockets, but rather in that one method of estimating the temperatures in the upper atmosphere is by measuring the speed of sound there.

While these strange and erratic changes of temperature have been taking place the density and the pressure of the air have fallen to values that are so low that they are almost meaningless if expressed in the ordinary units of mechanics; at a mere 100 km, for instance, **the density is less than one-millionth of that at ground level.**

IONOSPHERE AND EXOSPHERE

It is believed that at these heights there may be **great winds**, of hundreds, perhaps even a thousand kilometres per hour. The air above about the 70 km level is 'electrified' or ionised, that is to say it contains sufficient free electrons to affect

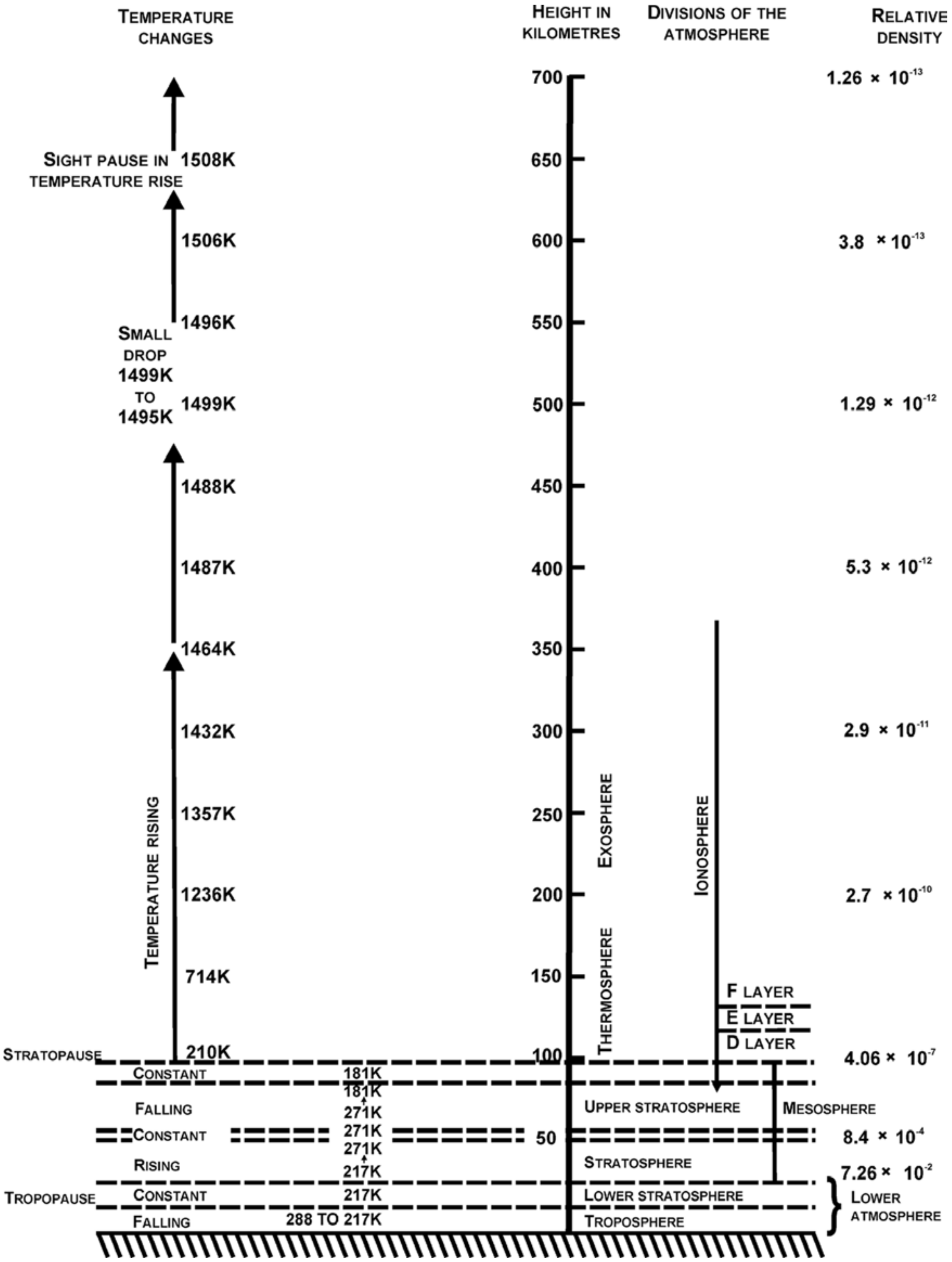

Fig.1.1. The upper atmosphere

The figures given are based on the US Standard Atmosphere, 1962, which was prepared under the sponsorship of NASA, the USAF and the US Weather Bureau

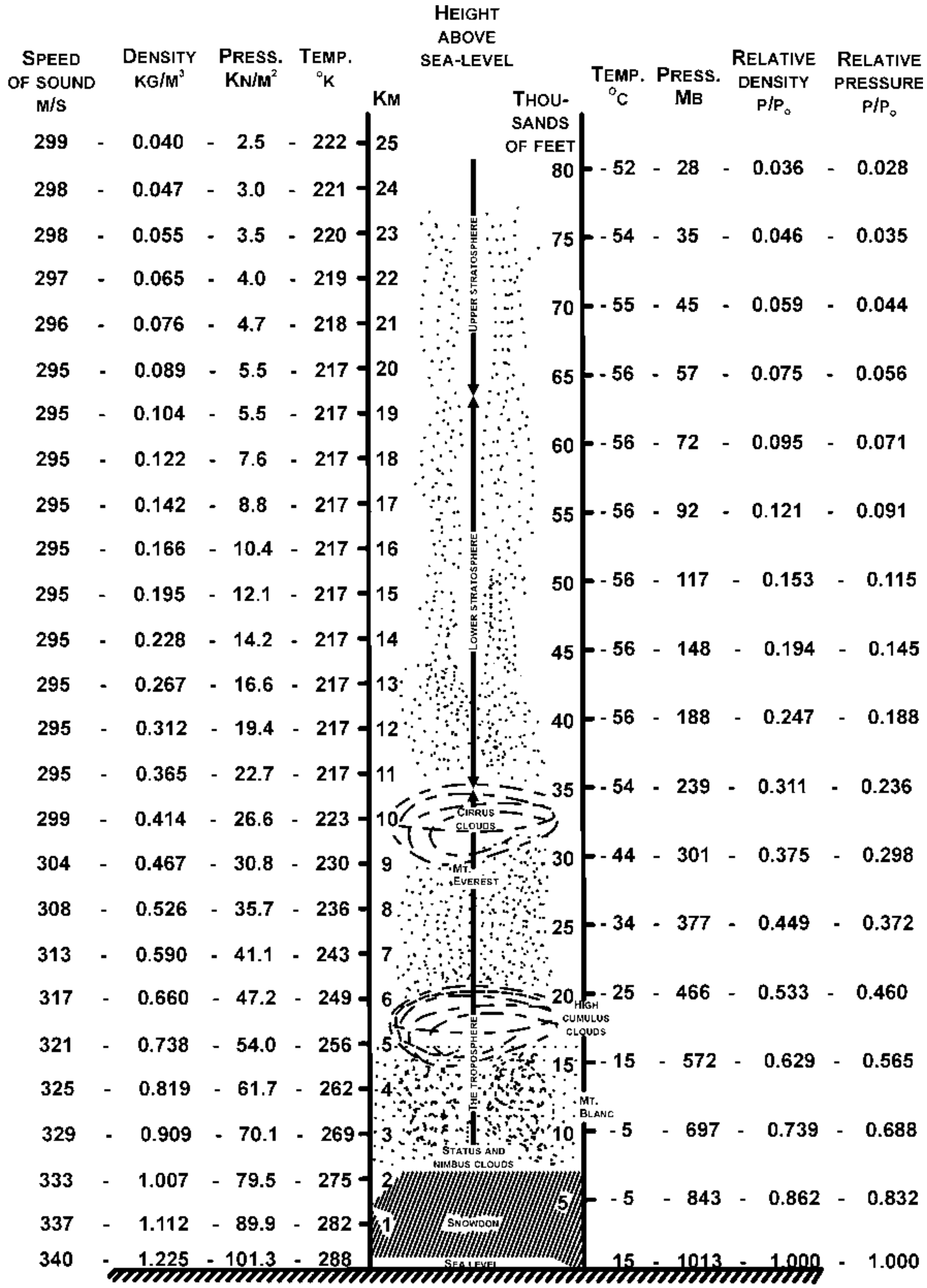

Fig.1.2. The international standard atmosphere

Based on the US Standard Atmosphere, 1962, which was prepared under the sponsorship of NASA, the USAF and the US Weather Bureau the propagation of radio waves. For this reason the portion of the atmosphere above this level is sometimes called the **ionosphere**, which really overlaps both the mesosphere and the thermosphere. Then there are the mysterious **cosmic rays** which come from outer space, and from which on the earth's surface we are protected by the atmosphere, but beyond this we know very little about them except that they may be the most dangerous hazard of all since they affect living tissues. Then there are the much more readily understandable **meteors**, 'shooting stars' as we usually call them, but actually particles of stone or iron which have travelled through outer space and may enter the earth's atmosphere at speeds of 100 kilometres per second, and which have masses of anything from a tiny fraction of a gram up to hundreds of kilograms. The larger ones are very rare, but some of these have actually survived the passage through the atmosphere without burning up, and have 'landed' on the earth causing craters of considerable size - these are called **meteorites**.

To prospective space travellers all this may sound rather alarming, but there are some redeeming features. The winds, for instance, wouldn't even 'stir the hair on one's head' for the simple reason that the air has practically no density, no substance. For the same reason the extreme temperatures are not 'felt' by a satellite or space-ship (what is felt is the temperature rise of the body itself, caused by the skin friction at the terrific speeds; it is this which burns up the meteors, it is this which has eventually caused the disintegration of many manlaunched satellites on re-entering the atmosphere - but all this has little or nothing to do with the actual temperature of the atmosphere). Then, as regards the very low densities and pressures, no-one is going to venture outside the vehicle, or walk in space, or even put his head out of the window to see whether the wind stirs the hair on his head, unless he is wearing a space-suit, and we have long ago learned to pressurise vehicles because this is required even for the modest heights in the lower atmosphere. Moreover, the strong outer casing of the vehicle which is required for pressurising will in itself give protection at least from the small and common meteors, and to some extent even from the cosmic rays, the greatest unknown. So altogether the prospect is not as bad as it might at first seem to be.

PROPERTIES OF ATMOSPHERE AND ITS VARIATION WITH ALTITUDE

a. Temperature

Another change which takes place as we travel upwards through the lower layers of the atmosphere is the gradual drop in temperature, a fact which unhappily disposes of one of the oldest legends about flying - that of Daedalus and his son Icarus, whose wings were attached by wax which melted because he flew too near the sun. In most parts of the world, the atmospheric temperature falls off at a steady rate called the lapse rate of about - 6.5°C for every 1000 metres increase in height up to about 11000 metres. Above 11000 metres, the temperature remains nearly constant until the outer regions of the atmosphere are reached.

b. Pressure

The weight of air above any surface produces a pressure at that surface - i.e. a force of so many newtons per square metre of surface. **The average pressure at sea-level due to the weight of the atmosphere is about 101 kN/m^2**, a pressure which causes the mercury in a barometer to rise about 760 mm. This pressure is sometimes referred to as 'one atmosphere', and high pressures are then spoken of in terms of 'atmospheres'. The higher we ascend in the atmosphere, the less will be the weight of air above us, and so the less will be the pressure

Decrease of pressure and density with altitude

The rate at which the pressure decreases is much greater near the earth's surface than at altitude. This is easily seen by reference to Fig. 1.2; between sea-level and 10000 ft (3480 m) the pressure has been reduced from 1013 mb to 697 mb, a drop of 316 mb; whereas for the corresponding increase of 10000 ft between 20000 ft (6096 m) and 30000 ft (9144 m), the decrease of pressure is from 466 mb to 301 mb, a drop of only 165 mb; and between 70000 ft (21336 m) and 80000 ft (24384 m) the drop is only 17 mb.

This is because air is compressible; the air near the earth's surface is compressed by the air above it, and as we go higher the pressure becomes less, the air becomes less dense, so that if we could see a cross-section of the atmosphere it would not appear homogeneous - i.e. of uniform density - but it would become thinner from the earth's surface upwards, the final change from atmosphere to space being so gradual as to be indistinguishable. In this respect air differs from liquids such as water; in liquids there is a definite dividing line or surface at the top; and beneath the surface of a liquid the pressure increases in direct proportion to the depth because the liquid, being practically incompressible, remains of the same density at all depths.

c. Density

Another property of air which is apt to give us misleading ideas when we first begin to study flight is its low density. The air feels thin, it is difficult for us to obtain any grip upon it, and if it has any mass at all we usually consider it as negligible for all practical purposes. Ask anyone who has not studied the question what is the mass of air in any ordinary room - you will probably receive answers varying from 'almost nothing' up to 'about 5 kilograms'. Yet the real answer will be nearer 150 kilograms, and in a large hall may be over a metric tonne! Again, most of us who have tried to dive have experienced the sensation of coming down 'flat' onto the surface of the water; since then we have treated water with respect, realising that it has substance, that it can exert forces which have to be reckoned with. We have probably had no such experience with air, yet if we ever try we shall find that the opening of a parachute after a long drop will cause just such a jerk as when we encountered the surface of the water. It is, of course, true that the density of air - i.e. **the mass per unit volume** - is low compared with water (the mass of a cubic metre of air at ground level is roughly

1.226 kg - whereas the mass of a cubic metre of water is a metric tonne, 1000 kg, nearly 800 times as much); **yet it is this very property of air - its density - which makes all flight possible,** or perhaps we should say **airborne** flight possible, because this does not apply to rockets. The balloon, the kite, the parachute, and the aeroplane - all of them are supported in the air by forces which are entirely dependent on its density; the less the density, the more difficult does flight become; and for all of them flight becomes impossible in a vacuum. So let us realise the fact that, however thin the air may seem to be, it possesses the property of density.

d. Viscosity

An important property of air in so far as it affects flight is its **viscosity**. This is a measure of the resistance of one layer of air to movement over the neighbouring layer; it is rather similar to the property of friction between solids. It is owing to viscosity that eddies are formed when the air is disturbed by a body passing through it, and these eddies are responsible for many of the phenomena of flight. Viscosity is possessed to a large degree by fluids such as treacle and certain oils, and although the property is much less noticeable in air, it is none the less of considerable importance.

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CHAPTER: 2
CLASSIFICATION OF AIRCRAFT

Aerostats
Aerostats are the floating crafts in atmosphere which are made lighter than air either by filling hot air or lighter gases. Their coarse of movement can by controlled manually or by remote. These are commonly used for sports, publicity and weather explorations. For example, Hot air baloons, airships and metrological gas baloons are aerostats.

Aerodynes
Aero dynes are the flying crafts. Which are powered by either their self propullsive force or launched into atmosphere by external force. Aeroplane, helicopters, gliders, rockets or missiles are the example of aerodynes. Which are commonly used for training, transportation and military purposes.

CLASSIFICATION OF AIRPLANES
(1) According to wings

a. Number of wings
1. Mono plane (One set of wing)
2. Bi Plane (Two set of wing)
3. Tri plane (Three set of wing).

b. (i) Position of wings
1. Conventional design : Horizontal tail located behind the wing.
2. No tail (Tail less)
3. Horizontal tail located above the vertical tail.
4. Canard Type - Tail is before the wing.

(ii) Position of the wings in resect to the axis of fuselage
1. Low wing : Boeing 737,Dakota.
2. Mid wing : Hunter
3. High wing : Fokker friendship, Lockheed, Hercules.

c. Shape of the wings
1. Delta wing (Concorde)
2. Diamond wing
3. Swept wing (Boeing 747)
4. Gull shaped.

(2) According to location and type of landing gear
a. Retractable-Boeing 737
b. Non Retractable-Puspak.
c. Tail Wheel (Non retractable) - Dakota.

(3) According to Power plants
a. No. of engines
1. Single engine : HAL-Ajeet
2. Two engine : Boeing 737
3. Multi engine : DC-10, Boeing 707

b. Type of engines
1. Piston engine : Dakota
2. Turbo Prop : HS-748
3. Turbo Jet : Concorde, MIG
4. Turbo Fan : Airbus, Boeing
5. Rocket : Missiles

c. Location of engines
1. Nose - HAL, HPT- 32
2. i. Rear Fuselage - HAL-Ajeet (Single engine)
   ii. Rear Fuselage (Two engine) - MIG 25, DC - 9
   iii. Rear Fuselage (Three engine) - DC-10

3. Jet engine submerged in the wing-HS Nimrod
4. Pylon mounted : Airbus.

(4) According to type of fuselage

a. Round : Boeing 707
b. Square : HAL-HPT - 32
c. Oval : HS. HAWK.

(5) According to Mission

a. **Civil**
b. **Cargo :** Payload high, economy in operation
c. **Military a/c :** Strategic [long range (6000 km), high speed, high endurance, needs aerodrome to land].
   i. Bombers : Tactical [long range (3000-5000 km), high speed, high endurance, fields or grass landing]. Tactical Interceptor : [High speed, High ceiling (300-400km/hr), can land at any field aerodrome].
   ii. Fighters : Interceptor [High rates of climb, High ceiling (500-600 km/hr), can land-particularly aerodrome].

PARTS OF AN AIRPLANE

a. **Plan view**
b. **Elevation**
c. **Side view**

Every aircraft is built for a definite assignment with the concrete aim of carrying out clearly defined tasks. This is the basis of the differences in the requirements of flight characteristics of aircraft, their load lifting capacity arrangements for accommodating the crew and the passengers and the "performance" of the aircraft. These differences, naturally, have an effect on the construction of aircraft and their aerodynamic configurations. They can be seen in the Fig.2.1.

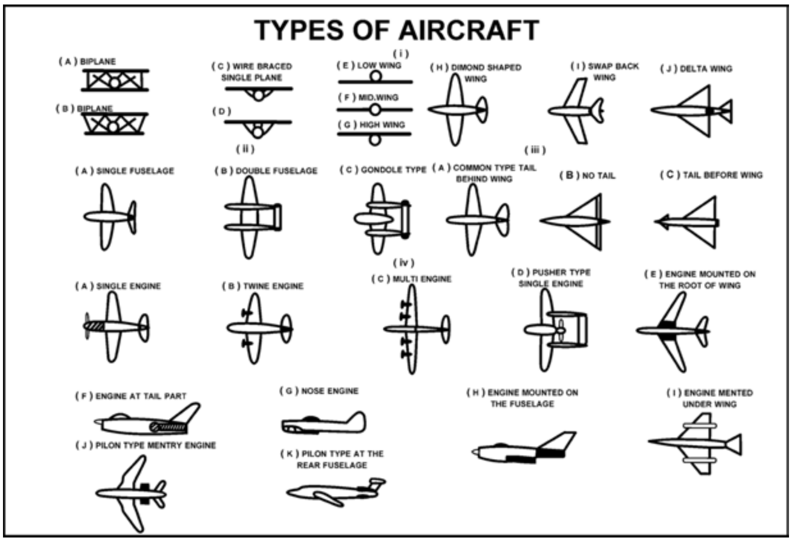

Fig.2.1. The most commonly used types of aircraft
I Shape and location of wing
II Type of fuselage
III Location of fin assembly
IV Location of power plants (engine)

Any aircraft, irrespective of its purpose, has the following basic parts : Wing creating lift, power plant creating thrust, fuselage comprising compartment for the crew, passengers, freight (and often fuel and power plant), tail unit in the form of fixed and movable planes used for stabilizing aircraft or changing its position in the air, chassis (landing gear) and control system in (Fig.2.2).

Wing

The wing creates lift due to the difference in pressure on its upper and lower surfaces. The pressure forces are distributed unevenly around the wing profile and they act directly on its surface. Their magnitude varies according to the velocity head and angle of attack. For every angle of attack there is a corresponding distribution and magnitude of pressure forces and, consequently, lift-to-drag ratio.

In acting on the wing covering the airstream tries to bend the wing in horizontal and vertical planes and twist it.

Besides the pressure forces exerted by the air, i.e. air load, on the wing so-called mass forces are set in motion due to

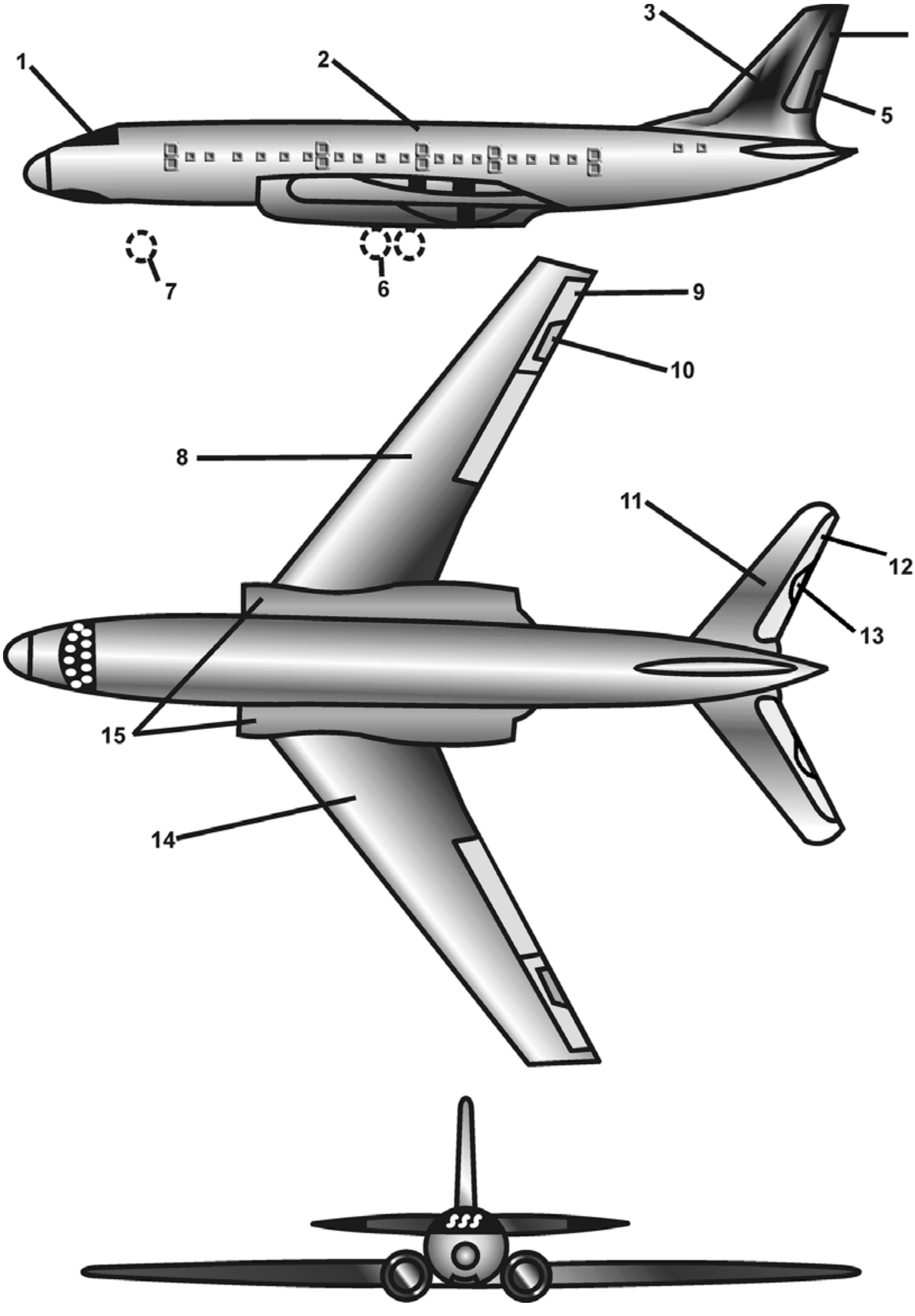

Fig.2.2. Scheme of aircraft.

1. Cockpit cannopy	*6. Main wheels*	*11. Stabilizer*
2. Fuselage	*7. Nose wheels*	*12. Elevator*
3. Tail fin	*8. Right wing*	*13. Elevator trimmer*
4. Rudder	*9. Aileron*	*14. Left wing*
5. Rudder trimmer	*10. Aileron trimmer*	*15. Engine*

the weight of the wing's structural elements as well as the aggregtes and loads accommodated in it and the inertia forces which are proportional to the mass of these elements. Moreover, at the points where the wing is joined with other parts of the aircraft reaction forces due to the wing's interaction with these parts appear.

The wing is so designed that at minimum weight it would withstand all the load acting on it during flight and maintain the necessary aerodynamic configuration that depends on the required flight data.

A wing consists of longitudinal elements: spars and stringers, and transverse elements: ribs and covering in (Fig.2.3).

A spar is something like a girder with upper and lower flanges connected together with webs or struts. It transmits to the wing joints vertical, also called shear forces and takes up the bending moment of these forces which is transmitted to its flanges. Wings are either single, double or multi-spar.

Stringers are longitudinal elements fixed to the upper and lower wing covering in order to give them stiffness and to take up together with the spar flanges the bending moment due to vertical forces.

Ribs are the basic elements forming the aerodynamic configuration of the wing section. The wing covering is fixed to them both directly and through stringers. They take up the air load and transmit it to the spars.

Taken together, the elements of the wing form a thin walled closed structure which consists of a frame and a covering functioning jointly.

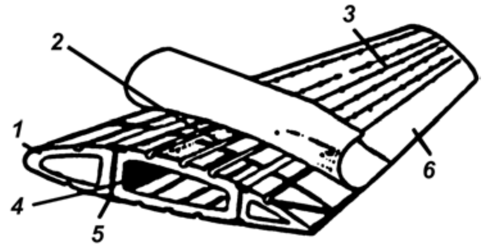

Fig.2.3.

The torque, appearing due to the fact that the resultant of all the forces does not pass through the shear centre of the wing section, is taken up by the wing covering and is transmitted through the last rib of the wing joint or directly to the fuselage sheathing if the wing is joined to it along the profile. The higher the aircraft's flight velocity, the greater the velocity head and, consequently, the load on the aircraft elements. In order to provide the required strength and rigidity for fast aircraft it becomes necessary to thicken the covering of the wing, tail plane and fuselage and to strengthen other structural elements.

With the reduction of relative thickness of the wing (ratio of maximum thickness of wing to its chord) and increase of load on it, the wing covering began to play a very big role in the design. Besides the direct air loading and torque it takes up a considerable part of longitudinal stresses due to bending moment.

It is known that the rods and other structural elements having a large length-to-thickness ratio loose their stability (bulge, bend) under longitudinal compression well before the load reaches a crippling value due to compression.

To prevent the wing covering from loosing stability at the gap between the ribs, it is reinforced by longitudinal elements-stringers. With the same purpose honeycomb and other constructions are used.

On the wing there are ailerons, which are devices for lateral control of the aircraft, and high-lift devices: slats, guided leading edges, split and plain flaps and devices for blowing or sucking the boundary layer.

The wing is the most important part of any aircraft. This is not so only because of its function but also because it accounts for 12-16% of the weight of the aircraft and up to 50% of the total resistance.

The wings of different types of aircraft differ from each other in plan (when seen from above), their location with respect to the fuselage, dihedral (front view), profile and specific loading (i.e. loading per square meter of the surface), constructional details and material.

High-Lift Devices

High-lift devices increase the lift and simultaneously the drag of the aircraft. The increase of lift is required during take-off in order to reduce the required take-off velocity of the aircraft and the length of the take-off run and during landing to decrease the approach speed and thus, as a result of increased resistance, to reduce the landing run.

It is possible to increase the lift of the wing at constant velocity by employing several methods. For example, by increasing the area of the wing or its angle of attack or by changing the wing camber so as to increase the value of coefficient C_y at the same angle of attack.

The first method, due to constructional complexity, is very rarely used in practice. The second method, i.e. increasing the critical angle of attack, which can be done with the help of slats, has little effect. The third method, increasing the camber (concavity) of the wing section, (Fig.2.4) is achieved by deflecting either the leading edge of the wing or its trailing edge(flap) or special flaps located on the lower surface of the wing, or by employing a combination of the above mentioned methods. This increases forces on the upper and lower surfaces of the wing as a result of which C_y is increased at the same angle of attack.

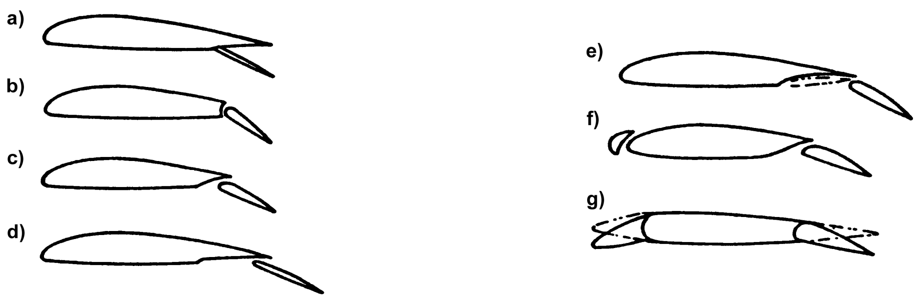

Fig.2.4. Various types of high-lift devices (a) split flap; (b) plain flap; (c) slotted flap; (d) zap flap; (e) fowler flap; (f) slat and flap; (g) guided leading edge and flap

It is also possible to increase the lift by blowing or sucking the boundary layers from the upper surface of the wing or jet flaps. The use of these methods, however, requires a large amount of energy.

The high-lift devices are so designed as to offer minimum possible drag while in the non-working position and to impart, while functioning, no variations in aircraft performance that the pilot would find hard to control.

Let us look at the operation of these devices one by one!

Deflection of the leading edge of the wing - A large angle of attack hinders the separation of the flow, making it possible to reach large angles of attack and consequently, high C_y. The increased profile camber(convexity) thus obtained leads to an additional increase of C_y. Since for profile with a sharp leading edge, used for the wings of fast aircraft, flow separation begins even at small angles of attack, such wings in most cases are designed with a deflecting leading edge.

Slats (leading edge) - Are small profile surfaces located as their name implies, on the leading edge of the wing. When the slat is pressed to the wing it forms part of the wing profile but when it moves away from the wing a profiled slot is formed between the slat and the wing. The air flowing through this slot moves along the upper surface of the wing, blowing the boundary layer. As a result the flow separation is delayed up to larger angles of attack and C_y increases.

Slats are usually used only to improve the lateral stability of aircraft during flight at large angles of attack. They are fixed with respect to the wing (with a fixed slot) and movable (can be pressed against the wing during flight with small angles of attack).

Movable flaps are operated either manually with the help of the mechanism provided or automatically in response to the suction force acting on the flap at large angles of attack.

A trailing edge flap on a wing makes it possible to increase C_y without increasing the angle of attack.

The simplest device is the split flap, which is part of the lower surface of the wing occupying 25-30% of the chord and up to 60% of the aircraft wing span. It can be deflected up to 55-60^0.

More complicated than split flaps air zap flaps which are deflected downward and at the same time move backward along the chord, increasing the effective area of wing.

At a fairly late stage plain and fowler flaps began to be used. A plain flap is the rear part of the wing profile that can deviate downward through a certain angle. Flaps can be either single or double. If the rear part of the profile not only deviates downward but also moves backward it is a fowler flap.

A narrow stream of air flowing through the wing and flaps blows the boundary layer from the upper surface of the flaps, thus securing an unseparated flow over them and, consequently, providing an opportunity to obtain large C_y.

Since the increased wing resistance is useful during the aircraft's landing run and dangerous during the take off run the angle of deviation for high lift devices during take-off is kept smaller than during landing.

Increase of the lift of a wing is also possible by increasing the intensity and regulating the airstream flowing over it. But in this case it is necessary to spend a certain amount of energy.

Theoretically the most effective devices are blowing or sucking the boundary layer from the upper surface of the wing.

Suction shown in (Figs.2.5 and 2.6) increases the velocity of flow and consequently the vacuum above the wing surface in the region in front of the point of suction. Blowing does the same practically over the whole chord of the wing. In blowing or sucking the boundary layer, with the increase in C_y, these takes place a simultaneous decrease in the wing drag and consequently the lift to drag ratio is increased, the faster the air is blown the larger the increase in C_y.

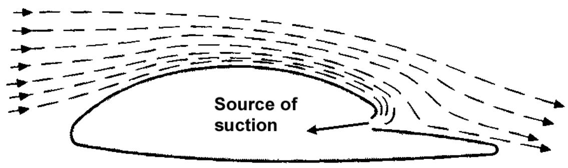

Fig.2.5. Control of boundary layer by suction.

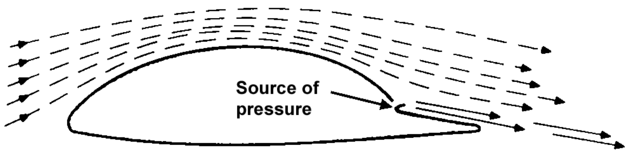

Fig.2.6. Control of boundary layer by pressure.

The jet flap is another means of increasing lift. In this method either air or gas is forced through a slot in the trailing edge of the wing at a certain angle with the chord. A jet from the slot plays the part of an original flap. A stream of air as it were flows around a fictitious wing of a larger chord and camber than those of the actual wing. The distribution of pressure over the actual wing and its supporting capacity, mainly in the region of the trailing edge, increases. The jet flap has not been widely used due to the weight of the gas feeding system and other operational complications.

Fuselage

The fuselage unites many parts of an aircraft : wing, tail plane, undercarriage and power plant. In it are accommodated the crew, passengers, equipment, freight and in some cases fuel, ammunition and engines. Fig.2.7.

There are considerable forces acting on a fuselage due to the aircraft parts joined to it, due to the weight of aerodynamic surface forces (pressure and vacuum). The magnitude of the latter forces at different places (cannopy, nose) can be as high as 7,000 kg/m^2.

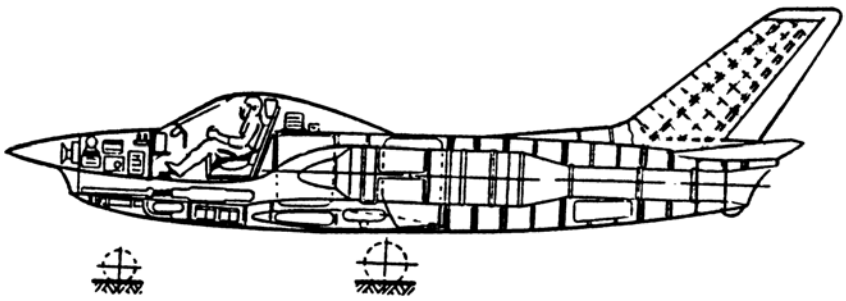

Fig.2.7. Fuselage arrangement of a fighter aircraft.

In addition, pressurized fuselage are loaded from the inside due to the excess pressure inside which is more than the air pressure outside.

The fuselage of a modern aircraft is a frame with a thin-walled covering. The frame is built from a group of longitudinal elements (longerons and stringers) and lateral ones (formers). Stringers and longerons are loaded with axial stresses (tensile and compressive) against bending moments of the fuselage. Stringers serve also as a reinforcement to the covering and increase its critical stresses (stresses at which it yields). Formers preserve the shape of the fuselage cross section. They seve as supports for Stringers and covering and take up the local aerodynamic loading. Reinforced formers transmit local concentrated forces to the covering.

The covering gives the fuselage a streamlined shape, it is subjected to normal (compressive and tensile) and tangential (shear) stresses arising during bending and twisting of the fuselage.

The fuselage usually has many big openings for access to equipment and freight, bomb bays, cabins, armaments, doors, under carriage, etc.

The nose of the fuselage of a supersonic aircraft is made sharp so that oblique shocks are formed and there is reduction in wave drag (these will be described later).

Livable conditions for passengers and crew at high altitudes of flight are secured by using pressurized cabins. A higher pressure of air with concentrated oxygen, as compared with the atmospheric pressure at the given altitude of flight, is built into these cabins. Normal temperature is maintained by thermal insulation and by heating and cooling equipment.

The most important parts of a pressurized cabin are the windows and the canopy. During flight through air at low temperatures the glass of cabin windows can become fogged or be covered with ice, obstructing visibility. Therefore either electrical or air heating of windows is installed or the windows are made of two panes of glass with an air gap between that is dried with the help of special cartridges. The proper tightness of riveted joints is achieved by using multibanked seams of specially treated strips.

Under carriage

The undercarriage permits an aircraft to park and move along the ground while taking off, landing and taxing. On modern aircraft the undercarriage can be retracted during flight.

The most widely used are the two types of tri-supported undercarriages. In the first type Fig.2.8(a) the main supports are located in front of the centre of gravity and in the rear there is a tail wheel. In the second Fig.2.8(b) type the main supports are behind the centre of gravity and in the front there is a nose wheel.

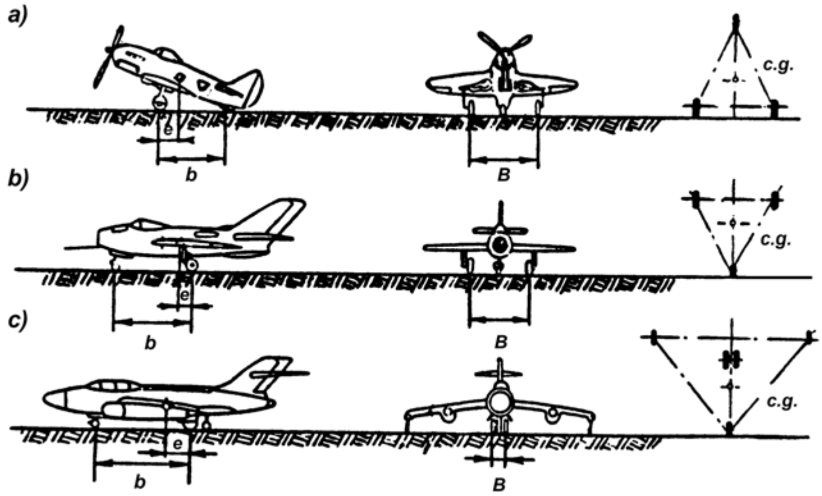

Fig.2.8. a) Stagger of main wheel; b) wheel base; c) (undercarriage) track.

In practice the undercarriage with the nose wheel is more widely used. The main reason for this is that the "tricycle undercarriage" (as the undercarriage with a nose wheel is called) provides stability to the aircraft during take off and landing runs and prevents it from nose tilting i.e. "nosingover".

The main supports of the undercarriage take up to 85% of the aircraft's weight. The tail or nose support carries nearly 15% of the weight. To facilitate movement of the aircraft on the ground it is made steerable by foot pedals. The main wheels retract into either the fuselage or the wing. For retractions a suitable space is left clear, its size depending on the number of wheels, their breadth and diameter.

The smaller the pressure in the tyres of the wheels the lower the unit pressure on the ground. The wheels then "stick" to the ground less, enabling the aircraft to take off and land on dirt landing strips. For such runways, however, the size of the wheels is increased.

Low pressure in the tyres is an asset in rainy weather when the ground is soaked. Good "traffic ability", (utilization factor) is the top priority for aircraft used in agricultural and ambulance aviation, passenger aircraft on local airlines and military aircraft in front-line combat.

Modern fast aircraft have thin wings and densely filled fuselages. In their case retraction of the wheels has become a difficult problem to solve. So they are fitted with wheels with high-pressure tyres and are obliged to operate from concrete runways.

When there are two or more wheels on each of the main supports they are mounted on special trolleys that ensure uniform load on the wheels during a change in the aircraft dip angle and deceleration.

The location of the undercarriage and the height from the ground of the aircraft's centre of gravity have a very important role in the design of aircraft. In the case of a "tricycle undercarriage" the main wheels are situated at such a distance behind the centre of gravity that its projection at take-off and landing angles of attack (plus a safety margin of 2-3^0) do not fall outside the line joining the right and left supports. This prevents the aircraft from nosing over and provides stability during the landing run.

The location of the main supports along the wing i.e. track width, depends on the arrangement of wheel retraction. The distance between the supports is determined taking into account the combined effect of track width and undercarriage height on aircraft stability against sideslip while landing in a cross-wind.

Aircraft with small track width, particularly with a twin-support, so-called "bicycle" undercarriage, are provided with additional retractable wheels at the wing tips.

The height of the undercarriage is kept to the minimum possible for economy in weight and simplicity in retraction.

Retraction and release of the undercarriage are carried out by systems which operate automatically when the pilot switches them on. These are either hydraulic, pneumatic or electro-mechanical. The systems are usually, duplicated as primary and emergency systems.

The undercarriage legs have shock-absorbers to dissipate shocks during the aircraft's movement on the ground and to damp vibrations. Nowadays oil-air (Oleoaerol, Oleo-pneumatic) shock absorbers are used exclusively in practice.

Aircraft Control

By aircraft controls are meant the devices and systems designed to turn control members (e.g. the rudder) in order to change the aircraft's position in the air, namely aircraft pitching, i.e. the inclination of the longitudinal axis to horizontal, rolling and turning.

Aircraft controls include : Control column for elevator and ailerons, control pedals for rudder, control levers for various accessories and aircraft systems, and system of cables, rods, rockers, ropes and hand levers.

Control is divided into hard control, in which the operating levers and aircraft control members are connected with the help of rods(pipes) with hinged ends of adjustable length, soft (rope) control and mixed control.

In order to change the position of the aircraft in the air it is necessary to turn it in a certain plane with respect to the centre of gravity. For this a controlling force situated at a certain distance from the centre of gravity is necessary. This is created by turning the aircraft control members.

Turning of the elevator or the whole stabilizer changes pitching, that of the ailerons changes rolling. Turning of the rudder changes the yaw angle, i.e. the angle between the plane of aircraft symmetry and the direction of its motion.

A forward movement of the control column (wheel) deviates the elevator downward, as a result of which a controlling force directed upward appears on the horizontal tail plane and gives rise to a diving moment (aircraft's nose goes down). Deviation of the control stick (wheel) toward the pilot gives rise to a pitching up moment (aircraft's nose goes up).

A movement of the control stick (wheel) to the right causes downward deviation of the left aileron and upward deviation of the right one. This gives rise to a pair of controlling forces : one, on the left wing directed upward and the other on the right wing directed downward. The aircraft will roll (turn) with respect to the axis of symmetry to the starboard wing side. On movement of the control stick to the left there will be a reverse effect.

Wheel-type control is usually used on big aircraft with two pilots. The wheel, by its large angle of rotation, enables the "gear ratio" to ailerons to be increased in comparison with that of the control stick, which the pilot cannot move more than 25^0. Due to this it is possible with the wheel to transmit a large force to the control member, but at a slower rate of deviation.

The control stick is so made that the pilot is able not only to operate it with one hand but also to carry out at the same time certain other functions such as applying wheel brakes, pressing the trigger of a gun and so on. The other hand is free to control the engines, release the undercarriage, regulate flaps and carry out various other operations.

Foot pedals are as rule arranged in an identical manner in all aircraft. By pressing a pedal with his foot the pilot turns the aircraft in the required direction. For example, by pressing the right pedal the aircraft is turned to starboard. Each pedal also has a device connected to the wheel brakes and to the steering system of the front or tail wheel.

Due to the development of aviation to high subsonic and supersonic speeds two new circumstances arose.

The first was the increase in the magnitude of the required controlling force with increasing flight speed. The effort necessary to control the aircraft was now such that the physical strength of a man was inadequate for it.

"Boosters" (amplifiers) were therefore installed in the aircraft. In the booster control the pilot does not turn the control members themselves by moving the column, wheel or pedals. He only moves booster vales joining the channels in their casing. Through these channels a special fluid under high pressure from the connected tank enters one or other recess of an actuating cylinder whose plunger is connected with the control member. Thus the effort required from the pilot is only to move a valve while the force necessary to turn the control member is produced by the actuating cylinder.

The second circumstance is the appearance on control surfaces of shocks moving forward or backward depending on the flight speed and angle of attack of the tail plane.

The shock after hinge moments, i.e. the moments of aerodynamic forces on the control member about the axis of the hinges on which it turns. This alteration is transmitted to the control stick in the form of a force that is undesirable and, in the case of a large force, inadmissible. The booster, due to its construction, cannot transmit forces from the control member to the control stick. Thus the use of boosters automatically solved these problems. Nowadays boosters are installed on the vast majority of modern aircraft.

Many types of aircraft, mainly those meant for long-range flights under constant regimes, namely passenger aircraft, transport aircraft, bombers, etc., include in their control systems autopilots. These are devices that automatically maintain the set regime of flight, i.e. its altitude, speed and course. After having set the flight regime the pilot switches on the autopilot, which then flies the aircraft.

When partial variation in the flight regime is necessary the pilot can produce it by feeding appropriate data to the autopilot without touching the control stick.

Stability and control members

Ailerons(elevons) situated on the wing and operating in conjunction and on the horizontal and vertical tail planes are the stability and control members.

Ailerons are movable parts of the wing operated by the pilot. They occupy 20-25% of its chord along the width starting from the trailing edge. They are located span wise nearer to the wing tips. The total area of both the ailerons is 8-10% of the net area of the wing. During their operation ailerons deviate in opposition(one up, one down or both central).

In order to carry out a roll the aircraft must turn through a certain angle about its axis of symmetry. This is possible only if both wings of the aircraft are subjected to lift forces differing in magnitude from what is achieved by only deviating the ailerons.

Thus by deviating the aileron on one wing downward the camber of the wing profile will increase and consequently its C_y coefficient. At the same time the aileron on the other wing will be deviated upward and the camber of its profile will decrease and so will its C_y. As a result there will arise a difference of moments of the aerodynamic forces with respect to the aircraft's axis of symmetry and it in turn will either increase or reduce the roll already present.

On tailless aircraft there are no stabilizers or elevators. Their functions are transferred to the ailerons. The ailerons are named elevons in this case. They provide not only lateral and longitudinal control of the aircraft but also balance.

This is achieved because the elevons can be deviated not only in opposite directions as ailerons but also simultaneously upward or downward, thus carrying out the functions of elevators. In size they are larger than ailerons.

Longitudinal control of the aircraft, i.e. pitching control, is accomplished by deviating the horizontal tail plane.

Generally the resultant of pressure forces acting on the surface of the wing, fuselage and other parts of the aircraft exposed to the airstream does not pass through the centre of mass of the aircraft. As a result a moment of aerodynamic forces arise with respect to the centre of mass.

On changing the angle of attack and flight speed the magnitude of the aerodynamic forces is changed as well as the position of the resultant forces and consequently the magnitude of the moment. For the aircraft to fly in a straight line it is necessary to apply to it a balancing moment equal in magnitude, but opposite in direction. To turn the aircraft about the lateral axis it is necessary to apply, in addition to the balancing moment, a control moment. Both these moments are created by the deviation of the horizontal tail plane situated at the rear of the fuselage. It consists of a fixed or movable stabilizer and elevators similar to the wing in construction.

An upward or downward deviation of the elevators changes the camber of the horizontal tail plane profile. This gives rise to a redistribution of pressure over its surface, as a result of which a control force is generated.

At the trailing edge of the elevators are situated small control surfaces-trim tabs which can be deviated upward or downward. A deviation of the trim tab causes corresponding deviation of the elevator. This movement is proportional in magnitude but opposite in direction. By operating the trim tab the pilot can deviate the elevator without touching the control stick. He can select such magnitude of trim tab deviation as will create the necessary balancing moment. The ailerons and rudders are also provided with trim tabs.

At transonic and supersonic speeds the flow around a profile distinctly changes. On the profile appear the shocks dividing the zones of supersonic and subsonic speeds of the airflow around it. This will be described later in detail.

A variation of pressure downstream from a shock cannot propagate upstream. Therefore if a shock has appeared in front of an elevator or on its leading edge the deviation of the elevator cannot have any effect on the pressure distribution upstream of it, i.e. on the stabilizer. In such a case the pilot can control the aircraft only by deviating the stabilizer.

The location of the horizontal tail plane with respect to the axis of the fuselage has many variants in aircraft construction practice. It is located either below the axis of the fuselage, along it or above on the fin.

All that has been said about the operation of a horizontal tail plane holds good for a vertical tail plane.

The vertical tail plane consists of a fin (one or more) and rudders. It creates a controlling force to turn the aircraft around the vertical axis (yaw moment) and also provides lateral trim and directional stability to the aircraft. To increase its effectiveness the vertical tail plane of twin or multiengine aircraft is often multiple so that it is situated in the slip-stream from the propellers.

Breaking Devices

In order to reduce airspeed the pilot decreases the engine thrust. However, due to the low value C_x that an aircraft has at high flight speeds its speed decreases comparatively slowly and over a long distance. In order to reduce the time and distance necessary to drop the speed many aircraft are provided with air brakes which increase C_x by opening or sliding out.

Air brakes are controllable surfaces in the form of flaps, sieves, etc. on fuselages or wings which can be slid into the airstream with the help of actuating cylinders to a magnitude and angle required by the pilot. The brake's aerodynamic resistance depends on their area and angle of inclination to the airstream.

To reduce speed during the landing run wheel brakes are used. The retarding force of a wheel is equal to the product of the load on it and its coefficient of friction with the ground. The load on the wheel is variable-it is minimal at the beginning of the landing run, since the wing still has a large lift force, and maximum at standstill. The retarding force varies correspondingly.

During landing modern fast aircraft possess such a large amount of kinetic energy that wheel brakes are unable to counteract it over the usable length of runway. Here brake parachutes releasable from the tail end of the aircraft fuselage after touchdown come to their help(Fig.2.9).

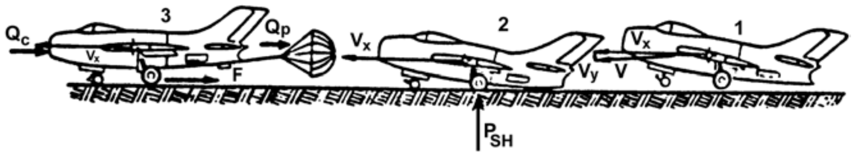

Fig.2.9. Forces acting on aircraft during landing

An aircraft performs its descent and landing with the engines running, i.e. with some thrust present.

Turbojet engines, in comparison with piston engines, have poorer "response", i.e. the capability of switching rapidly from idling to full power. Even at minimum revolutions they can produce quite a large thrust. Thus the use of turbojet engines worsened the landing characteristics of aircraft and accentuated the problem of brake effectiveness.

Many modern aircraft, therefore, have a special turbojet engine in the jet nozzles of which there is an arrangement for reversing thrust. With its help the jet of gases is turned forward in the direction of the aircraft's motion. Due to this a reaction force arises which slows the aircraft down. Thanks to thrust reversal two problems are solved together: firstly, in addition to the wheel and air brakes the pilot has at his disposal quite a large controllable retarding force, and secondly, a landing can take place at increased engine revolutions right up to the maximum which is convenient if the need arises for the aircraft to make a second approach. On aircraft with turbo-prop engines the reversed thrust of propellers is used for retardation.

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CHAPTER: 3
AEROFOILS AND WINGS
(SUBSONIC)

1. PLAN FORMS

If a horixonta wing is cut by a vertical plane parallel to the centre line of the vehicle, the resultant section is called the airfoil or aerofoil section. The generated lift and the stall characteristics of the wing depend strongly on the geometry of the airfoil sections that make up the wing. Geometric parameters that have an important effect on the aerodynamic characteristics of an airfoil section include

a. the leading-edge radius,
b. the mean camber line,
c. the maximum thickness and the thickness distribution of the profile, and
d. the trailing-edge angle.

The effect of these parameters, which are illustrated, in Fig.3.1. will be discussed after a brief introduction to airfoil section nomenclature.

2. SECTIONAL FORMS

"The gradual development of wing theory tended to isolate the wing-section problems from the effects of planform and led to a more systematic experimental approach. The tests made at Gottingen during World War -I contributed much to the development of modern types of wing sections. Upto about World War -II, most wing section in common use were derived from more or less direct extensions of the work at Gottingen. During this period, many families of wing sections were tested in the laboratories of various countries, but the work of the NACA was outstanding. The NACA investigations were further systematized by separation of the effects of camber and thickness distribution, and the experimental work was performed at higher Reynold's number than were generally obtained elsewhere".

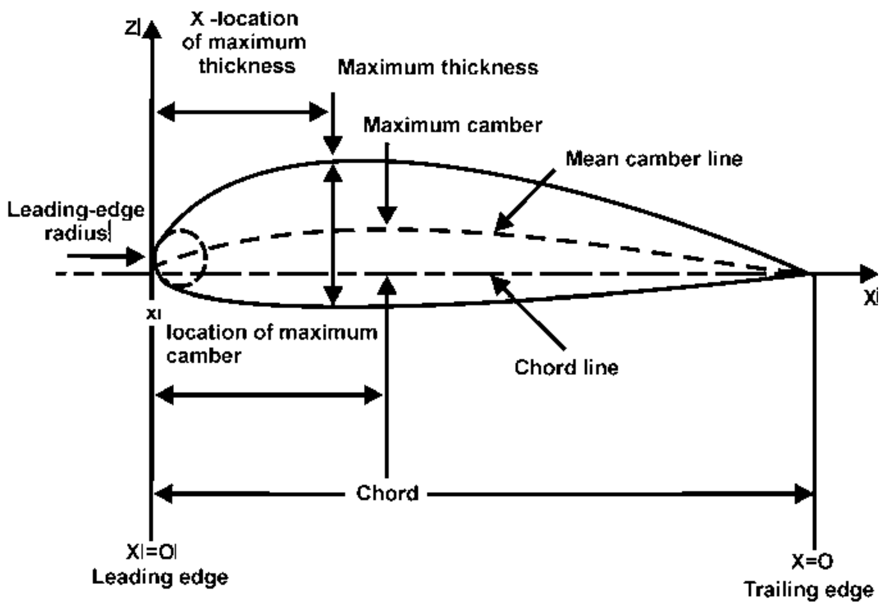

Fig.3.1.

As a result, the geometry of many airfoil sections is uniquely defined by the NACA designation for the airfoil. There are a variety of classifications, including NACA6 series wing sections. As an example, consider the NACA four-digit wing sections. The first integer indicates the maximum value of the main chamber-line ordinate in percent of the chord. The second integer indicates the distance from the leading edge to the maximum section thickness in percent of the chord. Thus, the NACA 0010 is a symmetric air foil section whose maximum thickness is 10% of the chord. The NACA 4412 airfoil section is a 12% thick airfoil which has 4% maximum chamber located at 40% of the chord.

A series of "standard" modifications are designated by suffix consisting of a dash followed by two digits. These modifications consist essentially of (1) changes of the leading edge radius from the normal value and (2) changes of the position of maximum thickness from the normal position (which is at 0.3c) Thus, NACA 0010 - 64

The first integer indicates the relative magnitude of the leading-edge radius (normal leading-edge radius is "6"; sharp leading edge is "0"). } The second integer of the modification indicates the location of the maximum thickness in tenths of chord.

However, because of the rapid improvements, both in computer hardware and computer software, and because of the broad use of sophisticated numerical codes, one often encounters airfoil sections being developed that are not described by the standard NACA geometries.

Leading-Edge Radius

The leading edge of airfoils used in subsonic applications is rounded, with a radius that is of the order of 1% of the chord length. The leading-edge radius of the airfoil section is the radius of a circle centred on a line tangent to the leading-edge camber connecting tangency points of the upper and the lower surfaces with the leading edge. The magnitude of the leading-edge radius has a significant effect on the stall (or boundary-layer separation) characteristics of the airfoil section.

Chord Line

The chord line is defined as the straight line connecting the leading and trailing edges. The center of the leading-edge radius is located such that the cambered section projects slightly forward of the leading-edge point. The geometric angle of attack is the angle between the chord line and the direction of the undisturbed, "free-stream" flow. For many airplanes the chord lines of the airfoil sections are inclined relative to the vehicle axis.

Mean Camber Line

The locus of the points midway between the upper surface and the lower surface, as measured perpendicular to the chord line, defines the mean camber line. The shape of the mean camber line is very important in determining the aerodynamic characteristics of an airfoil section. Cambered airfoils in a subsonic flow generate lift even when the section angle of attack is zero. Thus, an effect of camber is a change in the zero-lift angle of attack,α ox. While the symmetric sections have zero lift at zero lift at zero angle of attack, zero lift results for sections with positive camber when they are at negative angles of attack.

Furthermore, camber has beneficial effect on the maximum value of the section lift coefficient. If the maximum lift coefficient is high, the stall speed will be low, all other factors being the same. It should be noted, however, that the thickness and camber necessary for high maximum values for the sections lift coefficient produce low critical Mach numbers and high twisting moments at high speeds. Thus, one needs to consider the trade-offs in selecting a design value for a particular parameter.

Maximum Thickness and Thickness Distribution

The maximum thickness and the thickness distribution strongly influence the aerodynamic characteristics of the airfoil section. The maximum lift coefficient for an airfoil section increases as the maximum thickness of the airfoil increases. In addition the thicker airfoils benefit more from the use of high lift devices but have a lower critical Mach number.

The maximum local velocity to which a fluid particle accelerates as it flows around an airfoil section increases as the maximum thickness increases. Thus, the minimum pressure value is smallest for the thickest airfoil. As a result, the adverse pressure gradient associated with the deceleration of the flow from the location of this pressure minimum to the trailing edge is greatest for the thickest airfoil. As adverse pressure gradient becomes larger, the boundary layer becomes thicker (and is more likely to separate producing relatively large values for the form drag). Thus, the beneficial effects of increasing the maximum thickness are limited.

The thickness distribution for an airfoil affects the pressure distribution and the character of the boundary layer. As the location of the maximum thickness moves after the velocity gradient (and hence the pressure gradient) in the midchord region decreases. The resultant favorable pressure gradient in the midchord region promotes boundary layer stability and increases the possibility that the boundary layer remains laminar. Laminar boundary layers produce less skin friction drag than turbulent boundary layers but are also more likely to separate under the influence of an adverse pressure gradient.

Trailing-Edge Angle

The trailing edge angle affects the location of the aerodynamic center. The aerodynamic center of this aerofoil sections in a subsonic stream is theoretically located at the quarter chord.

Wing-Geometry Parameters

By placing the airfoil sections discussed in the preceding section in spanwise combinations, wings, horizontal tails, vertical tails, canards and or other lifting surfaces are formed. When the parameters that characterize the wing planform are introduced, attention must be directed to the existence of flow components in the spanwise direction. In other words, airfoil section properties relate to the resultant flow in three dimensions.

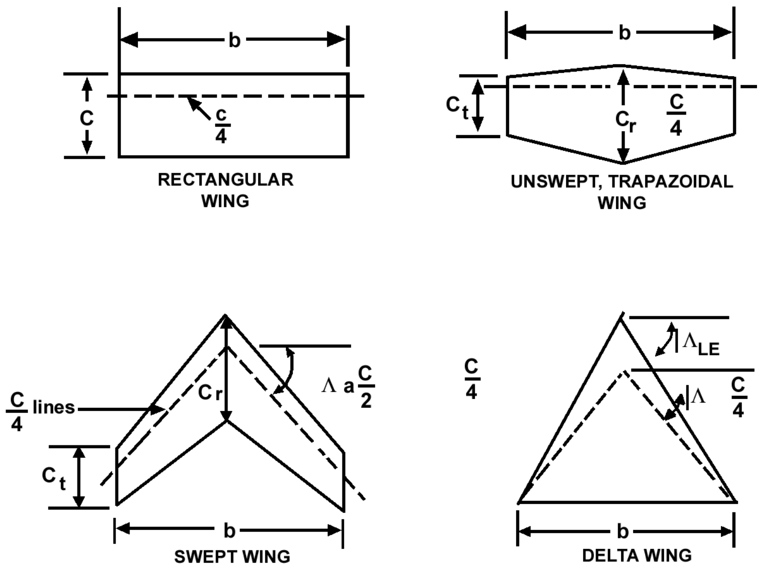

Fig.3.2.

In order to fully describe the planform of a wing, several terms are required. The terms that are pertinent to defining the aerodynamic characteristics of a wing are illustrated in Fig.3.2.

1. **The wing area,** S, is simply the plan surface area of the wing. Although a portion of the area may be covered by fuselage or nacelles, the pressure carry over on these surfaces allows legitimate consideration of the entire plan area. The plan area of the wing including continuation with the fuselage is the 'gross wing area' S_G. The plan area of the exposed wing (i.e. excluding the continuation within the fuselage) is the 'net wing area' S_N.

2. **The wing span,** b, is measured tip to tip.

3. **The average chord,** $\bar{c}$ is the geometric average. The product of the span and the average chord is the wing area $(b \times \bar{c} = S)$.

4. **The aspect ratio,** AR, is the ratio of the span and the average chord. For a rectangular wing, the aspect ratio is simply

$$AR = \frac{b}{c}$$

For a non-rectangular wing,

$$AR = \frac{b^2}{S}$$

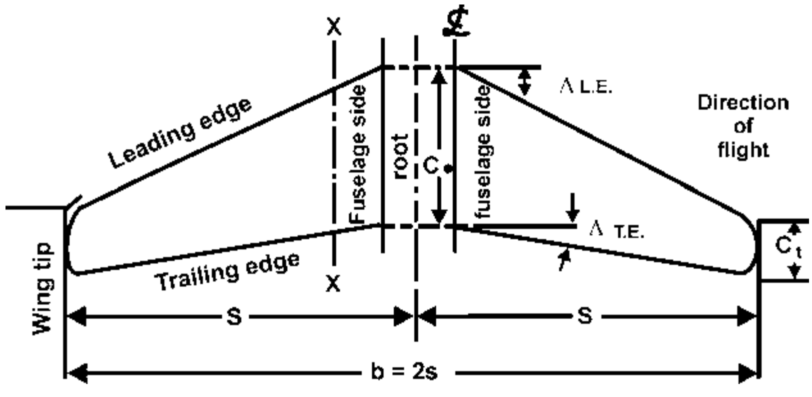

Fig.3.3.

The aspect ratio is a fineness ratio of the wing and is useful in determining the aerodynamic characteristics and structural weight. Typical aspect ratios vary from 35 for a high-performance sailplane to 2 for a supersonic jet fighter.

5. **The root chord,** c_r, is the chord at the wing centre line, and the tip chord, c_t, is measured at the tip.

6. Considering the wing planform to have straight lines for the leading and trailing edges, the taper ratio, λ is the ratio of the tip chord to the root chord.

$$\lambda = \frac{c_t}{c_r}$$

The taper ratio affects the lift distribution and the structural weight of the wing. A rectangular wing has a taper ratio of 1.0 while the pointed tip delta wing has a taper ratio of 0.0.

7. **The sweep angle,** Λ is usually measured as the angle between the line of 25 percent chord and a perpendicular to the root chord. Sweep angles of the leading edge or of the trailing edge are often presented with the parameters, since they are of interest for many applications. The sweep of a wing causes definite changes in the maximum lift, in the stall characteristics, and in the effects of compressibility.

8. **The mean aerodynamic chord,** (m.a.c.) is used together with S to non-dimensionalize the pitching moments. Thus, the mean aerodynamic chord represents an average chord which, when multiplied by the product of the average chord which, when multiplied by the product of the average section moment coefficient, the dynamic pressure, and the wing area, gives the moment for the entire wing. The mean aerodynamic chord is given by

$$\bar{c} = \frac{1}{S}\int_{-0.5b}^{+0.5b} [c(y)]^2 dy$$

9. **The dihedral angle** is the angle between a horizontal plane containing the root chord and a plane midway between the upper and lower surfaces of the wing. If the wing lies below the horizontal plane, it is termed an anhedral angle. The dihedral angle affects the lateral stability characteristics of the airplane (Fig. 3.4.).

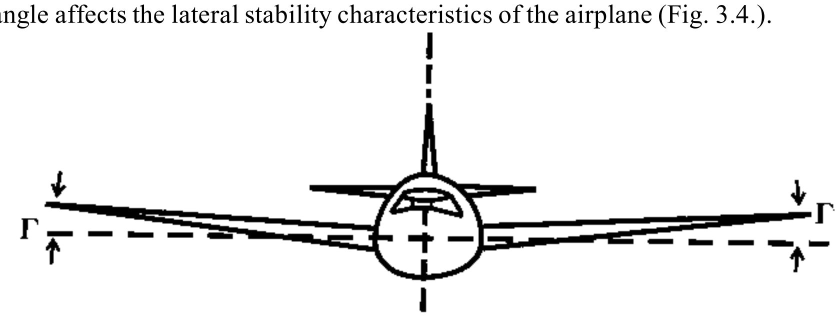

Fig.3.4. (a)

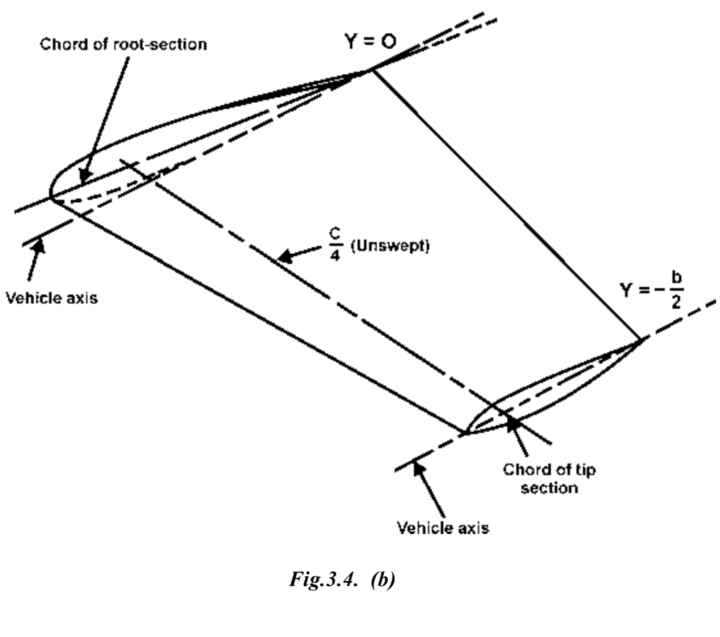

Fig.3.4. (b)

10. **Geometric twist defines** the situation where the chord lines for the spanwise distribution of airfoil sections do not all lie in the same plane. Thus, there is a spanwise variation in the geometric angle of incidence for the sections. The chord of the root section of the wing shown in the sketch of Fig.3.4b is inclined 4° relative to the vehicle axis. The chord of the tip section, however is parallel to the vehicle axis. In this case, were the incidence of the airfoil sections relative to the vehicle axis decrease toward the tip, the wing has "wash out". The wings of numerous subsonic aircraft have wash out to control the spanwise lift distribution and, hence, the boundary-layer separation (i.e., stall) characteristics. If the angle of incidence increases toward the tip, the wing has "wash in".

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CHAPTER: 4
LIFT AND ANGLE OF ATTACK (α)

LIFTING SURFACES

If air flows past an aerofoil, a flat plate or indeed almost any shape that is inclined to the direction of flow, we find that the pressure of air on the top surface is reduced while that underneath is increased. This difference in pressure results in a net force on the plate trying to push it both upwards and backwards. In the case of a simple flat plate, you might imagine that the net force would act at right angles to the plate. This is not so, because there is also a tangential force caused by the different pressures that act on the small leading and trailing edge face areas. This tangential force though small, is by no means negligible. Rather surprisingly, the pressure at the leading edge is normally very low, and at small angles of inclination, the tangential force will act in the direction shown in (Fig.4.1and Fig.4.2). The reasons for the low pressure at the leading edge will be shown later. Note, that although the tangential force may be directed towards the front of the plate, the resultant of the tangential and normal forces must always be tilted back relative to the local flow direction.

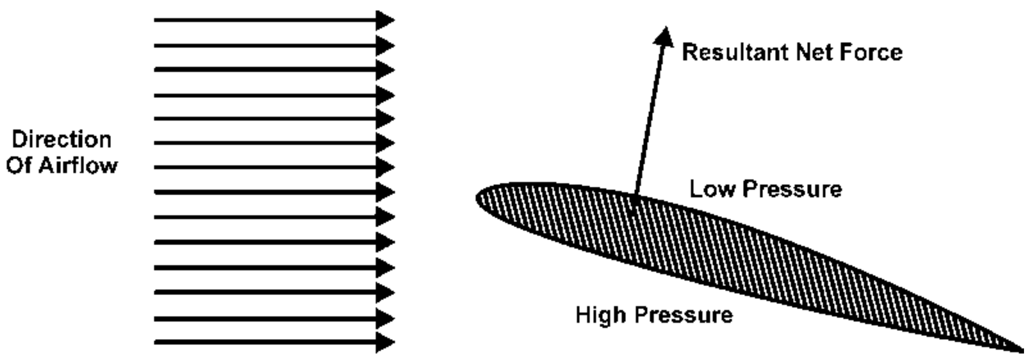

Fig.4.1. Resultant force on an aerofoil due to pressure difference

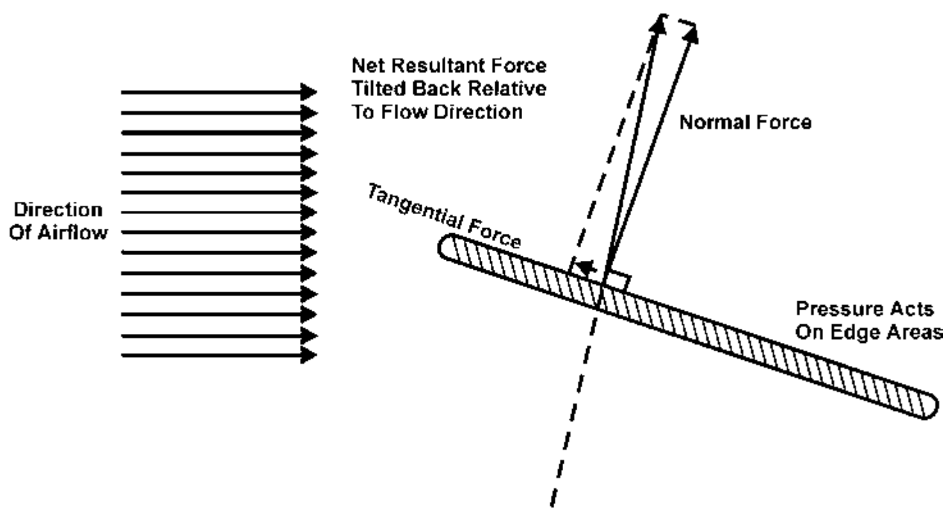

Fig.4.2. Forces due to pressure differences in a flat plate

LIFT

The resultant or net force on the lifting surface may be conveniently split into two components relative to the airflow direction as follows -

1. The component at right angles to the direction of the airflow, called LIFT (Fig.4.3).
2. The component parallel to the direction of the airflow, called DRAG (Fig.4.4).

The use of the term 'lift' is apt to be misleading, for under certain conditions of flight, such as a vertical nose dive, it may act horizontally, and cases may even arise where it acts vertically downwards.

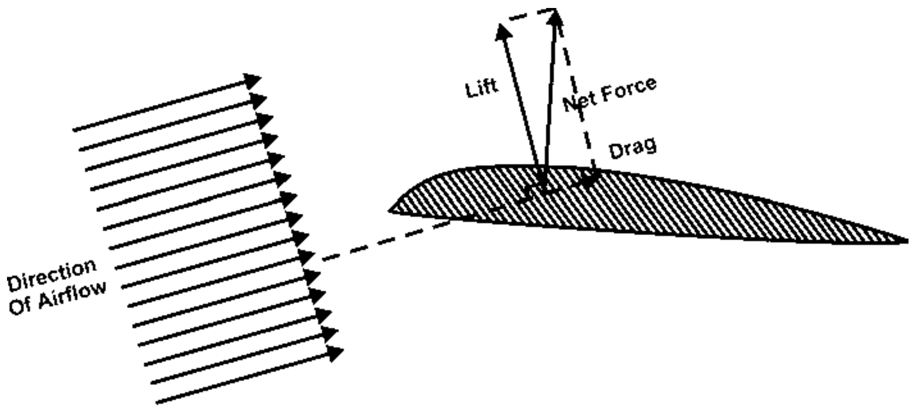

Fig.4.3. Lift and drag shown for the case of a descending aircraft

Airflow and pressure over aerofoil

It was soon discovered that a much greater lift, especially when compared with the drag, could be produced by using a curved surface instead of a flat one, and thus the modern aerofoil was evolved. The curved surface had the additional advantage that it provided a certain amount of thickness which was necessary for structural strength.

Experiments have shown that the air flows over an aerofoil (Fig.4.4) much more smoothly than over a flat plate.

In (Fig.4.4), which shows the flow of air over a typical aerofoil, the following results should be noticed -

1. There is a slight upflow before reaching the aerofoil.
2. There is a downflow after passing the aerofoil. This downflow should not be confused with the downwash produced by the trailing vortices as described later.
3. The air does not strike the aerofoil cleanly on the nose, but actually divides at a point just behind it on the underside.
4. The streamlines are closer together above the aerofoil where the pressure is decreased.

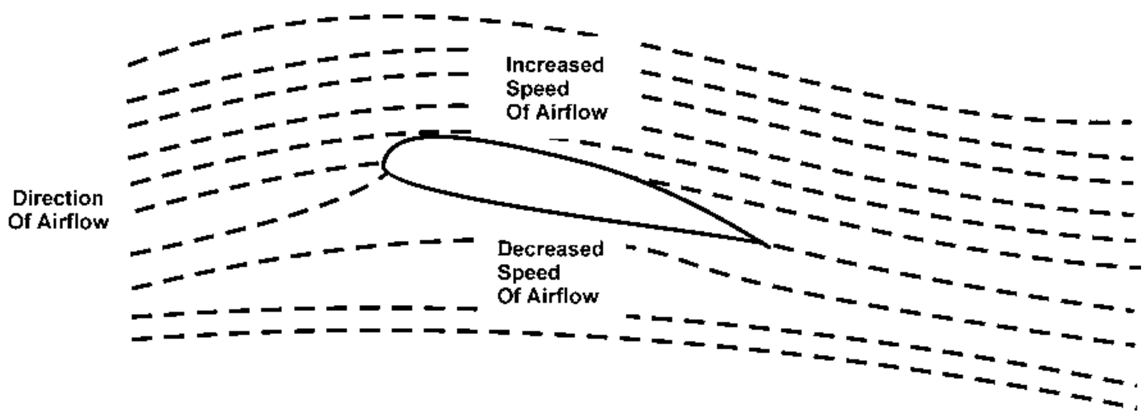

Fig.4.4. Airflow over an aerofoil inclined at a small angle

This last fact is at first puzzling, because, as in the venturi tube, it may lead us to think that the air above the aerofoil is compressed, and that therefore we should expect an increased pressure. The explanation is that the air over the top surface acts as though it were passing through a kind of bottleneck, similar to a venturi tube, and that therefore its velocity must increase at the narrower portions, i.e. at the highest points of the curved aerofoil.

The increase in kinetic energy due to the increase in velocity is accompanied by a corresponding decrease in static pressure. This is, in fact, an excellent example of Bernoulli's Theorem.

Another way of looking at it is to consider the curvature of the streamlines. In order that any particular particle of air may be deflected on this curved path, a force must act upon it towards the centre of the curve, so that it follows that the pressure on the outside of the particle must be greater than that on the inside; in other words, the pressure decreases as we move down towards the top surface of the aerofoil. This point of view is interesting because it emphasises the importance of curving the streamlines.

Chord line and angle of attack

It has already been mentioned that the angle of inclination to the airflow is of great importance. On a curved aerofoil it is not particularly easy to define this angle, since we must first decide on some straight line in the aerofoil section from which we can ensure the angle to the direction of the airflow. Unfortunately, owing to the large variety of shapes used as aerofoil sections it is not easy to define this chord line to suit all aerofoils. Nearly all modern aerofoils have a convex under-surface; and the chord must be specially defined, although it is usually taken as the line joining the leading edge to the trailing edge. This is the centre in the particular case of symmetrical aerofoils.

We call the angle between the chord of the aerofoil and the direction of the airflow the angle of attackα (Fig.4.5).

This angle is often known as the angle of incidence; that term was avoided in early editions of this book because it was apt to be confused with the riggers' angle of incidence, i.e. the angle between the chord of the aerofoil and some fixed datum line in the aeroplane. Now that aircraft are no longer 'rigged' (in the old sense) there is no objection to the term angle of incidence; but by the same token there is no objection either to angle of attack, many pilots and others have become accustomed to it; it is almost universally used in America, and so we shall continue to use it in this edition.

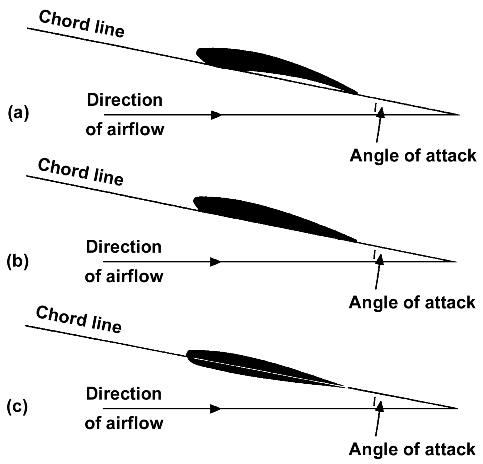

Fig.4.5. Chord line and angle of attack
(a) Aerofoil with concave undersurface.
(b) Aerofoil with flat undersurface.
(c) Aerofoil with convex undersurface.

Note: If we wish to be precise we must be careful in the definition of the term 'angle of attack', because, as has already been noticed, the direction of the airflow is changed by the presence of the aerofoil itself, so that the direction of the airflow which actually passes over the surface of the aerofoil is not the same as that of the airflow at a considerable distance from the aerofoil. We shall consider the direction of the airflow to be that of the air stream at such a distance that it is undisturbed by the presence of the aerofoil.

Line of zero lift

Now an aerofoil may provide lift even when it is inclined at a slightly negative angle to the airflow. And one may well ask how an aerofoil which is inclined at a negative angle can produce lift? The idea seems absurd, but the explanation of the riddle is simply that the aerofoil is not really inclined at a negative angle. Our curious chord may be at a negative angle, but the curved surfaces of the aerofoil are inclined at various angles, positive and negative, the net effect being that of a slightly positive angle, which produces lift.

If we tilt the nose of the aerofoil downwards until it produces no lift, it will be in an exactly similar position to that of a flat plate placed edgewise to the airflow and producing no lift, and if we now draw a straight line through the aerofoil parallel to the airflow (Fig.4.6) it will be the inclination of this line which settles whether the aerofoil provides lift or not.

Such a line is called the line of zero lift or neutral lift line, and would in some senses be a better definition of the chord line, but it can only be found by wind tunnel experiments for each aerofoil, and, even when it has been found, it is awkward from the point of view of practical measurements.

Nor is it of much significance in practical flight, except perhaps in a dive when the angle of attack may approach the no lift condition.

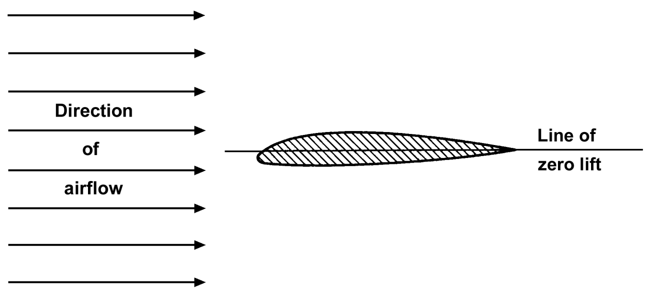

Fig.4.6. Line of zero lift

Note that for an aerofoil of symmetrical shape zero lift corresponds to zero angle of attack.

Pressure plotting

As the angle of attack is altered the lift and drag change very rapidly, and experiments show that this is due to changes in the distribution of pressure over the aerofoil. These experiments are carried out by the method known as 'pressure plotting' (Fig.4.7), in which small holes in the aerofoil surface (*a, b, c, d,* etc.) are connected to glass manometer tubes (*a, b, c, d,* etc.) containing water or other liquid; where there is a suction on the aerofoil the liquid in the corresponding tubes is sucked up, where there is an increased pressure the liquid is depressed. Such experiments have been made both on models in wind tunnels and on aeroplanes in flight, and the results are most interesting and instructive.

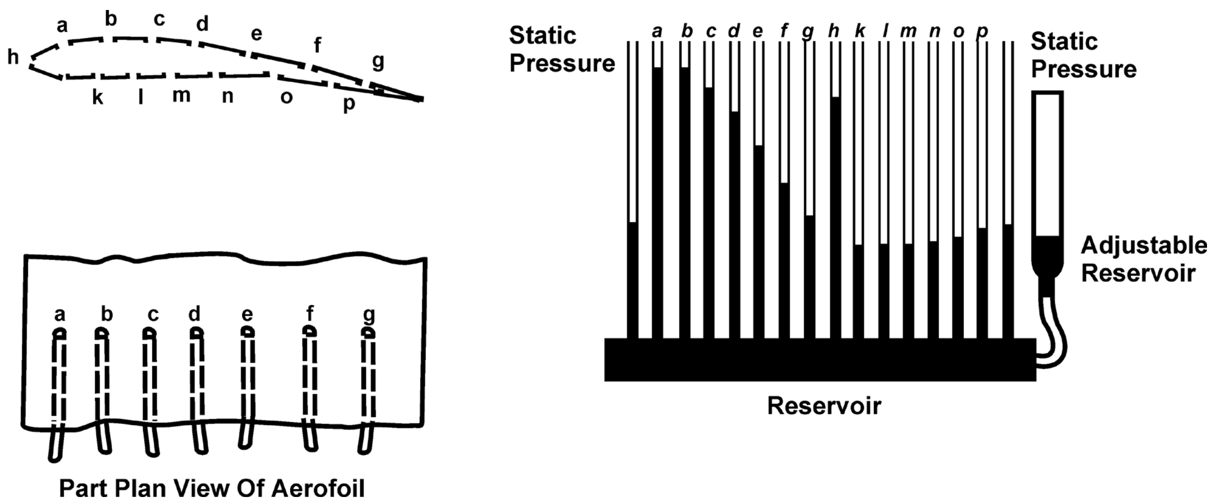

Fig.4.7. Pressure Plotting

Nowadays, simple glass manometers are seldom used for 'pressure plotting' purpose except in elementary teaching laboratories. For more serious research work, pressure transducers are employed. These are devices that produce an electrical output that is proportional to the applied pressure. The output from such transducers may then be fed through an interface to a computer. The tedious process of reading the pressures and plotting the distribution can then be left to the computer which may also be used to calculate the resulting lift and pitching moment.

Pressure distribution

Figure 4.8 shows the pressure distribution, obtained in this manner, over an aerofoil at an angle of attack of 4°. Two points are particularly noticeable, namely -

1. The decrease in pressure on the upper surface is greater than the increase on the lower surface.
2. The pressure is not evenly distributed, both the decreased pressure on the upper surface and the increased pressure on the lower surface being most marked over the front portion of the aerofoil.

Both these discoveries are of extreme importance.

The first shows that, although both surfaces contribute, it is the upper surface, by means of its decreased pressure, which provides the greater part of the lift; at some angles as much as four-fifths.

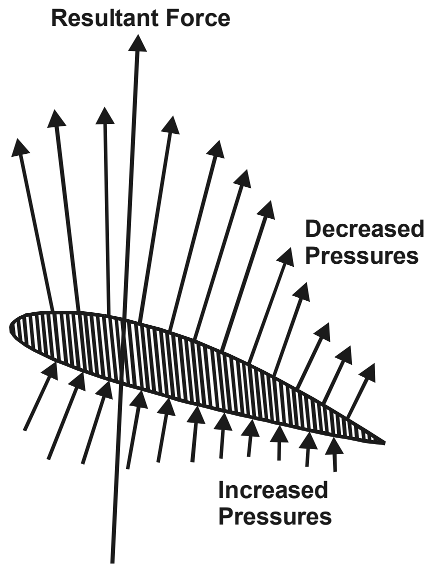

Fig.4.8. Pressure distribution over an aerofoil

The student is at first startled by this fact, as he feels that it is contrary to his ideas of common sense; but, as so often happens, once he has learnt the truth, he is inclined to exaggerate it, and to refer to the area above the aerofoil as a 'partial vacuum' or even a 'vacuum'. Although, by a slight stretch of imagination, we might allow the term 'partial vacuum', the term 'vacuum' is hopelessly misleading. We find that the greatest height to which water in a manometer is sucked up when air flows over an ordinary aerofoil at the ordinary speeds of flight is about 120 to 150 mm; now, if there were a 'vacuum' over the top surface, the water would be sucked up about 10 m, i.e. 10000 mm. Or, looking at it another way, suppose that there were a 'vacuum' over the top surface of an aerofoil and that the pressure underneath was increased from 100 kN/m^2 to 120 kN/m^2, then we would have an average upward pressure on the aerofoil of 120 kN/m^2. The actual average lift obtained from an aeroplane wing is from about 1/2up to 5 kN/m^2. Take a piece of cardboard of about 100 cm^2, or 1/10th of a square metre, and place a weight of 100 N on it; lift this up and it will give you some idea of the average lift provided by one-tenth of a square metre of aeroplane wing, and the type of load that has to be carried by the skin. You will not want to repeat the experiment with more than 10000 N on the cardboard!

The reason why the pressure distribution diagram has not been completed round the leading edge is because the changes of pressure are very sudden in this region and cannot conveniently be represented on a diagram. The increased pressure on the underside continues until we reach a point head-on into the wind where the air is brought to rest and the increase of pressure is $\frac{1}{2}\rho V^2$, or q, as recorded on a pitot tube. The point at which this happens is called the stagnation point, and its position round the leading edge varies slightly as the angle of attack of the aerofoil is changed but is always just behind the nose on the underside of positive angles of attack. After the stagnation point there is a very sudden drop to zero, followed by an equally sudden change to the decreased pressure of the upper surface, and rather surprisingly on the nose.

LIFT CURVE (C_L - α CURVE)

Let us first see how the lift coefficient changes with the angle of attack (Fig.4.9).

We notice that when the angle of attack has reached 0^o there is already a definite lift coefficient and therefore a definite lift; this is a property of most cambered aerofoils. A flat plate, or a symmetrical aerofoil, will of course give no lift when there is no angle of attack.

Then between 0^o and about 12^o the graph is practically a straight line, meaning that as the angle of attack increases there is a steady increase in the lift; whereas above 12^o, although the lift still increases for a few degrees, the increase is now comparatively small and the graph is curving to form a top, or maximum point.

At about 15^o the lift coefficient reaches a maximum, and above this angle it begins to decrease, the graph now curving downwards.

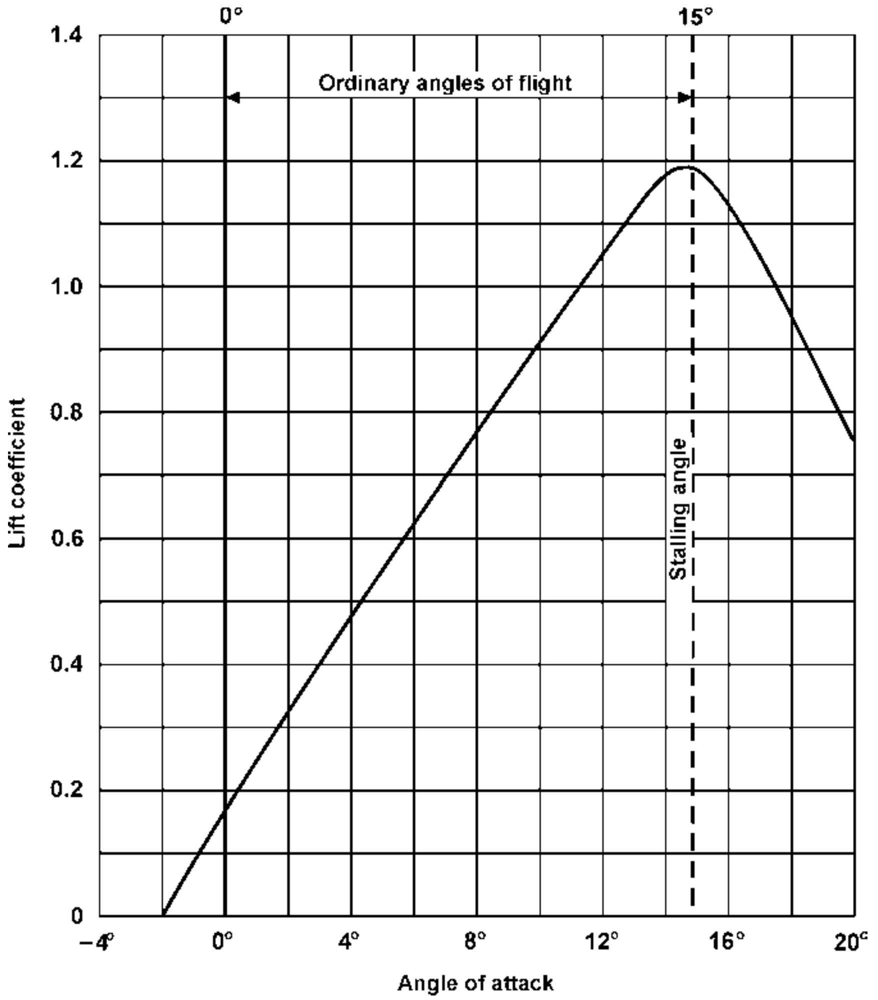

Fig.4.9. Lift curve

STALLING ANGLE: DETAILS OF STALLING

This last discovery is perhaps the most important factor in the understanding of the why and wherefore of flight. It means that whereas at small angles any increase in the angle at which the aerofoil strikes the air will result in an increase in lift, when a certain angle is reached any further increase of angle will result in a loss of lift.

This angle is called the **stalling angle of the aerofoil**, and, rather curiously, perhaps, we find that the shape of the aerofoil makes little difference to the angle at which this stalling takes place, although it may affect considerably the amount of lift obtained from the aerofoil at that angle.

Now, what is the cause of this comparatively sudden breakdown of lift? The student will be well advised to take the first available opportunity of watching, or trying for himself, some simple experiment to see what happens. Although, naturally, the best demonstration can be given in wind tunnels with proper apparatus for the purpose, perfectly satisfactory experiments can be made by using paper or wooden model aerofoils and inserting them in any fairly steady flow of air or water, or moving them through air or water. The movement of the fluid is emphasised by introducing wool streamers or cigarette smoke in the case of air and coloured streams in the case of water.

Contrary to what might be expected, the relative speed at which the aerofoil moves through the fluid makes very little difference to the angle at which stalling takes place; in fact, an aerofoil stalls at a certain angle, not at a certain speed. (It is not correct to talk about the stalling speed of an aerofoil, but it will be seen in a later chapter why we talk about the stalling speed of an aeroplane). Now what happens? While the angle at which the aerofoil strikes the fluid is comparatively small, the fluid is deflected by the aerofoil, and the flow is of a streamline and steady nature ; but suddenly, when the critical angle of about 15° is reached, there is a complete change in the nature of the flow. The airflow breaks away or separates from the top surface forming vortices similar to those behind a flat plate placed at right angles to the wind; there is therefore very little lift. Some experiments actually show that the fluid which has flowed beneath the under-surface doubles back round the trailing edge and proceeds to flow forward over the upper surface. In short, the streamline flow has broken down and what is called separation or 'stalling' has taken its place, with consequent loss in lift (Fig.4.10).

Anyone who has steered a boat will be familiar with the same kind of phenomenon when the rudder is put too far over, and yachtsmen also experience 'stalling' when their sails are set at too large an angle to the relative wind. There are, in fact, many examples of stalling in addition to that to the aerofoil.

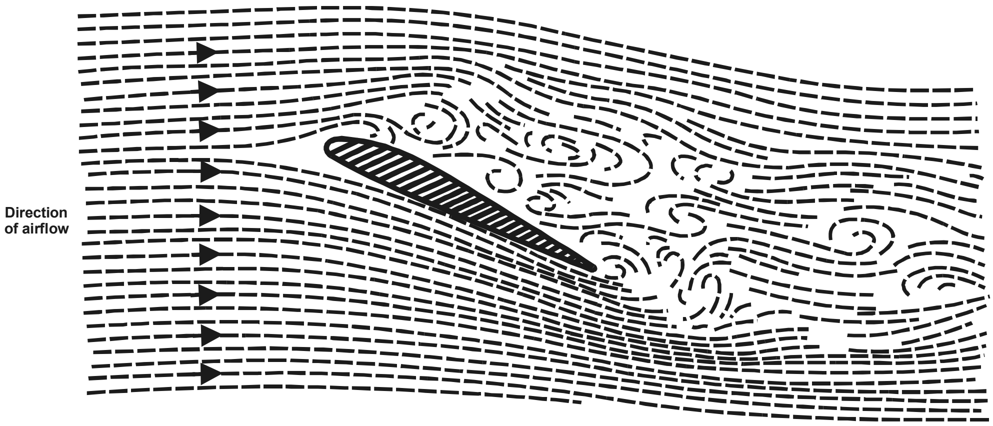

Fig.4.10. Stalling of an aerofoil

What happens is made even more clear if we look again at the results of pressure plotting . We notice that up to the critical angle considerable suction has been built up over the top surface, especially near the leading edge, whereas when we reach the stalling angle the suction near the leading edge disappears, and this accounts for the loss in lift, because the pressure on other parts of the aerofoil remains much the same as before the critical angle.

Some students are apt to think that all the lift disappears after the critical angle; this is not so, as will easily be seen by reference to either the lift curve or to the pressure plotting diagrams. The aerofoil will, in fact, give some lift up to an angle of attack of 90^o. Modern interceptor aircraft are sometimes flown at very high angles of attack during violent manoeuvres, so the upper portion of the graph is nowadays quite important.

The stalling angle, then, is that angle of attack at which the lift coefficient of an aerofoil is a maximum, and beyond which it begins to decrease owing to the airflow becoming separated instead of streamlined.

AERODYNAMIC FORCES AND MOMENT COEFFICIENTS

I Centre of pressure

The second thing that we learn from the pressure distribution diagram - namely, that both decreases and increases of pressure are greatest near the leading edge of the aerofoil - means that if all the distributed forces due to pressure were replaced by a single resultant force, this single force would act less than halfway back along the chord. The position on the chord at which this resultant force acts is called the centre of pressure (Fig.4.11). The idea of a centre of pressure is very similar to that of a centre of gravity of a body whose weight is unevently distributed, and it should therefore present no difficulty to the student who understands ordinary mechanics.

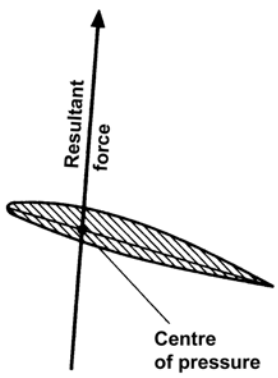

Fig.4.11. Centre of pressure

To sum up; we have, on average, a decreased pressure above the aerofoil and an increased pressure below. The decrease of pressure above is greater than the increase below, and in both cases, the effect is greater near the leading edge(Fig. 4.8).

All this is important when we consider the structure of the wing; for instance, we shall realise that the top surface or 'skin' must be held down on to the ribs, while the bottom skin will simply be pressed up against them.

Total resultant force on an aerofoil

If we add up the distributed forces due to pressure over an aerofoil, and replace it by the total resultant force acting at the centre of pressure, we find that this force is not at right angles to the chord line nor at right angles to the flight direction. Near the tips of swept wings it can sometimes be inclined forward relative to the latter line due to rather complicated three dimensional effects, but over most of the wing, and on average, it must always be inclined backwards, otherwise we would have a forwards component, or negative drag, and hence perpetual motion.

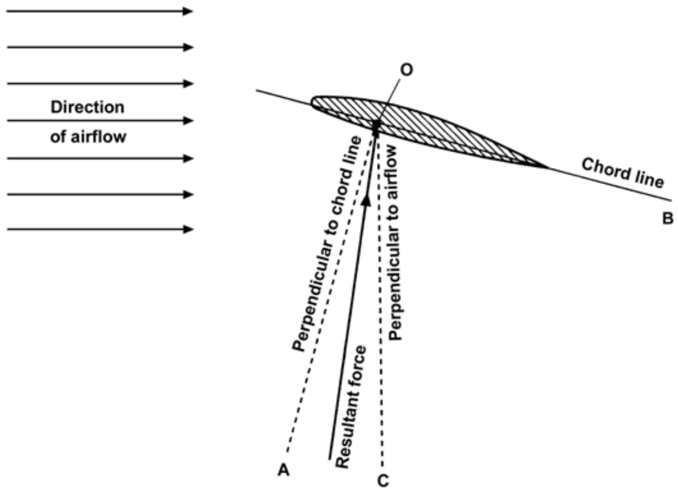

Fig.4.12. Inclination of resultant force

Although the force must on average be inclined backwards relative to the flight direction as in Fig.4.12 it can often be inclined forwards relative to the chord line normal. (Fig.4.12) illustrates the situation. You will see from this figure that there can be a component of the force that is trying to bend the wings forward. This may come as a surprise, because you might have expected that the wings would always be bent rearwards.

Movement of centre of pressure

Pressure plotting experiments also show that as the angle of attack is altered the distribution of pressure over the aerofoil changes considerably, and in consequence there will be a movement of the centre of pressure. The position of the center of pressure is usually defined as being a certain proportion of the chord from the leading edge. (Fig. 4.13) illustrates typical pressure distribution over an aerofoil at varying angles of attack. In these diagrams only the lift component of the total pressure has been plotted - the drag component has hardly any effect on the position of the centre of pressure. It will be noticed that at a negative angle, and even at 0^o, the pressure on the upper surface near the leading edge is increased above normal, and that on the lower surface is decreased; this causes the loop in the pressure diagram, which means that this portion of the aerofoil is being pushed downwards, while the rear portion is being pushed upwards, so that the whole aerofoil tends to turn over nose first.

So, even at the angle of zero lift, when the upward and downward forces are equal, there is a nose-down pitching movement on the aerofoil; as will be seen later this is a matter of considerable significance. Putting it another way, at these negative angles the centre of pressure is a long way back - the only place where we could put one force which would have the same moment or turning effect as the distributed pressure would be a long way behind the trailing edge, in fact at zero lift it could not provide a pitching moment at all unless it were an infinite distance back - which is absurd. Perhaps a more sensible way of putting it is to say that there is a couple acting on the aerofoil, and a couple has no resultant and has the same moment about any point.

As the angle of attack is increased up to 16^o, the centre of pressure gradually moves forward until it is less than one-third of the chord from the leading edge; above this angle it begins to move backwards again.

Now during flight, for reasons which we shall see later, the angle of attack is usually between 2^o and 8^o and is very rarely below 0^o or above 16^o. So, for the ordinary angles of flight, as the angle of attack of the aerofoil is increased, the centre of pressure tends to move forward.

Lift a poker at its centre of gravity and it will lie horizontal; move the position at which you lift it forwards towards the knob and the rear end of the poker will drop: this is because the centre of lift has moved forwards as compared with the centre of gravity. Therefore if the aerofoil is in balance or 'trimmed' at one angle of attack, so that the resultant force passes through the centre of gravity, then the forward movement of the centre of pressure on the aerofoil as the angle of attack is increased will tend to drop still farther the trailing edge of the aerofoil; in other words, the angle of attack will increase even more, and this will in turn cause the centre of pressure to move farther forward, and so on. This is called **instability**, and it is one of the problems of flight.

If we were to take the wing off a model aeroplane and try to make it glide without any fuselage or tail, we would find that it would either turn over nose first or its nose would go up in the air and it would turn over on to its back. This is because the wing is unstable, and although we might be able to weight it so that it would start on its glide correctly, it would very soon meet some disturbance in the air which would cause it to turn over one way or the other.

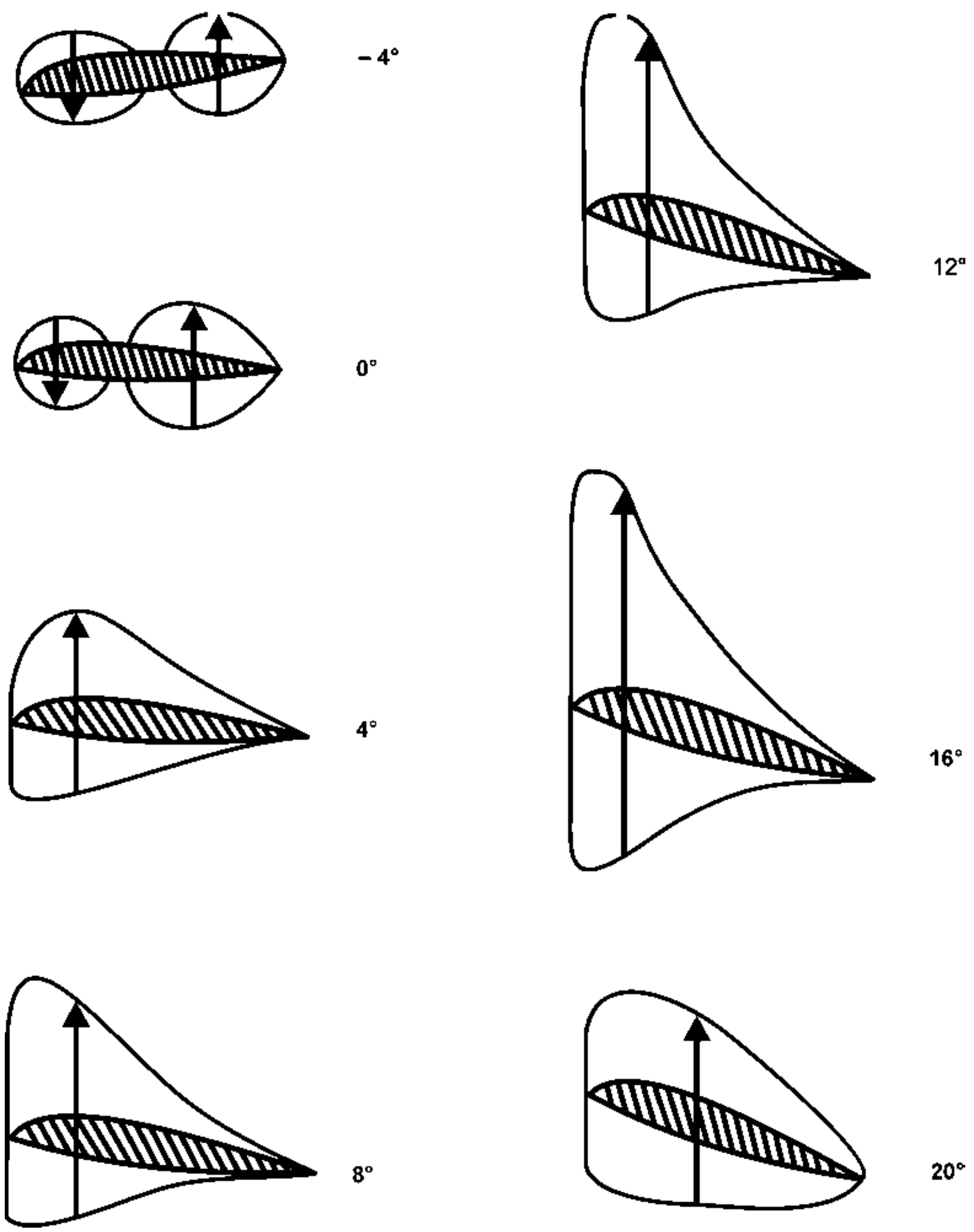

Fig.4.13. How the lift distribution changes with angle of attack

Curiously enough, in the case of a flat plate, an increase of the angle of attack over the same angles causes the centre of pressure to move backwards; this tends to dip the nose of the plate back again to its original position, and so makes the flat plate stable. For this reason it is possible to take a flat piece of stiff paper or cardboard, and, after properly weighting it, to make it glide across the room. If it is not weighted the centre of pressure will always be in front of the centre of gravity, and this will cause the piece of paper to revolve rapidly.

The unstable movement of the centre of pressure is a disadvantage of the ordinary curved aerofoil, and in a later chapter we shall consider the steps which are taken to counteract it. Attempts have been made to devise aerofoil shapes which have not got this unpleasant characteristic, and it has been found possible to design an aerofoil in which the centre of pressure remains practically stationary over the angles of attack used in ordinary flight. The chief feature in such aerofoils is that the under-surface is convex, and that there is sometimes a reflex curvature towards the trailing edge (see Fig.4.14); nearly all modern aerofoil sections have in fact got convex camber on the lower surface. Unfortunately, attempts to improve the stability of the aerofoil may often tend to spoil other important characteristics.

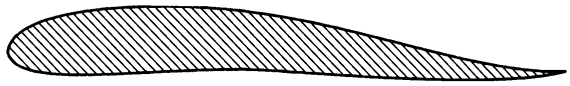

Fig.4.14. Reflex curve near trailing edge

■ Pitching Moment of an aerofoil

Now the ultimate object of the aerofoil is to obtain the lift necessary to keep the aeroplane in the air; in order to obtain this lift it must be propelled through the air at a definite velocity and it must be set at a definite angle of attack to the flow of air past it. We have already discovered that we cannot obtain a purely vertical force on the aerofoil; in other words, we can only obtain lift at the expense of a certain amount of drag. The latter is a necessary evil, and it must be reduced to the minimum so as to reduce the power required to pull the aerofoil through the air, or alternatively to increase the velocity which we can obtain from a given engine power. Our next task, therefore, is to investigate how much lift and how much drag we shall obtain from different shaped aerofoils at various angles of attack and at various velocities. The task is one of appalling magnitude; there is no limit to the number of aerofoil shapes which we might test, and in spite of thousands of experiments carried out in wind tunnels and by full-scale tests in the air, it is still impossible to say that we have discovered the best-shaped aerofoil for any particular purpose. However, modern theoretical methods make it possible to predict the behaviour of aerofoil sections. Such methods can even be used to design aerofoils to give specified characteristics.

In wind-tunnel work it is the usual practice to measure lift and drag separately, rather than to measure the total resultant force and then split it up into two components. The aerofoil is set at various angles of attack to the airflow, and, the lift, drag and pitching moment are measured on a balance.

The results of the experiments show that within certain limitations the lift, drag and pitching moment of an aerofoil depend on -

a. The shape of the aerofoil.
b. The plan area of the aerofoil.
c. The square of the velocity.
d. The density of the air.

Notice the dimilarity of these conclusions to those obtained when measuring drag, and in both cases there are similar limitations to the conclusions arrived at.

The reader should notice that whereas when measuring drag we considered the frontal area of the body concerned, on aerofoils we take the plan area. This is more convenient because the main force with which we are concerned, i.e. the lift, is at right angles to the direction of motion and very nearly at right angles to the aerofoils themselves, and therefore this force will depend on the plan area rather than the front elevation. The actual plan area will alter as the angle of attack is changed and therefore it is more convenient to refer results to the maximum plan area (the area projected on the plane of the chord), so that the area will remain constant whatever the angle of attack may be. Unfortunately it is customary to use the same symbol S both for the plan area of a wing and the frontal area of any other body.

In so far as the above conclusions are true, we can express them as formulae in the forms -

$$\text{Lift} = C_L . \frac{1}{2}\rho V^2 . \ S \text{ or } C_L . q . S$$

$$\text{Drag} = C_D . \frac{1}{2}\rho V^2 . \ S \text{ or } C_D . q . S$$

$$\text{Pitching Moment} = C_M \cdot \frac{1}{2}\rho V^2 \cdot Sc \text{ or } C_M \cdot q \cdot Sc$$

Since the pitching moment is a moment, i.e. a force X distance, and since $\frac{1}{2}\rho V^2 \cdot S$ represents a force, it is necessary to introduce a length into the equation - this is in the form of the chord, c, measured in metres.

The pitching moment is positive when it tends to push the nose upwards, negative when the nose tends to go downwards - as at zero lift.

The symbols C_L, C_D and C_M are called the lift coefficient, drag coefficient and pitching moment coefficient of the aerofoil respectively; they depend on the shape of the aerofoil, and they alter with changes in the angle of attack. The air density is represented by ρ in kilograms per cubic metre, S is the plan area of the wing in square metres, V is the air speed, in metres per second, c the chord of the aerofoil in metres; the method of writing the formulae in terms of $\frac{1}{2}\rho V^2$, or q,

Aerofoil characteristics
The easiest way of setting out the results of experiments on aerofoil sections is to draw curves showing how -

a. the lift coefficient,
b. the drag coefficient,
c. the ratio of lift to drag, and
d. the position of the centre of pressure, or the pitching moment coefficient, alter as the angle of attack is increased over the ordinary angle of flight.

It is much more satisfactory to plot the coefficients of lift, drag and pitching moment rather than the total lift, drag and pitching moment, because the coefficients are practically independent of the air density, the scale of the aerofoil and the velocity used in the experiment, whereas the total lift, drag and moment depend on the actual conditions at the time of the experiment. In other words, suppose we take a particular aerofoil section and test it on different scales at different velocities in various wind tunnels throughout the world, and also full-scale in actual flight, we should in each case obtain the same curves showing how the coefficients change with angle of attack.

It must be admitted that, in practice, the curves obtained from these various experiments do not exactly coincide; this is because the theories which have led us to adopt the formula lift $= C_L \cdot \frac{1}{2}\rho V^2 \cdot S$ are not exactly true for very much the same reasons as those we mentioned when dealing with drag - for instance, scale effect and the interference of wind-tunnel walls. As a result of the large number of experiments which have been performed, it is possible to make allowances for these errors and so obtain good accuracy whatever the conditions of the experiment.

Now let us look at the curves to see what they mean, for a graph which is properly understood can convey a great deal of information in a compact and practical form.

The Drag Curve
Now for the drag coefficient curve (Fig.4.15). Here we find much what we might expect. The drag is least at about 0^o, or even a small negative angle, and increases on both sides of this angle; up to about 6^o, however, the increase in drag is not very rapid, then it gradually becomes more and more rapid, especially after the stalling angle when the airflow separates.

The Lift/Drag ratio curve
Next we come to a very interesting curve (Fig.4.16), that which shows the relation between the lift and drag at various angles of attack.

In a former paragraph we came to the conclusion that we want as much lift, but as little drag, as it is possible to obtain from the aerofoil. Now from the lift curve we find that we shall get most lift at about 15^o, from the drag curve least drag at about 0^o, but both of these are at the extreme range of possible angles, and at neither of them do we really get the best conditions for flight, i.e. the best lift in comparison to drag, the best lift/drag ratio.

If the reader has available the lift curve and the drag curve for any aerofoil, he can easily plot the lift/drag curve for himself by reading C_L off the lift curve at each angle and dividing it by the C_D at the same angle. It should be noted that it makes no difference whether we plot L/D or C_L/C_D, as both will give the same

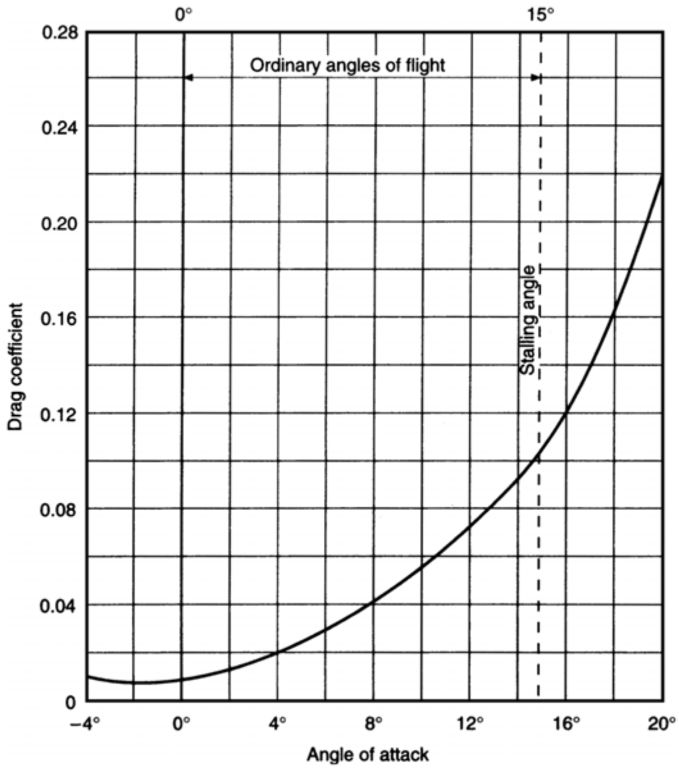

Fig.4.15. Drag curve.

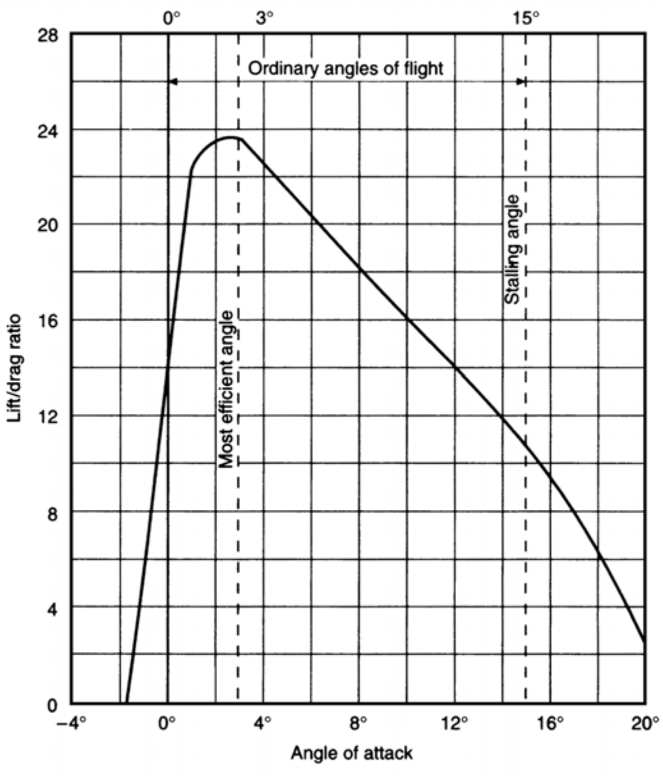

Fig.4.16. Lift/drag curve.

numerical value, since $L = C_L \cdot \frac{1}{2}\rho V^2 \cdot S \quad D = C_D \cdot \frac{1}{2}\rho V^2 \cdot S$.

We find that the lift/drag ratio increases very rapidly up to about 3° or 4°, at which angles the lift is nearly 24 times the drag (some aerofoils give an even greater maximum ratio of lift to drag); the ratio then gradually falls off because, although the lift is still increasing, the drag is increasing even more rapidly, until at the stalling angle the lift may be only 10 or 12 times as great as the drag, and after the stalling angle the ratio falls still further until it reaches 0 at 90°.

The chief point of interest about the lift/drag curve is the fact that this ratio is greatest at an angle of attack of about 3° or 4°; in other words, it is at this angle that the aerofoil gives its best all-around results - i.e. it is most able to do what we chiefly require of it, namely to give as much lift as possible consistent with a small drag.

The centre of pressure and moment coefficient

Lastly, let us examine the curves (Fig.4.17) which show how the centre of pressure moves, and what happens to the pitching moment coefficient, as the angle of attack is increased.

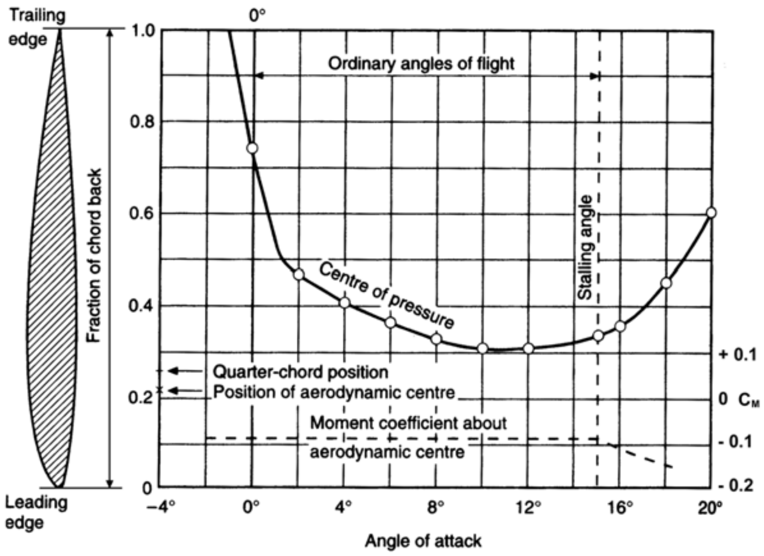

Fig.4.17. Centre of pressure and moment coefficient curves.

The centre of pressure curve merely confirms what we have already learnt about the movement of the centre of pressure on an ordinary aerofoil. After having been a long way back at negative angles, at 0° it is about 0.70 of the chord from the leading edge, at 4° it is 0.40 of the chord back, and at 12° 0.30 of the chord; in other words, the centre of pressure gradually moves forward as the angle is increased over the ordinary angles of flight; and this tends towards instability. After 12° it begins to move back again, but this is not of great importance since these angles are not often used in flight.

It is easy to understand the effect of the movement of the centre of pressure, and for that reason it has perhaps been given more emphasis in this book than it would be in more advanced books on the subject.

It is important to remember that the pitching moment, and its coefficient, depend not only on the lift (or more correctly on the resultant force) and on the position of the centre of pressure, but also on the point about which we are considering the moment - which we shall call the reference point. There is, of course, no moment about the centre of pressure itself - that, after all, is the meaning of centre of pressure - but, as we have seen, the centre of pressure is not a fixed point. If we take as our point of reference some fixed point on the chord we shall find that the pitching moment - which was already slightly nose-down (i.e. slightly negative) at the angle of zero lift -increases or decreases as near as matters in proportion to the angle of attack, i.e. the graph is a straight line, like that of the lift coefficient, over the ordinary angles of flight. About the leading edge, for instance, it becomes more and more nose-down as the angle is increased; but about a point near the trailing edge, although starting at the same slightly nose-down moment at zero lift, it becomes less nose-down, and finally nose-up, with increase of angle (Fig.4.18).

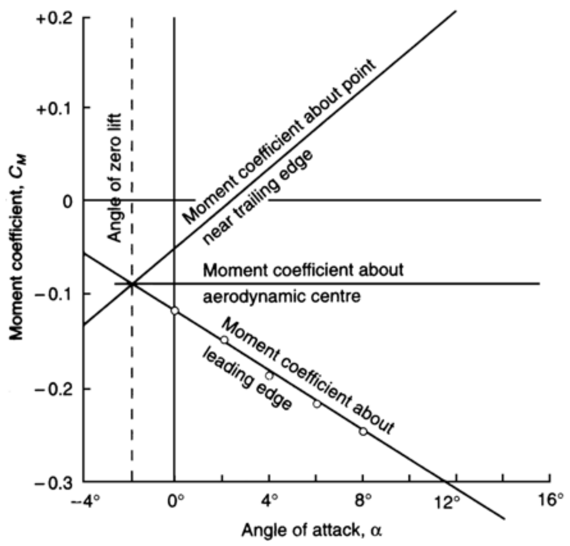

Fig.4.18. Moment coefficient about different reference points.

The reader may be surprised at the increasing nose-down moment about the leading edge, because is not the centre of pressure moving forward? Yes, but the movement is small and the increasing lift has more effect on the pitching moment. The intelligent reader may be even more surprised to hear that an increasing nose-down tendency is a requirement for the pitching stability of the aircraft, for have we not said that the movement of the centre of pressure was an unstable one? Yes, this is a surprising subject, but the answer to the apparent paradox emphasises once again the importance of the point of reference; in considering the stability of the whole aircraft our point of reference must be the centre of gravity, and the centre of gravity is always, or nearly always, behind the leading edge of the wing, so the change of pitching moment with angle of attack is more like that about the trailing edge - which is definitely unstable.

Aerodynamic Centre

But something else of considerable importance arises from the differing effects of different reference points. For if about the leading edge there is a steady increase, and about a point near the trailing edge a steady decrease in the nose-down pitching moment, there must be some point on the chord about which there is no change in the pitching moment as the angle of attack is increased, about which the moment remains at the small negative nose-down value that it had at the zero lift angle (Figs.4.17 and 4.18).

This point is called the aerodynamic centre of the wing.

So we have two possible ways of thinking about the effects of increase of angle of attack on the pitching moment of an aerofoil, or later of the whole aeroplane; one is to think of the lift changing, and its point of application (centre of pressure) changing; the other is to think of the point of application (aerodynamic centre) being fixed, and only the lift changing (Fig.4.19). Both are sound theoretically; the conception of a moving centre of pressure may sound easier at first, but for the aircraft as a whole it is simpler to consider the lift as always acting at the aerodynamic centre. In both methods we really ought to consider the total force rather than just the lift, but the drag is small in comparison and, for most purposes, it is sufficiently accurate to consider the lift alone.

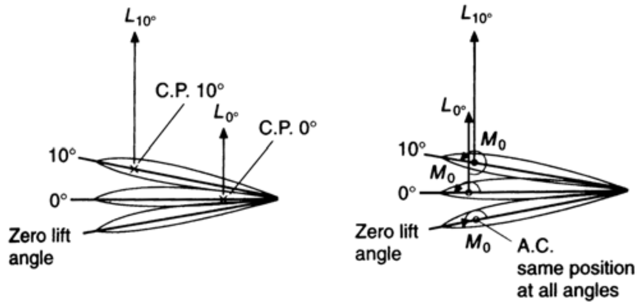

Fig.4.19. Centre of pressure and aerodynamic centre.

At subsonic speeds the aerodynamic centre is usually about one-quarter of the chord from the leading edge, and theoretical considerations confirm this. In practice, however, it differs slightly according to the aerofoil section, usually being ahead of the quarter-chord point in older type sections, and slightly aft in more modern low drag types.

The graph in Fig.4.18 (it can hardly be called a curve) shows how nearly the moment coefficient, about the aerodynamic centre, remains constant on our aerofoil at its small zero-lift negative value of about -0.09. This is further confirmed by the figures of C_M given in Appendix 1 for a variety of aerofoil shapes.

The graphs tell us all we want to know about a particular wing section; they give us the 'characteristics' of the section, and from them we can work out the effectiveness of a wing on which this section is used.

For example, to find the lift, drag and pitching moment per unit span (about the aerodynamic centre) of an aerofoil of this section, of chord 2 metres at 6° angle of attack, and flying at 100 knots at standard sea-level conditions.

From Figs.4.9, 4.15 and 4.17, we find that at 6°-

C_L = 0.6
C_D = 0.028
C_M = -0.09 about aerodynamic centre
100 knots = 51.6 m/s

Since $\frac{1}{2}\rho V^2$ (or q) is common to the lift, drag and moment formulae, we can first work out its value -

$$q = \frac{1}{2}\rho V^2 = \frac{1}{2}\times 1.225\times 51.6\times 51.6 = 1631\,\mathrm{N/m^2}$$

So lift = $C_L \cdot q \cdot S = 0.6\times 1631\times 2 = 1957\,\mathrm{N}$

drag = $C_D \cdot q \cdot S = 0.028\times 1631\times 2 = 91.3\,\mathrm{N}$

pitching moment = $C_M \cdot q \cdot Sc = -0.09\times 1631\times 20\times 2$

$= -5872\,\mathrm{N\text{-}m}$

But where is the aerodynamic centre on this aerofoil?

At zero lift there is only a pure moment, or couple, acting on the aerofoil, and since the moment of a couple is the same about any point, this moment, and its coefficient, must be equal to that about the aerodynamic centre, which we shall call $C_{M\cdot AC}$ (sometimes written as C_{MO}), and this by definition will remain the same whatever the angle of attack.

For all practical purposes we can assume that the aerodynamic centre is on the chord line, though it may be very slightly above or below. So let us suppose that it is on the chord line, and at distance x from the leading edge, and that the angle of attack is α° (Fig.4.20).

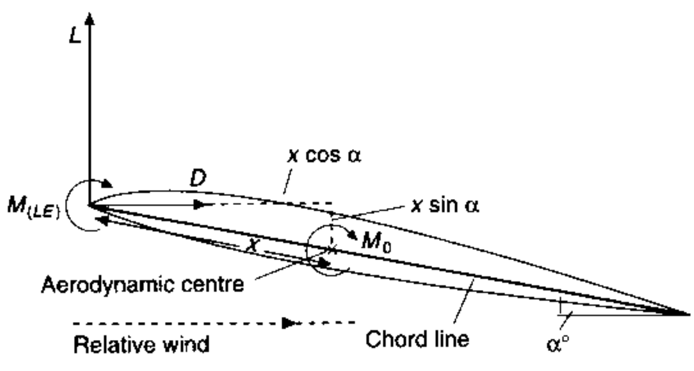

Fig.4.20. To find aerodynamic centre.

The moment about the aerodynamic centre, i.e. $C_{M \cdot AC} . q . Sc$, will be equal to the moment about the leading edge (which we call $C_{M \cdot LE} . q . Sc$) plus the moments of L and D about the aerodynamic centre; the leverage being x cos α and x sin α respectively.
So

$$C_{M.AC} . q . Sc = C_{M.LE} . q . Sc + C_L . q . S . x . \cos\alpha + C_D . q . S . x . \sin\alpha$$

and, dividing all through by q.S,

$$C_{M.AC} . c = C_{M.LE} . c + C_L . x . \cos\alpha + C_D . x . \sin\alpha$$

$$\therefore x = c . (C_{M.AC} - C_{M.LE}) / (C_L \cos\alpha + C_D \sin\alpha)$$

or, expressed as a fraction of the chord,

$$x/c = (C_{M.AC} - C_{M.LE}) / (C_L \cos\alpha + C_D \sin\alpha)$$

But the moment coefficient about the leading edge for this aerofoil at 6^o is -0.22 (see Fig.4.18), and $C_{M.Ac}$ is -0.09 (Fig.4.17),

C_L = 0.6, cos 6^o = 0.994, C_D = 0.028, sin 6^o = 0.10

So

$$x / c = (-0.09 + 0.22)/(0.6 \times 0.994 + 0.028 \times 1.10)$$
$$= 0.13/(0.60+0.003)$$
$$= \mathbf{0.216}$$

which means that the aerodynamic centre is 0.216 of the chord, or 0.432 metres, behind the leading edge, and so in this instance is forward of the quarter-chord (0.25) point.

Notice that at small angles, such as 6^o, cos α is approx 1, sin α is nearly 0, so we can approximate by forgetting about the drag and saying that x/c = $(C_{M.AC} - C_{M.LE})/C_L$ approx.

About the centre of pressure there is no moment, so

(Distance of C.P. from L.E.)/c = $-C_{M.LE} / (C_L \cos\alpha + C_D \sin\alpha)$
= $-C_{M.LE} / C_L$ approx.
= + 0.22/0.60 = **0.37**

thus confirming the position of the C.P. as shown in Fig.4.17.

All this has been explained rather fully at this stage; its real significance in regard to the stability of the aircraft will be revealed later.

■■■

CHAPTER: 5
AIRCRAFT CONTROL SURFACES

PRIMARY CONTROL SURFACES

The primary control surfaces of an airplane include the ailerons, rudder, and elevator. Secondary control surfaces include tabs, flaps, spoilers, and slats. The construction of the control surfaces is similar to that of the stabilizers ; however, the movable surfaces usually are somewhat lighter in construction. They often have a spar at the forward edge to provide rigidity and to this spar are attached the ribs and the covering. Hinges for attachment are also secured to the spar. Where it is necessary to attach tabs to the trailing edges of control surfaces, additional structure is added to provide for transmission of the tab loads to the surface.

Control surfaces may be constructed of any combination of materials, with the more common combination being a sheet-metal structure (usually an aluminium alloy) covered with metal skin or fabric, a steel structure covered with fabric, or a wood structure covered with plywood or fabric. Each of these types of construction is treated by some method to inhibit the deterioration of the structure and the covering and includes drain holes to prevent water from becoming trapped inside the structure and causing the control surfaces to be thrown out of balance. Methods of joining the components may include metal fasteners as well as adhesives and bonding agents.

Some aircraft are using composite and bonded structures which include the use of honeycomb internal components. These structures are often sealed from the atmosphere and therefore do not include drainage openings in their design.

(i) Aileron

Ailerons are primary flight control surfaces utilized to provide lateral (roll) control of aircraft; that is they control aircraft movement about the longitudinal axis. They are usually mounted on the trailing edge of the wing near the wing tip. Large jet aircraft often employ two sets of ailerons, one set being approximately midwing or immediately outboard of the inboard flaps, and the other set being in the conventional location near the wing tips. The outboard ailerons become active whenever the flaps are extended beyond a fixed setting. As the flaps are retracted, the outboard aileron control system is "locked out" and fairs with the basic wing shape. Thus, during cruising operations at comparatively high speeds, only the inboard ailerons are used for control. The outboard ailerons are active during landing or other slow flight operations. (Fig.5.1) shows a transport aircraft wing with this aileron configuration.

Ailerons for light aircraft are usually constructed with a single spar to which ribs are attached. The majority of currently manufactured aircraft are of all-metal construction with aluminium alloy skin riveted or bonded to the internal structure.

Aileron control systems operated by the pilot through mechanical connections require the use of balancing mechanisms so that the pilot can overcome the air loads imposed on the ailerons during flight. Balancing of the ailerons can be achieved by extending part of the aileron structure ahead of the hinge line and shaping this area so that the airstream strikes the extension and helps to move the surface. This is known as aerodynamic balancing. Another method which may be used is to place a weight ahead of the hinge line to counteract the flight loads. This is known as static balancing. Some aircraft may use a combination of these techniques.

Aircraft such as jet transports use hydraulically operated ailerons and may not employ these forms of balancing. If the transport control system is designed to allow the pilot to operate the ailerons without hydraulic assistance, then some method of balancing or control by control tabs is used.

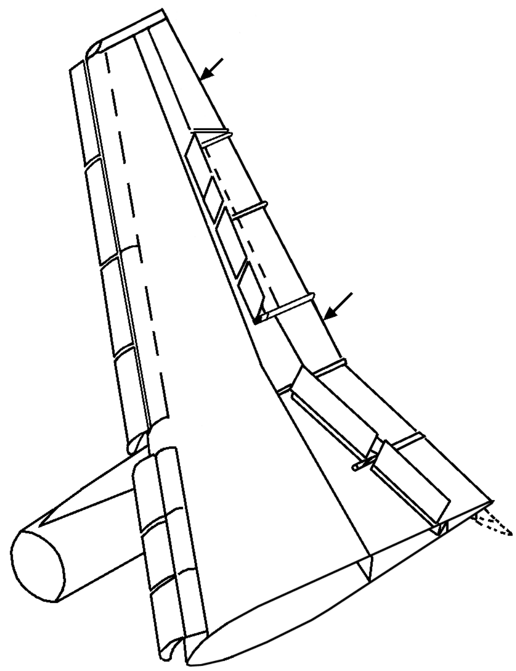

Fig.5.1. The L-1011, uses two ailerons on each wing

The geometry of the control system for the ailerons affects the amount that the ailerons move above or below the neutral setting. (The neutral setting fairs the ailerons with the wing contour). Some aircraft have their ailerons operating symmetrically; that is, they move up the same amount that they move down. Other aircraft have the ailerons operating asymmetrically; that is, the upward-moving aileron moves further than the downward moving aileron. This asymmetrical operation is used in some aircraft designs to reduce the amount of rudder pressure required when making turns. This reduces what is know as "adverse aileron yaw", which is caused by the downward-moving aileron creating an increase in aerodynamic drag, and results in the airplane yawing away from the direction of the desired turn. Aircraft having this arrangement are sometimes said to have differential ailerons.

(ii) Elevators

Elevators are the control surfaces which govern the movement (pitch) of the aircraft around the lateral axis. They are normally attached to hinges on the rear spar of the horizontal stabilizer. The construction of an elevator is similary to that of other control surfaces, and the design of the elevator may be unbalanced or balanced aerodynamically and/or statically. Typical elevator installations for light aircraft and transports are shown in (Fig.5.2 and 5.3).

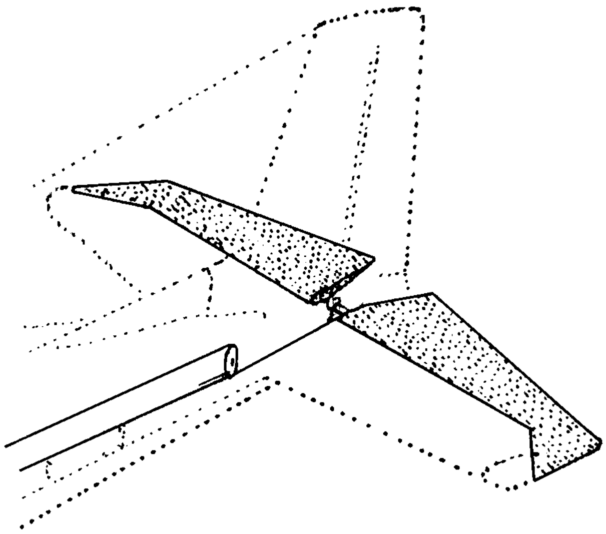

Fig.5.2. The elevator of a light aircraft

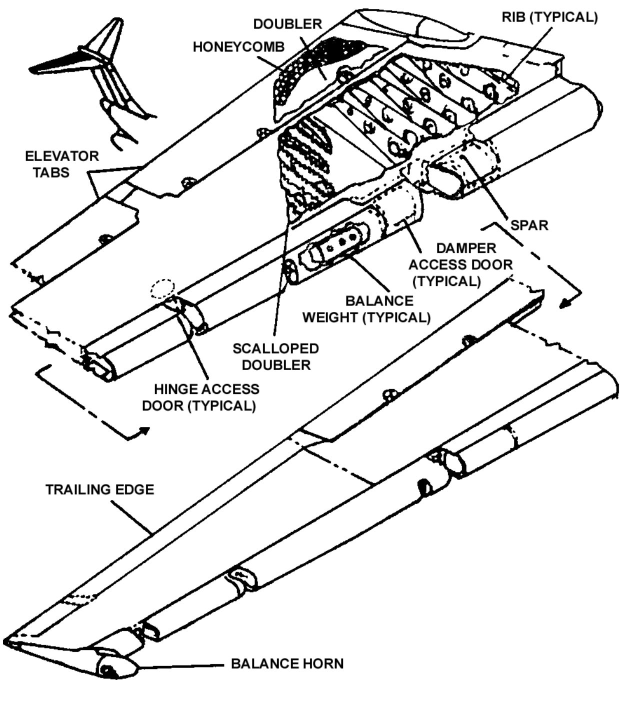

Fig.5.3. The elevator of a DC-9

(iii) Rudder

The rudder is the flight control surface that controls the aircraft movement about its vertical axis. The rudder is constructed very much like other flight control surfaces with spars, ribs and skin.

Rudders are usually balanced both statically and aerodynamically to provide for greater ease of operation and to eliminate the possibility of flutter. It should be noted that some light-aircraft rudders do not use any balancing method. Different rudders for light aircraft are shown in (Fig.5.4) below.

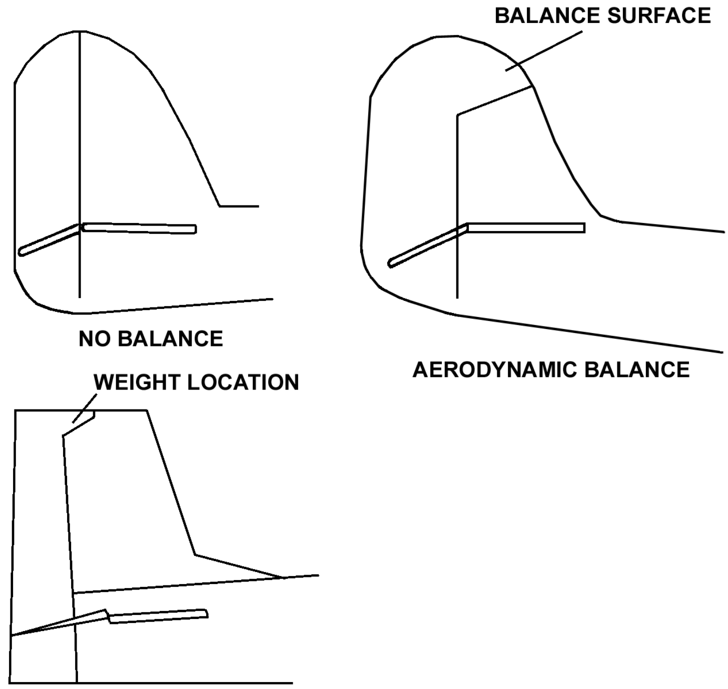

Fig.5.4. Different rudder configuration

Rudders for transport aircraft vary in basic structural and operational design. Some are single structural units operated by one or more control systems. Others are designed with two operational segments which are controlled by different operating systems and provide a desired level of redundancy.

A single-unit rudder is capable of being operated by three different hydraulic systems in the aircraft. A rudder with an upper and a lower segment of which each segment can be operated by a different hydraulic systems.

The rudder with of two segments consists of upper and lower, and each segment consists of a forward and aft section. The forward rudder sections are attached to hinge brackets mounted on the rear spar of the vertical stabilizer. The aft rudder sections are supported by hinge brackets attached to the rear spar of the forward sections. The aft rudder sections are hinged to the forward sections and connected by pushrods to the vertical stabilizer structure. This provides aft-section displacement proportional to forward-section displacement, thus increasing the aerodynamic efficiency of the rudders. Trim and control tabs are not required with this type of rudder design because their functions are performed by the aft sections of the rudder.

SECONDARY CONTROL SURFACES

Because aircraft often are capable of operating over a wide speed range and with different weight distributions, secondary flight controls, also called auxiliary flight controls, have been developed. Some of these surfaces called tabs, allow the flight controls. Other surfaces fall in a group termed high-lift devices which includes flaps, slats and slots. These allow the lift and drag characteristics of the aircraft wing to be changed to allow-slow speed flight for takeoff and landing and high-speed flight for cruising. Still a third group of surfaces are used to reduce lift and generate drag. This group includes spoilers and speed brakes.

The number and complexity of the secondary control surfaces on a particular aircraft depends on the type of operation and flight speeds for which the aircraft is designed. (Fig.5.5) below shows the secondary flight control surfaces found on a typical jet transport aircraft.

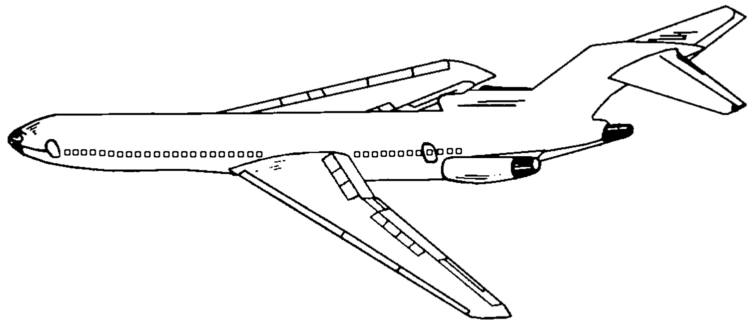

Fig.5.5. The location of secondary flight controls on a Boeing 727

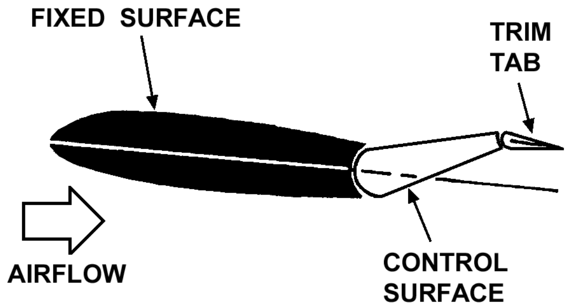

Fig.5.6. Trim tabs must be adjusted opposite to the desired movement of the surface being controlled

Tabs

Tabs are small secondary flight control surfaces set into the trailing edges of the primary surfaces. These are used to reduce the pilot's work load required to hold the aircraft in some constant attitude by "loading" the control surface in a position to maintain the desired attitude. They may also be used to aid the pilot in returning a control surface to a neutral or trimmed center position (Fig.5.6).

(i) Trim Tabs

The term trim tabs describes small secondary flight-control surfaces set into the trailing edges of the primary control surfaces. Tabs are used to reduce the work load required to hold the aircraft in some constant attitude by "loading" the control surface to a neutral or trimmed-center position. (Fig.5.7) demonstrates the tab action. Tabs can be fixed or variable, and the variable tabs can be designed to operate in several different manners.

Fixed Trim Tabs

A fixed trim tab, such as is shown in (Fig.5.7), is normally a piece of sheet metal attached to the trailing edge of a control surface. This fixed tab is adjusted on the ground by bending it in the appropriate direction to eliminate cabin flight control forces for a specific flight condition. The fixed tab is normally adjusted for zero control forces while in cruising flight. Adjustment of the tab is a trial and error process where the aircraft must be flown and the trim tab adjusted based on the pilot's report. The aircraft must then again be flown to see if further adjustment is necessary. Fixed tabs are normally found on light aircraft and are used to adjust rudders and ailerons.

Controllable Trim Tabs

A controllable trim tab is illustrated in (Fig.5.7) Controllable tabs are adjusted by means of control wheels, knobs, or cranks in the cockpit, and an indicator is supplied to denote the position of the tab.

Controllable trim tabs are found on most aircraft with at least the elevator tab being controlled. These tabs are normally operated mechanically, electrically or hydraulically. When the trim-control system is activated, the trim tab is deflected in direction opposite to the desired movement of the control surface. When the trim tab is deflected into the airstream the air tries to push the tab back flush with the control surface. Since the control mechanism prevents the tab from being pushed back flush, the whole control surface is moved.

(ii) Servo Tabs

The servo tabs, sometimes referred to as the flight tabs, are used primarily on the large main control surfaces. A servo tab is one that is directly operated by the primary controls of the airplane. In response to movement of the cockpit control, only the servo tab moves. The force of the airflow on the servo tab then moves the primary control surface. The servo tab, illustrated in (Fig.5.7) , is used to reduce the effort required to move the controls on a large airplane.

(iii) Balance Tabs

A balance tab is linked to the airplane in such a manner that a movement of the main control surface will give an opposite

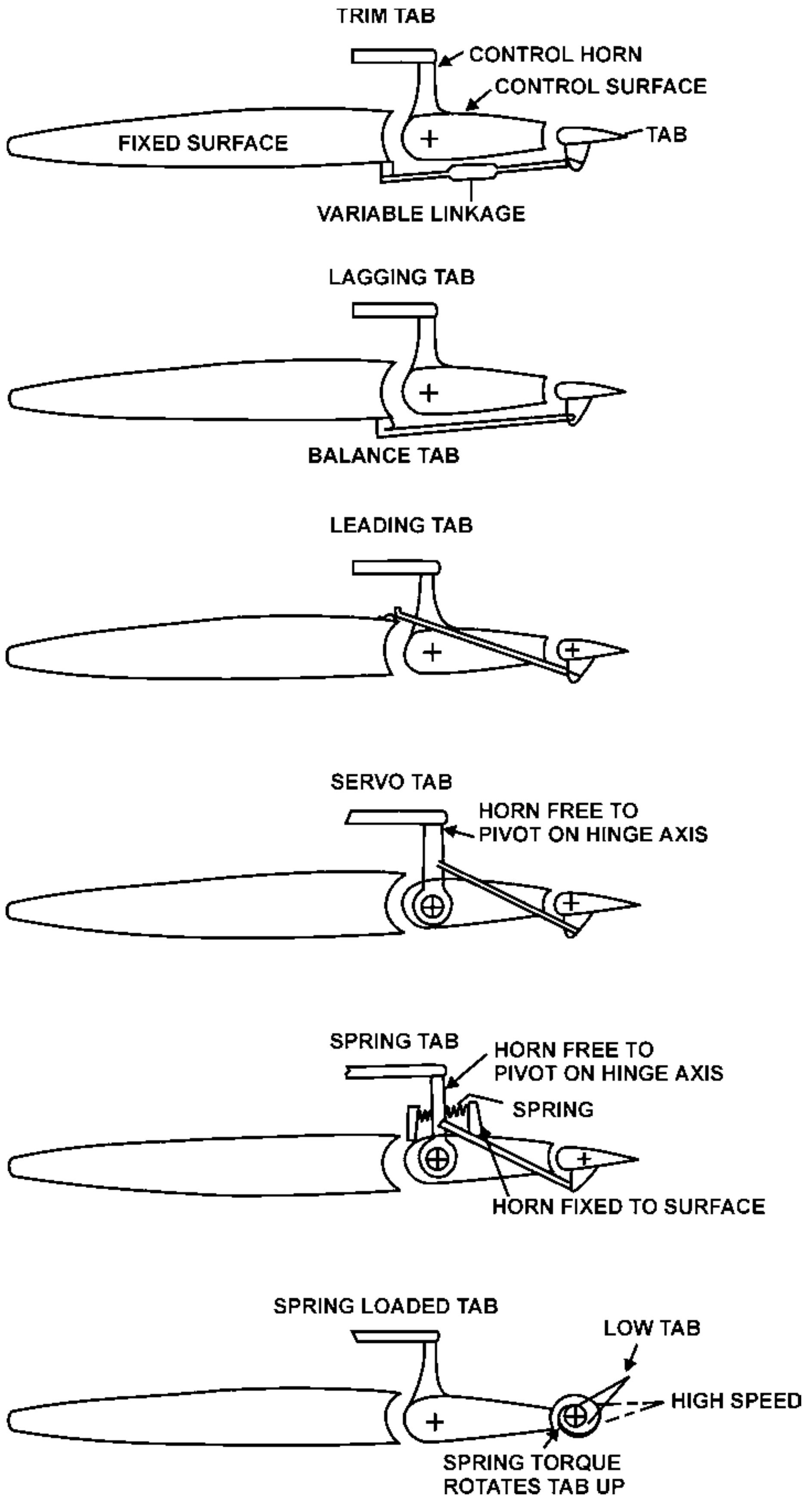

Fig.5.7. Various types of trim tabs

movement to the tab. Thus, the balance tab will assist in moving the main control surface. Balance tabs are particularly useful in reducing the effort required to move the control surfaces of a large airplane. A balance tab is illustrated in Fig. 5.7.

(iv) Spring Tabs

The spring tabs, like some servo tabs, are usually found on large aircraft that require considerable force to move a control surface. The purpose of the spring tab is to provide a boost, thereby aiding in the movement of a control surface. On the spring tab, illustrated in Figure 5.7, the control horn is connected to the control surface by springs.

AUXILIARY CONTROL SURFACES

(i) Slots

How do multi-element aerofoils greatly augment lift without suffering the adverse effects of boundary-layer separation? Hitherto the conventional explanation was that, since the slot connects the high-pressure region on the lower surface of a wing to the relatively low-pressure region on the top surface, it therefore acts as a blowing type of boundary-layer control. This explanation is to be found in a large number of technical reports and textbooks, and as such is one of the most widespread misconceptions in aerodynamics. It can be traced back to no less an authority than Prandtl who wrote:

"The air coming out of a slot blows into the boundary layer on the top of the wing and imparts fresh momentum to the particles in it, which have been slowed down by the action of viscosity. Owing to this help the particles are able to reach the sharp rear edge without breaking away."

(ii) Flaps

The history of flaps is longer, and just as varied, as that of slots. The plain or camber flap works on the same principle as an aileron or other control surface; it is truly a 'variable camber'. Such flaps were used as early as the 1914-1918 war, and the original idea was the same as with slots, to decrease landing speed with flaps down, and retain maximum speed with flaps up. Their early use was almost exclusively for deck-landing purposes. It seemed at first as though the invention of slots, which followed a few years after that war, might sound the death-knell of flaps. Far from it - if anything it has been the other way round, for flaps have become a necessity on modern aircraft. Flaps, like slots, can increase lift - honours are about even in this respect so far as the plain (or camber) flap, or split flap is concerned. But these flaps can also increase drag - not, like slots, at high speed when it is not wanted, but at low speed when it is wanted. But the main difference between the effects of flaps and slots is shown in Fig.5.8; from this it will be seen that whereas slots merely prolong the lift curve to higher values of the maximum lift coefficient, when the angle of attack of the main portion of the aerofoil is beyond the normal stalling angle, the high-lift type of flap increases the lift coefficient available throughout the whole range of angles of attack.

However it is no longer appropriate to compare the relative merits of slots and flaps because in modern aircraft it is usual to combine the two in some form or other; and in this way to get the best of both devices (Fig.5.9). There are a large number of possible combinations, but Fig.5.10 is an attempt to sum up the main varieties, and to describe the effect they have on the maximum lift coefficient, on the angle of the main aerofoil when maximum lift is obtained, why they improve the lift, what effects they have on the drag, how they affect the pitching moment, and so on.

From this figure it will be seen that the simpler flaps such as the camber flap, split flap and single slotted flap give a good increase in maximum lift coefficient at a reasonable angle of attack of the main aerofoil, and therefore a reasonable attitude of the aeroplane for landing; they also increase drag which is an advantage in the approach and landing.

The more complicated types such as the Zap and Fowler flap, and the double-or treble-slotted flap, give an even greater increase in maximum lift coefficient, but still at a reasonable angle of attack; while the even more complicated combinations of slots and flaps give yet greater maximum lift coefficients, but usually at larger angles of attack, and of course at the expense of considerable complication (Fig.5.11).

Blown and jet flaps are in a class of their own since they depend on power to produce the blowing, and this may be a serious disadvantage in the event of power failure. The true jet flap isn't a flap at all, but simply an efflux of air, or a jet stream in the form of a sheet of air ejected under pressure at or near the trailing edge of the aerofoil. This helps to control the boundary layer, and if the sheet of air can be deflected the reaction of the jet will also contribute directly to the lift.

The Krueger and other types of nose flap are used mainly for increasing lift for landing and take-off on otherwise high-speed aerofoils.

Spoilers, air brakes, dive brakes, lift dumpers and suchlike are a special category in that their main purpose is to increase drag, or to destroy lift, or both; moreover, they need not necessarily be associated with the aerofoils (Fig.5.12). They are used for various purposes on different types of aircraft; to spoil the L/D ratio and so steepen the gliding angle on high-performance sailplanes and other 'clean' aircraft; to check the speed before turning or manoeuvring; to assist both lateral and longitudinal control; to 'kill' the lift and provide a quick pull-up after landing; and on really high-speed aircraft to prevent the speed from reaching some critical value as in a dive. They will be considered later as appropriate to their various functions.

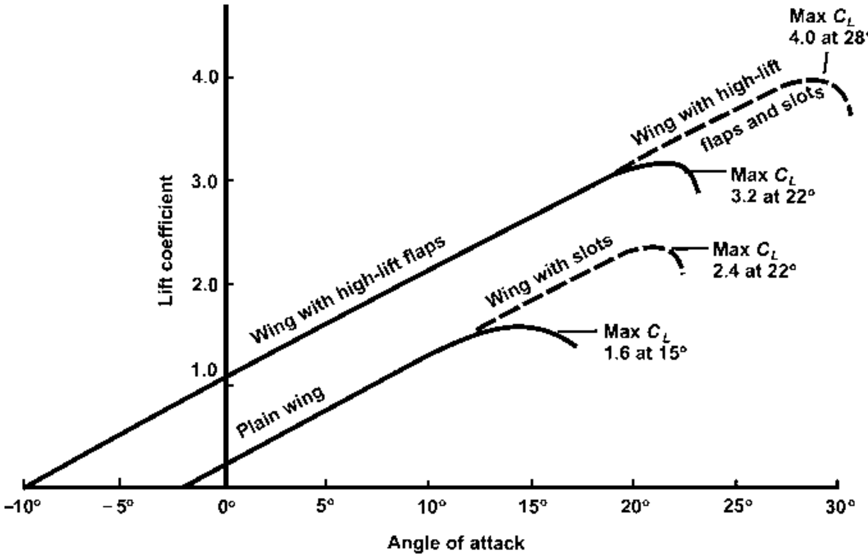

Fig.5.8. Effect of flaps and slots on maximum lift coefficient and stalling angle

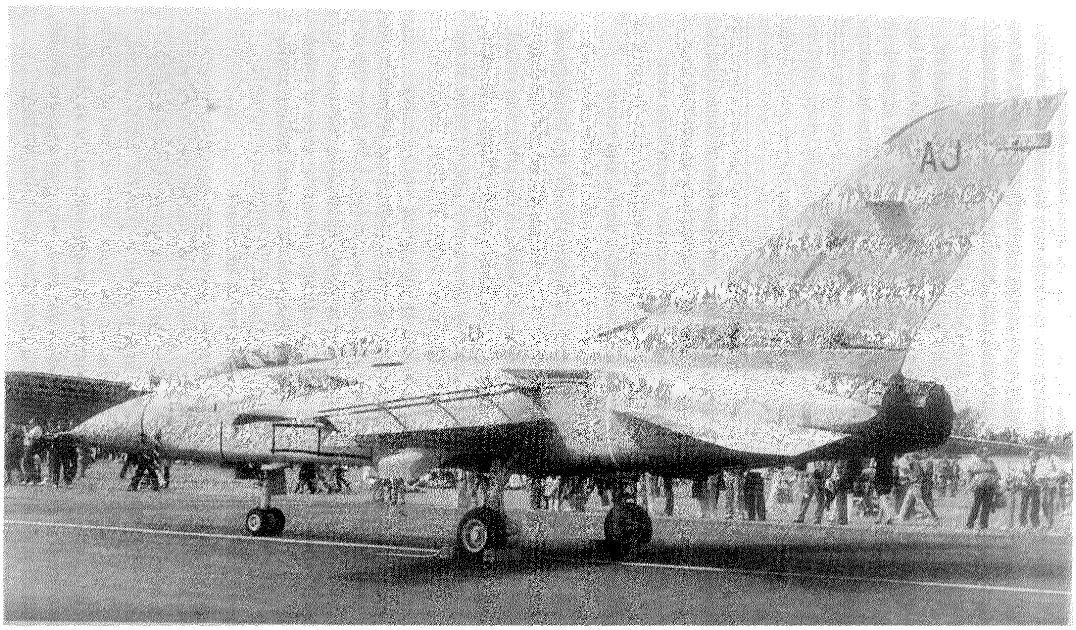

Fig.5.9. Flaps and slats (opposite)
Double-slotted flaps and leading edge slats are used on the Tornado. Because the flaps extend across the entire span, there is no room for ailerons, instead, the slab tailplane surfaces can move differentially as well as collectively, and this 'taileron' serves both for roll and pitch control.

HIGH LIFT DEVICES

High-Lift Devices	Increase of Maximum Lift	Angle of Basic Aerofoil at Max. Lift	Remarks
 BASIC AEROFOIL	–	15°	EFFECTS OF ALL HIGH-LIFT DEVICES DEPEND ON SHAPE OF BASIC AEROFOIL.
 PLAIN OR CAMBER FLAP	50%	12°	INCREASE CAMBER. MUCH DRAG WHEN FULLY LOWERED. NOSE-DOWN PITCHING MOMENT.
 SPLIT FLAP	60%	14°	INCREASE CAMBER. EVEN MORE DRAG THAN PLAIN FLAP. NOSE-DOWN PITCHING MOMENT.
 ZAP FLAP	90%	13°	INCREASE CAMBER AND WING AREA. MUCH DRAG. NOSE-DOWN PITCHING MOMENT.
 SLOTTED FLAP	65%	16°	CONTROL OF BOUNDARY LAYER. INCREASE CAMBER. STALLING DELAYED. NOT SO MUCH DRAG.
 DOUBLE-SLOTTED FLAP	70%	18°	SAME AS SINGLE-SLOTTED FLAP ONLY MORE SO. TREBLE SLOTS SOMETIMES USED.
 FOWLER FLAP	90%	15°	INCREASE CAMBER AND WING AREA. BEST FLAPS FOR LIFT. COMPLICATED MECHANISM. NOSE-DOWN PITCHING MOMENT.

High-Lift Devices	Increase of Maximum Lift	Angle of Basic Aerofoil at Max. Lift	Remarks
 DOUBLE-SLOTTED FOWLER FLAP	100%	20°	SAME AS FOWLER FLAP ONLY MORE SO. TREBLE SLOTS SOMETIMES USED..
 KRUEGER FLAP	50%	25°	NOSE-FLAP HINGING ABOUT LEADING EDGE. REDUCES LIFT AT SMALL DEFLECTIONS. NOSE-UP PITCHING MOMENT.
 SLOTTED WING	40%	20°	CONTROLS BOUNDARY LAYER SLIGHT EXTRA DRAG AT HIGH SPEEDS.
 FIXED SLAT	50%	20°	CONTROLS BOUNDARY LAYER EXTRA DRAG AT HIGH SPEEDS. NOSE-UP PITCHING MOMENT.
 MOVABLE SLAT	60%	22°	CONTROLS BOUNDARY LAYER. INCREASES CAMBER AND AREA. GREATER ANGLES OF ATTACK. NOSE-UP PITCHING MOMENT.
 SLAT AND SLOTTED FLAP	75%	25°	MORE CONTROL OF BOUNDARY LAYER. INCREASED CAMBER AND AREA. PITCHING MOMENT CAN BE NEUTRALIZED.
 SLAT AND DOUBLE-SLOTTED FOWLER FLAP	120%	28°	COMPLICATED MECHANISMS. THE BEST COMBINATION FOR LIFT; TREBLE SLOTS MAY BE USED. PITCHING MOMENT CAN BE NEUTRALIZED.
 BLOWN FLAP	80%	16°	EFFECT DEPENDS VERY MUCH ON DETAILS OF ARRANGEMENT.
 JET FLAP	60%	?	DEPENDS EVEN MORE ON ANGLE AND VELOCITY OF JET.

Fig.5.10. High lift devices

Note: Since the effects of these devices depend upon the shape of the basic aerofoil, and the exact design of the devices themselves, the values given can only be considered as approximations. To simplify the diagram the aerofoils and the flaps have been set at small angles, and not at the angles giving maximum lift.

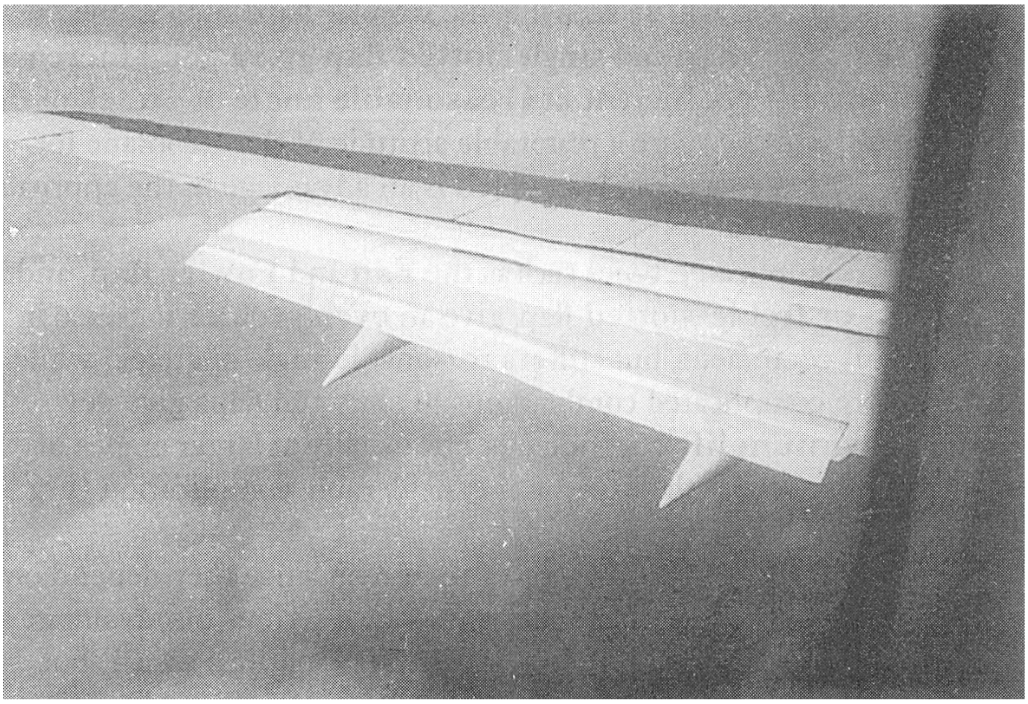

Fig.5.11. Multi-element slotted flaps
Three-element slotted Fowler-type flaps extend rearwards and down as this Boeing 737 prepares to land.

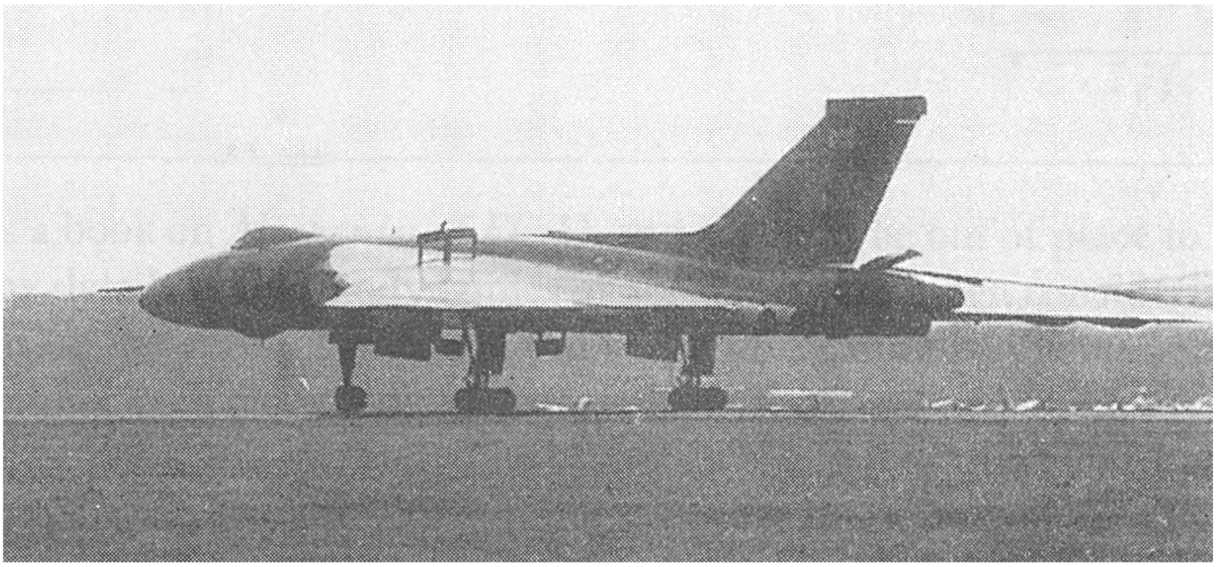

Fig.5.12. Speed brakes
Speed brakes on the wings of the last Vulcan bomber (now sadly retired). The cables of a braking parachute can also just be seen trailing from the rear

(iii) Spoiler

Spoilers, also called "lift dumpers" are control surfaces which are used to reduce or "spoil" the lift on a wing. Spoilers are located on the upper surface of wings and are one of two basic configurations. The more common configuration on jet transports, is to have a flat panel spoiler laying flush with the surface of the wing and hinged at the forward edge. When the spoilers are deployed, the surface rises up and reduces the lift. The other configuration shown in Fig.5.13 is common among sailplanes and has the spoiler located inside the wing structure. When the spoilers are deployed they rise vertically form the wing and spoil the lift.

Flight spoilers are used in flight to reduce the amount of lift that the wing is generating to allow controlled descents without gaining excessiver air speed. Depending on the aircraft design, the spoilers may also be operated by the pilot's control wheel or stick. When the pilot moves the control left or right for a roll movement, the spoilers on the wing toward the center of the turn (upward-moving aileron) move upward and aid in rolling the aircraft into the turn. In some aircraft designs, the spoilers are the primary flight control for roll.

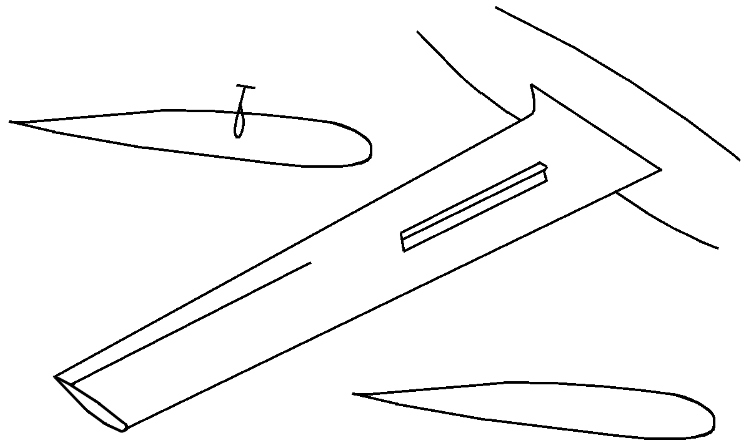

Fig.5.13. Some aircraft, such as sailplanes, have the spoilers arranged so that they rise vertically out of the wing

Ground spoilers are only used when the aircraft is on the ground and are used along with the flight spoilers to greatly reduce the wing's lift upon landing. They also increase the aerodynamic drag of the aircraft's after landing to aid in slowing the aircraft.

Spoilers can be controlled by the pilot through a manual control lever, by an automatic flight control system, or by an automatic system activated upon landing. The typical relative location of flight and ground spoilers is shown in Fig.5.14.

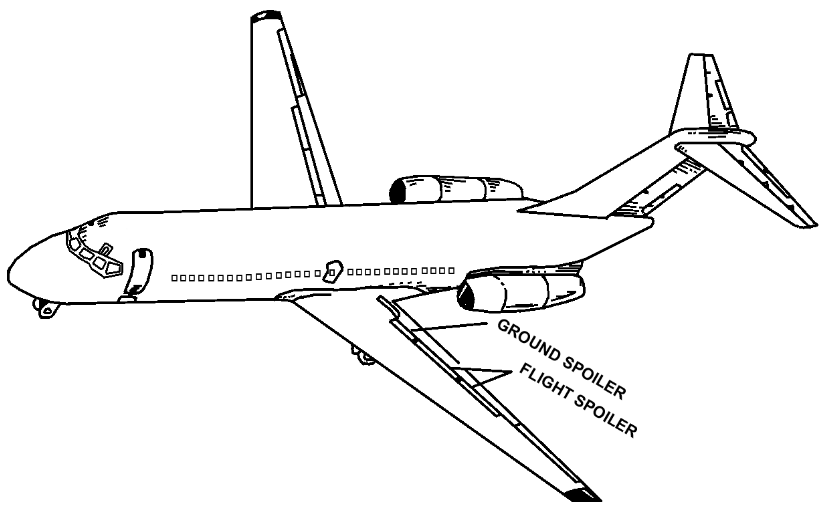

Fig.5.14. Flight spoilers are normally located outboard of ground spoilers

(iv) Speed Brakes

Speed brakes, also called dive brakes, are large drag panels used to aid in control of the speed of an aircraft. They may be located on the fuselage or on the wings. If on the fuselage, a speed brake is located on the top or the bottom of the structure. If speed brakes are deployed as a pair, one is on each side of the fuselage. If located on the wings, speed brakes are deployed symmetrically from the top and the bottom of the wing surface to control the speed of the aircraft as well as to act as spoilers to decrease the lift of the wings. (See Fig.5.15).

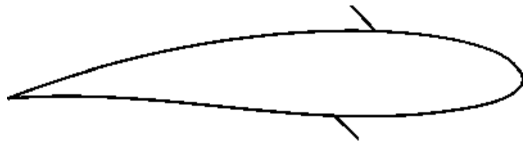

Fig.5.15. Speed brakes on a wing open on the top and bottom of the wing

On some aircraft designs, particularly gliders and sailplanes there may not be any clear distinction between a spoiler and a divebrake because one control surface may serve the purpose of both actions, i.e., to decrease lift and increase drag.

TYPES OF SLOTS

Although there is a large variety of high-lift devices nearly all of them can be classed as slots.

Slots may be subdivided into -

i. Fixed slots
ii. Controlled slots
iii. Automatic slots
iv. Blown slots

We can also classify the effects of both slots and flaps on the characteristics of an aerofoil by saying that their use may cause one or more of the following -

a. Increase of Lift
b. Increase of Drag
c. Change of Stalling Angle
d. Decrease of Lift
e. Change of Trim

If a small auxiliary aerofoil, called a slat, is placed in front of the main aerofoil, with a suitable gap or slot in between the two (Fig.5.16), the maximum lift coefficient of the aerofoil may be increased by as much as 60 per cent (Fig.5.8). Moreover the stalling angle may be increased from 15^0 to 22^0 or more, not always an advantage as we shall discover when we consider the problems of landing. An alternative to the separate slat, simpler but not so effective, is to cut one or more slots in the basic aerofoil itself, forming as it were a slotted wing.

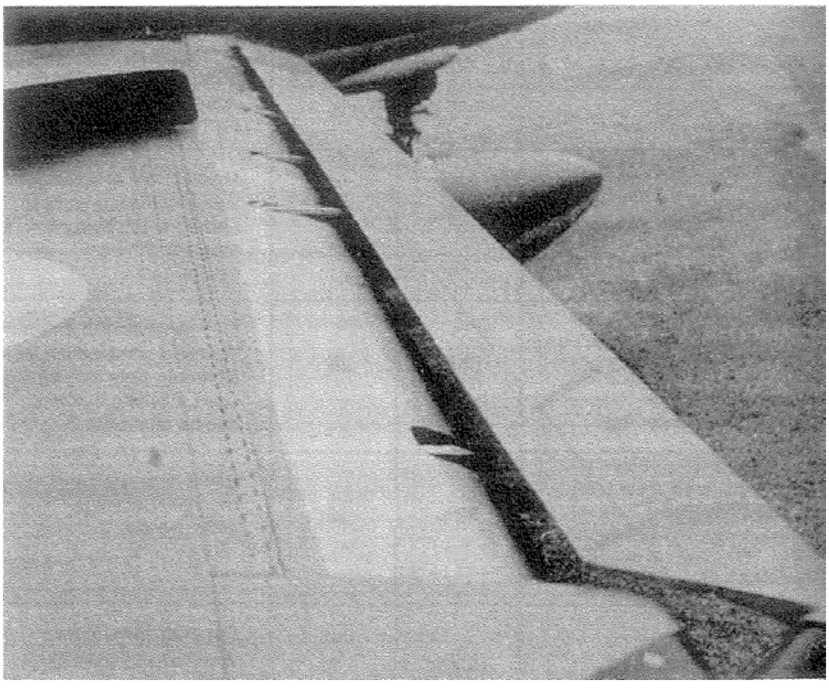

Fig.5.16. Leading edge slat and slot
(By courtesy of Fiat Aviazione, Torino, Italy)

The reason behind these results is clearly shown in Fig.5.17. Stalling is caused by the breakdown of the steady streamline airflow. On a slotted wing the air flows through the gap in such a way as to keep the airflow smooth, following the contour

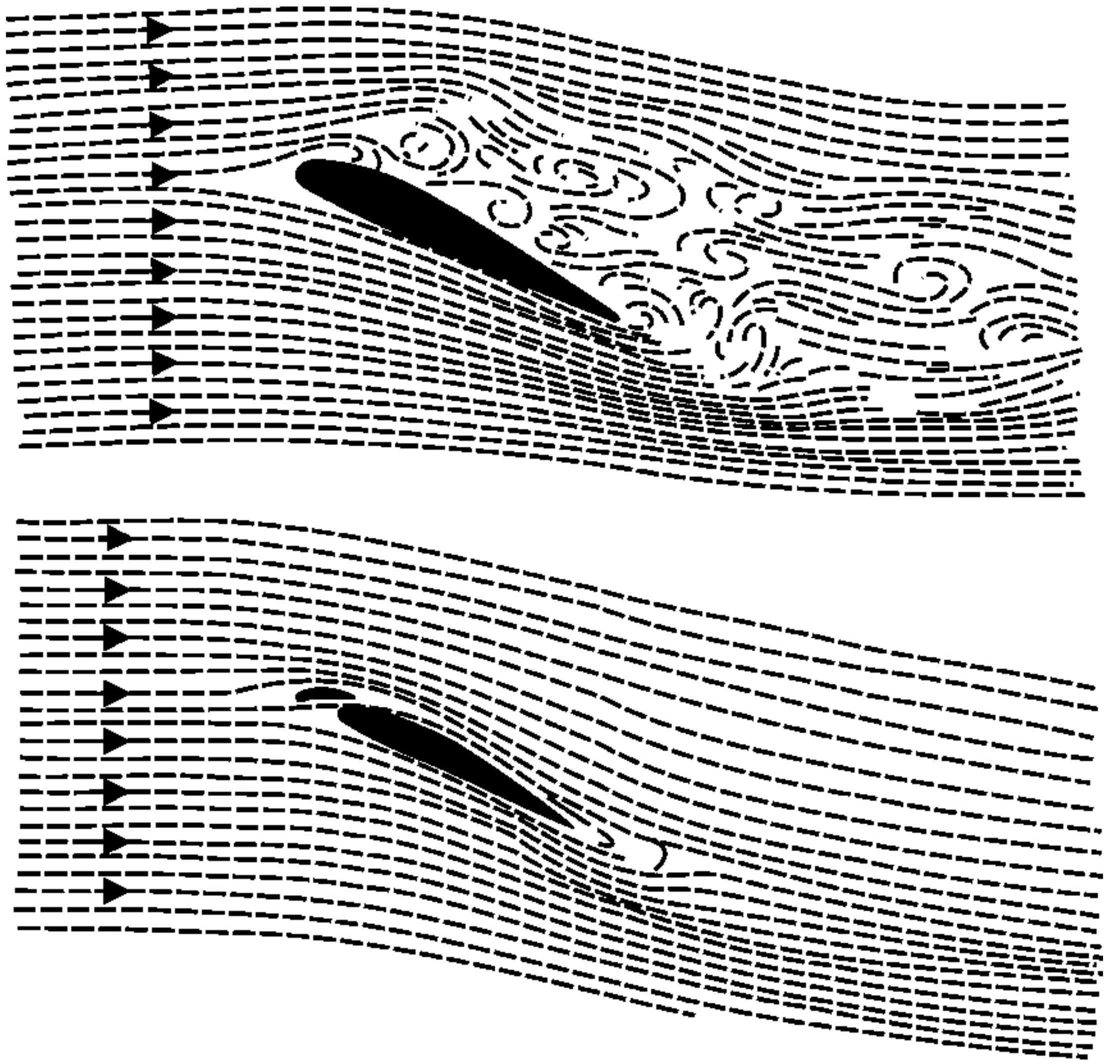

Fig.5.17. Effect of slot on airflow over an aerofoil at large angle of attack

of the surface of the aerofoil, and continuing to provide lift until a much greater angle is reached. Numerous experiments confirm this conclusion. It is, in effect, **a form of boundary layer control** as described earlier.

The extra lift enables us to obtain a lower landing or stalling speed, and this was the original idea. If the slots are permanently open, i.e. **fixed slots**, the extra drag at high speed is a disadvantage, so most slots in commercial use are **controlled slots**, that is to say, the slat is moved backwards and forwards by a control mechanism; and so can be closed for high-speed flight and opened for low speeds. In the early days experiments were made which revealed that, if left to itself, the slat would move forward of its own accord. So **automatic slots** came into their own; in these the slat is moved by the action of air pressure, i.e. by making use of that forward and upward suction near the leading edge. Fig.5.18

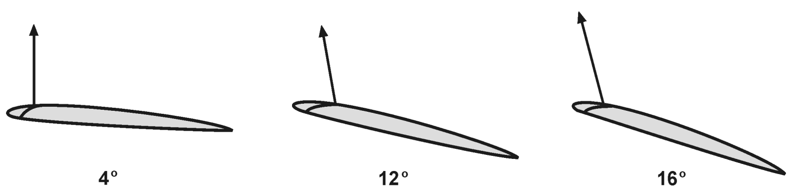

Fig.5.18. Direction of force on slat at varying angles of attack

shows how the force on the slat inclines forward as the stalling angle is reached. The opening of the slot may be delayed or hastened by 'vents' at the trailing or leading edge of the slat respectively (Fig.5.19), and there may be some kind of spring or tensioning device to prevent juddering, which may be otherwise likely to occur.

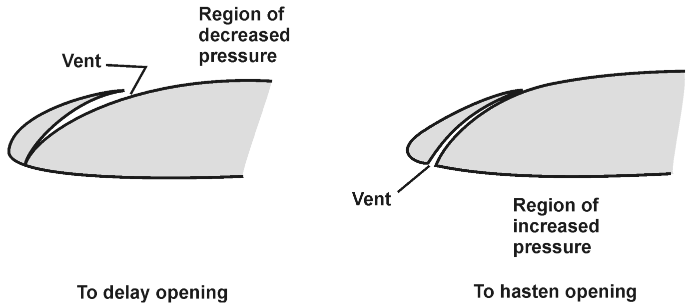

Fig.5.19. Effect of vents on opening of automatic slots

Before leaving the subject of slots - for the time being, at any rate - there are a couple of interesting points which may be worth mentioning. Firstly, the value of the slot in maintaining a smooth airflow over the top surface of the wing can be materially enhanced by blowing air through the gap between slat and wing; this may be called a **blown slot.** Secondly, what might be called the 'slot idea' may be extended to other parts of the aircraft. Specially shaped cowlings can be used to smooth the airflow over an engine, and fillets may be used at exposed joints, and other awkward places, to prevent the airflow from becoming turbulent.

TYPES OF FLAPS

A wing flap is defined as hinged, pivoted, or sliding airfoil, usually attached near the trailing edge of the wing. The purpose of wing flaps is to change the camber of the wing and in some cases to increase the area of the wing, thus permitting the aircraft to operate at lower flight speeds for landing and takeoff. The flaps effectively increase the lift of the wings and, in some cases, greatly increase the drag, particularly when fully extended. Various configurations for wing flaps are shown in Fig.5.20.

I Leadingedge flaps

While flaps are generally located on the trailing edge of a wing, they can also be placed on the leading edge. Leading-edge flaps are normally used only on large transport-category aircraft that need large amounts of additional lift for landing. A leading-edge flap is a high-lift device which reduces the severity of the pressure peak above the wing at high angles of attack. This enables the wing to operate at higher angles of attack than would be possible without the flap.

a. Krueger flap

Another method for providing a leading-edge flap is to design an extendable surface known as the Krueger flap that ordinarily fits smoothly into the lower part of the leading edge. When the flap is required, the surface extends forward and downward, as shown in the second drawing of Figure.5.21.

b. Droop snoot

One method for providing a wing flap is to design the wing with a leading edge that can be drooped, as shown in the top drawing of Figure.5.21.

II Trailing edge flaps

The trailing edge flap is simply a small auxiliary aerofoil, located near the rear of a main aerofoil, and which can be deflected about a given line, where it is hinged. This deflection causes a change in the geometry of the aerofoil, and hence in its aerodynamic characteristics. In the case of a flap designed as a high-lift device, usually only downward deflection is possible, though the amount of deflection is variable. In the case of a flap designed as a control surface, deflection in both senses is possible, though the range of deflection is usually much less. The main types of trailing edge flap are described in principle in the following paragraphs.

a. Camber flap or Plain flap

The camber flap, in effect, acts as if the trailing edge of the wing were deflected downward to change the camber of the wing. Thus increasing both lift and drag. If the flap is moved downward sufficiently, it becomes an effective air brake. The plain flap may be hinged to the wing at the lower side, or it may have the hinge line midway between the lower and upper surfaces.

b. Split flap

The split flap, when retracted, forms the lower surface of the wing trailing edge. When extended, the flap moves downward and provides an effect similar to that of the pain flap. Plain flaps and split flaps may be attached to wing with three or more separate hinges, or they may be attached at the lower surface with a continuous piano hinge.

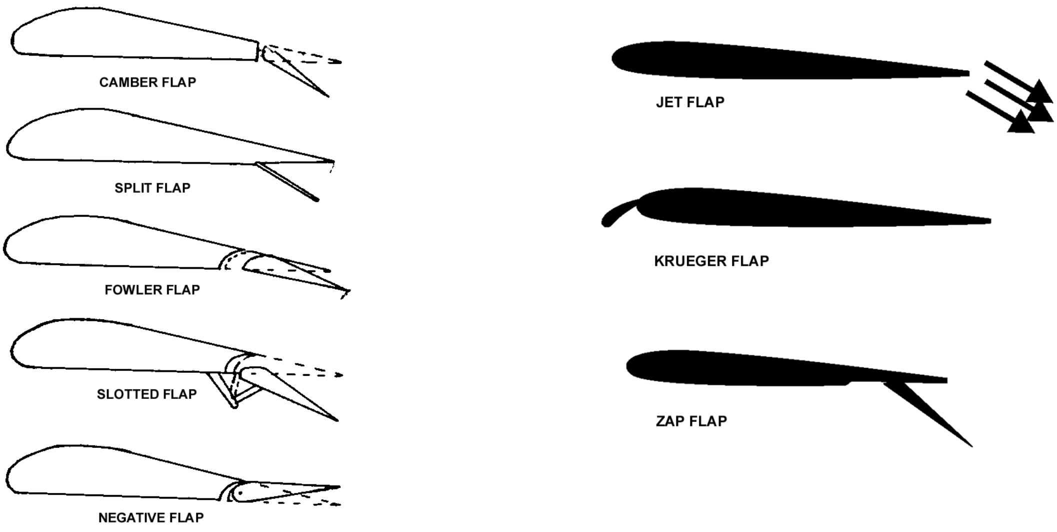

Fig.5.20. Configuration for wing flaps

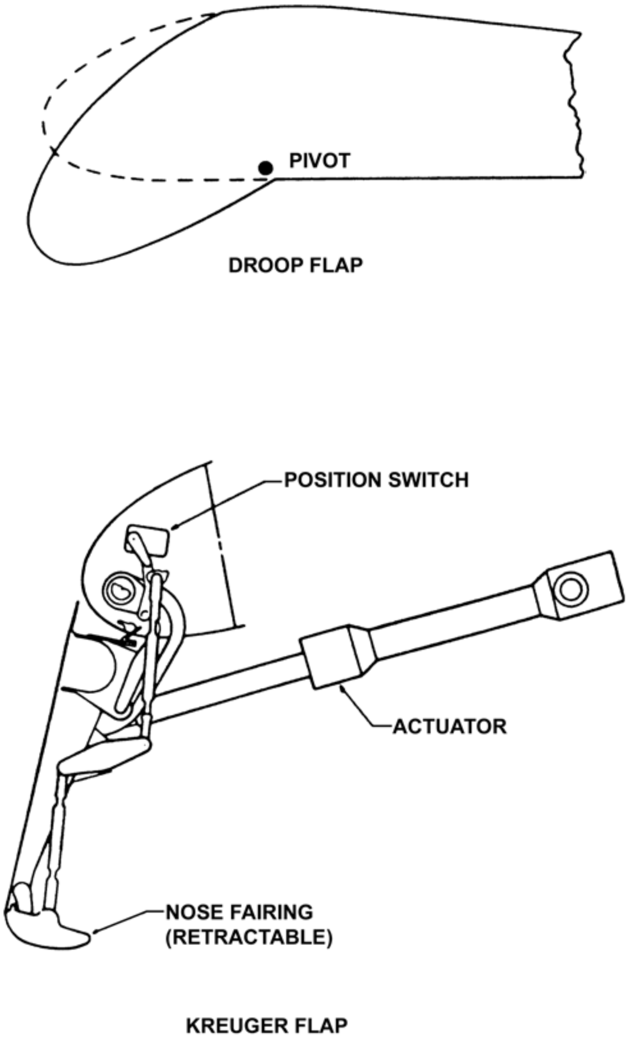

Fig.5.21. Types of leading-edge flaps

c. Fowler flap

The fowler flap and others with similar operation are designed to increase substantially the wing area as the flap is extended, the flap forms the trailing edge of the wing. As this type of flap is extended, it is moved rearward, often by means of a worm gear, and is supported in the correct position by means of curved tracks. The effect of the Flower flap, when extended is to greatly reduce the stalling speed of the aircraft by the increase in wing area and change in wing chamber.

d. Zap flap

Zap flap is a combination both Split flap and Fowler flap.

e. Slotted flap

A slotted flap is similar to a plain flap except that as the flap is extended, a gap develops between the wing and the flap. The leading edge of the flap is designed so that air entering this gap flows smoothly through the gap and aids in holding the airflow on the surface. This increases the lift of the wing with the flap extended.

f. Jet flap

The jet flap consists of a very high speed jet of air blown out through a narrow slit in the trailing edge of the wing. The jet, deflected slightly downwards, divides the upper surface flow from the lower surface flow, and produces an effect on the flow over the wing just like that which would be produced by a very large physical trailing edge flap. There is an additional increment due to the downward component of the momentum of the jet. Experiments with such a device have produced very high lift coefficients.

Some aircraft designs incorporate combinations of the fowler and slotted flaps to greatly increase the lift and drag of the wing. When the flap is initially extended, it moves aft on its track. Once past a certain point on the track, further aft movement is accompanied by a downward deflection which opens up the slot between the flap and the wing. Many jet transport aircraft use this basic design with several slot openings being used to improve the airflow over the wing and flap surfaces. Fig.5.22 illustrates this type of flap combination.

A few aircraft, particularly sailplanes, incorporate a negative flap capability into the flap control design so that the flap can be raised above its neutral position. This changes the airfoil shape and allows the aircraft to fly at a higher speed with reduced drag.

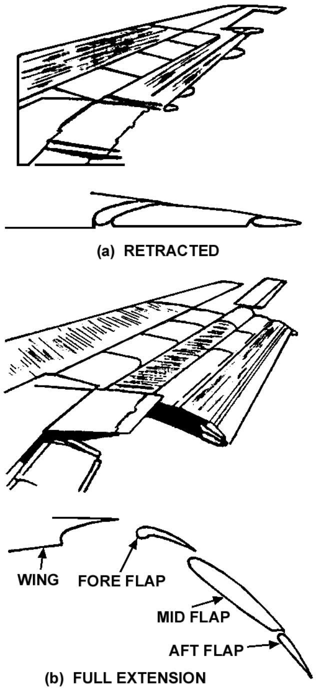

Fig.5.22. The retracted and extended position of the flap segment in a typical flap system

BOUNDARY LAYER SUCTION

While blowing energizes the boundary layer in order to prevent separation, the principle of suction is the removal of the slowly moving air in the boundary layer, so that there in no layer to separate. Small holes, flush with the surface, are made in the surface of the aerofoil upstream of the separation point, and the air in the boundary layer is sucked into the wing through these holes. However, from this point onwards the boundary layer will re-form and thicken, and separation may still occur at some point downstream. To prevent this a series of suction holes must be made at various chord wise positions, as indicated in Fig.5.23. The logical extension of this idea is the use of a porous wing surface, with suction applied every- where on the surface.

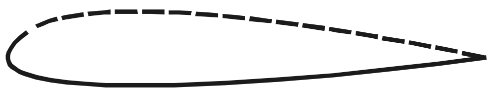

Fig.5.23.

In addition to preventing separation, suction may also be used to prevent transition, and hence to keep drag low. Such a device would appear to be of particular interest in conjunction with the use of low drag wing sections. The principle behind the design of a low drag section is the maintenance of laminar flow. The disadvantage inherent in such designs is that separation occurs very readily when the incidence is increased by even a fairly small amount above the design value. Suction could be helpful both in maintaining laminar flow and in preventing separation.

BOUNDARY LAYER BLOWING

The principle of boundary layer blowing is similar to that of the leading edge slot. Highspeed air is blown into the boundary layer through a narrow slit in the upper surface of the aerofoil, where it re-energizes the boundary layer and prevents separation. Since the velocity of the air fed in this way is so much higher than the speed of the air passing through a leading edge slot, or a slotted flap, blowing will generally prove much more effective. The stall can be delayed almost indefinitely by this means. In addition, The jet of air has the effect almost indefinitely by this means. In addition, the jet of air had the effect of increasing the circulation round the wing, thus giving a direct lift increment at all incidence

The slot may be near the nose of the aerofoil, so that the blowing affect the whole of the upper surface, as in Fig.5.24. Alternatively, the slot maybe situated just upstream of the nose of a plain flap. In this position, the upstream of the slot will be affected to some extent by induction, but the main object is to prevent separation of the flow over the upper surface of the flap. Thus device is known as the blown flap, and is illustrated in Fig.5.25.

Fig.5.24.

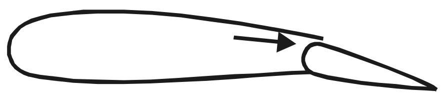

Fig.5.25.

The circulation effect is still present, though less important. There is some advantage in this device compared with that of slot placed further forward in that, in the latter case, the effect of blowing may be lessened by the time the flow reaches the rear of the aerofoil, where separation is most likely.

VORTEX GENERATORS

Many devices are used by the designer to control the separation or breakaway of the airflow from the surface of the wing - all these devices, in one way or another, over one part of the wing or another, have this in common, that they are intended to prevent or delay this breakaway. How? Well, that depends to some extent on the device, and we will consider vortex generators first (Fig.5.26).

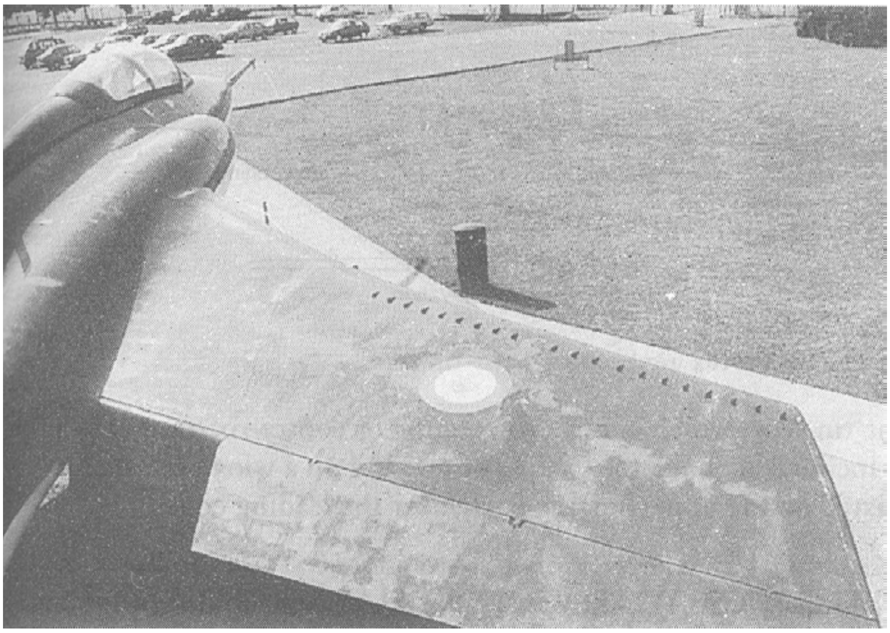

Fig.5.26. Vortex generators

The fundamental reason for the breakaway is that the boundary layer becomes sluggish over the rear part of the wing section, flowing as it is against the pressure gradient. The formation of a shock wave makes matters worse; the speed in the boundary layer is still subsonic which means that pressure can be transmitted up stream, causing the boundary layer to thicken and, if the pressure rise is too steep, to break away from the surface. Now vortex generators are small plates or wedges, projecting an inch or so from the top surface of the wing, i.e. three or four times the thickness of the boundary layer. Their purpose is to put new life into a sluggish boundary layer; this they do by shedding small lively vortices which act as scavengers, making the boundary layer turbulent and causing it to mix with and acquire extra energy from the surrounding faster air, thus helping it to go farther along the surface before being slowed up and separating from the surface. In this way the small drag which they create is far more than compensated by the considerable boundary layer drag which they save, and in fact they may also weaken the shock waves and so reduce shock drag also; and the vorticity which they generate can actually serve to prevent buffeting of the aircraft as a whole - a clever idea indeed,

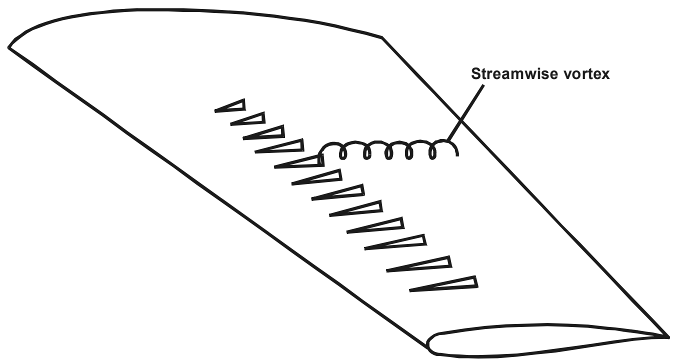

Fig.5.27. Vortex generators: bent-tin type.

and so simple. The net effect is very much the same as blowing or sucking the boundary layer, but the device is so much lighter in weight and simpler. The greater the value of the thickness/chord ratio the more necessary does some such device become.

There are various types of vortex generator; Fig.5.27 illustrates the bent-tin type, which may be co-rotating or contra-rotating. The plates are inclined at about 15^{o} to the airflow, and on a wing are usually situated on the upper surface fairly near the leading edge.

■■■

CHAPTER: 6
DRAG

INTRODUCTION

Whenever a body is moved though air, or other viscous fluid, there is produced a definite resistance to its motion. In aeronautical work this resistance is usually referred to as drag.

And efforts must be made to reduce the resistance of every part of an aeroplane to a minimum, provided strength and other essential factors can be maintained. For this reason many thousands of experiments have been carried out to investigate the problems of air resistance; in fact, in this, as in almost every branch of this subject, our knowledge is founded mainly on the mean results of accumulated experimental data.

Experiments on air resistance have performed in two ways-
1. By study of the nature of the flow of air and other fluids past solid bodies.
2. By measuring the actual forces produced on the bodies by the passage of the fluid.

The data produced by these two methods show that there is a direct connection between the two, and generally speaking, it can be said that the greater the formation of eddies, or turbulence the greater is the resistance too the passage of aircraft through air.

In experimental work it is usual to allow the fluid to flow past the body rather than to move the body through the fluid. The former method has the great advantage that the body is at rest, and consequently the measurement of any forces upon it is comparatively simple. Furthermore, since we are only concerned with the relative motion of the body and the fluid, the true facts of the case are fully reproduced provided we can obtain a flow of the fluid which would be as steady as the corresponding motion of the body through the fluid.

TYPES OF DRAG

Lines which show the direction of the flow of the fluid at any particular moment are called streamlines, and a body so shaped as to produce the least possible eddy or vortex motion is said to be of streamline shape.

By introducing smoke into the airflow in wing tunnels, and coloured jets into water tank experiments. The results of these experiments show that we may divide the resistance of a body passing through a fluid into two parts i.e. Form Drag and skin friction.

These two between then form a large part of the total drag of an aeroplane in the high subsonic range, the major part. The sum of the two is sometimes called profile Drag but this term will be avoided since it is apt to give an impression of being another name for Form drag, whereas it really includes skin friction.

The total drag of an aeroplane is sometimes divided in another way in which the drag of the wings or lifting surfaces, wing drag, is separated from the drag of those parts which do not contribute towards the lift, the drag of the latter being called parasite drag.

i Profile drag

This is the portion of the resistance which is due to the fact that when a viscous fluid flows past a solid object, vortices are formed, and we no longer get a smooth streamline flow. The extreme example of this type of resistance is a flat plate placed at right angles to the wind. The resistance is very large and is almost entirely due to the formation of vortices, the skin fiction being negligible in comparison.

Experiments show that not only is the pressure in front of the plate greater than the atmospheric pressure, but that the pressure behind is less than that of the atmosphere, causing a kind of "sucking" effect on the plate (Fig.6.1).

This is 'the drag arising from the resolved components or the tangential stresses on the surface of the body.

At all points on the surface of a body past which a viscous fluid is flowing there is a traction along the surface in the direction of flow. This traction is due directly to viscosity. The traction at each point has a component acting in the direction of the undisturbed stream (or in the direction opposite to the direction of the opposite to the direction of flight). The total effect of these components, taken over the whole of the exposed surface of the body, is the surface friction drag (skin-friction drag).

Surface friction drag cannot exist in an inviscid fluid.

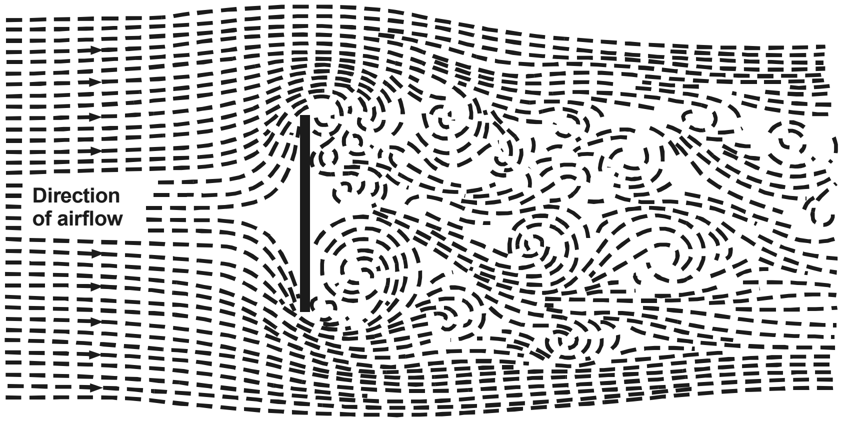

Fig.6.1.

ii Induce Drag

The trailing vortices which are shed from near the tips of a finite wing contain energy associated with the rotational velocities. This energy is abstracted from the airflow, so that some power must equal the rate of flow of energy associated with the trailing vortices. This is equivalent to saying that there is now a further drag force on the wing, to be added to its profile drag. This is known as induced drag, and it forms a very important part of the total drag of a finite wing. Boundary Layer Drag (Profile Drag)- the boundary layer drag is the sum of the surface friction drag and the boundary layer normal-pressure drag (form drag).

iii Interference Drag

In general, with any aircraft configuration, there is, in addition to the drag of the drag of the various components, an extra amount of drag due to interference between these components. Principally between wing and the fuselage. This is especially important at high speeds. Experiment shows that a large part of the transonic drag rise for a complete aircraft is due to interference. It also shows that interference drag at transonic speeds may be minimized by ensuring that the distribution along the length of the aircraft axis of its total cross-sectional area follows a certain smooth pattern.

With some early high-speed aircraft designs this was not the case. The area increase rapidly in the region of the wing, and again in the vicinity of the tail, and decrease elsewhere, giving an area distribution something like the one sketched in Fig.6.2(a). On later aircraft, the fuselage was wasted, i.e., the area reduced in the region of the wing and again near the tail, so that there was no-hump in the area distribution, giving a distribution like that of Fig.6.2(b). There is an optimum area distribution, and the minimization of transonic interference drag requires that the aircraft should be designed to fit this distribution as closely as possible. This requirement is known as the transonic area rule. In practice, of course, no aircraft has this optimum distribution, but any reasonably smooth area distribution helps to keep down the transonic drag rise.

iv Wave Drag

The sudden extra drag which is such a marked feature of the shock stall has two main components. First the energy dissipated in the shock wave itself is reflected in additional drag (wave drag) on the aerofoil. Secondly, the shock wave may be accompanied by separation, or at any rate a thickening of and increase in turbulence level in the boundary layer. Either of these will modify both the pressure on the surface and the skin friction behind the shock wave.

So this shock drag may be considered as being made up of two parts, i.e. the wave- making resistance, or wave drag, and the drag caused by the thick turbulent layer or region of separation which we will call boundary layer drag.

As the shock wave and the thickened turbulent boundary layer or separation are like the chicken and the egg-we do not know which comes first; what we do know is that when one comes so does the other. That is not to say tat they are by any means the same thing, or that they have the same effects, or that a device which reduces one will necessarily reduce the other.

ANOTHER WAY OF CLASSIFICATION OF DRAG

i. Parasite drag

The other components of the aircraft, e.g., the fuselage, engine nacelles, tail unit, etc., contribute an additional amount of drag. Since these components are 'non-lifting', i.e., they do not contribute significantly to the useful lift of the aircraft, this drag is called "parasite drag". The parasite drag coefficient is often assumed to be constant.

ii. Wing drag or Active drag

The total drag of a wing is the sum of profile drag and induced drag. If CD represents the total drag coefficient, then

$$C_D = C_{Do} + C_{Di}$$

$$\text{Thus } C_D = C_{Do} + \frac{kC_L^2}{\pi A}$$

We have seen that it is often valid to assume that C_{Do} is constant, at least over a range of low incidence. If this assumption is made, then the relation between C_D and C^2_L is linear. If a graph is drawn in which C_D is plotted against C_L^2, curve like the one in Fig.6.2(c) is obtained. At high lift coefficients, the relation is no longer linear, because the profile drag increases rapidly as the stall is approached.

The slope of the linear portion of the curve is K/πA, and the intercept on the vertical axis is C_{Do}. Thus from a plot of C_D against

C_L^2 the induced drag factor and the profile drag coefficient is not constant, but is more correctly represented by

$$C_{Do}=C_{Dz}+b'C_L2, \qquad C_D = C_{DZ} + b'C_L^2 + (K/\pi A)\, C_L^2 \quad - (1)$$

$$C_D = a + b\, C_L^2 \qquad (2)$$

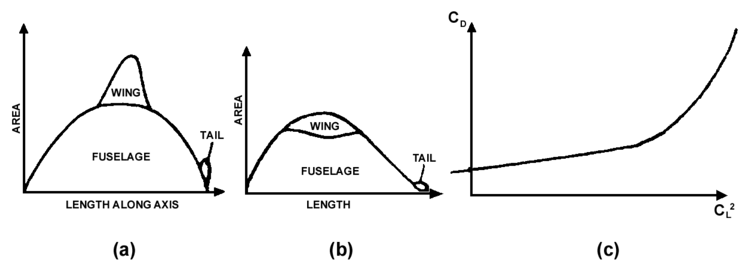

Fig.6.2.

where a and b are constants. The relation between C_D and CL^2 is still linear, but the slope of the line will now be greater than k/πA- in fact, it will be b' + k/πA, However, b' is usually small in comparison with k/πA, and the induced drag factor is generally estimated, with reasonable accuracy, from this curve.

If a complete aircraft is considered, the drag relation for the wing, as given in equation 2 may change again. The other components of the aircraft, e.g., the fuselage, engine nacelles, tail unit, etc., contribute an additional amount of drag. Since these components are 'non-lifting', i.e., they do not contribute significantly to the useful lift of the aircraft, this drag is called parasite drag. The parasite drag coefficient is often assumed to be constant. Term in equation. In fact, however, it also increase with the square of the lift coefficient, so that there is also some increase in the second term. The relationship remains of the form

$$C_D = a + bC_L^2,$$

but the following points should be noted:

(a) The constant 'a' does not now consist only of the profile drag coefficient of the wing, but will be much increased. In fact, the parasite drag may be a good deal more than the profile drag. However, 'a' may still be referred to as the drag coefficient at zero lift.

(b) The $C_D \sim C_L^2$ curve will still be linear over a considerable range. The slope, still given by b, will now generally be appreciably larger than that due to the induced drag of the wing alone. However, this slope may still be used, as before to determine what is often called the induced drag factor of the complete aircraft. Typical values for this will be much greater than those quoted above for a conventional wing. Values of 2 or 3 are often obtained.

COEFFICIENT OF DRAG VS ANGLE OF ATTACK

Now for the drag coefficient curve (Fig.6.3). Here we find much what we might expect. The drag is least at about 0°, or even a small negative angle, and increases on both sides of this angle; up to about 6°, however, the increase in drag is not very rapid, then it gradually becomes more and more rapid, especially after the stalling angle when the airflow separates.

PROFILE DRAG DETAILS

i Form drag reduction by stream lining

Form drag is kept to a minimum by preventing boundary-layer separation and in this respect has already been discussed in the previous sections. Streamlining is vitally important for reducing form drag. It is worth noting that at high Reynolds numbers a circular cylinder has roughly the same overall drag as a streamlined aerofoil with a chord length equal to 100 cylinder radii. Form drag is overwhelmingly the main contribution to the overall drag for bluff bodies like the cylinder, whereas the predominant contribution in the case of the streamlined body is skin-friction drag, form drag being less than ten per cent of the overall drag. For bluff bodies even minimal streamlining can be very effective.

ii Skin friction drag reduction by smoothening the surface

In broad terms skin-friction drag can be reduced in one of two ways. Either laminar flow is maintained by postponing transition, this is the so-called laminar-flow technology, or ways are found to reduce the surface shear stress generated by the turbulent boundary layer.

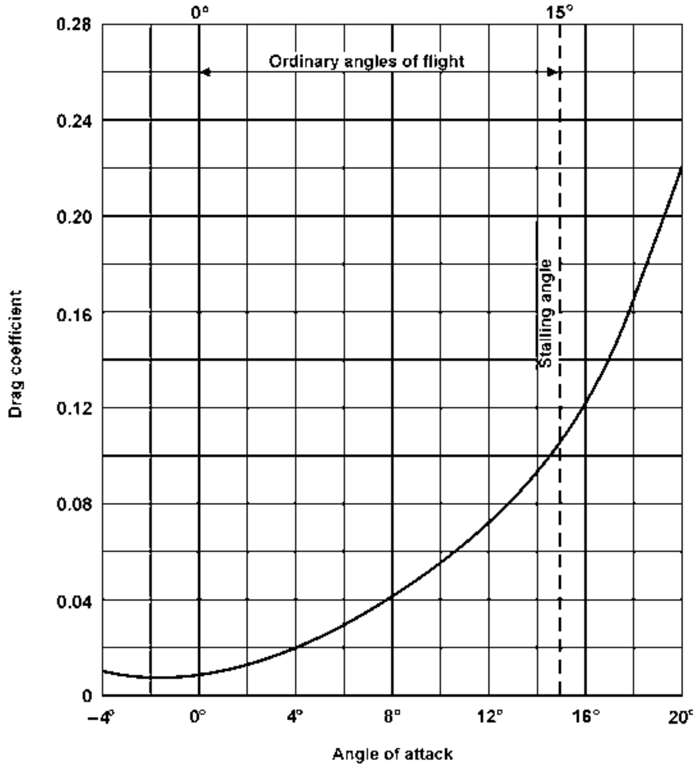

Fig.6.3. Drag curve

The laminar-flow technology has already been discussed briefly in Section 6.3. Laminar flow can be maintained passively by prolonging the favourable or constant-pressure region over the wing surface. Active control of transition requires the use of suction, either distributed or through discrete spanwise slots (in a similar way as discussed in the previous section for the prevention of separation). Often the suction is used in conjunction with favourable pressure distributions. The basic principles of maintaining laminar flow by means of suction have been known for at least the past thirty-five to forty years. However, problems with practical implementation on aircraft, either real or perceptual, have prevented the wide-spread use of the technology. It seems increasingly likely, however, that the considerable gains in efficiency which would result from the use of laminar-flow technology, will ensure that it will be much more widely exploited on commercial aircraft in the near future. Other methods for maintaining laminar flow have been developed but, as yet, have not been seriously considered for practical application in aviation.

INDUCED DRAG (VORTEX DRAG) DETAILS

i. To reduce induced drag: Saw tooth or dog tooth leading edge

There it was shown that, in accordance with the classic wing theory, induced drag falls as the aspect ration of the wing is increased. It was also shown that for a given aspect ratio elliptic-shaped wings have the lowest induced drag. Over the past fifteen years the winglet has been developed as a device for reducing induced drag without increasing aspect ratio. A typical example is illustrated in Fig. Winglets of this type have now been fitted to many types of commercial aircraft.

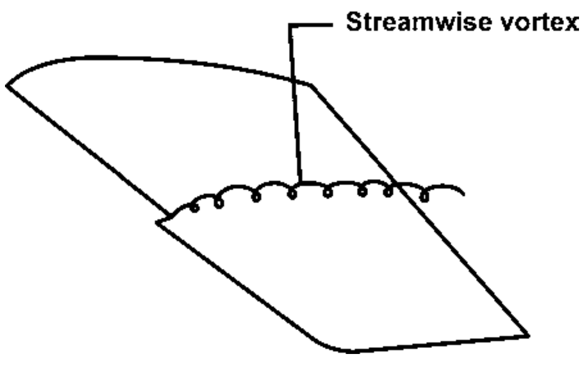

Fig.6.4 Saw tooth

The physical principle behind the winglet is illustrated in Fig.8.19b and 8.19c. On all subsonic wings there is a tendency for a secondary flow to develop from the high-pressure region below the wing round the wing tip to the relatively low-pressure region on the upper surface (Fig.8.19b). This is part of the process of forming the trailing vortices. If a winglet of the appropriate design and orientation is fitted to the wing-tip, the secondary flow causes the winglet to be at an effective angle of incidence, giving rise to lift and drag components L_W and D_W have components in the direction of the drag of the aircraft as a whole. L_W provides a component counter to the aircraft drag, while D_W provides a component which augments the aircraft drag. For a well-designed winglet the contribution of L_W predominates, resulting in a net reduction in overall drag, or a thrust, equal to ΔT (Fig.8.19c).

ii. Wing let and Wing fence

Ideally, air would always flow chordwise over a wing; however, as has been discussed, air will tend to flow spanwise toward the tip. Spanwise flow is particularly a problem on swept wings. This spanwise flow of air may be partially controlled by the use of a wing (flow) fence, such as is illustrated in Figure _. A wing fence is a stationary vane, projecting from the upper surface of an airfoil, which is used to prevent the spanwise flow of air. Flow fences are often located in alignment with trailing-edge control surfaces, such as ailerons, to improve the chordwise flow and thereby the effectiveness of the control surfaces.

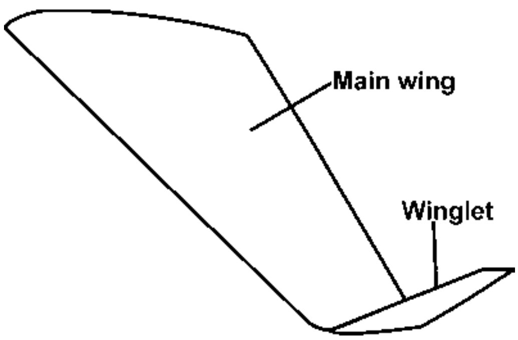

Fig.6.5. Wing let

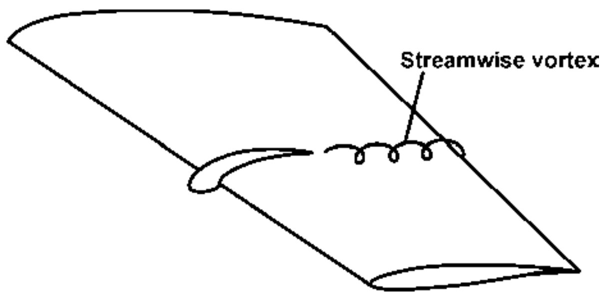

Fig.6.6. Wing fence

iii Wing tip fence

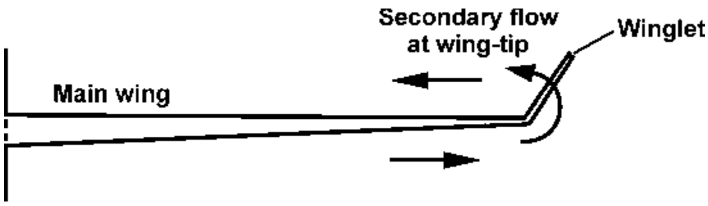

Fig.6.7. Wing tip fence

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CHAPTER: 7
AIRCRAFT PERFORMANCE

I POWER AVAILABLE AND REQUIRED CURVES
The Thrust Required for steady (unaccelerated) straight and level flight is of course equal to the drag (i.e. T = D) and so the Thrust Required curve is identical to the familiar Drag Curve (shown opposite).

Points to be noted from the Thrust Required or Drag Curve (Fig.7.1) are:

* High thrust is required at high speeds and low angles of attack to overcome what is mainly parasite drag;
* Minimum thrust is required at the minimum drag speed (which is also the best L/D ratio speed, since L= W in straight and level flight and D is at its minimum value);
* High thrust is required at low speeds and high angles of attack to overcome what is mainly induced drag (caused in the production of Lift).

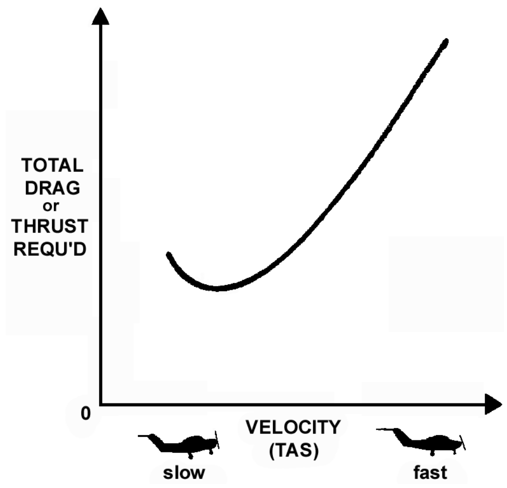


Fig.7.1. The thrust required curve (or the drag curve)

The engine-propeller combination is a power-producer (rather than a thrust-producer like a jet engine). The fuel flow (in litres/hr or gallons/hr) of an engine/propeller combination is a function of power produced (rather than thrust).

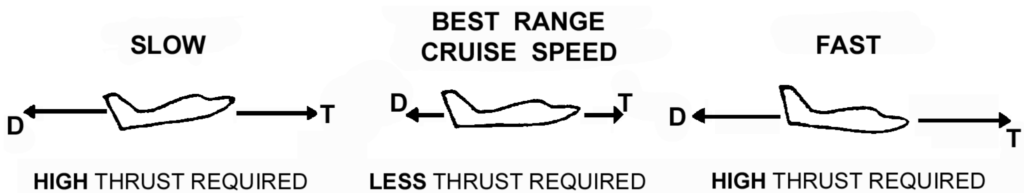


Fig.7.2. Both low speed and high speed require high thrust

Power is defined as the rate of doing work, i.e. the speed at which an applied force moves a body. Therefore the power required for flight depends on the product of :

* thrust required; and
* flight velocity (True Air Speed).

We can develop a Power Required Curve from the Thrust Required curve (shown previously) by multiplying:
the thrust required at a point on the curve x the TAS at that point, to give us the Power Required to maintain level flight at that speed. The graph will appear as shown.

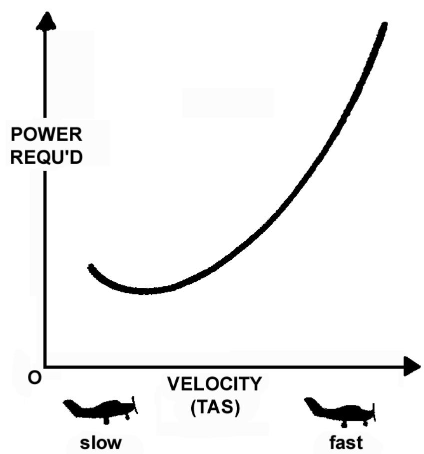


Fig.7.3. The power required curve

These graphs are easy to understand if you take it slowly. If you want to fly at a particular velocity (TAS) then, by reading up from that TAS on the airspeed axis, the Power Curve will tell you the power that the engine/propeller must deliver. This power will supply sufficient thrust to balance the drag and maintain speed straight and level.

In straight and level flight you would set the attitude for the desired airspeed (different airspeeds require different angles of attack) and adjust the power to maintain this speed. There is no need, of course, for the Pilot to actually refer to the graphs.

Maximum Level Flight Speed
Maximum level flight speed for the aeroplane occurs when the power available from the engine/propeller matches the

power required to produce enough thrust to balance the drag at the high speed. At higher speeds, there is insufficient power available.

Minimum Level Flight Speed
At low speeds (slower than the minimum power speed), higher power from the engine/propeller is required to provide thrust to balance the higher drag (mainly induced drag).

The minimum level flight speed is usually not determined by the power capabilities of the power plant, but rather by the aerodynamic capabilities of the aeroplane. As airspeed reduces, the stalling angle is reached, or some condition of instability or control difficulty usually occurs, prior to any power limitation of the power plant.

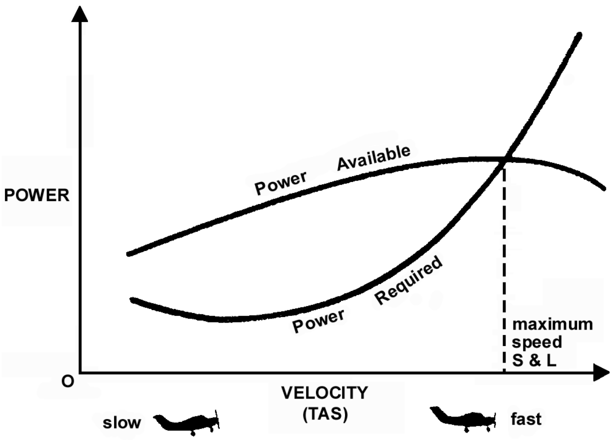


Fig.7.4. Maximum level flight speed

II STEADY LEVEL FLIGHT
In steady straight and level flight the aeroplane is in equilibrium. This means that all the forces acting on it are in balance and there is no resultant force to accelerate or decelerate it. Acceleration is a change in velocity, which means a change in speed or a change in direction, or both. In straight and level flight, the aeroplane is not forced to change either speed or direction.

The four main forces acting on the aeroplane are Lift, Weight, Thrust and Drag. We assume that Thrust acts in the direction of flight, as indicated in Fig.7.5.

Each of the four main forces has its own point of action:
the Lift through the Centre of Pressure;
the Weight through the Centre of Gravity;
the Thrust and the Drag in opposite senses, parallel to the direction of flight, through points that vary with aircraft attitude and design.

We make the assumption that the Thrust force from the engine/propeller is acting in the direction of flight, even though this is not a always the case. For instance, at a high angle of attack and slow speed the aircraft has a nose-high attitude with the propeller shaft inclined upwards to the horizontal direction of flight. This assumption that Thrust acts in the direction of flight simplifies our discussion considerably.

In straight and level flight.

LIFT = WEIGHT and THRUST = DRAG.

The Lift/Weight forces are much larger than the Thrust/Drag forces.

For in-depth study and revision of each of these forces you should refer back to their individual chapters.

Pitching Moments
The CP and the CG vary in position-the CP changing with angle of attack and the CG with fuel burn-off and passenger or cargo movement. The result is that the Lift: Weight combination sets up a couple which will cause a nose-down or a nose-up pitching moment, depending upon whether the Lift acts behind or in front of the CG.

Similarly the effect of the Thrust : Drag couple depends upon whether the Thrust line is below the Drag line (as is usually the case) or vice versa.

The usual design is to have the CP behind the CG, so that the L : W couple is nose-down, and the thrust line lower than the drag line, so that the T : D couple is nose-up. Any loss of power will weaken the nose-up couple, and consequently the nose-down Lift ; Weight couple will pitch the aeroplane into a descent, thereby maintaining flying speed-a fairly safe arrangement.

The Lift: Weight couple and the Thrust: Drag couple should balance each other in straight and level flight so that there is no residual moment acting to pitch the aeroplane either nose-up or nose-down.

This ideal situation rarely exists between the four main force and so the tailplane/elevator is designed into the aeroplane to produce a balancing force. This force may be an up or down force, depending upon the relationship that exists at the time between the Lift : Weight nose-down couple and the Thrust : Drag nose-up couple.

If you have to exert a steady pressure on the control column, so that the elevator produces the required balancing force, then you can trim this pressure off with the elevator trim wheel. Hold the desired attitude, and then trim to relieve the load.

Variation of Speed in Level Flight
For level flight, Lift = Weight. From our now (hopefully) familiar lift formula:

$$L = C_{Lift} \text{ x } 1/2 \text{ rho V-squared x s}$$

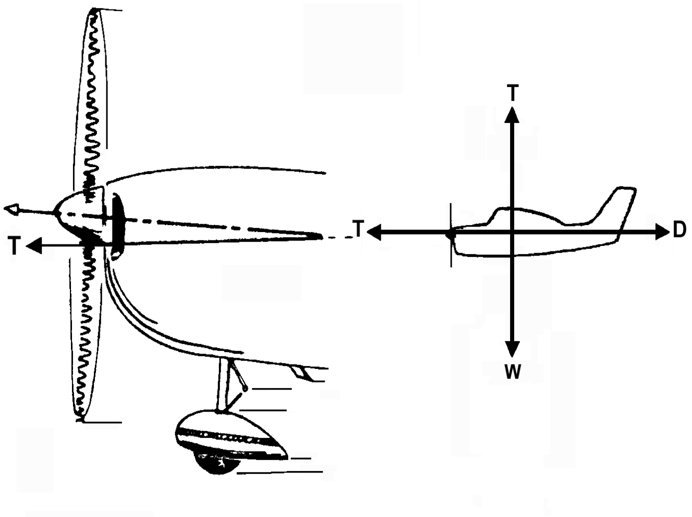


Fig.7.5. The four main forces

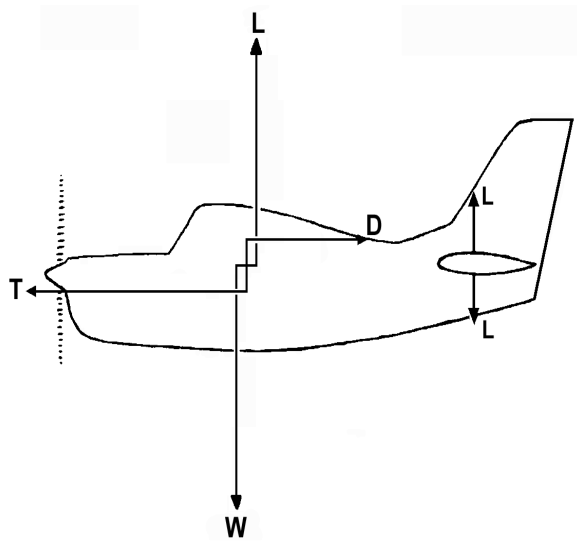


Fig.7.6. The tailplane provides the final balancing moment

Fig.7.7. Indicated air speed varies inversely with angle of attack

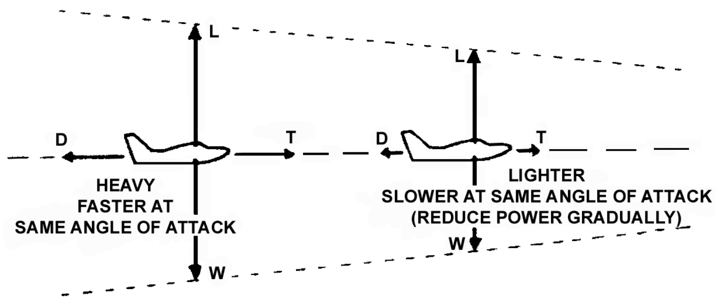


Fig.7.8. For same AoA lighter aeroplane must fly slower

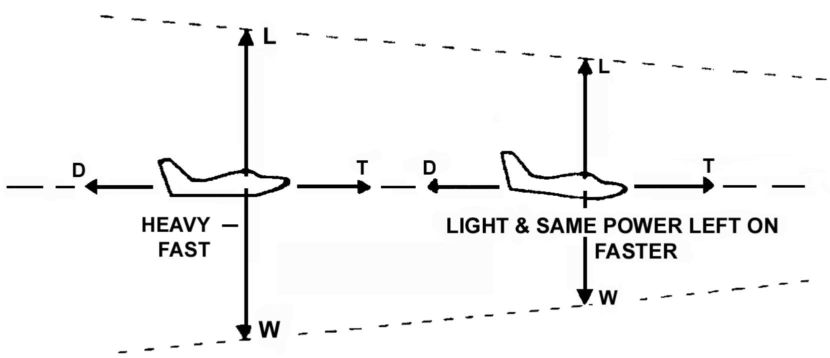


Fig.7.9. Same power - lighter aeroplane has lower angle of attack and flies faster

We can see that if the speed factor V (the TAS) is reduced, then the lift coefficient C_{Lift} (angle of attack) must be increased to retain the balance of Lift = Weight.

V is the True Air Speed (TAS) - the speed of the aircraft relative to the air mass that it is passing through. The Air Speed Indicator in the cockpit, however, does not display TAS- it displays the Indicated Air Speed (IAS), which depends upon the dynamic pressure (1/2 rho V^2).

We need to be care full in our discussion not to become confused between TAS and IAS. Where you see V, think of True Air Speed (TAS), and where you see the formula 1/2 rho V-squared, think of Dynamic Pressure and Indicated Air Speed (IAS).

TAS determines the distance travelled through the air.

IAS determines the aerodynamic effects-the Lift and the Drag.

ATTITUDE IN LEVEL FLIGHT

To obtain the required Lift, at low speed a high angle of attack (high C_{Lift}) is required; while at high speed only a small angle of attack (low C_{Lift}) is needed. Since we are considering level flight, the Pilot sees these angles as an aeroplane pitch attitude relative to the horizon-nose-up at low speeds and fairly nose-level at high speeds.

THE EFFECT OF WEIGHT ON LEVEL FLIGHT

In a normal flight the Weight gradually reduces as fuel is burned-off. If the aeroplane is to fly level, the Lift produced must gradually decrease as the weight reduces. If there is a sudden decrease in Weight, say be half dozen parachutists leaping out, then to maintain straight and level flight the Lift must reduce by a corresponding amount. The C_{Lift} (angle of attack) or the airspeed must be reduced so that Lift generated is less.

Suppose that the aeroplane is flying at a particular angle of attack, say at that for the best L/D ratio (about 4 degrees). To maintain this most efficient angle of attack (C_{Lift} for best L/D ratio) as the weight reduces, the velocity factor ‘V’ must be reduced to lower the Lift produced so that it still balances the Weight.

So, if the height and the angle of attack are kept constant, then the airspeed will have to be reduced. The power (Thrust) will be adjusted to balance the Drag. For most efficient flying (best L/D ratio), the cruising speed will decrease with decreasing Weight.

If the power is kept constant and you want to maintain height as the weight decreases, the lift must be decreased by lowering the angle of attack (decreasing the C_{Lift}). Therefore the speed will increase until the power produced by the engine/propeller is equalled by the power required to overcome the drag.

If you want to keep the speed constant and maintain height, then as the weight reduces you must reduce the lift produced, and you do this by decreasing C_{Lift} (angle of attack). In cruising flight this will mean less drag, and therefore the power required from the engine/propeller is less. If the power is not reduced as the weight decreases, the airspeed will tend to increase.

If your aim is to maintain a constant airspeed, then you would raise the nose a little to avoid the airspeed increasing. Without any reduction in power, the aeroplane would commence a climb and gradually a new set of equilibrium conditions (balance of forces) would establish themselves for a steady climb-no longer level flight. (This is covered in the next chapter-Climbing.)

A very practical relationship for a Pilot to remember is that :

POWER + ATTITUDE = PERFORMANCE
(airspeed or rate of climb)

If you have excess power, then you can adjust the attitude so that the height remains the same and the airspeed increases; or you can hold the attitude for the same airspeed and accept a rate of climb.

Sometimes the Weight increases in flight, for instance by the formation of ice on the structure. An increased Weight will mean that increased Lift is required to maintain level flight-and once again the above discussion applies, but in reverse.

Ice-accretion means more than just a weight addition. If ice forms on the wings, especially on the upper surface near the leading edge, it will cause a drastic decrease in the lift-producing qualities (C_{Lift} for a particular angle of attack) of the wing. There will also be a significant increase in drag. The aeroplane must be flown at a greater angle of attack to return CLift to its original value and so the speed will decrease unless power is added.
If ice forms on the propeller blades, it diminishes their thrust producing qualities. Icing means reduced performance all round, so avoid it if at all possible.

III CLIMB AND GLIDE

i) Climbing

As an aeroplane Climbs. It is gaining potential energy (the position, in this case due to altitude). There are two ways in which an aeroplane can do this:

Climbing can be a temporary gain in height for a loss in airspeed, or it can be a long term steady climb.

1. Climbing by exchanging the kinetic energy of motion ($1/2mv^2$) for potential energy (mgh), i.e. by converting a high velocity ‘v’ to an increase in height `h’ by zooming the aeroplane. Zooming is only a transient (temporary) process, as the velocity cannot be decreased below flying speed. Of course, the greater the speed range of the aeroplane and the greater the need for a rapid increase in altitude, the greater the value and capability of zooming, e.g. a jet fighter being pursued at high speed can gain altitude rapidly with a zoom, or an aerobatics glider that converts the kinetic energy of a dive into potential energy at the top of a loop.
2. Climbing by converting propulsive energy in excess of that needed for straight and level flight to potential energy. The propulsive energy of course comes from fuel energy which is converted to propulsive energy via the engine and propeller. In this way a steady climb can be maintained. It is the steady climb that is of importance to us.

FORCES IN THE CLIMB

We make the assumption that, normal steady en route climb, the thrust force acts in the direction of flight, directly opposite the Drag force.

The Lift force acts perpendicular to the direction of flight. The weight force acts vertically, but note how, in the climb, it has a component that acts in the direction opposing flight. If the pilot maintains a steady climb at constant indicated Air speed, the engine- propeller must supply sufficient thrust to:
a) overcome the Drag force;
b) help lift the weight of the aeroplane at a vertical speed (known as rate of climb).

In a steady climb there is no acceleration. The system of forces in equilibrium and consequently the resultant force acting on the aeroplane is zero. An interesting point is that, when climbing, the Lift force (developed aerodynamically by the wing at 90 degrees to the direction of flight) is marginally less than the weight .

The equilibrium is possible because the excess force of a Thrust minus Drag has a vertical component to help balance the weight force.

IN A CLIMB
Thrust is greater than Drag.
Lift is less than weight.

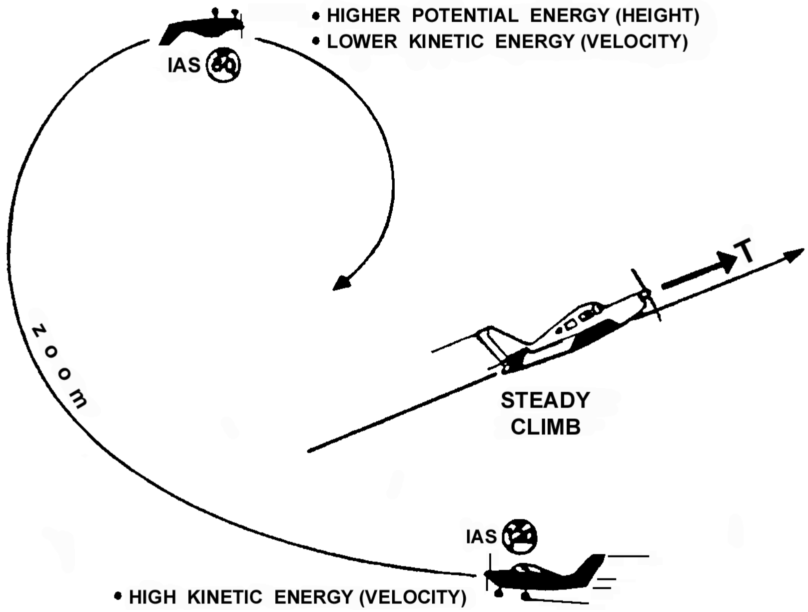


Fig.7.10. Climbing steadily, or zooming

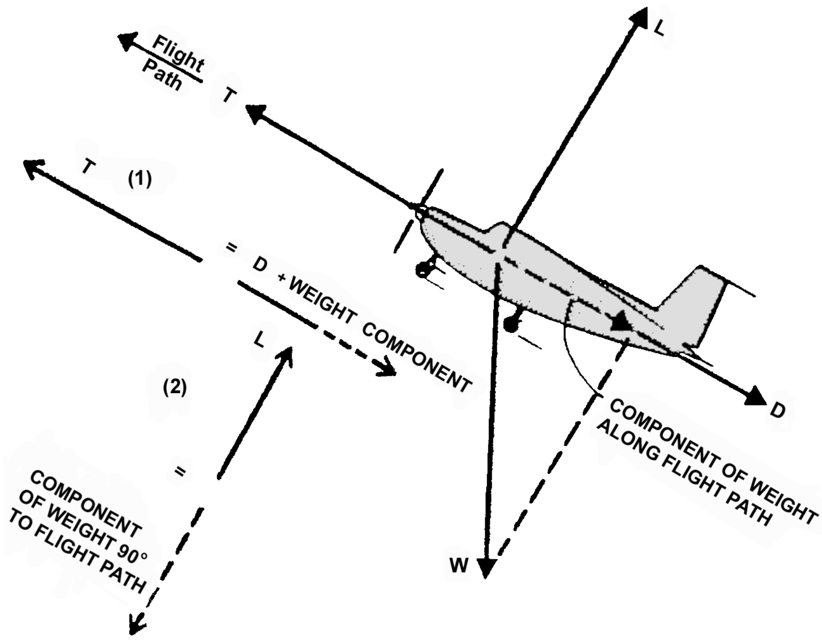


Fig.7.11. The four forces in a steady climb

ANGLE OF CLIMB - (GRADIENT OF CLIMB)

The angle of climb depends directly upon the excess Thrust (the Thrust force in excess of the Drag force) and the weight. A heavy aeroplane will as well as when it is lighter. The higher the weight, the poorer the climb performance.

The lower the weight (w), the greater the angle of climb. A light aeroplane can climb more steeply than a heavy one. Thrust is used to overcome Drag. If the engine/propeller can provide Thrust in excess of that needed to balance the Drag, then aeroplane is capable of climbing.

The greater the Thrust (T), the greater the angle of climb. The Drag (D), the greater the angle of climb. For good climb-gradient capability, the aeroplane should generally be kept in a low drag configuration, e.g. flaps up. This is a very important consideration for takeoff. Flap for takeoff decreases the takeoff run prior to liftoff, but once in flight the angle of climb may be less due to the higher Drag with flaps down.

Since the pilot normally cannot vary the weight significantly in flight, the only way he can improve the angle of climb is to make sure the aeroplane is clean (low drag), and to fly at speed which gives the maximum excess thrust force.

RATE OF CLIMB

The vertical velocity is given the name of climb, it is usually expressed in feet per minute. A Rate of climb (RoC) of 500 fpm means that the aeroplane will gain a 500 ft of altitude i one minute. Rate of climb is shown is the cockpit on the vertical speed indicator (VSI).

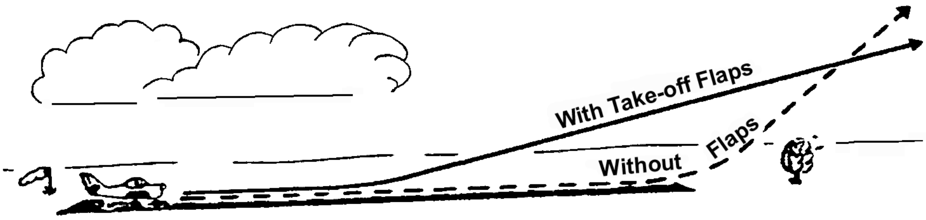


Fig.7.12. Climb gradient may be less with flaps extended

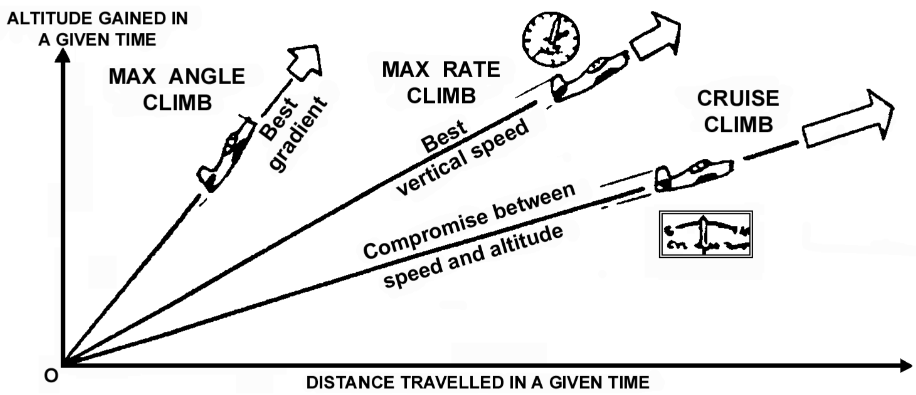


Fig.7.13. Max angle of climb, max rate of climb, cruise climb, use the one that fits the situation

The grater the excess power, the greater the rate of climb, The lower the weight, the greater the rate of climb. The maximum rate of climb usually occurs at a speed somewhere near that for the best Lift/Drag ratio, and is faster that the speed for maximum angle of climb (gradient).

The best rate of climb speed will gain altitude in the shortest time.

THE VARIOUS CLIMB SPEEDS

When mention is made of climb performance, then you must think of both angle (gradient) and rate. The pilot will decide which type of climb he wants:

* a Maximum Gradient (Angle) Climb to clear obstacles -height gained for horizontal distance travelled is the consideration here. Maximum gradient speed (also known as V_x) is the lowest of the three climb speeds. It is usually carried out at high power and for only sufficient time to clear obstacles. The low speed leads to less cooling and consequently higher engine temperatures.
* a maximum Rate Climb to gain height in the shortest time-to get to cruise altitude as soon as possible. Max rate climb speed (V_Y) is usually somewhere near the speed for the best Lift \ Drag ratio.
* a Cruise Climb is a compromise climb that allows for a high speed (to hasten your arrival at the destination) as well as allowing the aeroplane to gain height and reach the cruise altitude without too much delay. It also allows for better engine cooling due to the higher speed. The cruise climb will be a shallower climb at a higher airspeed.

Refer to your pilot's operating Handbook or flight manual for the various climb speeds for your particular aeroplane. Typically, max Gradient climb speed (V_x) is about 10 kt less than max Rate Climb speed (V_y).

CLIMB PERFORMANCE
Increased weight decreases climb performance.

Weight Temperatures decrease Climb Performance because of lower air density.

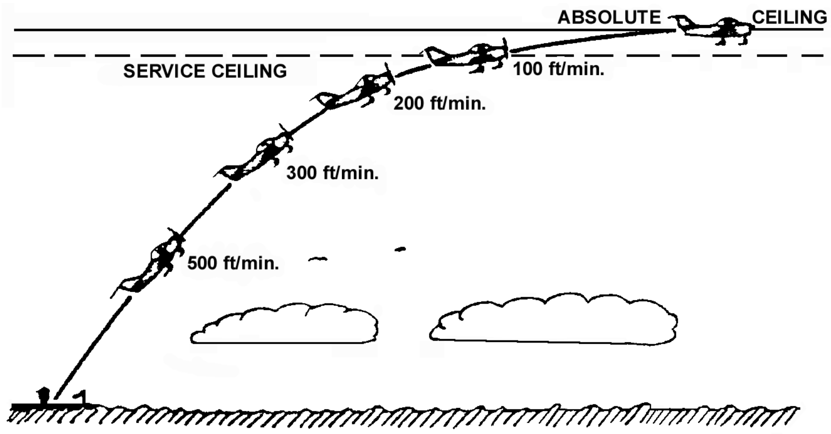


Fig. 7.14. Climb performance decreases with altitude

MAXIMUM RATE OF CLIMB

CONDITIONS
Flaps Up
Full Throttle

NOTE:
Mixture leaned above 3000 feet for maximum RPM.

WEIGHT LBS	PRESS ALT FT	CLIMB SPEED KIAS	RATE OF CLIMB - FPM			
			-20°C	0°C	20°C	40°C
1670	S.L.	67	835	765	700	630
	2000	66	735	670	600	535
	4000	65	635	570	505	445
	6000	63	535	475	415	355
	8000	62	440	380	320	265
	10,000	61	340	285	230	175
	12,000	60	245	190	135	85

Note that RoC decreases with temperature

Note that climbing IAS for best RoC decreases with altitude

Note that RoC decreases with altitude

Fig. 7.15. A typical climb performance table

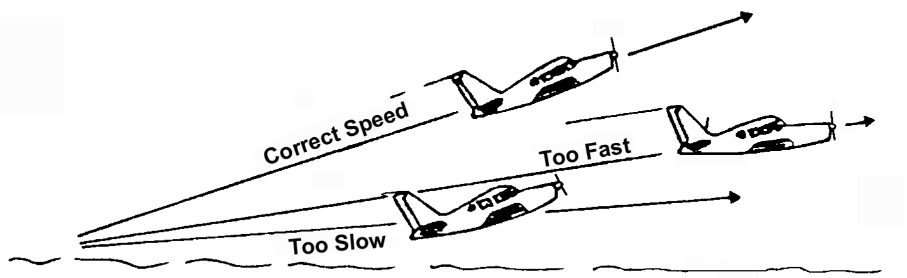


Fig. 7.16. Fly at the correct climb speed for the best performance

Increasing Altitude decreases Climb Performance. Power Available from the engine/propeller decreases with altitude. Even though sea-level performance can be maintained to high altitudes with supercharging, sooner or later power available starts to fall off. The climb performance, the rate of climb, and the angle of climb capability, will therefore all decrease with altitude.

The altitude at which the climb performance falls close to zero and a steady climb can no longer be maintained is known as the ceiling. In technical terms, the Service Ceiling is the altitude at which the steady rate of climb has fallen to just 100 ft/min; the Absolute Ceiling is the slightly higher altitude at which the Steady rate of climb achievable at speed is zero (and there therefore almost impossible to climb to).

The aeroplane's Flight Manual and pilot's Operating Handbook will normally contain a table or graph with climb performance details.
Note that:

* Climb performance decreases as air density decreases (i.e. at high altitudes and high air temperatures); and
* Climbing IAS for best performance decreases as altitude is gained.

Flying Too Fast Decreases Climb performance. You fly faster than the recommended speeds, say at the speed where the Thrust = Drag, and no excess power to give you a rate of climb. The aeroplane can only maintain level flight. At higher speeds, there would be a thrust deficiency and a power deficiency, causing the aeroplane to have an angle of descent and a rate of descent, rather than a climb.

Flying Too Slowly Decreases Climb Performance. Flying slower than the recommended speeds will cause the excess thrust and excess power to be less than optimum (due to the high drag and high angles of attack that it must overcome) and so climb performance will be decreased. At low speed the engine/propeller loses efficiency and produces less thrust. The aeroplanes at low speed has a high drag (mainly induced drag). Eventually the aeroplane will come up against the stall if flown too slowly.

Climbing flight is possible in the speed range where the engine-propeller can produce sufficient power to provide excess thrust (i.e. Thrust in excess of Drag.) On the low speed side you may be limited by the stalling angle.

High Ambient Temperatures Decrease Climb Performance. If the temperature is high then the air density (ρ) is less. The engine/propeller and the airframe will both be less efficient and so the performance capability of the aeroplane is less on a hot day than on a cold day.

THE EFFECT OF A STEADY WIND ON THE CLIMB

The aeroplane flies in the medium of air and it 'sees' only the air. Rate of climb will not be affected by a steady wind. Similarly, the angle of climb through the air will not be affected by steady wind. However, if we consider the angle of climb (or the gradient of climb) over the ground, i.e. the flight path, a headwind increases the effective climb gradient over the ground and a tailwind decreases the effective climb gradient over the ground.

Taking-off into wind has obvious advantages for obstacle clearance-it improves your clearance of obstacles on the ground.

ii) Gliding

If the aeroplane is descending, with no Thrust being produced by the engine/propeller, only three of the four main forces will be acting on the aeroplane-Weight, Lift and Drag-and in a steady glide these forces will be equilibrium as the resultant force acting on the aeroplane is zero.

Suppose that the aeroplane is in steady straight and level flight and the Thrust is reduced to zero. The Drag force is now unbalanced and will act to decelerate the aeroplane-unless a descent is commenced where the component

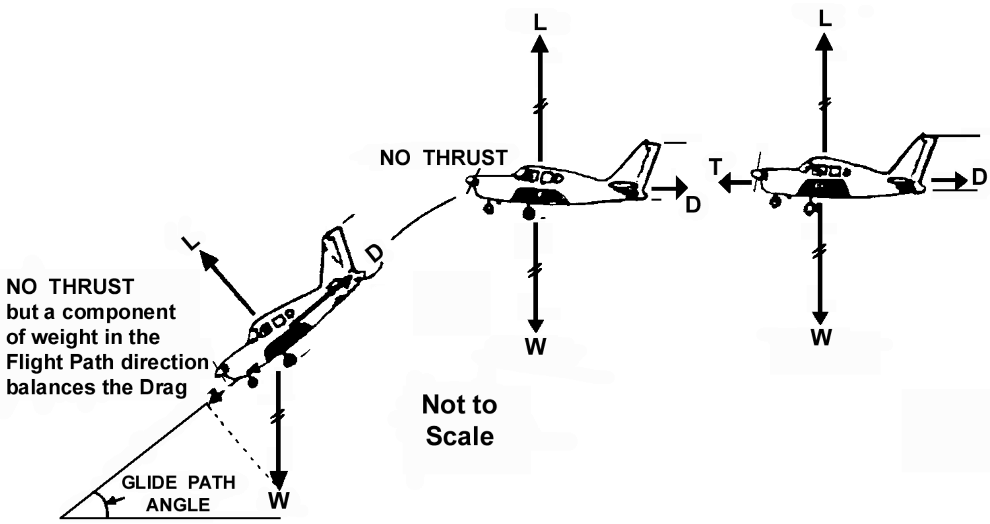


Fig.7.17. In a glide, a component of weight balances the drag

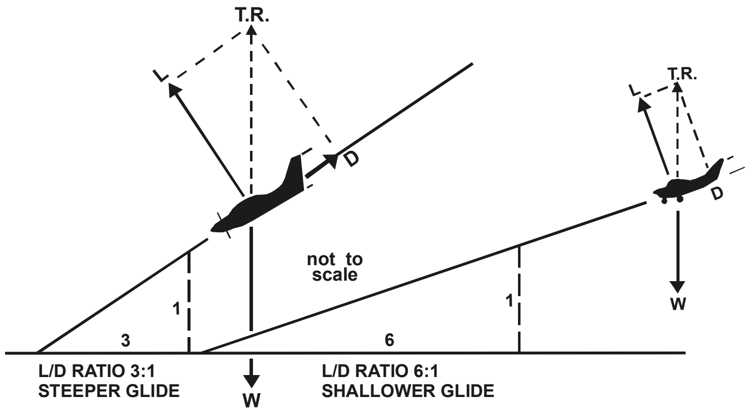


Fig.7.18. Lift and drag balance the weight in a steady glide

of the Weight force acting in the direction of the flight path is sufficient to balance the Drag. This effect allows the aeroplane to maintain airspeed by descending and converting potential energy due to its altitude into kinetic energy (motion).

Resolving the forces in the flight path direction shows that a component of the Weight force acts along the flight path in a descent, balancing Drag and contributing to the aeroplane's speed.

Resolving the forces vertically, the Weight is now balanced by the total Reaction (i.e. the resultant of the Lift and Drag).

Notice that the greater the Drag force, the steeper is the glide. The shallowest glide is obtained when, for the required Lift Drag is least, i.e. at the best Lift/Drag ratio.

If the L/D is high, the angle of descent is shallow, i.e. a flat angle and the aeroplane will glide a long way.

If the L/D is poor (low), with a lot of Drag being produced for the required Lift, then the aeroplane will have a large angle of descent, i.e. a steep glide angle and will therefore not glide very far.

Two points can be made here :

1. An aerodynamically efficient aeroplane is one which can be flown at a high Lift/Drag ratio. It has the capability of gliding further for the same loss of height than an aeroplane that is flown with a lower L/D ratio.

2. The same aeroplane will glide furthest through still air when it is flown at the angle of attack (and airspeed) that gives its best L/D ratio. This angle of attack is usually about 4 degrees.

Because the pilot cannot read angle of attack in the cockpit, flying at the recommended gliding or descent speed (in the Pilot's Operating Handbook) will ensure that the aeroplane is somewhere near this most efficient angle of attack.

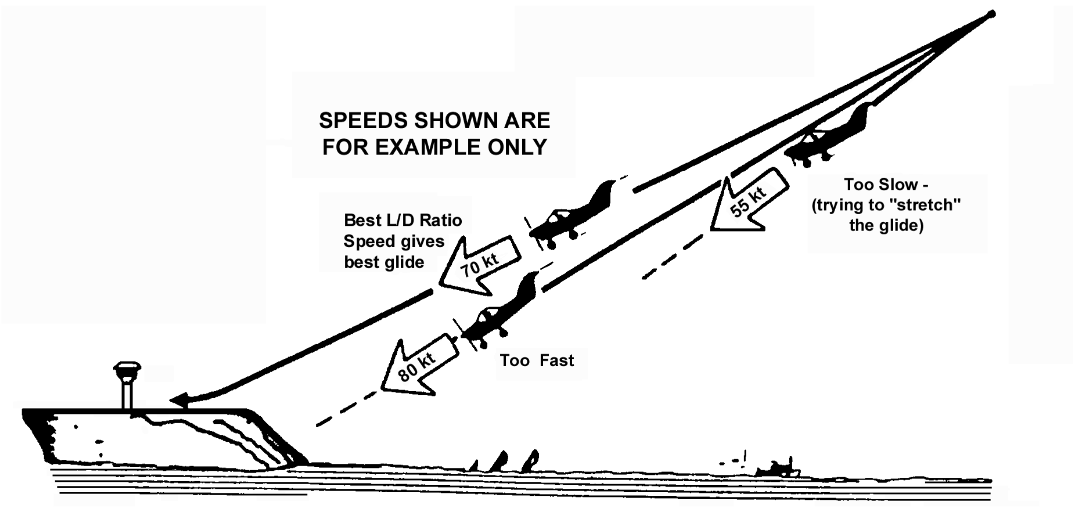


Fig.7.19. The flatest glide is achieved at best L/D airspeed

The Wrong Airspeed (too fast or too slow) Steepens the Glide
If the aeroplane is flown at a smaller angle of attack (and therefore faster), the L/D ratio will be less and the aeroplane will not glide as far-it will 'dive' towards the ground faster and at a steeper angle.

If the aeroplane is flown at a greater angle of attack (lower airspeed) than that for the best L/D ratio the L/D ratio will be less and therefore the optimum glide angle will not be achieved. This may be deceptive for the Pilot-the nose attitude may be quite high, yet aeroplane is descending steeply.

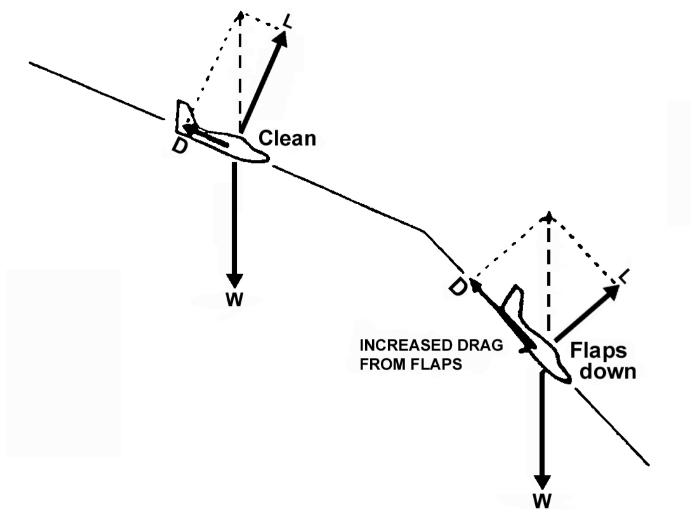


Fig.7.20. Flaps steepen the glide

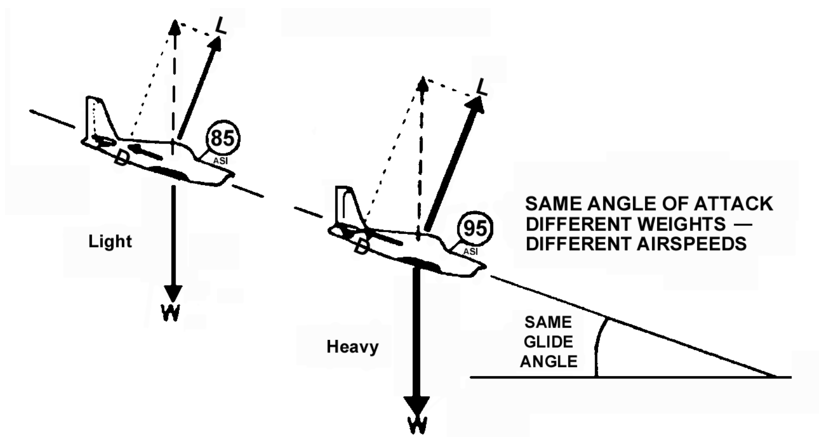


Fig.7.21. Best glide angle is the same at all weights (Best L/D) but airspeed must be lower at lower weights

To Glide the Furthest in Still Air, fly at the Recommended Airspeed (and therefore Angle Attack) that gives the Best Lift/Drag Ratio.

If you are gliding at the recommended airspeed and it looks like you will not reach the desired point, do not raise the nose to increase the glide distance. It will not work. The higher nose attitude may give the appearance of stretching the glide, but in fact it will decrease your gliding distance.

Flaps Steepen the Glide (i.e. increase the Glide Angle)
Any flap settings will increase the Drag more than the Lift and consequently the L/D ratio is lower. This gives a steeper glide.

The Smaller Flap settings increases Lift significantly, with only a small increase in Drag-hence the name Lift Flaps sometimes given to low flap settings.

The Larger Flap Settings give large increase in Drag with only a small increase in the Lift-hence the name Drag Flaps for the larger flap settings. Large flap settings will give a much steeper glide. (The lower nose attitude required with flap extended give the Pilot much better visibility.)

Reduced Weight does not change the Glide Angle, but reduces the best Gliding Speed
If the Weight is less, the aircraft will have a lower airspeed at any particular angle of attack compared to when is heavy.

At the angle of attack for the best L/D ratio (and therefore for the best glide), the airspeed will be lower but the glide angle the same. This also means that the rate of descent for the aeroplane when it is lighter will be less.

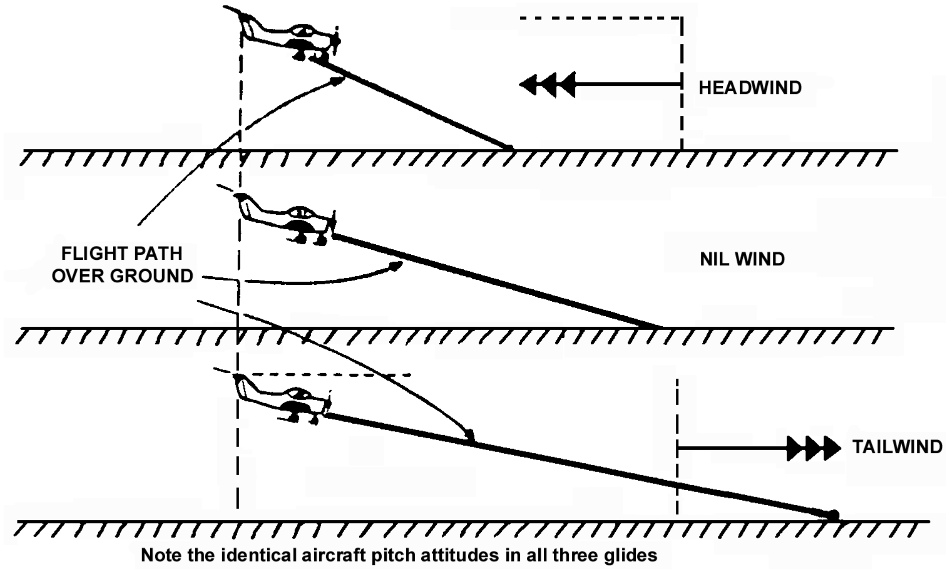


Fig.7.22. More ground is covered gliding with a tailwind and less with headwind

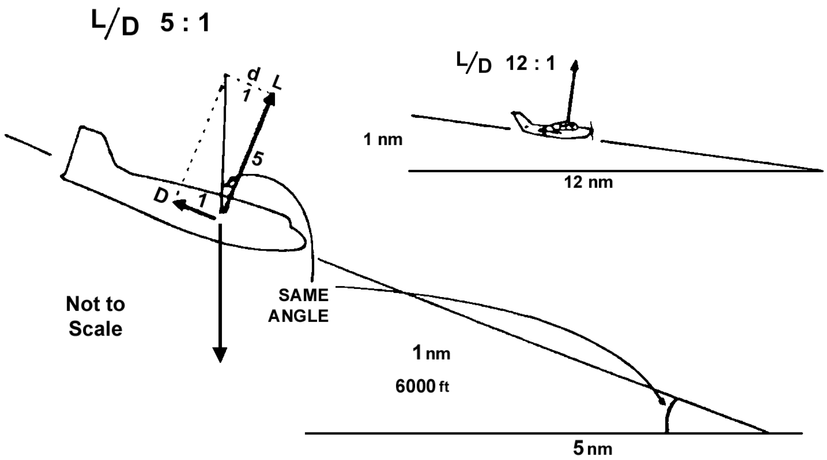


Fig.7.23. a&b. Air distance/Altitude is the same ratio as lift/drag

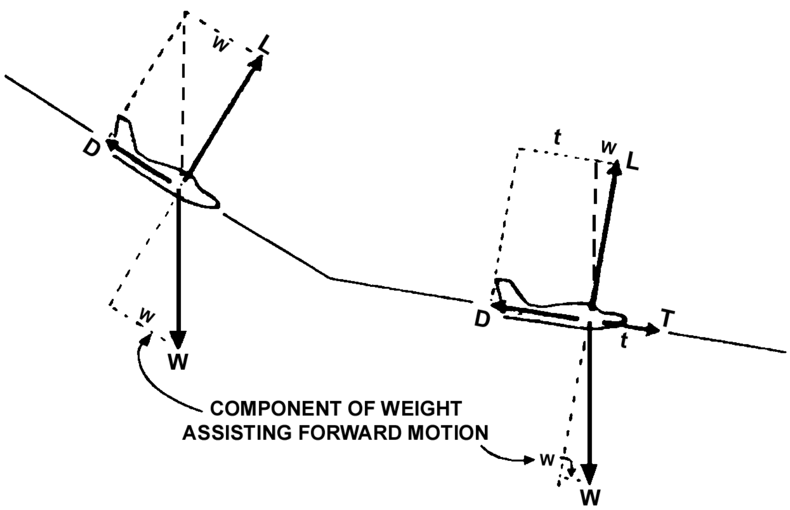


Fig.7.24. Adding power will flatten the descent

The recommended gliding speed (stated in the Flight Manual and the Pilot's Operating Handbook is based on maximum all-up-weight. The variation in weight for most training aircraft is not sufficiently great to significantly affect the glide if the recommended glide speed is used at all times-even though, theoretically, a slightly lower glide speed could be used when lightly-loaded.

The recommended descent speed in your Pilots Operating Handbook will be suitable for all normal weights of your light training aircraft.

GLIDING DISTANCE OVER THE GROUND

A Headwind reduces the Gliding Distance over the Ground, though it does not affect the gliding distance through the air, nor does it affect the of descent.

The aeroplane 'sees' only the air in which it is flying. In the case illustrated below we can see three identical glides through an air mass-same airspeed, same nose attitude, same angle of attack, same rate of descent (therefore some time taken to reach the ground) in all three cases. The only difference is that the air mass is moving over the ground in three different ways and carrying the aeroplane with it. The ground distance covered differs.

A Tailwind Increases the Gliding Distance over the Ground (even though it does not affect the gliding distance relative to the air mass nor the rate of descent).

ESTIMATION OF GLIDING DISTANCE IN STILL AIR

If you refer to the diagram of the forces acting in a glide you will see that, for the best L/D ratio, the gliding distance is furthest.

If the L/D ratio is 5:1, the aeroplane will glide 5 times as it will descend. If you are 1 nautical mile high (about 6000 ft), you will glide for about 5 nautical miles. If you are at about 12,000 If (2nm). you will glide approximately 10 nm.

An aeroplane with a L/D ratio of 12:1 will glide 12 times further horizontally in still air than the height it descends.

FLATTEN THE DESCENT BY INCREASING POWER

If the engine/propeller is producing power, then the thrust force will help overcome part of the Drag force. The result is that the aeroplane will have a shallower descent angle and a lower rate of descent than in the power-off glide. Of course, with sufficient power, the descent angle may be zero, i.e. the aeroplane will fly level. With even more power, the aeroplane may climb.

If you are sinking beneath your desired flight path, the correct procedure is to apply some power and not to just raise the nose (which, as we saw, simply worsened the situation by steepening the glide). Any change in power will require some small adjustments to the nose attitude for the desired airspeed to be maintained.

STEEPEN THE DESCENT BY REDUCING POWER OR INCREASING DRAG

If you are descending above your descent path, there are two things that you can do:

* reduce the thrust, and/or
* increase the drag (extend the flaps, lower landing gear). Usually when you extend the flaps, a lower nose attitude is required.

IV MAXIMUM RANGE AND ENDURANCE

i) Maximum Range

Maximum Range in still air is achieved at the TAS which allows :

* maximum air distance for a given fuel burn-off; or conversely
* minimum fuel burn-off for a given air distance (i.e. the lowest fuel burn-off/air distance ratio).

By converting burn-off and air distance to rates, this ratio becomes fuel burn-off per unit time/air distance per unit time, i.e. fuel flow/TAS, and maximum range will be achieved at the TAS for which this ratio is least. This occurs at the point on the Power vs TAS Curve where the tangent from the origin meets the curve. At all other points, the ratio power/TAS is greater.

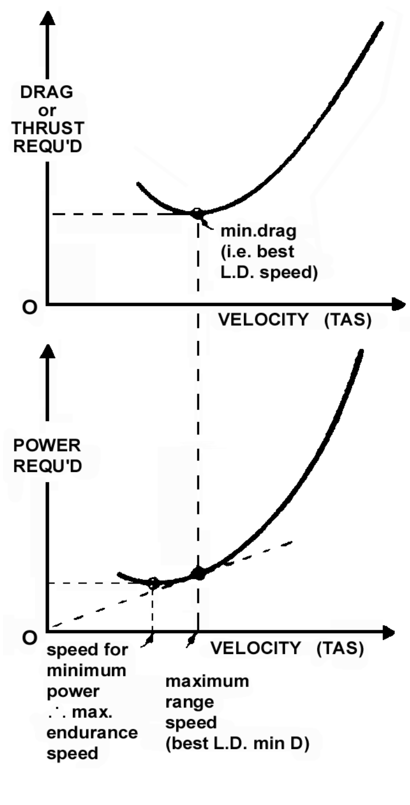


Fig.7.25. Maximum endurance speed and maximum range speed

Power is defined as force x velocity so:
Power Required = Thrust Required x TAS = Drag x TAS (since Thrust = Drag)

Therefore the Power/TAS ratio =

(Drag x TAS) / TAS = Drag, and will of course have a minimum value when Drag is minimum, i.e. Maximum Range TAS is the TAS for minimum total Drag.

To sum up, the maximum range speed shows up on the Drag Curve at the Minimum drag point (which, as explained earlier, is also the point of maximum L/D ratio).

THE REQUIRED PERFORMANCE IS OBTAINED USING POWER PLUS ATTITUDE

Whilst in flight the Pilot of course does not refer to these graphs. Instead he adjusts both the power from the engine-propeller and the pitch attitude of the aeroplane to achieve the desired performance.

POWER + ATTITUDE = PERFORMANCE

ii) Maximum Endurance

Maximum endurance means either:

* the maximum time in flight for given amount of fuel; or
* a given time in flight for the minimum amount of fuel.

It is appropriate to fly at maximum endurance speed when the speed over the ground is not significant, for instance, when:

* holding overhead or near an aerodrome waiting to land; or
* carrying out a search in a specific area.

Since fuel flow for an engine-propeller combination depends upon power set, minimum fuel flow (and therefore maximum endurance) will occur when minimum power to maintain steady straight and level flight is required.

V TAKEOFF AND LANDING

Takeoff and landing performance is a condition of accelerated motion. For instance, during takeoff the airplane starts at zero velocity and accelerates to the takeoff velocity to become airborne. During landing, the airplane touches down at the landing speed and decelerates (or accelerates negatively) to the zero velocity of the stop.

From basic physics, the relationship of velocity, acceleration, and distance for uniformly accelerated motion is defined by the following equation:

$$S = \frac{V^2}{2a}$$

where

S = acceleration distance, ft.

V = final velocity, ft. per sec., after accelerating uniformly from zero velocity

a = acceleration ft. per sec.^{2}

This equation could relate the takeoff distance in terms of the takeoff velocity and acceleration when the airplane is accelerated uniformly from zero velocity to the final takeoff velocity. Also, this expression could relate the landing distance in terms of the landing velocity and deceleration when the airplane is accelerated (negatively) from the landing velocity to a complete stop.

VI ABSOLUTE CEILING & SERVICE CEILING

At the absolute ceiling, then, the maximum excess power is zero, and so is the maximum rate of climb. In fact, the absolute ceiling may be only a theoretical measure of the maximum altitude at which the aircraft can operate. There are two reasons for this. It is clearly undesirable, fro, the handling point of view, to fly at an altitude at which there is only one possible speed. Any disturbance in speed, occasioned perhaps by a gust, leads to a non-equilibrium condition. Control is lost, and may be difficult to recover. Again, the time taken to climb to the absolute ceiling using the steady state techniques we have been studying is generally infinite, as we shall see later. This may mean that in practice the aircraft cannot reach its absolute ceiling. However, all the foregoing analysis, on which this conclusion is based, is concerned with steady rates of climb, i.e., there are no accelerations or retardations involved. It is possible for some jet aircraft to zoom to a height which is even greater than their absolute ceiling, exchanging kinetic energy for potential energy and, of course, losing speed in the process. This is an extremely undesirable manoeuvre, since, at the end of it, the aircraft is in a condition in which it is out of the pilot's control.

The absolute ceiling, then, is not necessarily the highest altitude which the aircraft can reach under its own power, but the highest altitude in which it can be flown steadily, straight and level. However, since it is generally unattainable, as we have seen, it is necessary to have some more practical measure of the greatest height at which the aircraft can normally be expected to operate. It is conventional to define for this purpose the altitude at which the maximum rate of climb is some generally specified figure, greater than zero, small enough for the given altitude to be quite close to the absolute ceiling and so a reasonable measure of maximum operating height, but large enough for the given altitude to be attainable in a reasonable time. This altitude is known as the service ceiling of the aircraft. In the past, the figure for maximum rate of climb at the service ceiling has been specified as 100 ft/min for a piston-engined aircraft. In S.I. units, this is roughly 0.5 m/s, so that we could define the service ceiling for a piston-engined aircraft as the altitude at which its maximum rate of climb is 0.5 m/s. The maximum rate of climb of a jet-engined aircraft is generally much higher than that of a piston-engined aircraft, and if the same definition were used in the case of a jet, the service ceiling would be still too close to the absolute ceiling. The service ceiling of a jet has often been defined as the altitude at which its maximum rate of climb is 500 ft/min, and we may now amend this to read the altitude at which maximum rate of climb is 2.5 m/s.

The service ceiling, then, can be reached in a reasonable time, and at this altitude there is a reasonable, though still narrow, range of speeds within which the aircraft can operate. The curves of Fig.15.23 illustrate this.

As we have seen, the greater the weight the smaller will be the maximum rate of climb at any altitude. Thus, the greater the weight, the lower will be both the service and absolute ceilings.

To determine the ceilings of the aircraft, the power required and power available curves are drawn at various altitudes, and from these the maximum rate of climb is determined at each height. A graph is then prepared of altitude against maximum rate of climb, and generally takes the form depicted in Fig.15.24. This curve, prepared from results which apply to various altitudes from sea level to some height near the expected ceilings, can be extrapolated as shown, in order to read off the altitudes at which the maximum rate of climb is respectively 0.5 m/s (or perhaps 2.5 m/s)a and zero. These are the service and absolute ceilings.

The effect of increasing weight is to reduce $V_{c_{max}}$ at all altitudes, and hence to lower the curve of h against $V_{c_{max}}$. This is also illustrated in Fig.15.24, which shows how both ceilings are lowered as a result of increased weight.

■■■

CHAPTER: 8
BANKED TURN

EQUATION FOR COORDINATED TURN
A moving body tends to continue moving in a straight line at a constant speed, i.e. at a constant Velocity (as stated in Newton's First Law of Motion). To change this state (either to change the speed or to change the direction, i.e. to accelerate the body) a force must be exerted on the Newton's Second Law Motion).

A body constrained to travel in a curved path has a natural tendency to travel in a straight line (and therefore to fly off at a tangent). To keep it on its curved path, a force must continually act on the body forcing it towards the centre of the turn. This force is called the Centripetal Force.

Holding a stone tied to a string, your hand supplies a Lift force equal and opposite to the Weight of the stone. If you swing the stone in a circle, your hand supplies not only a vertical force to balance the Weight but also a centripetal force to deed the stone turning. The force exerted through the string is greater and you will certainly feel the increase.

To turn an aeroplane, some sort of force towards the centre of the turn needs to be generated. This can be done by banking the aeroplane and tilting the Lift force so that it a sideways component.

Flying straight and level, the Lift force on the wings balances the Weight of the aeroplane. If you turn the aeroplane, the wings still need to supply a vertical force to balance the Weight (unless you want to descend) plus a centripetal force towards the centre of the turn to keep the turn going.

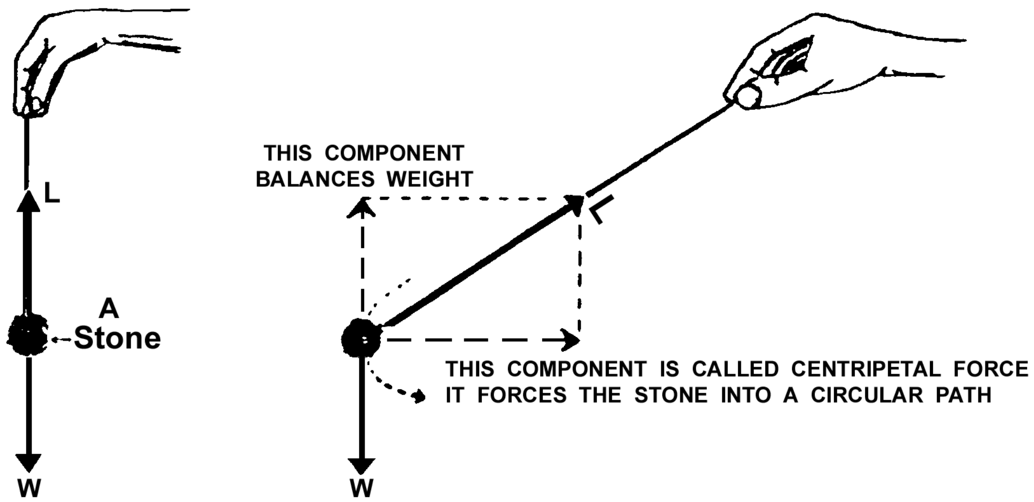


Fig.8.1. Centripetal force pulls a body into a turn

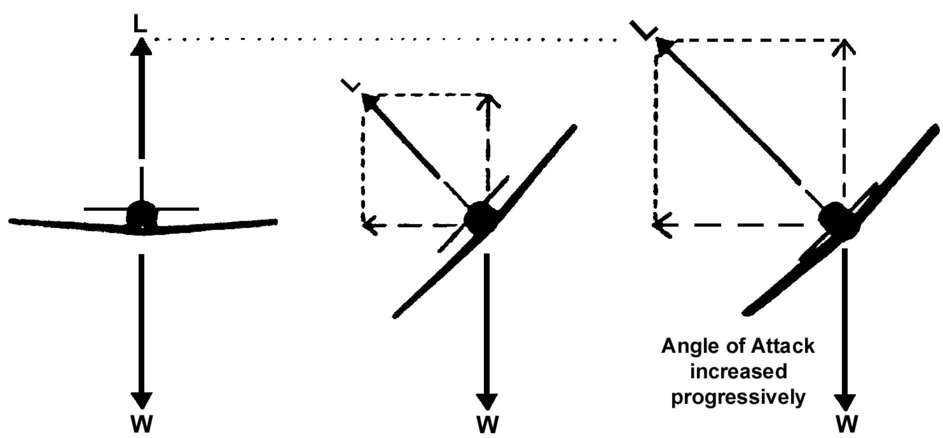


Fig.8.2. By banking, lift from the wings provides a centripetal component

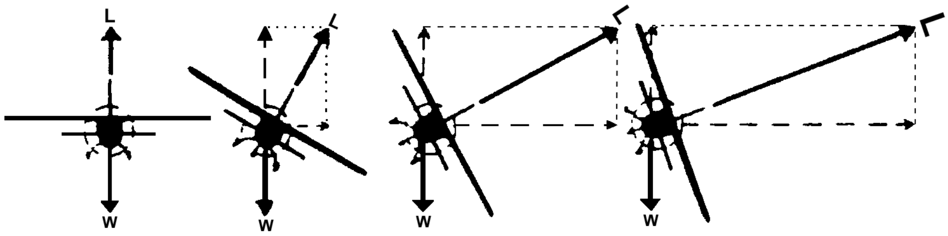


Fig.8.3. To maintain altitude - the steeper the bank, the greater the lift force required from the wings

In an ordinary turn (Fig.8.4) the inward centripetal force is provided by the aeroplane banking (like a car on a racing track) so that the total lift on the wings, in addition to lifting the aeroplane, can supply a component towards the centre of the turn (Fig.8.5).

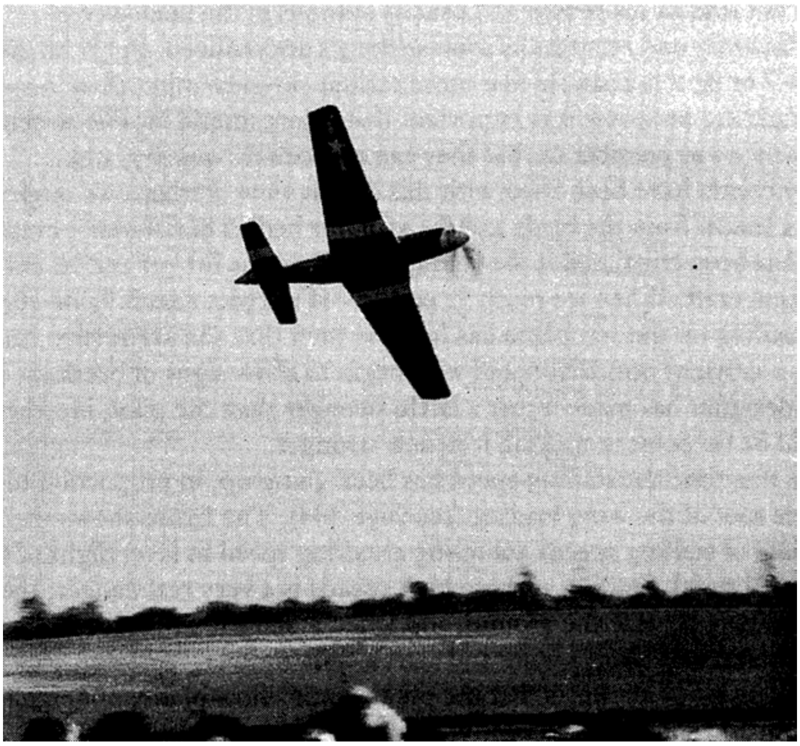


Fig.8.4. By bank

Suppose an aeroplane of weight W newtons to be travelling at a velocity of V metres per second on the circumference of a circle of radius r metres, then the acceleration towards the centre of the circle is V^2/r metres per second.

Therefore the force required towards the centre
$= WV^2/gr$ newtons.

If the wings of the aeroplane are banked at an angle of θ to the horizontal, and if this angle is such that the aeroplane has no tendency to slip either inwards or outwards, then the lift L newtons will act at right angle to the wings, and it must provide a vertical component, equal to W newtons, to balance the weight, and an inward component, of WV^2/gr newtons, to provide the acceleration towards the centre.

$\tan\theta = (WV^2/gr) \div W = V^2/gr$

This simple formula shows that there is a correct angle of bank, θ, for any turn of radius r metres at a velocity of V m/s, and that this angle of bank is quite independent of the weight of the aeroplane.

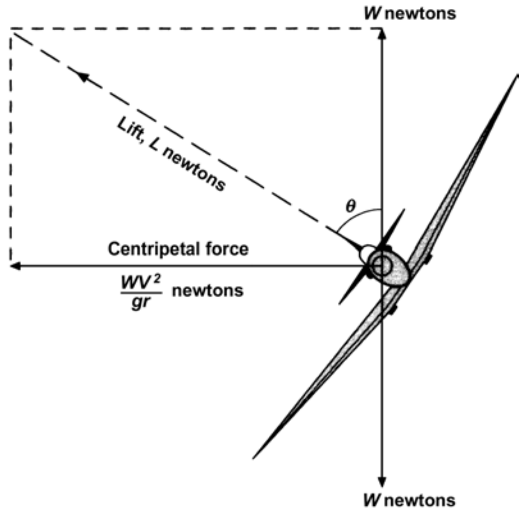


Fig.8.5. By bank

Consider a numerical example -

Find the correct angle of bank for an aeroplane travelling on a circle of radius 120 m at a velocity of 53 m/s (take the value of g as 9.81m/s^2).

V = 53 m/s

r = 120 m

$\tan \theta = V^2/gr$ = (53 X 53) / (9.81X 120) = 2.38

$\therefore \theta = 67^{o}$ approx.

What would be the effect if the velocity were doubled, i.e. 106 m/s?

$\tan \theta$ would be 4 X 2.38 = 9.52

$\therefore \theta = 83^{o}$ approx.

What would be the effect if the velocity were 53 m/s as in the first example, but the radius was doubled to 240 m instead of 120 m?

$\tan \theta$ would be 2.38/2 = 1.19

$\therefore \theta = 49^{o}$ approx.

Thus we see that an increase in velocity needs an increase in the angle of bank, whereas if the radius of the turn is increased the angle of bank may be reduced, all of which is what we might expect from experiences of cornering by other means of transport. Fig.8.6. and 8.7. shows the correct angle of bank for varying speeds and radii; notice again how the speed has more effect on the angle than does the radius of turn.

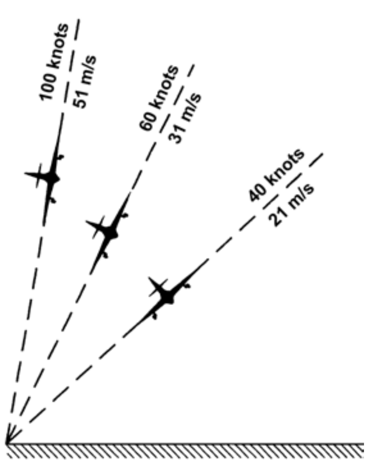


Fig.8.6. Correct angles of bank

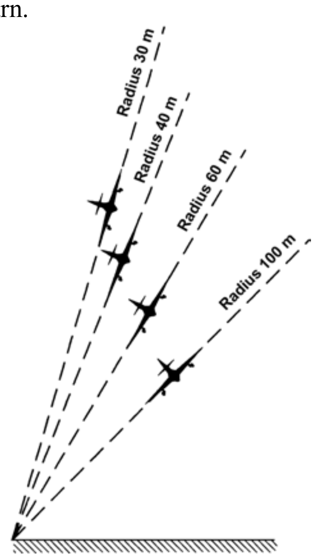


Fig.8.7. Correct angles of bank

Side Slipping

During a roll (Fig.8.8) the aeroplane rotates laterally through 360°, but the actual path is in the nature of a horizontal corkscrew, there being varying degrees of pitch and yaw. In the so-called slow roll the loads in the 180° position are reversed, as in inverted flight, whereas in the other extreme, the barrel roll, which is a cross between a roll and a loop, the loads are never reversed.

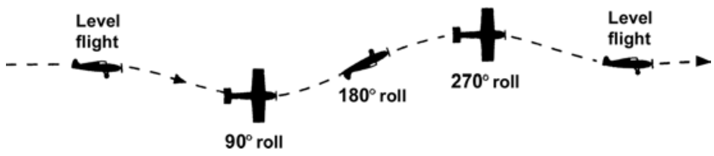


Fig.8.8. A roll

In a sideslip (Fig.8.9) there will be considerable wind pressure on all the side surfaces of the aeroplane, notably the fuselage, the fin and the rudder, while if the planes have a dihedral angle the pressure on the wings will tend to bring the machine on to an even keel. The sideslip is a useful manoeuvre for losing height or for compensating a sideways drift just prior to landing, but, as already mentioned, modern types of aircraft do not take very kindly to sidslipping.

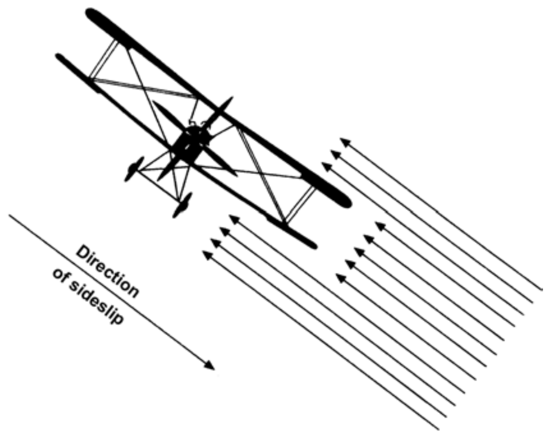


Fig.8.9. A sideslip in an old-type biplane

The small side area means that they drop very quickly if the sideslip is at all steep, and the directional stability is so strong that it may be impossible to hold the nose of the machine up (by means of the rudder), and the dropping of the nose causes even more increase of speed.

Skidding

During a turn shown in (Fig.8.5). When the centripetal force and lift force are not balanced, the aircraft will start move away from its original course, that is called skidding of aircraft.

■■■

CHAPTER: 9
PRINCIPLES OF STABILITY AND CONTROL

STATIC STABILITY

When a mechanical system, such as an aircraft, is very slightly displaced from equilibrium, a non-equilibrium force system is generally created. If the sense of the resultant force or moment is such that it tends to restore the system to its original condition, then the system is said to be statically stable. If, however, the sense of the resultant forc or moment is such that it tends to cause the system to be displaced still further from its original condition, the system is statically unstable. If the displacement causes no new force or moment to be created, so that the system has no tendency either to return to or to deviate further from its original condition, then the condition is a neutral one.

As a simple illustration, consider a marble resting at the topmost point of a smooth convex surface, or at the lowest point of a smooth concave surface, or on a smooth horizontal plane. These are three equilibrium conditions, illustrated in the three diagrams of Fig. 9.1.

Fig.9.1.

If, in the first case, the marble is slightly displaced, the normal reaction of the surface is no longer vertical, but has a sideways component which tends to push the marble further away from the central position. The condition is statically unstable. In the second case, when the marble is slightly displaced, the normal reaction of the surface is again tilted, but its sideways component tends to restore the marble to the central position, so that this is a statically stable condition. In the third case, when the marble is slightly displaced, the reaction of the surface remains vertical, and there is no resultant force on the marble, which therefore tends to remain in its new position. This condition is one of neutral staic stability.

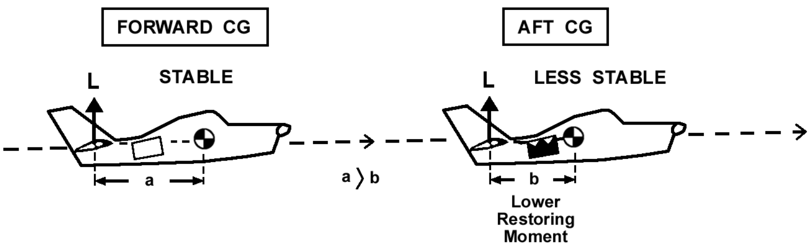
It is important to make the following points concerning the concept of static stability:

a. The concept only has real significance in the context of a system which is initially in equilibrium, although, as we shall see, it may be extended to apply to a steady manoeuvre, such as a steady, level turn.
b. It is concerned only with the tendency of the system to return to equilibrium, or otherwise, and not with what subsequently does happen to the system.
c. It is concerned only with the tendency created by infinitesimal displacement from equilibrium, and not with the consequences of relatively large displacements. For instance, a marble at the point A on a surface such as that depicted in Fig.9.2 would, if displaced to B, have no tendency to return to A, nor to move further from A. Nevertheless, its condition at A is statically unstable.

Fig.9.2.

a. Longitudinal Stability

Longitudinal Stability is in the pitching plane and about the lateral axis. To be longitudinally stable, an aircraft must have a natural or inbuilt tendency to return to the same attitude in pitch after any disturbance. If the angle of attack is suddenly increased by a disturbance, then forces will be produced that will lower the nose and decrease the angle of attack.



stability

Fig.9.3. A forward CG greater longitudinal

The longitudinally stable aeroplane tends to maintain the trimmed condition of flight and is therefore easy for a Pilot to fly in pitch.

b. Lateral Stbility

Lateral stability is the natural or inbuilt ability of the aeroplane to recover from a disturbance in the lateral plane, i.e. rolling about the longitudinal axis without any Pilot input.

A disturbance in roll will cause one wing to drop and the other to rise.

When the aircraft is banked, the lift vector is inclined and produces a sideslip into the turn. As well as the forward motion through the air the aeroplane slips sideways due to the Lift and Weight not being directly opposed, causing a resultant sideways force on the aeroplane.

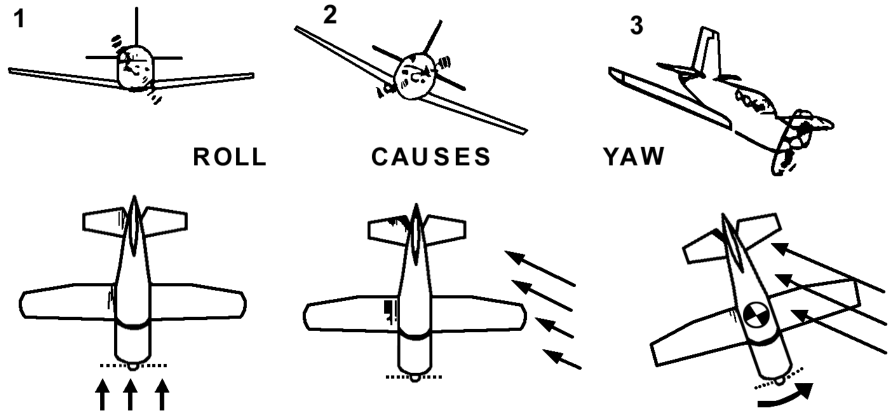


Fig.9.4. An initial roll *Sideslip* *Yaw*

As a result of this sideslip, the aeroplane is subjected to a sideways component of relative airflow and forces are generated that produce a rolling moment to restore the aeroplane to its original wings-level position.

The main contributor to Lateral Stability is the Wing.

c. Directional Stability

Directional stability of an aeroplane is its natural or inbuilt ability to recover from a disturbance in the yawing plane, i.e. about the normal axis. It refers to an aeroplane's ability to 'weathercock' its nose into any crosswind, (i.e. a wind with a component from the side).

If the aircraft is disturbed from its straight path by the nose or tail being pushed to one side (i.e. yawed) then, due to its inertia, the aircraft will initially keep moving in the original direction.

The aircraft will now be moving somewhat sideways through the air, with its side or keel-surfaces exposed to the airflow.

The vertical fin (or tail or vertical stabilizer) is simply a symmetrical aerofoil. As it is now experiencing an angle of attack, it will generate a sideways Lift force which tends to take the fin back to its original position. This restores the nose to its original position.

The powerful moment (turning effect) of the vertical fin, due to its large area and the length of the moment arm between it and the Centre of Gravity, restores the nose to its original position. The greater the fin area and keel surface area behind the CG, and the greater the moment arm, the greater the directional stability of the aeroplane. Thus a forward CG is preferable to an aft CG as it gives a longer moment arm for the fin or vertical stabilizer.

A secondary effect of power or thrust is that caused by the slipstream. Propeller slipstream can affect the airflow over the fin, and therefore the fin's effectiveness as a directional stabiliser.

Changes in power cause changes in the slipstream and can lead to large changes in directional trim requirements.

FACTORS AFFECTING LONGITUDINAL STABILITY

We shall start with longitudinal stability, since this can be considered independently of the other two. In order to obtain stability in pitching, we must ensure that if the angle of attack is temporarily increased, forces will act in such a way as to depress the nose and thus decrease the angle of attack once again. To a great extent

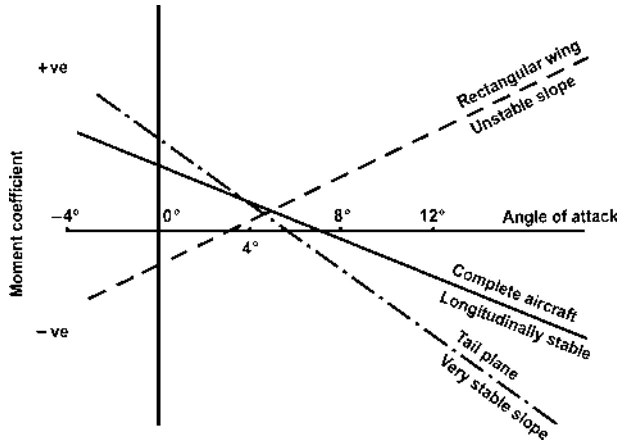


Fig.9.5.

we have already tackled this problem while dealing with the pitching moment, and the movement of the centre of pressure on aerofoils. We have seen that an ordinary upswept wing with a cambered aerofoil section cannot be balanced or 'trimmed' to give positive lift and at the same time be stable in the sense that a positive increase in incidence produces a nose-down pitching moment about the centre of gravity.

The position as regards the wing itself can be improved to some extent by sweepback, by wash-out (i.e. by decreasing the angle of incidence) towards the wing tips, by change in wing section towards the tips (very common in modern types of aircraft), and by a reflex curvature towards the trailing edge of the wing section.

But it is not only the wing that affects the longitudinal stability of the aircraft as a whole, and in general it can be said that this is dependent on four factors :-

1. The position of the centre of gravity, which must not be too far back; this is probably the most important consideration.

2. The pitching moment on the main planes; this, as we have seen, usually tends towards instability, though it can be modified by the means mentioned.

3. The pitching moment on the fuselage or body of the aeroplane; this too is apt to tend towards instability.

4. The tail plane -- its area, the angle at which it is set, its aspect ratio, and its distance from the centre of gravity. This is nearly always a stabilising influence (Fig.9.5)

i. Position of C.G. and C.P.

The further forward the CG of the aircraft the greater the moment arm for the tailplane, and therefore the greater the turning effect of the tailplane lift force. This has a very stabilising effect longitudinally. The position of the CG can be marginally controlled by the Pilot by the disposition of payload and fuel, usually done prior to flight. A forward CG leads to increased longitudinal stability and an aft movement of the CG leads to reduced longitudinal stability.

Limits are laid down for the range within which the CG must lie for safe flight and a prudent Pilot always loads his aeroplane and checks the trim sheet to ensure that this is so. If the CG is behind the legally allowable aft limit, the restoring moment of the tailplane in pitch may be insufficient for longitudinal stability. The same example of a dart is useful here. A CG further forward leads to more stability.

The more stable the aeroplane, the greater the control force the Pilot must exert to control or move the aeroplane in manoeuvres, which can become tiring. Also, if the CG is too far forward, the elevator may not be sufficiently effective at low speeds to flare the nose-heavy aeroplane for landing.

The pitching moment on the main planes; this, as we have seen, usually tends towards instability, though it can be modified by the means mentioned.

The pitching moment on the fuselage or body of the aeroplane; this to is apt to tend towards instability.

ii Area and Location of tail plane

Let us consider a situation that is constantly occurring in flight. If a disturbance, such as a gust, changes the attitude of the aircraft by pitching it nose-up, the aircraft, due to its inertia, will continue initially on its original flight path and therefore present itself to the relative airflow at an increased angle of attack.

Changes in Tailplane Lift lead to Longitudinal Stability.

With the same initial pitch-up caused by the disturbance and the aeroplane at first continuing in the original direction due to its inertia, the tailplane will be presented to the relative airflow at a greater angle of attack. This will cause the tailplane to produce a greater upwards Lift force (or decreased downwards force) than before the disturbance. The increased tail-up lift of course gives a nose-down pitching moment, tending to return the aeroplane to its original trimmed condition.

Because of the great length of the moment arm between the centre of gravity and the tailplane, the Lift force produced by the tailplane need not be large for its turning effect to be quite powerful. As the tail is raised and the nose pitches back down, the original angle of attack is restored, the extra upwards lift force from the tailplane disappears and things are back to where they were prior to the disturbance. As shown in Fig.9.6, the tailplane has a similar stabilising effect following an uninvited nose-down pitch.

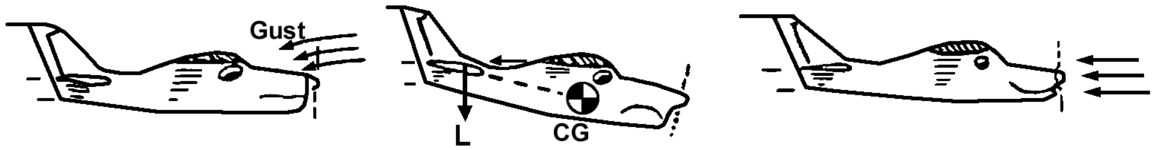


Fig.9.6. Longitudinal stability following an uninvited nose down pitch

In this way the changes caused in the tailplane Lift force have led to longitudinal stability.

A good example of the stabilising effect of a tailplane is the passage of a dart or an arrow through the air, in which the tail-fins act as a tailplane to maintain longitudinal stability.

iii Sweep back

The effect of wing sweepback on rolling moment due to sideslip arises in much the same way as its effect on yawing moment due to sideslip. The aerodynamic force on a wing is largely determined by the wind velocity component normal to its leading edge. This component is greater on the forward wing, in the case of a swept wing in a condition of sideslip. The aircraft depicted in Fig.9.7 is sideslipping to the right, and the normal component of velocity is

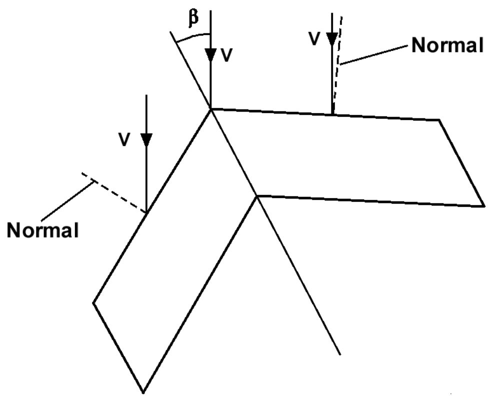


Fig.9.7.

greater on the starboard wing than on the port wing, so the starboard wing carries more lift. As a result, there is a rolling moment to port, which is stabilizing. Again, sweep-forward would have a de-stabilizing effect.

iv Longitudinal Dihedral

The tail plane is usually set at an angle less than that of the main planes, the angle between the chord of the tail plane and the chord of the main planes being known as the "longitudinal dihedral" (Fig.9.8). This longitudinal dihedral is a practical characteristic of most types of aeroplane, but so many considerations enter into the problem that it cannot be said that an aeroplane which does not possess this feature is necessarily unstable longitudinally.

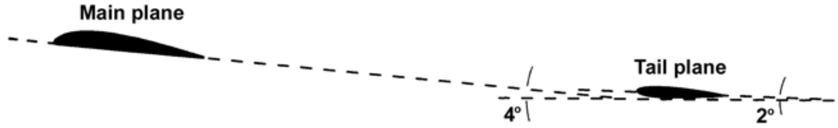


Fig.9.8. Longitudinal dihedral angle.

In any case, it is the actual angle at which the tail plane strikes the airflow, which matters; therefore we must not forget the downwash from the main planes. This downwash, if the tail plane is in the stream, will cause the actual angle of attack to be less than the angle at which the tail plane is set (Fig.9.9). For this reason, even if the tail plane is set at the same angle as the main planes, there will in effect be a longitudinal dihedral angle, and this may help the aeroplane to be longitudinally stable.

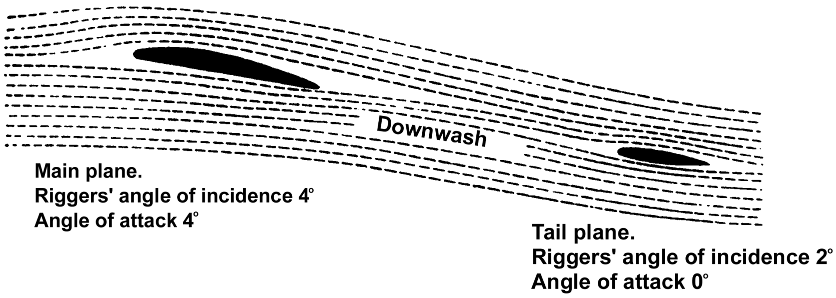


Fig.9.9. Effect of downwash on the tail plane.

Suppose an aeroplane to be flying so that the angle of attack of the main planes is 4° and the angle of attack of the tail plane is 2°; a sudden gust causes the nose to rise, inclining the longitudinal axis of the aeroplane by 1°. What will happen? The momentum of the aeroplane will cause it temporarily to continue moving practically in its original direction and at its previous speed. Therefore the angle of attack of the main planes will become nearly 5° and of the tail plane nearly 3°. The pitching moment (about the centre of gravity) of the main planes will probably have a nose-up, i.e. unstable tendency, but that of the tail plane, with its long leverage about the centre of gravity, will definitely have a nose-down tendency. If the restoring moment caused by the tail plane is greater than the upsetting moment caused by the main planes, and possibly the fuselage, then the aircraft will be stable.

This puts the whole thing in a nutshell, but unfortunately it is not quite so easy to analyse the practical characteristics which will bring about such a state of affairs; however the forward position of the centre of gravity and the area and leverage of the tail plane will probably have the greatest influence.

It is interesting to note that a tail plane plays much the same part, though more effectively, in providing longitudinal stability, as does reflex curvature on a wing, or sweepback with wash-out of incidence towards the tips.

When the tail plane is in front of the main planes (Fig.9.10) there will probably still be a longitudinal dihedral, which means that this front surface must have greater angle than the main planes. The latter will naturally still be at an

Fig.9.10. Tail-first.

efficient angle, such as 4°, so that the front surface may be at, say, 6° or 8°. Thus it is working at a very inefficient angle and will stall some few degrees sooner than the main planes. This fact is claimed by the enthusiasts for this type of design as its main advantage, since the stalling of the front surface will prevent the nose being raised any farther, and therefore the main planes will never reach the stalling angle.

In the tail-less type, in which there is no separate surface either in front or behind, the wings must be heavily swept back, and there is a 'wash-out' or decrease in the angle of incidence as the wing tip is approached, so that these wing tips do, in effect, act in exactly the same way as the ordinary tail plane (Figs.9.11 and 9.12).

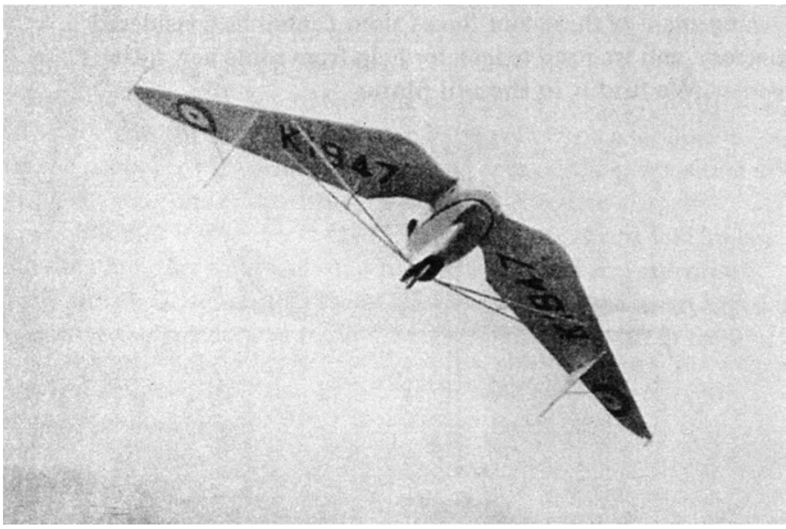


Fig.9.11. Tail-less-old type.

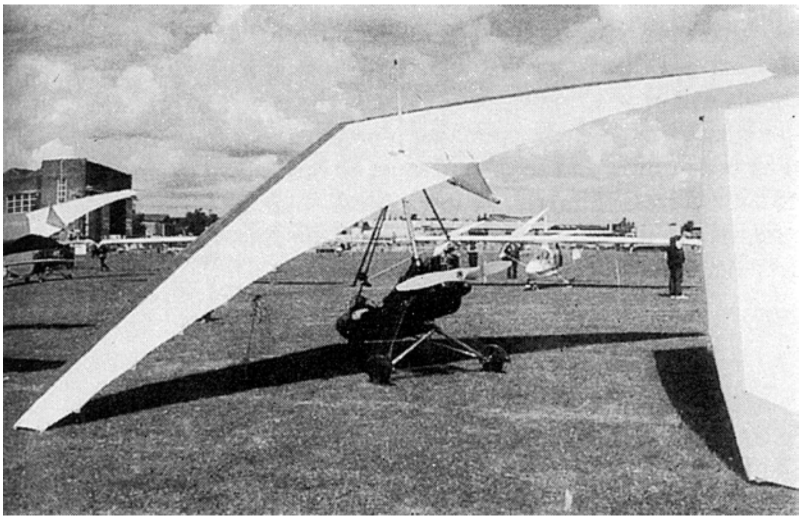


Fig.9.12. Tail-less-new type.

STATIC MARGIN AND NEUTRAL POINT

Neutral point as the furthest aft position of the centre of gravity for which the aircraft is not statically unstable. It follows that the neutral point is the centre of gravity position for which $dC_m/dC_L = 0$, giving neutral stability. It is clear from this that increments in total lift due to change of incidence act through the neutral point, since they produce no change in pitching moment about that point. Thus the neutral point of the aeroplane is analogous to the aerodynamic centre of the wing alone.

Thus, for static stability, the centre of gravity must lie ahead of the neutral point, and the distance of the centre of gravity ahead of the neutral point is a direct measure of the longitudinal static stability. This distance, expressed as a fraction of the wing mean chord, is called the static margin, or sometimes the C.G. margin. See Fig.16.13.

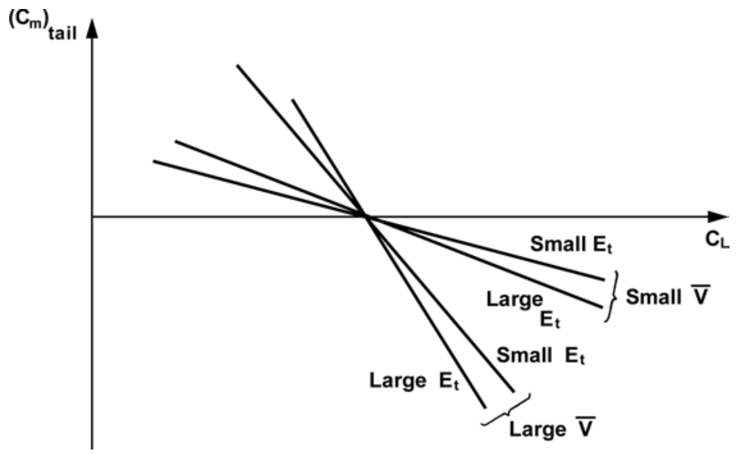


Fig.9.13.

In fact, an aircraft possesses two neutral points, according to whether it is the stick-fixed or stick-free stability which is under discussion. In the following paragraphs we shall study methods of determining the neutral point in both these conditions.

FACTORS AFFECTING LATERAL STABILITY

Lateral and directional stability will first be considered separately; then we shall try to see how they affect each other.

To secure lateral stability we must so arrange things that when a slight roll takes place the forces acting on the aeroplane tend to restore it to an even keel.

In all aeroplanes, when flying at a small angle of attack, there is a resistance to roll because the angle of attack, and so the lift, will increase on the down-going wing, and decrease on the up-going wing. But this righting effect will only last while the aeroplane is actually rolling. It must also be emphasised that this only happens while the angle of attack is small; if the angle of attack is near the stalling angle, then the increased angle on the falling wing may cause a decrease in lift, and the decreased angle on the other side an increase; thus the new forces will tend to roll the aeroplane still further, this being the cause of auto-rotation previously mentioned (Fig.9.14).

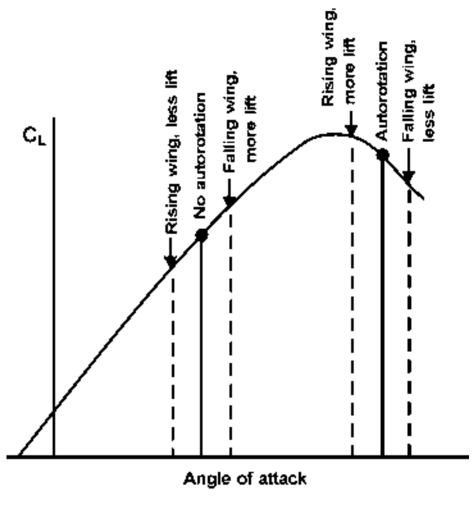


Fig.9.14. Cause of autorotation.

But the real test of stability is what happens after the roll has taken place.

i. Dihedral

The most common method of obtaining lateral stability is by the use of a dihedral angle on the main planes (Fig.9.15 & 16). Dihedral angle is taken as being the angle between each plane and the horizontal, not the total angle between the two planes, which is really the geometrical meaning of dihedral angle. If the planes are inclined upwards towards the wing tips, the dihedral is positive; if downwards, it is negative and called anhedral (Fig.9.17); the latter arrangement is used in practice for reasons of dynamic stability.

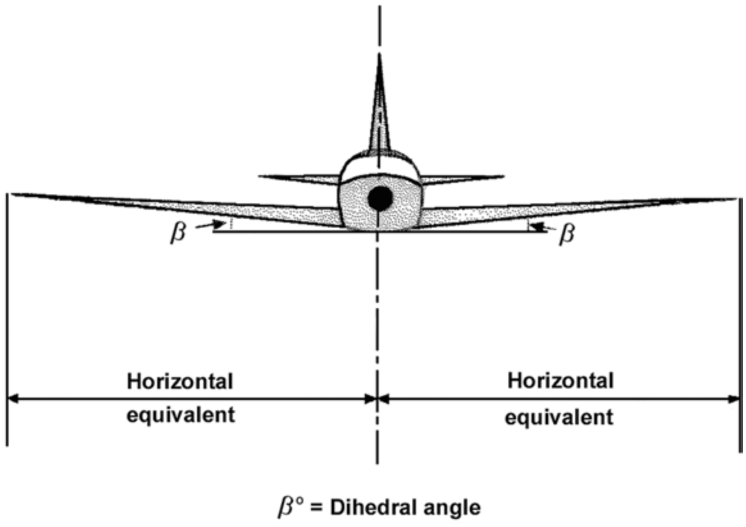


Fig.9.15. Lateral dihedral angle.

Fig.9.16. Dihedral.

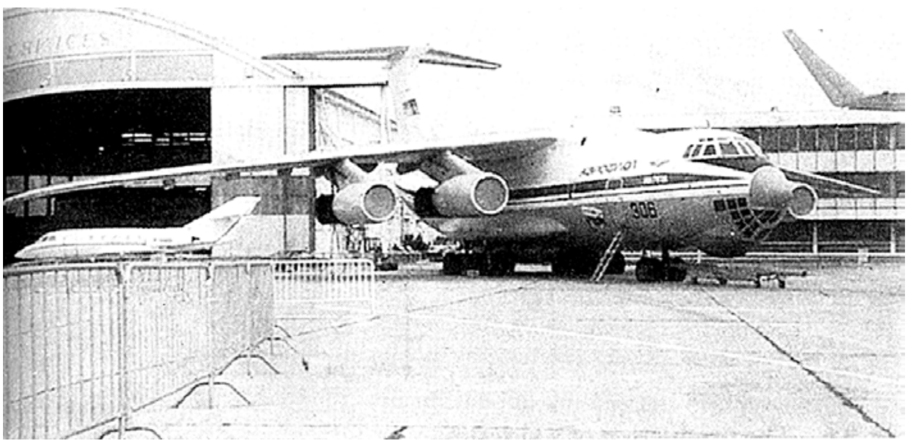


Fig.9.17. Anhedral.

The effect of the dihedral angle in securing lateral stability is sometimes dismissed by saying that if one wing tip drops the horizontal equivalent on that wing is increased and therefore the lift is increased, whereas the horizontal equivalent and the lift of the wing which rises is decreased, therefore obviously the forces will tend to right the aeroplane.

ii. High Wing

If the wings are placed in a high position and the centre of gravity is correspondingly low, the lateral stability can be enhanced. When an aircraft sideslips, the lift on the lower wing becomes greater than that on the higher one. Furthermore, a small sideways drag force is introduced. In consequence, the resultant force on the wing will be in the general direction indicated in Fig.9.18. You will see that this force does not now pass through the centre of gravity so there will be a small moment which will tend to roll the aircraft back to a level condition. This will occur even on a low-wing aircraft, but is more effective with a high wing because the moment arm is greater. For this reason a high-wing aircraft requires less dihedral than a low-wing type.

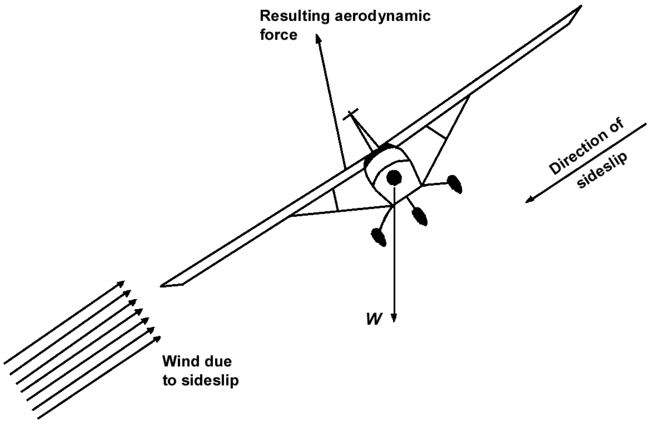


Fig.9.18.

iii. Sweep Back

The effect of wing sweepback on lateral static stability may be considerable, and it is mainly because of the large positive contribution associated with this feature that highly swept wings on high performance aircraft may be mounted with anhedral, as observed above, to ensure that the aircraft is not too stable in this mode.

FACTORS AFFECTING DIRECTIONAL STABILITY

Consider first directional effects, and, in particular the static stability of an aircraft in the directional mode. When an aircraft is yawed, a side force is set up as a result of the asymmetry in attitude. This side force will, in general, give rise to a yawing moment, which may be added to as a result of other effects which we shall examine. If the overall yawing moment, N, which results from a small, involuntary displacement in yaw, then the aircraft is statically stable in yaw. Thus the requirement for positive directional static stability is that positive yaw should result in negative yawing moment, i.e., that $dN/d\psi < 0$. If $dN/d\psi > 0$, the aircraft is statically unstable, and if $dN/d\psi = 0$, it is neutrally stable, in the directional mode. The various possibilities are depicted in the curves of Fig.9.19. Graphs showing the variation of N with sideslip as well as with yaw are included since these constitute alternative ways of presenting the same data. It is assumed that when $\psi = 0$ the condition is symmetrical, so that N = 0, and that the variation of N with ψ is linear.

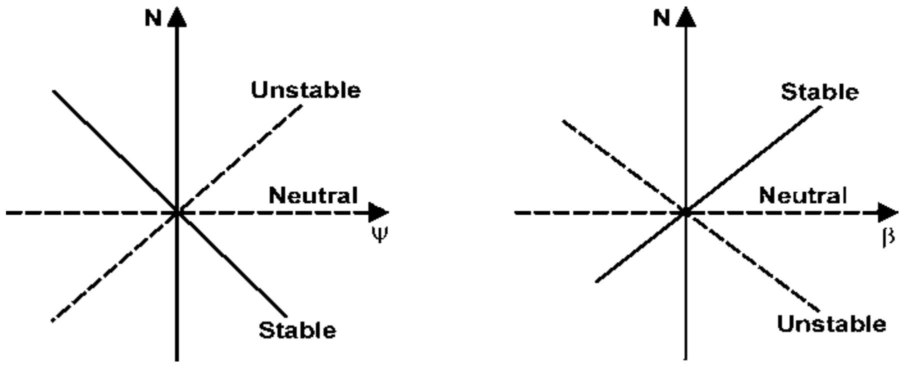


Fig.9.19.

Several components of the aeroplane contribute substantially to the directional static stability, in a positive or negative way.

i. Vertical Stabilizer

Just as a horizontal tailplane gives longitudinal static stability to an aircraft, so a vertical tail fin provides directional static stability, and it is the component of the aircraft which provides the principal stabilizing effect in this mode. Fig.9.20 shows a section of a vertical tail fin on an aircraft which is sideslipping to the left, i.e., yawed to the right. The fin is at incidence, and experiences a side force to the right. This creates a yawing moment to port, which is a stabilizing moment. The side force on the tail fin may still be relatively small compared with that on the fuselage, which is de-stabilizing, but because its line of action is far aft of the aircraft centre of gravity, the yawing moment it creates is relatively large, and gives overall stability to the fuselage-fin combination.

Thus the principle behind the effect of the tail fin as a stabilizer is just the same as in the case of a dorsal or ventral fin. However, because it is much larger, and, in particular, has a much higher aspect ratio, it is effective at low angles of yaw. It remains effective until the angle of yaw is such that the fin incidence approaches its stalling value, but above this value the side force on the fin decreases with increasing angle of yaw, and the fin ceases to be effective as a stabilizer. It is at this point that the dorsal or ventral fin becomes important. Because it stalls at a very much higher incidence, it takes over the stabilizing role of the tail fin at large angles of yaw. These effects are illustrated in the curves of Fig.9.21. The fuselage by itself

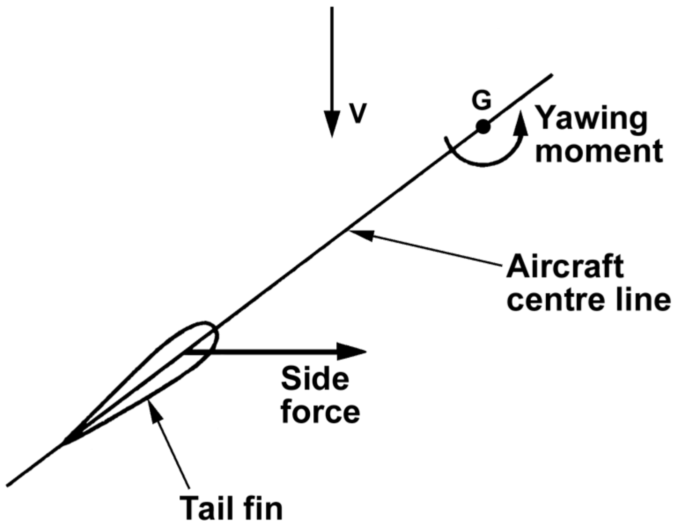


Fig.9.20.

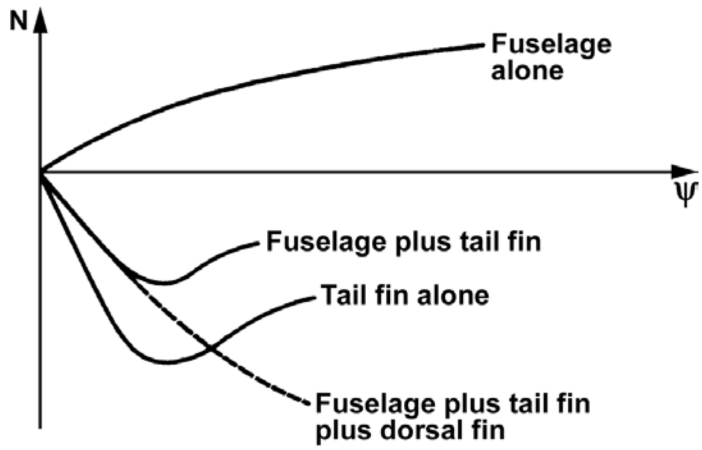


Fig.9.21.

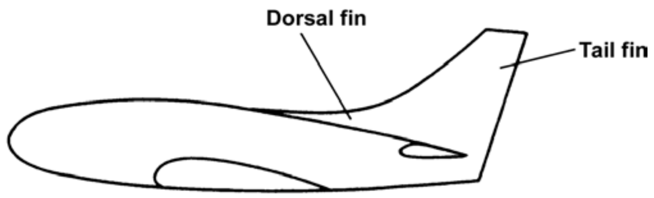


Fig.9.22.

is seen to be basically unstable, i.e., the N ~ ψ curve has positive slope. The tail fin by itself is stable, and forms a stable combination with the fuselage, until it stalls. The addition of a dorsal fin stabilizes the fuselage-fin combination at higher incidences. In practice, the tail fin is often so designed that it constitutes a combination of high aspect ratio fin with low aspect ratio fin, as sketched in side elevation in Fig.9.22.

ii. Sweep Back

The tilting of the lift vector on each wing, associated with wing dihedral, is responsible for a minor de-stabilizing contribution to the yawing moment due to yaw. However, this contribution is insignificant compared with the effect of wing sweepback.

Consider a yawed swept wing, such as that illustrated in Fig.9.23. The wind vector relative to any wing can be resolved into components normal and parallel to its leading edge, or to the line of aerodynamic centres

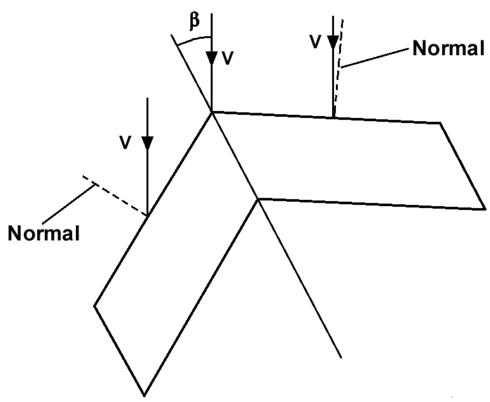


Fig.9.23.

in the case of a wing whose leading and trailing edges are not parallel. It is largely the normal component which determines the drag of the wing. In the case of a swept wing in the yawed condition illustrated, the inclination of the forward, i.e., starboard, wing to the wind direction is greater than that of the rearward wing, so that there is a higher normal component of wind velocity, and hence more drag, on the starboard side. The result of this discrepancy in drag on the two wings is a yawing moment to starboard, which tends to eliminate the sideslip. Thus this is a stabilizing effect, and may be quite important if the sweepback angle is quite large.

Clearly, sweep forward would have a de-stabilizing effect. A straight wing without dihedral has no significant effect on directional static stability.

iii. Vertical Fin

To overcome this instability in the fuselage, it is possible to incorporate into its design dorsal or ventral fins. A dorsal fin is a small aerofoil, of very low aspect ratio, mounted on top of the fuselage near the rear. A ventral fin is similar, but mounted below. Such fins are depicted in Fig.9.24. The effect of the fins, when the aircraft is yawed to starboard, say, is to create a side force to the right. The line of action of this force is well aft of the centre of gravity of the aircraft,

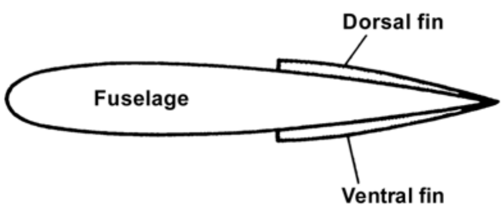


Fig.9.24.

so the yawing moment which results is to port, and this is a stabilizing effect. However, at small angles of yaw, the fins are at low incidence, and because they are small, and, in particular, their aspect ratio is very low, resulting in small lift-curve slope, the side force they create at such small angles will be very small, and they will be ineffective. Thus a fuselage which is unstable in yaw, will, when fitted with a dorsal or ventral fin, remain unstable at low angles of yaw. However, at relatively high angles of yaw, the fins become more effective; because of their low aspect ratio they do not tend to stall, and this increase in effectiveness continues, so that the combination of fuselage with dorsal or ventral fin is stable at large angles of yaw. The curves of Fig.9.25, which are plotted for a fuselage alone and for a fuselage fitted with a moderately large dorsal or ventral fin, indicate these effects.

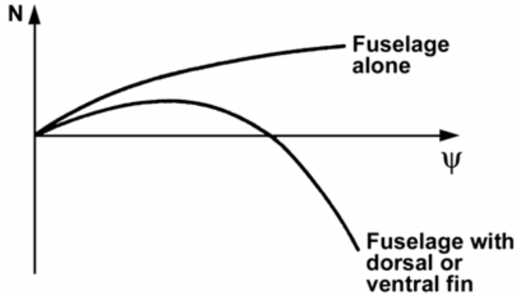


Fig.9.25.

We may note here that, while dorsal and ventral fins contribute in exactly the same way to directional static stability, a dorsal fin contributes positively to lateral static stability, while a ventral fin is de-stabilizing in this mode, as we shall see later. For this reason, the dorsal fin is much the more common feature.

SPIRAL DIVERGENCE, DIRECTIONAL DIVERGENCE AND DUTCH ROLL

Although we shall not present here any general discussion of dynamic stability in the lateral and directional modes, we shall look now at some special dynamic effects which may arise if an aircraft does not have satisfactory stability characteristics.

SPIRAL DIVERGENCE

If, on the other hand, an aircraft has too much directional stability, and too little lateral stability in proportion, a divergence of quite a different kind may ensue. A disturbance in yaw to starboard, say, would produce a stabilizing yawing moment which would quickly eliminate the motion in yaw. However, a rolling moment to starboard will also be created, causing the aircraft to roll to starboard, and consequently to lose height as a result of the tilting of the lift vector. The aircraft will then enter a spiral dive which, if there is insufficient lateral stability, may get tighter and tighter. This kind of instability is known as spiral divergence.

DIRECTIONAL DIVERGENCE

The first of these is a form of instability known as directional divergence, and is a simple divergence in yaw which may occur if an aircraft is directionally statically unstable, i.e., if $dN/d\psi > 0$. Any disturbance in yaw then gives rise to a yawing moment in the same sense as the original displacement. Thus, if an aircraft flying straight and level experiences a small displacement in yaw to star-board, say, the result will be a yawing moment to starboard which will cause the displacement to increase. Further, in the resulting yawed attitude, the aircraft will be subject to a side force to star-board, which will cause it to deviate from its original flight path. The side force will, at least up to a point, increase with angle of yaw, so that the flight path will continue to curve away from the original direction. If the aircraft is laterally statically stable, this directional divergence will occur without any significant degree of banking, and will continue until a limiting condition is reached in which the angle of yaw is so large that $dN/d\psi$ is no longer positive. In this condition, the aircraft would still be flying along a curved path at a very high angle of sideslip. This kind of divergence is not difficult to correct in flight, but it is avoided in design simply by building in sufficient positive directional static stability.

DUTCH ROLL

A third possible motion, which may ensue if the aircraft has positive directional static stability, but not so much, in relation to the lateral stability, as may lead to spiral divergence, is the lateral-directional oscillation, commonly known as the Dutch roll. Here a disturbance in yaw leads to an oscillation in yaw which, though damped, is only lightly so, and is therefore quite persistent. The displacement in yaw also gives rise to a rolling moment, and so initiates an oscillation in roll. If there is a high degree of lateral stability, such oscillations are quickly damped out. However, every period of the continuing oscillation in yaw acts in such a manner as to force further displacement in roll, and the resulting motion is a combination of oscillations in roll and yaw which have the same frequency, though they are out of phase. This is the Dutch roll. It is usually only lightly damped, and may even be unstable.

In practice, design is a matter of compromise. The aircraft is first designed to have enough lateral stability so as not to be likely to suffer from spiral divergence. Enough directional stability is then built in to ensure that, in the first place, there will be no tendency towards directional divergence, and, in the second place, the relationship between the directional and static stability is such that the Dutch roll is adequately damped. As a result of this chosen ratio there may now be some renewed tendency towards spiral divergence, but this will tend to occur slowly and it is certainly preferable to an unstable Dutch roll.

The precise nature of any of these motions is determined by the various aerodynamic derivatives of a particular aircraft, and the relationships between them. The Dutch roll is often initiated deliberately in flight tests, since the measured data concerning amplitude, frequency and phase of the oscillations make it possible to estimate the values of several of these derivatives.

AERODYNAMIC BALANCING

The forces which are necessary to move the various controls, i.e., the elevator, ailerons or rudder, may in some cases be very large, especially in the case of large aircraft, with large control surface, and/or at high speeds. This implies that the control surface hinge moments are too large to be acceptable from the pilot's point of view. By careful design it is possible to make use of aerodynamic forces to help to deflect the control surface thus reducing the restoring hinge moment and the corresponding stick force. An aerodynamic device which performs this function is known as an aerodynamic balance. There is several different types of aerodynamic balance, of which the most common are listed and described in the following paragraphs.

i. Set back the hinge

Downward deflection of an elevator, say, will normally produce a lift force, L, acting on the elevator through its centre of pressure P, as depicted in section in Fig.9.26. If the elevator hinges is at A, this lift force produces a nose-down hinge moment, which is the restoring moment which the pilot has to counterbalance in order to keep the elevator down. If the hinge is set further back at B, say, then for the same elevator deflection and elevator lift increment, the hinge moment will be much less. Thus aerodynamic balance is provided by the simple process of setting back the hinge. It would appear that the hinge could be set as near to as one would wish to P, and so provide any required amount of aerodynamic balancing. However, P is not necessarily a fixed point, and in general moves forward with increasing elevator deflection. If the hinge is set too far back, there is a danger that under some conditions P might be forward of the hinge, so that the resulting hinge moment would be nose-up. This would represent an unstable condition, in which the hinge moment co-efficient b_2 is positive. Such a condition known as overbalance, and must be avoided. There is thus a limit to the amount of balance one can achieve by setting back the hinge, and there are also structural reasons why the hinge should not be set too far back.

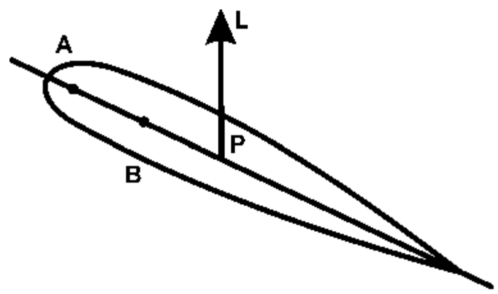


Fig.9.26.

ii. Sealed nose balance

With this type of balance, sketched in Fig.9.27 below a plate projects forward from the nose of the control surface. This plate, or tongue, is joined to the main part of the wing, tailplane or fin by a loose fold of impermeable fabric, which constitutes a seal between the regions above and below the control, or on the two sides of the control surface, in the

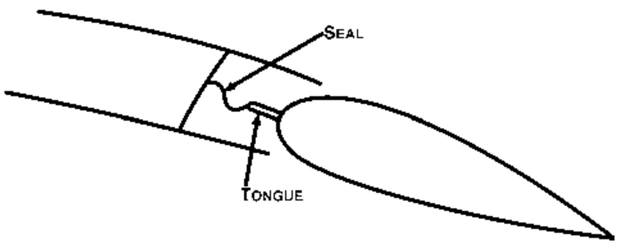


Fig.9.27.

case of a rudder. The space above the tongue in the case of an elevator, say, is open to the air above to the tailplane and similarly the space below the tongue is open to the air below the tail. When the elevator is deflected downwards lift is increased on the elevator. This is the result of generally reduced pressure on the upper surface, increased pressure on the lower surface. In particular, the pressure just above the tongue is low, and just below it is high, so that there is an upware force on the tongue which provides a nose-up hinge moment, which is the required balancing moment.

iii. Geared tabs

In this case, the control surface is fitted with a tab in such a way that when the main control moves downwards, say, the tab is automatically deflected upwards. This tab deflection produces a downward force on the tab, and hence a nose-up control hinge moment, which helps to keep the control down. Fig.9.28 shows the device in the neutral position and with the control deflected downwards. The tab is geared to the main control, so that its deflection is always proportional to the control deflection, in accordance with a particular gear ratio. Thus the balancing effect increases with control deflection, but there is little danger of overbalancing.

Another related device is the so-called servo-tab, which is designed to provide all the hinge moment required to deflect the control. The principle is that when the pilot moves the control column this controls the tab directly. The deflected tab then creates a hinge moment which deflects the control. This is not strictly an aerodynamic balance, although the stick force necessary to deflect the tab is much less than that which would be required to deflect the main control directly, so that the effect is the same. There are also various kinds of spring tabs which , in effect , distribute the stick force

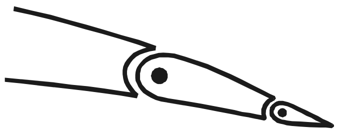


Fig.9.28 (a)

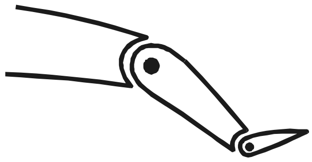


Fig.9.28 (b)

between the tab and the main control in varying rations. Thus, the servo-tab is at one extreme and the geared balance tab is at the other. The ratio in which the load is distributed between tab and main control is simply determined by the mechanical linkages in the control circuit. The correct design of these linkages yields the required value for the hinge moment coefficient b_2 , i.e., the correct amount of aerodynamic balancing.

iv. Frise aileron

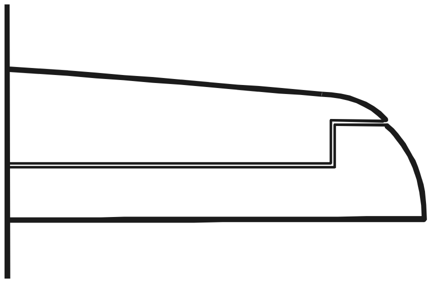
The Frise type of balance consists in designing the shape of the control surface in such a manner that when it is deflected upwards the nose projects below the level of the main surface of the wing, say. The consequence, is a pressure on the nose which provides a nose-down hinge moment, and so reduces the restoring nose-up moment. Such a design has already been described before in connection with its use to offset adverse aileron yaw, and depicted in Fig.9.29.

Fig.9.29.

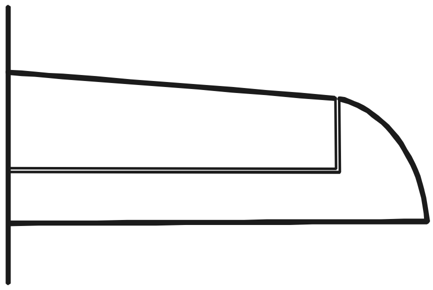
The danger with this type of balance is that if large control deflections are used over balance is almost certain to result. For this reason, it is only used on ailerons, where two geared controls, the port aileron and the starboard aileron, move in opposite directions. The design is such that when the control moves downwards the nose does not project above the main wing surface, so that the downgoing aileron is not balanced at all. Thus, even if the upgoing aileron is overbalanced for large deflections, the combination of the two ailerons will generally remain stable. As we have seen, the extra drag on the side where the lift is reduced by upward aileron deflection also performs a useful function in offsetting adverse aileron yaw.

v. Horn balance

A horn balance consists simply in part of the control surface projecting far forward of the hinge line, so that it can be regarded as effectively as a heavily set-back hinge over part of the span of the control surface. When an elevator, say, is deflected downwards, there is a lift force on the horn which, being forward of the hinge line, provides a nose-up hinge moment to partly offset the nose-down moment on the rest of the elevator. The horn may be shielded or unshielded, the two designs being such as those depicted in plan view in Fig.9.30 (a) and (b).



(a)



(a)

Fig.9.30.

AEROELASTIC EFFECTS

We have hitherto made the assumption that the aircraft structure is perfectly rigid, i.e., that it cannot be deformed, no matter what loads are applied to it. This is, of course, an idealization which is not realized in practice. In fact, the airframe is subjected to aerodynamic loads which depend on its external shape. The consequence of these loads is to strain the airframe, so that its shape is slightly modified in some respects, and this change in shape leads to a modification of the aerodynamic loads. This interaction between the aerodynamic loads and the elastic strain of the airframe is known as aeroelasticity. At relatively low speeds, the aerodynamic forces are relatively small, and the resulting, strain of the airframe produces only negligible effects. At higher speeds, the aerodynamic loads and the consequent strain are correspondingly greater and aeroelastic effects may be important, though it must be emphasized that they do not necessarily become important at the same speeds as those at which compressibility effects first arise. We shall not present a general discussion of aeroelasticity but we shall give an account of three important phenomena which could arise as a result of aeroelastic effects.

I Wing Torsional Divergence

Consider a wing at incidence. The pressure distribution, with the main loads located near the nose, is such as to cause the wing to twist in the nose-up sense. Because the structure is not perfectly rigid, it does in fact twist, and its shape becomes distorted relative to the wing root section. It twists about an axis known as the torsional axis of the wing, which is usually aft of the line of aerodynamic centres. This twist, by increasing the effective incidence, creates a lift increment which then acts forward of the torsional axis, so that the effect is unstable, in the sense that the more it twists the bigger is the twisting moment tending to cause it to twist still further. However, the tendency to twist is, of course, resisted by elastic forces due to the stiffness of the structure. This resistance to twist increases rapidly with the amount of twist, or strain, until it balances the aerodynamic twisting couple and equilibrium, is reached.

However, as the speed increases, the aerodynamic forces increase rapidly, in proportion to V^2, and therefore so also does the twisting moment. The elastic stiffness is not affected by speed, and so the amount of twist increases with speed. Eventually, a speed is reached at which the elastic resistance to twist is only just sufficient to counteract the twisting moment, and equilibrium is only achieved with the wing at breaking point. This speed is a critical speed called the wing torsional divergence speed, and any increase in speed above this value will result in structural failure-the wing will break off.

There is no real remedy for this situation. It is therefore essential to be able to predict this critical divergence speed, and to make the airframe strong and stiff enough to ensure that it is higher, by a substantial safety margin, than any speed which will ever be achieved in any condition in flight.

Similar considerations apply to wing bending, and there is a critical wing flexural divergence speed which, similarly, must be made so high that it will not be reached in flight.

II Control Surface Flutter

Consider now an elevator which is hinged along a line which is well forward of the centre of gravity of the elevator, as depicted in Fig.9.31. Suppose the aircraft is in a steady condition when a sudden disturbance causes the tail, and with it the elevator, to be displaced upwards. Because of its inertia, the elevator will tend to rotate about its hinge line in such a way that downward deflection results. This will produce more lift on the tail, and thus the fuselage, which is an elastic structure, will bend, the tail moving further upwards. This will cause the elevator to deflect still further downwards, and so the process continues. Resistance to this motion is, of course, provided by the elastic stiffness of the control circuit, which, if the stick is fixed, opposes the motion of the elevator. Similarly, the stiffness of the fuselage provides resistance to the tendency to bend. However, an oscillation may develop which, in certain circumstances is undamped, depending on the interaction of the inertia, aerodynamic and elastic effects described above. This oscillation is known as elevator flutter, and, if not checked, will ultimately cause structural failure.

There is generally a simple remedy. Suppose that the distribution of the mass of the elevator could be altered, so that its centre of gravity lies on the hinge line. Then the initial disturbance of the hinge line will not create any elevator deflection, and flutter will not occur. Further, if the centre of gravity of the elevator is forward of the hinge line, then upward movement of the hinge will cause the elevator to be deflected upwards, creating a downward increment in tail lift which, so far from creating the kind of instability which gives rise to flutter, acts as a damping factor and helps to eliminate the effects of the disturbance.

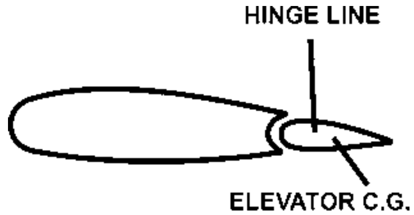


Fig.9.31.

The careful arrangement of the mass distribution of the elevator, in order to eliminate the inertia-aeroelastic coupling which produces flutter, is called mass-balancing. It is not to be confused with aerodynamic balancing, to which it is totally unrelated. In the case of the simple kind of flutter just described, the object can be achieved by simply attaching a tongue to the nose of the control, and placing a mass on the end of it. This has the desired effect of moving the elevator centre of gravity forward. To eliminate the possibility of flutter even after a much more general kind of disturbance, more complete mass-balancing, involving the moments and products of inertia about the relevant axes, may be necessary.

The aileron may be caused to flutter in just the same way as the elevator, with the bending wing playing the role corresponding to that of the bending fuselage. An example of a more complicated motion which might occur would be a combination of wing torsional and flexural oscillations due to aileron inertia. Again, the remedy lies in mass-balancing of the aileron.

a. Mass balancing

Control surfaces are often balanced in quite a different sense. A mass is fitted in front of the hinge. This is partly to provide a mechanical balancing of the mass of the control surface behind the hinge but may also be partly to help prevent an effect known as 'flutter' which is liable to occur at high speeds (Fig.9.32). This flutter is a vibration which is caused by the combined effects of the changes in pressure distribution over the surface as the angle of attack is altered, and the elastic forces set up by the distortion of the structure itself. All structures are distored when loads are applied. If

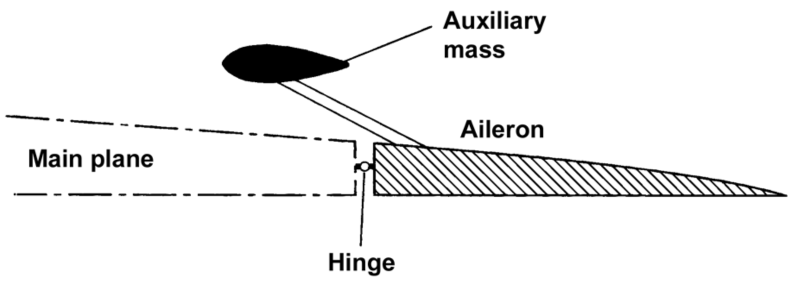


Fig.9.32.

the structure is elastic, as all good structures must be, it will tend to spring back as soon as the load is removed, or changes its point of application. In short, a distorted structure is like a spring that has been wound up and is ready to spring back. An aeroplane wing or fuselage can be distorted in two ways, by bending and by twisting, and each distortion can result in an independent vibration. Like all vibrations, this flutter is liable to become dangerous if the two effects add up. The flutter may affect the control surfaces such as an aileron, or the main planes, or both. The whole problem is very complicated, but we do know of two features which help to prevent it - **a rigid structure and mass balance of the control surfaces.** When the old types of aerodynamic balance were used, e.g. the inset hinge or horn balance, the mass could be concealed inside the forward portion of the control surface and thus two birds were killed with one stone; but when the tap type of balance is used alone the mass must be placed on a special arm sticking out in front of the control surface. In general, however, the problems of flutter are best tackled by increasing the rigidity of the structure and control-system components.

Large aircraft and military types now invariably have powered controls and these are much less sensitive to problems of flutter as the actuating system is very rigid.

Perhaps it should be emphasised that the **mass** is **not** simply a weight for the purpose of balancing the control surface statically, e.g. to keep the aileron floating when the control mechanism is not connected; it may have this effect, but it also serves to alter the moments of inertia of the surface, and thus alter the period of vibration and the liability of flutter. It may help to make this clear if we realise that mass balance is just as effective on a rudder, where the weight is not involved, as on an elevator or aileron.

On old military biplane aircraft, the exact distribution of mass on the control surfaces was so important that strict orders had to be introduced concerning the application of paint and dope to these surfaces. It is for this reason that the red, white and blue stripes which used to be painted on the rudders of Royal Air Force machines were removed (they were later restored, but only on the fixed fin), and why the circles on the wings were not allowed to overlap the ailerons. Rumour has it that when has it that when this order was first promulgated, some units in their eagerness to comply with the order, but ignorant as to its purpose, painted over the circles and stripes with further coats of dope!

b. Control reversal

Consider a wing fitted with an aileron. The aileron is designed to produce positive increments in wing lift when it is deflected downwards. But downward deflection modifies the pressure distribution over wing and aileron in such a way that this increment in lift act well aft of the wing torsional axis, so that as a result the wing is twisted in the nose-down sense. This causes a reduction in effective incidence, and so a reduction in lift. However, the elastic stiffness of the wing creates a resistance to the wing twist, so that in the equilibrium condition the reduction in lift due to twist is generally less than the increase due directly to aileron deflection. Thus downward aileron deflection gives increase in lift overall, as expected, although the aileron effectiveness is reduced as a result of the twisting of the wing.

However, with increase in speed, the twisting moment, which arises from the aerodynamic loads, increase rapidly, while the elastic resistance is not changed, so that the wing twist more and more, for the same aileron deflection. There is a critical speed at which the loss in lift due to twist at a given aileron deflection is only just equal to the increase in lift due to that deflection and the aileron is then totally ineffective. Above this speed, aileron deflection downwards will actually results in reduced lift overall and deflection, of the ailerons will result in a roll in the opposite sense to that which results from the same deflection at lower speeds. This phenomenon is known as aileron reversal and the speed at which it occurs is called the aileron reversal speed. This control result which can also occur in relation to the elevator and rudder, must be avoided, so that it is a design requirement that control reversal speeds must be higher than any physical speed to be achieved in flight.

ADVERSE YAW
Consider first the effect of aileron deflection on yawing moment. Suppose that the starboard aileron goes up, and the port aileron down, in order to initiate a turn to starboard. The result of the downward deflection on the port wing is to increase the profile drag of that wing, while, conversely, the profile drag is slightly reduced on the starboard side. There is also more induced drag on the port side because the lift on that side is higher. The result is a yawing moment to port, i.e., in the sense opposite to that of the desired turn. This phenomenon is known as adverse aileron yaw, and would be measured in terms of the aerodynamic dericative N_{ξ}, which is clearly positive, from the above argument. It may be alleviated by the use of ailerons of special design, known as Frise ailerons. The typical section shape of such ailerons is depicted in Fig.9.33(a), which shows an aileron in the neutral, up and down positions. When an aileron is deflected upwards, as in Fig.9.33(b), the nose projects below the main wing surface, spoiling the flow and increasing

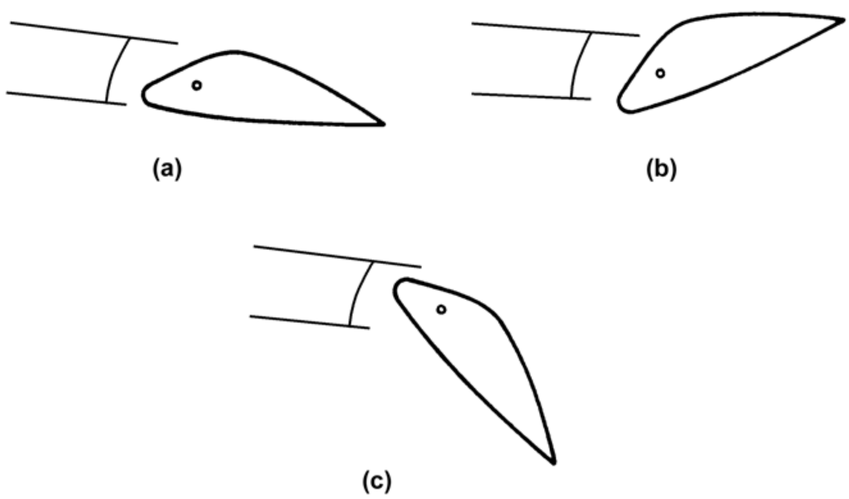


Fig.9.33.

the drag. When it is deflected downwards, as in Fig.9.33(c), there is no such projection, and no drag increment. The increased drag on the side on which the aileron is deflected upwards tends to offfset to some extent the adverse aileron yaw.

'Frise', or other specially shaped ailerons (Fig.9.34). This is a patented device, the idea being so to shape the aileron that when it is moved downwards the complete top surface of the main plane and the aileron will have a smooth, uninterrupted contour causing very little drag, but when it is moved upwards the aileron, which is of the balanced variety, will project below the bottom surface of the main plane and cause excessive drag. This method has the great advantage of being simple, and it undoubtedly serves to decrease the bad yawing effect of the ailerons, and therefore it is often used. Unfortunately, its effects are not drastic enough.
Differential ailerons (Fig.9.35). Here, again, is a delightfully simple device suffering only from the same defect that, although it provides a step in the right direction, it does not go far enough to satisfy our needs. Instead of the two ailerons moving equally up and down, a simple mechanical arrangement of the controls causes the aileron which moves upwards to move through a larger angle than the aileron which moves downwards, the idea being to increase the drag and decrease the lift on the wing with the up-going aileron, while at the same time the down-going aileron, owing to its smaller movement, will not cause excessive drag.

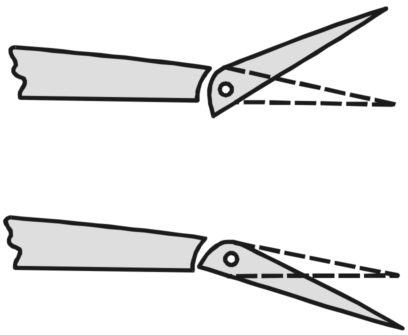


Fig.9.34. Frise ailerons

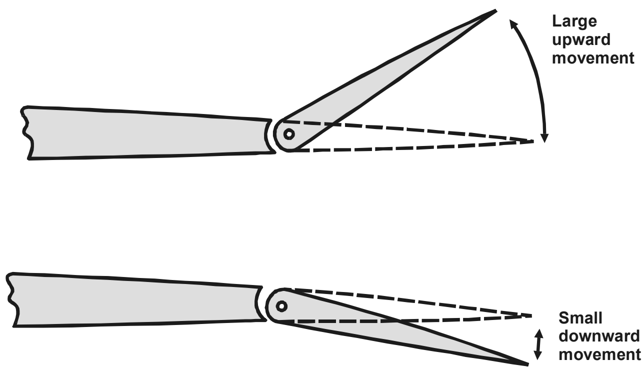


Fig.9.35. Differential ailerons

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CHAPTER: 10
TRANSONIC FLIGHT
[SUBSONIC, SONIC AND SUPERSONIC FLOW PATTERNS]

We have already talked about flight at subsonic, transonic, and supersonic speeds, and it should now be clear that the problems of flight are quite different in these three regions, but the dividing lines between the regions are of necessity somewhat vague. Fig.10.1 shows the subsonic region as being below a Mach Number of 0.8, the transonic region from M 0.8 to M 1.2, and the supersonic region above M 1.2. There are arguments in favour of considering the transonic region as starting earlier, say at a Mach Number of about 0.7 or near the point marked in the figure as the critical Mach Number, and extending up to say a Mach Number of 1.6 or even 2.0. In terms of sea-level speeds this would mean defining subsonic speeds as being below 450 knots, transonic speeds as 450 up to 1000 or even 1200 knots, and supersonic speeds above that.

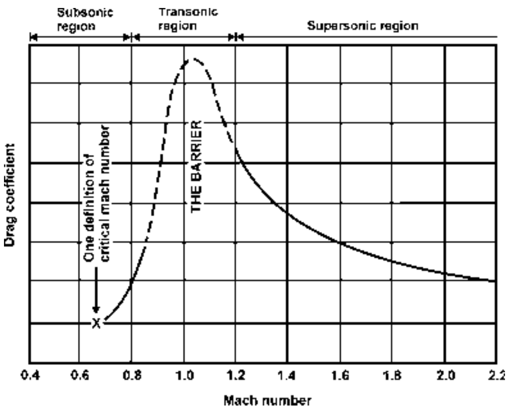


Fig.10.1. Devel

Perhaps the best definition of the three regions is to say that the subsonic region is that in which all the airflow over all parts of the aeroplane is subsonic, the transonic region is that in which some of the airflow is subsonic and some supersonic, and the supersonic region is that in which all the airflow is supersonic. Once again we are in trouble if we take our definition too literally. Even at very high speeds we may have local pockets of subsonic flow - just in front of a blunt nose for example. So the space shuttle would only be transonic even at the fastest point of re-entry! Also with this definition we are none the wiser as to the speeds or Mach Numbers at which each regions begins or ends; the beginnings and endings will of course be quite different for different aeroplanes.

In this chapter our main concern is with speeds in the transonic range, and particularly in the narrow range between Mach Numbers of 0.8 and 1.2. This range, as is probably already evident, presents us with some of the most baffling but fascinating problems of flight; it is the range in which most of the change takes place, the change from apparent incompressibility to actual compressibility, the gradual substitution of supersonic flow for subsonic flow; it is the range about which we are even now most ignorant.

FORMATION OF SHOCK WAVES

In an earlier paragraph we described how a shock wave is formed at a speed of about three-quarters of the speed of sound, i.e. at about M = 0.75. On a symmetrical wing at zero angle of attack the incipient shock wave appears on both top and bottom surfaces simultaneously, approximately at right angles to the surfaces, and, as one would expect, at about the point of maximum camber (Fig. 10.2b). On a wing at a small angle of attack, even if the aerofoil section is symmetrical, the incipient shock wave appears first on the top surface only (Fig. 10.3) - again as one would expect, because it is on the top surface that the speed of the airflow first approaches the speed of sound.

Fig.10.2 shows how the shock-wave pattern changes (on a symmetrically shaped sharp-nosed aerofoil at zero angle of attack) as the speed of airflow is increased from subsonic, through the transonic range to fully supersonic flow.

Between the formation of the incipient wave (at a Mach Number of about 0.75 or 0.8) and the time when the wing as a whole is moving through the air at a speed of sound (M = 1.0), the shock wave tends to move backwards, but in doing so becomes stronger and extends farther out from the surface, while there is even more violent turbulence behind it (Fig.10.2c). At a speed just above that of sound another wave appears, in the form of a bow wave, some distance ahead of the leading edge; and the original wave, which is now at the trailing edge, tends to become curved, and shaped rather like a fish tail (Fig.10.2d). As the speed is further increased the bow wave attaches itself to the leading edge, and the angles formed between both waves and the surfaces become more acute (Fig.10.2e). Still further increases of speed have little effect on the general shock-wave pattern - but here we are trespassing on supersonic flight, which is the subject of the next chapter.

At each wave there is a sudden increase of pressure, density and temperature, a decrease in velocity, and a slight change in direction of the airflow. The thickness of a shock wave, through which these changes take place, is only of the order of 2 to 3 thousandths of a millimetre - they look thicker on photographs because it is not possible to get a perfectly plane shock wave in the experiment. The changes at the shock wave are irreversible, which is another way of saying that the high pressure behind the wave cannot be communicated to the lower pressure in front - messages can only go down stream. It is interesting to note, however, that the incipient waves only extend a short distance from the surface, and

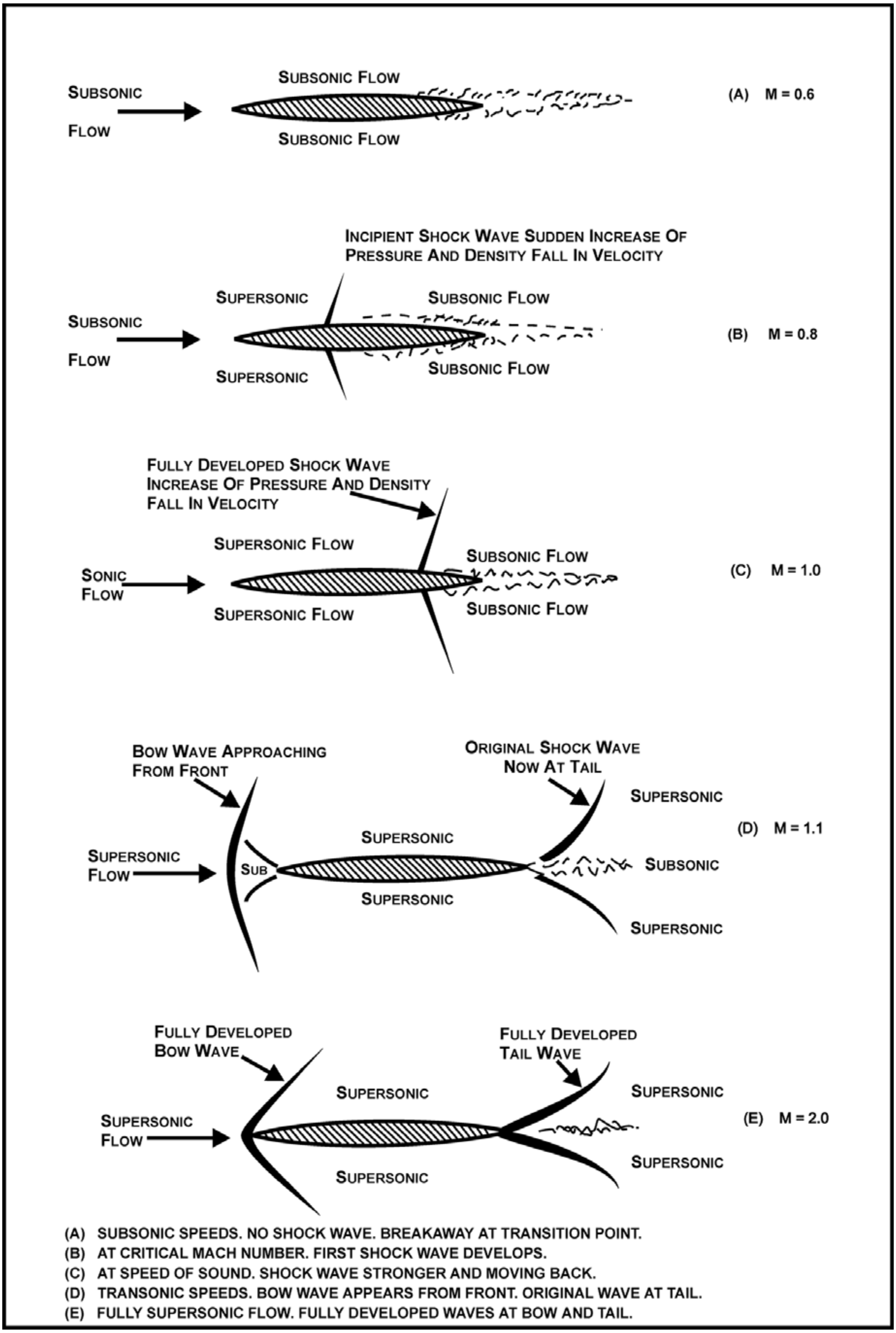


Fig.10.2. Development of shock waves at increasing Mach numbers

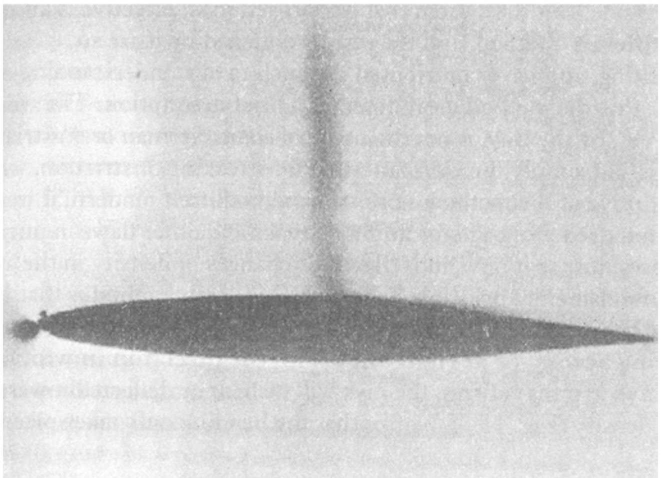
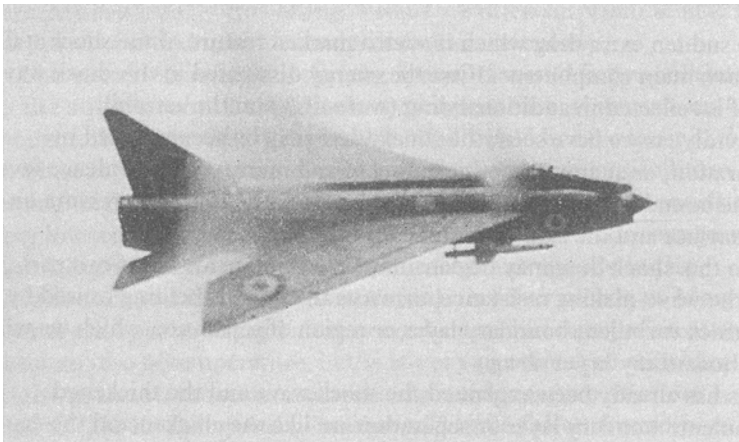


Fig.10.3. Incipient shock wave
(By courtesy of the Shell Petroleum Co Ltd)
An incipient shock wave (taken by schlieren photography) has formed on the upper surface; the light areas near the leading edge are expansion regions, separated by the stagnation area which appears as a dark blob at the nose



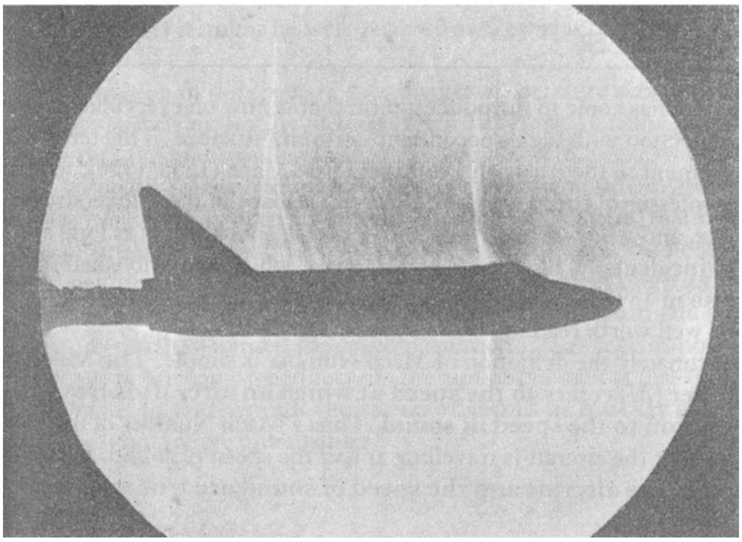


Fig.10.4. Shock waves
(By courtesy of the former British Aircraft Corporation, Preston)
Top: Lightning at M 0.98; low pressure regions above canopy and wing cause condensation, the evaporation to the rear marks the shock wave.
Bottom: Schlieren photograph of model of Lightning at M 0.98; note the extraordinarily close resemblance to actual flight.

leaks are possible round the ends of the waves; as speed increases the waves extend and there is less and less possibility of such leaks. It is interesting to note, too, that the decrease in velocity, which occurs behind the shock wave, means that when an aircraft is moving through the air, and a shock wave is formed, the air behind the shock wave begins to move in the direction in which the aircraft is travelling.

In addition to showing the shock-wave patterns, Fig.10.2 also indicates the areas in which the flow is subsonic or supersonic. In (a) at M = 0.6 it is all subsonic (clearly we are still in the subsonic region); at M = 0.8 the flow immediately in front of the shock wave is supersonic, but all the remainder is subsonic (we are now in the transonic region with both types of flow); at M = 1 the area of supersonic flow has increased but the flow behind the shock wave is still subsonic (as we shall learn later it is always subsonic behind a shock wave that is at right angles to the flow, it can only be supersonic behind an inclined or oblique shock wave); at M = 1.1 nearly all the flow is supersonic, but there are still small regions of subsonic flow, immediately in front of the leading edge at what is called the stagnation point where the flow is brought to rest, and immediately behind the trailing edge (we are still in the transonic region, but not for much longer); at M = 2 the flow is all supersonic - we are through the barrier (though to be strictly correct, unless the bow wave is actually attached to the leading edge, which will only happen if the edge is very sharp, there will still be a small area of subsonic flow at the stagnation point between the bow wave and the leading edge; and of course in the boundary layer itself the air immediately next to the surface is at rest relative to the surface, and most of the remainder of the airflow in the boundary layer is subsonic).

The figure also shows how the extent of the separated region, or thickened boundary layer tends to decrease with increasing Mach Number, and this suggests that as wave drag becomes relatively less so. This may also give a clue to the decrease in drag coefficient as we pass through the barrier (Fig.10.1).

The reader should now be able to draw for himself the shock patterns, corresponding to those of Fig.10.2 for an aerofoil inclined at a small angle of attack, and the exercise in doing so will help him to appreciate how and why shock waves are formed.

Fig.10.4 is a remarkable example of condensation and shock waves on an aeroplane in flight with, below, a schlieren photograph of shock waves on a model of the same aircraft. More shock waves on an aerofoil are shown in Fig.10.5.

When supersonic flow is clearly established, all changes in velocity, pressure, density, flow direction, etc., take place quite suddenly and in relatively confined areas. The areas of flow change are generally distinct and the phenomena are referred to as “wave” formations. All compression waves occur suddenly and are wasteful of energy. Hence, the compression waves are distinguished by the sudden “shock” type of behaviour. All expansion waves are not so sudden in their occurrence and are not wasteful of energy like the compression shock waves. Various types of waves can occur in supersonic flow and the nature of the wave formed depends upon the airstream and the shape of the object causing the flow change. Essentially, there are three fundamental types of waves formed in supersonic flow: 1. the*oblique* shock wave (compression), 2. the *normal* shock wave (compression), 3. the *expansion* wave (no shock).

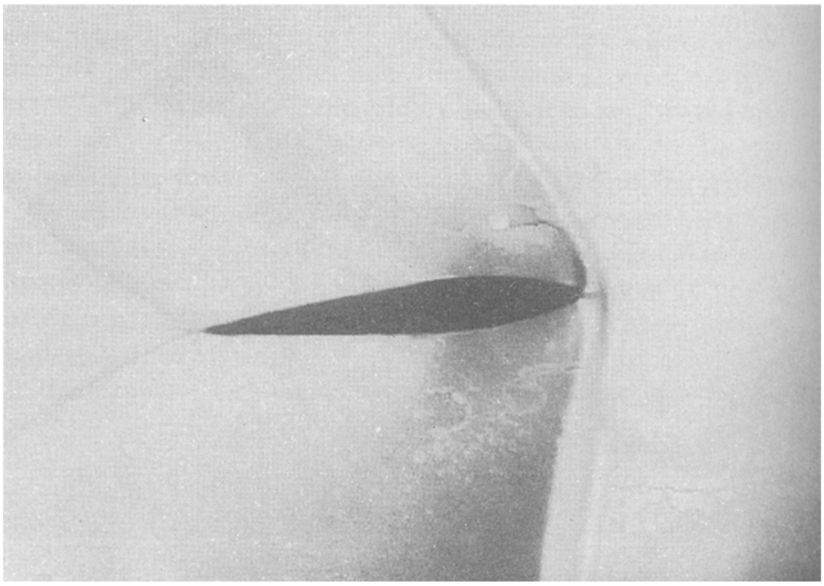


Fig.10.5. Shock waves
Shock waves from an aerofoil at incidence to the flow. Note the stronger leading edge shock on the underside. The wave in the top left-hand corner is a reflection from the wind-tunnel wall.

i. Normal Shock Wave

If a bluntnosed object is placed in a supersonic airstream the shock wave which is formed will be detached from the leading edge. This detached wave also occurs when a wedge or cone angle exceeds some critical value. Whenever the shock wave forms perpendicular to the upstream flow, the shock wave is termed a "normal" shock wave and the now immediately behind the wave is *subsonic*. Any relatively blunt object in a supersonic airstream will form a normal shock wave immediately ahead of the leading edge slowing the airstream to subsonic so the airstream may feel the presence of the blunt nose and flow around it. Once past the blunt nose the airstream may remain subsonic or accelerate back to supersonic depending on the shape of the nose and the Mach number of the free stream.

In addition to the formation of normal shock waves described above, this same type of wave may be formed in an entirely different manner when there is no object in the supersonic airstream. It is particular that whenever a supersonic airstream is slowed to subsonic without a change in direction a normal shock wave will form as a boundary between the supersonic and subsonic regions. This is an important fact since aircraft usually encounter some "compressibility effects" before the flight speed is sonic. Fig.10.6 illustrates the manner in which an airfoil at high subsonic speeds has local flow velocities which are supersonic. As the local supersonic flow moves aft, a normal shock wave forms slowing the flow to subsonic. The transition of flow from subsonic to supersonic is smooth and is not accompanied by shock waves if the transition is made gradually with a smooth surface. The transition of flow from supersonic to subsonic without direction change *always* forms a normal shock wave.

A supersonic airstream passing through a normal shock wave will experience these changes:

1. The airstream is slowed to subsonic; the local Mach number behind the wave is approximately equal to the reciprocal of the Mach number ahead of the wave-e.g., if Mach number ahead of the wave is 1.25 the Mach number of the flow behind wave is approximately 0.80.
2. The airflow direction immediately behind the wave is encharged.
3. The static pressure of the airstream behind the wave is increased greatly.
4. The density of the airstream behind the wave is increased greatly.
5. The energy of the airstream (indicated by total pressure-dynamic plus static) is greatly reduced. The normal shockwave is very wasteful of energy.

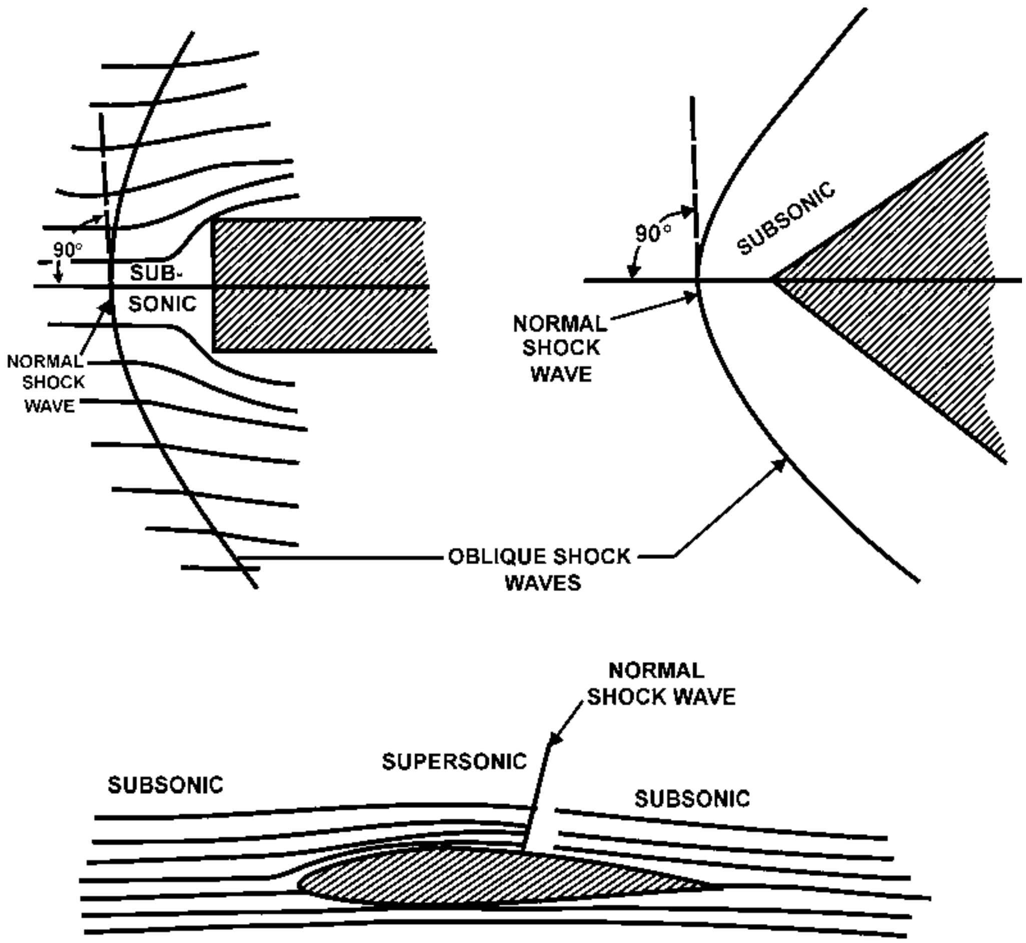


Fig.10.6. Normal Shock Wave Formation

ii. Oblique Shock Wave

Consider the case where a supersonic airstream is turned into the preceding airflow. Such would be the case of a supersonic flow "into a corner" as shown in Fig.10.7. A supersonic airstream passing through the Oblique shock wave will experience these changes:

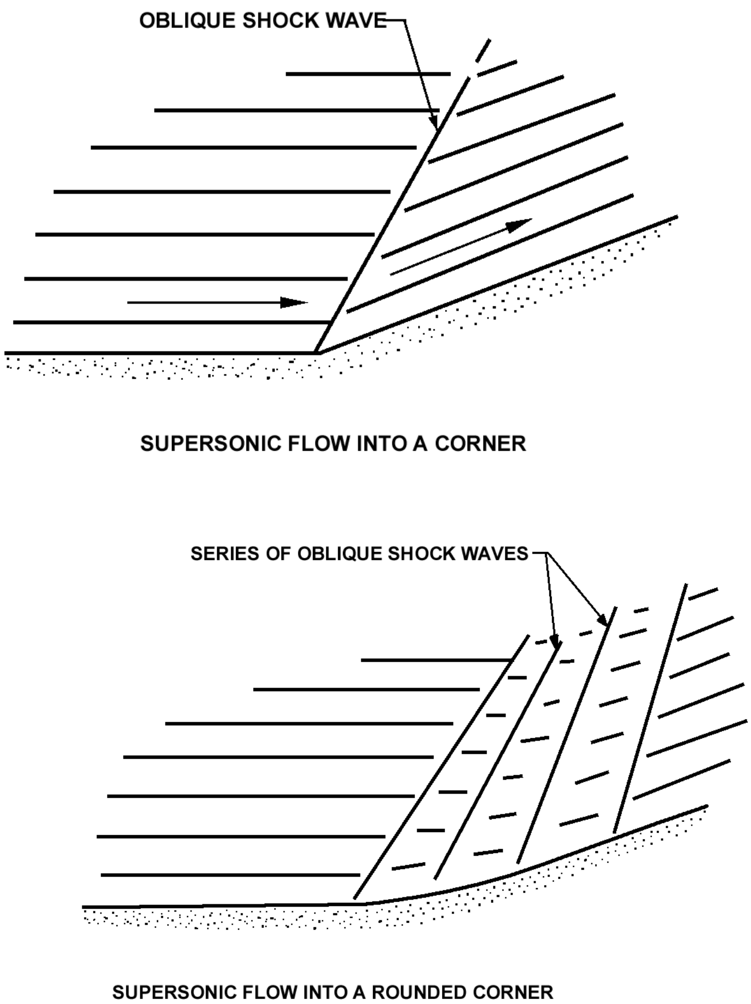


Fig.10.7. Oblique Shock Wave Formation

1. The airstream is slowed down; the velocity and Mach number behind the wave are reduced but the flow is still supersonic.
2. The flow direction is changed to flow along the surface.
3. The static pressure of the airstream behind the wave is increased.
4. The density of the airstream behind the wave is increased.
5. Some of the available energy of the airstream (indicated by the sum of dynamic and atmospheric pressure) is dissipated and turned into unavailable heat energy. Hence, the shock wave is wasteful of energy.

A typical case of oblique shock wave formation is that of a wedge pointed into a supersonic airstream. The oblique shock wave will form on each surface of the wedge and the inclination of the shock wave will be a function of the free stream Mach number and the wedge angle. As the free stream Mach number increases, the shock wave angle decreases; as the wedge angle increases the shock wave angle increases, and, if the wedge angle is increased to some critical amount, the shock wave will detach from the leading edge of the wedge. It is important to note that detachment of the shock wave will produce *subsonic* flow-immediately after the central portion of the shock wave. Fig.10.8 illustrates these typical flow patterns and the effect of Mach number and wedge angle.

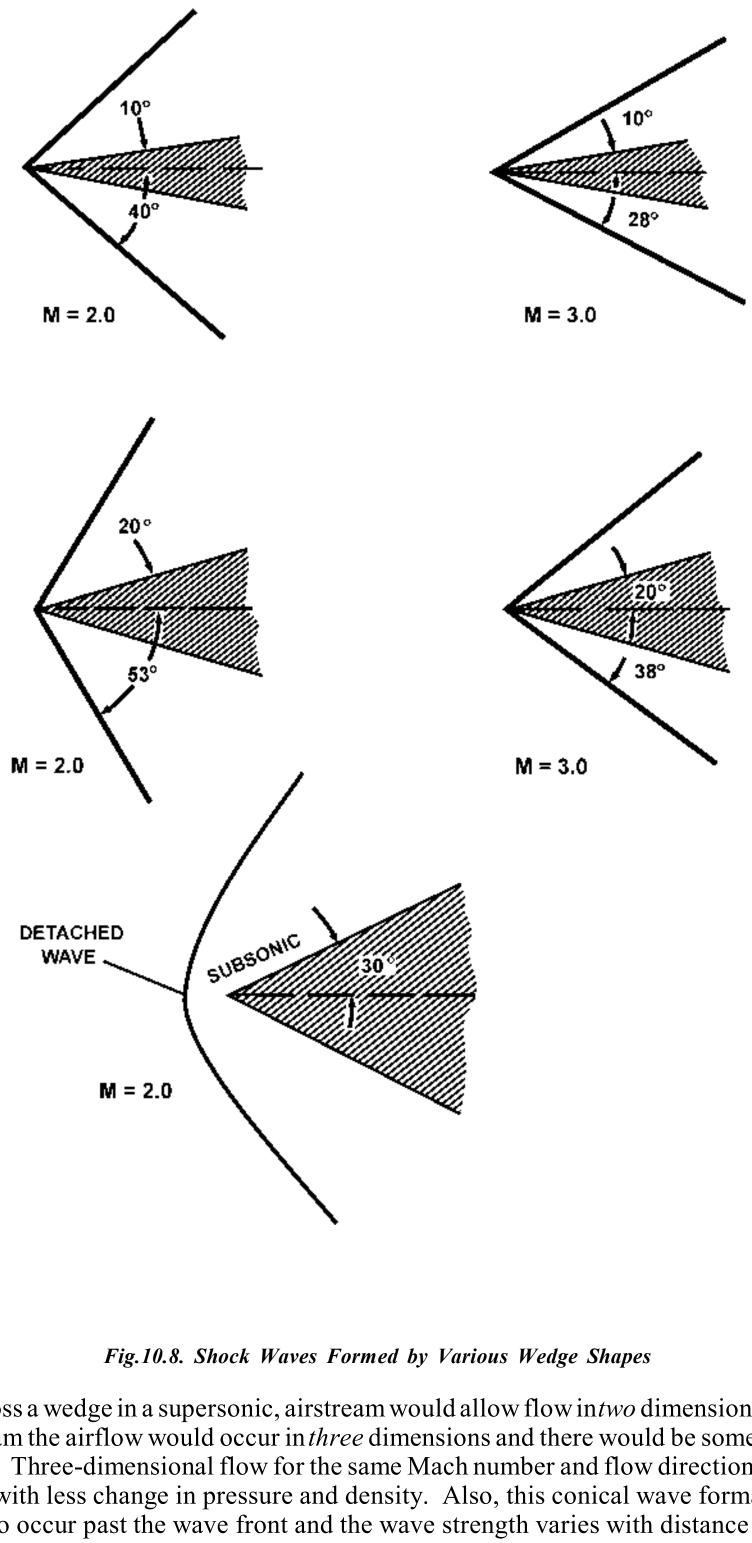


Fig.10.8. Shock Waves Formed by Various Wedge Shapes

The previous flow across a wedge in a supersonic, airstream would allow flow in *two* dimensions. If a cone were placed in a supersonic airstream the airflow would occur in *three* dimensions and there would be some noticeable differences in flow characteristics. Three-dimensional flow for the same Mach number and flow direction change would produce a weaker shock wave with less change in pressure and density. Also, this conical wave formation allows changes in airflow that continue to occur past the wave front and the wave strength varies with distance away from the surface. Fig.10.9 depicts the typical three-dimensional flow past a cone.

Oblique shock waves can be reflected like any pressure wave and this effect is shown in Fig.10.9. This reflection appears logical and necessary since the original wave changes the flow direction toward the wall and the reflected wave creates the subsequent flow change to cause the flow to remain parallel to the wall surface. This reflection phenomenon places definite restrictions on the size of a model in a wind tunnel since a wave reflected back to the model would cause a pressure distribution not typical of free flight.

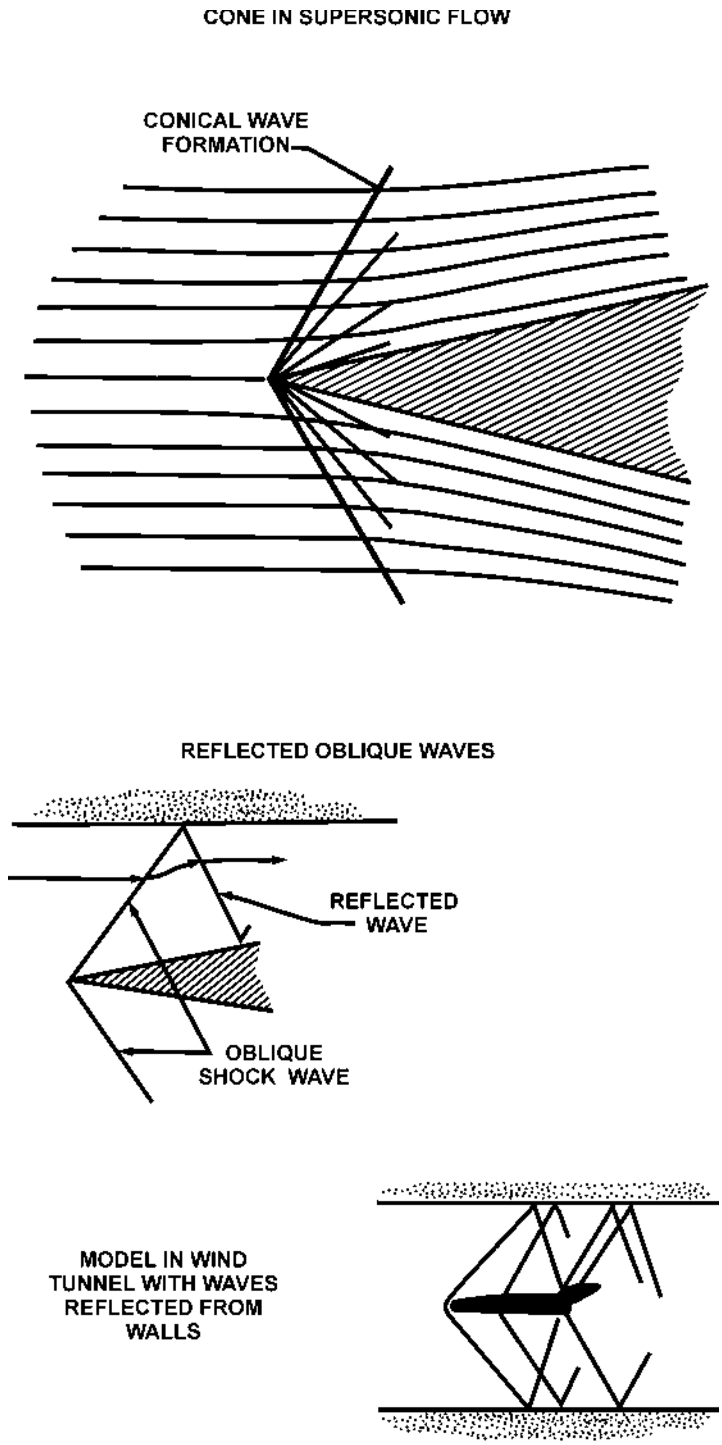


Fig.10.9. Three Dimensional and Reflected Shock Waves

iii. Bow Wave

Consider then a supersonic stream approaching the leading edge of a subsonic, i.e., round-nosed, aerofoil. In order to flow round the nose, the air has to turn through a right angle. At supersonic speeds, this impossible, and therefore a normal shock is created ahead of the nose. Behind this shock, the flow is subsonic, and can therefore flow round the nose. Within a short distance, however, the flow again accelerates to supersonic speeds. The flow pattern is then as indicated in Fig. 10.10. This shock ahead of the leading edge, called a detached bow shock.

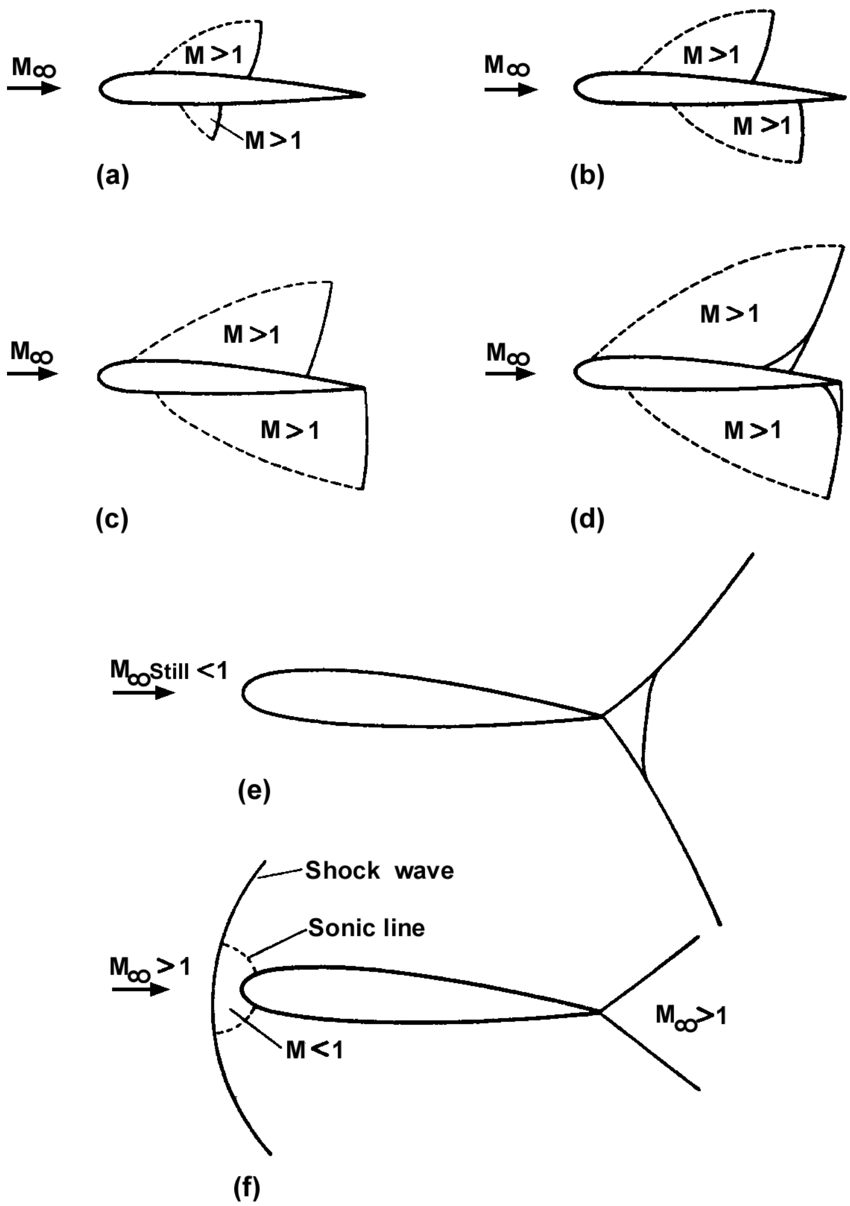


Fig.10.10.

iv. Expansion Wave

If a supersonic airstream were turned away from the preceding flow an expansion wave would form. The flow "around a corner" shown in Fig.10.11 will not cause sharp, sudden changes in the airflow except at the corner itself and thus is not actually a "shock" wave. A supersonic airstream passing through an expansion wave will experience these changes:

1. The airstream is accelerated; the velocity and Mach number behind the wave are greater.
2. The flow direction is changed to flow along the surface-provided separation does not occur.
3. The static pressure of the airstream behind the wave is decreased.
4. The density of the airstream behind the wave is decreased.
5. Since the flow changes in a rather gradual manner there is no "shock" and no loss of energy in the airstream. The expansion wave does not dissipate airstream energy.

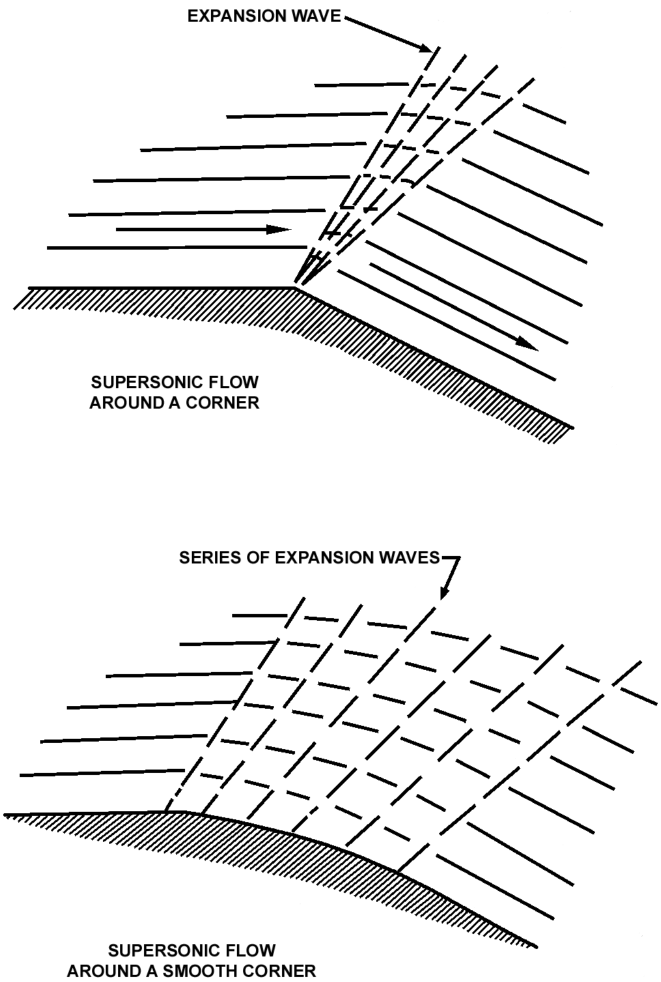


Fig.10.11. Expansion Wave Formation

The expansion wave in three dimensions is a slightly different case and the principal difference is the tendency for the static pressure to continue to increase past the wave.

The following table is provided to summarize the characteristics of the three principal wave forms encountered with supersonic flow.

Sections In Supersonic Flow

In order to appreciate the effect of these various wave forms on the aerodynamic characteristics in supersonic flow, inspect Fig.10.12. Parts (a) and (b) show the wave pattern and resulting pressure distribution for a thin flat plate at a positive angle of attack. The airstream moving over the upper surface passes through an expansion wave at the leading edge and then an oblique shock wave at the trailing edge. Thus, a uniform suction pressure exists over the upper surface. The airstream moving under neath the flat plate passes through an oblique shock wave at the leading edge then an expansion wave at the trailing edge. This produces a uniform positive pressure on the underside of the section. This distribution of pressure on the surface will produce a net lift and incur a subsequent drag due to lift from the inclination of the resultant lift from a perpendicular to the free stream.

Parts (c) and (d) of Fig.10.12 show the wave pattern and resulting pressure distribution for a double wedge airfoil at zero lift. The airstream moving over the surface passes through an oblique shock, an expansion wave, and another oblique shock. The resulting pressure distribution on the surfaces produces no net lift, but the increased pressure on the forward half of the chord along with the decreased pressure on the aft half of the chord produces a “wave” drag. This wave drag is caused by the components of pressure forces which are parallel to the free stream direction. The wave drag is in addition to the drag due to friction, separation, lift, etc., and can be a very considerable part of the total drag at high supersonic speeds.

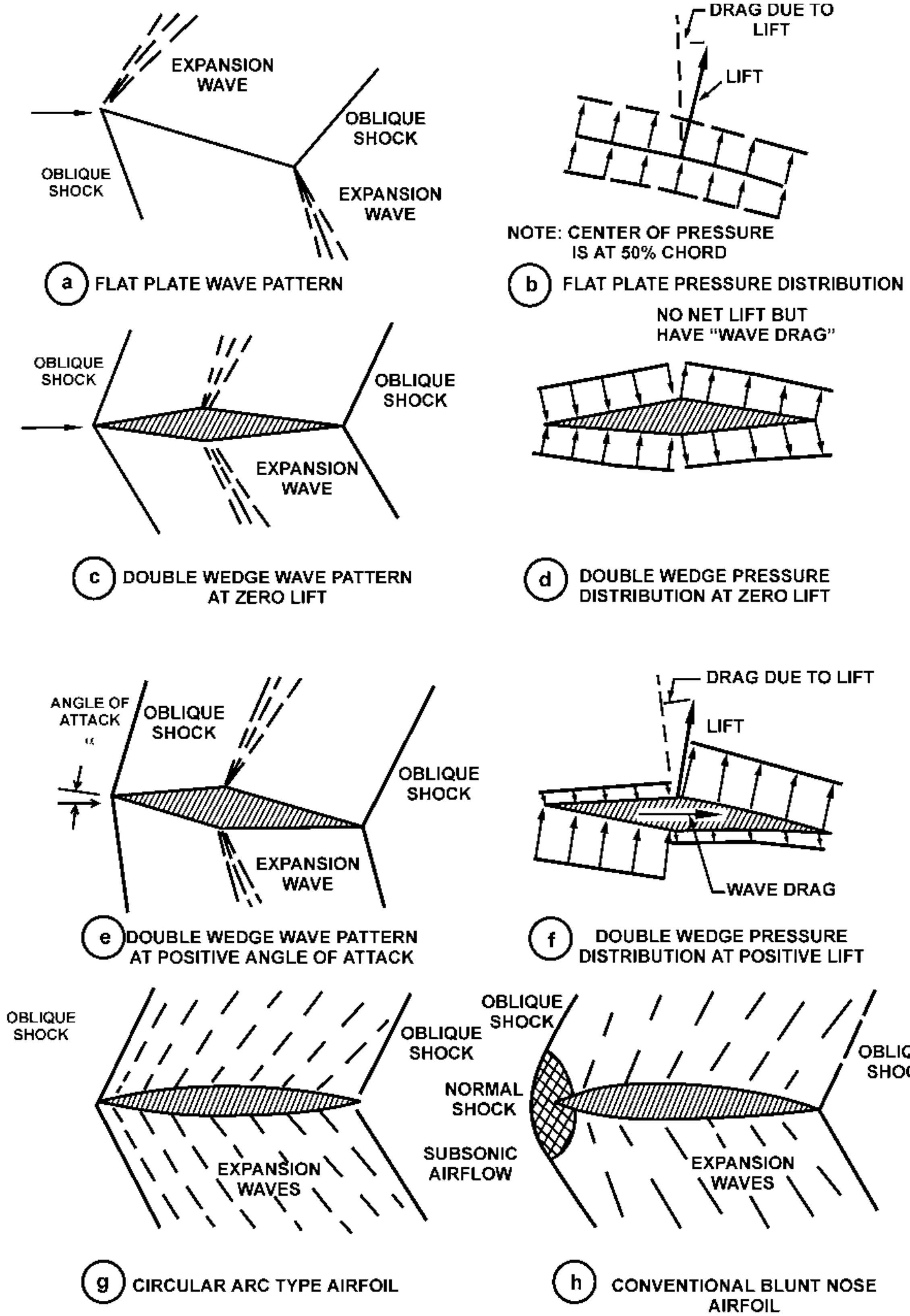


Fig.10.12. Typical Supersonic Flow Patterns and Distribution of Pressure

Parts (e) and (f) of Fig.10.12 illustrate the wave pattern and resulting pressure distribution for the double wedge airfoil at a small positive angle of attack. The net pressure distribution produces an inclined lift with drag due to lift which is in addition to the wave drag at zero lift. Part (g) cf Fig.10.12 shows the wave pattern for a circular arc airfoil. After the airflow traverses the oblique shock wave at the leading edge, the airflow undergoes a gradual but continual expansion until the trailing edge shock wave is encountered. Part (h) of Fig.10.12 illustrates the wave pattern on a conventional blunt nose airfoil in supersonic flow. When the nose is blunt the wave must detach and become a normal shock wave immediately ahead of the leading edge. Of course, this wave form produces an area of subsonic airflow at the leading edge with very high pressure and density behind the detached wave.

The drawings of Fig.10.12 illustrate the typical patterns of supersonic flow and point out these facts concerning aerodynamic surfaces in two dimensional supersonic flow:

1. All changes in velocity, pressure, density and flow direction will take place quite suddenly through the various wave forms. The shape of the object and the required flow direction change dictate the type and strength of the wave formed.
2. As always, lift results from the distribution of pressure on a surface and is the net force perpendicular to the free stream direction. Any component of the lift in a direction parallel to the windstream will be drag due to lift.
3. In supersonic flight, the zero lift drag of an airfoil of some finite thickness will include a "wave drag". The thickness of the airfoil will have an extremely powerful effect on this wave drag since the wave drag varies as the square of the thickness ratio- if the thickness is reduced 50 percent, the wave drag is reduced 75 percent. The leading edges of supersonic shapes must be sharp or the wave formed at the leading edge will be a strong detached shock wave.
4. Once the flow on the airfoil is supersonic, the aerodynamic center of the surface will be located approximately at the 50 percent chord position. As this contrasts with the subsonic location for the aerodynamic center of the 25 percent chord position, significant changes in aerodynamic trim and stability may be encountered in transonic flight.

CRITICAL MACH NUMBER

When an aerofoil at low incidence is immersed in a flow at high subsonic speeds, the flow round the aerofoil is accelerated locally so that the Mach number at some points is higher than the value in the free stream. This is especially so on the upper surface, where velocities are generally higher than on the lower surface. If the free stream Mach number is increased sufficiently, there will come a stage when the local Mach number reaches unity at some point on the upper surface, though the free stream is still subsonic. The free stream Mach number at which this first occurs is called the critical Mach number, is denoted by M_{crit}, and is the lower limit of the transonic flow regime. For values of M_∞ just greater than M_{crit}, there will then be a small region of M_∞ supersonic flow on the upper surface, terminated by a normal shock, as indicated in Fig.10.13.

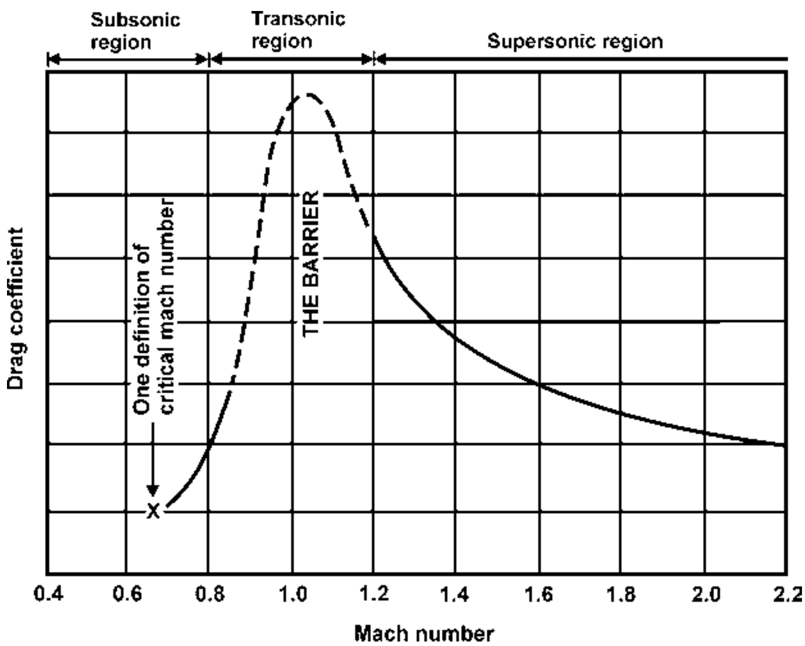


Fig.10.13. Transonic drag rise

The value of M_{crit} depends on the design of the aerofoil. We shall discuss later the use of various design features intended to raise the critical Mach number. But its value also varies with incidence. At high incidence, the accelerations on the upper surface are greater than at low incidence, so that sonic velocities will be achieved locally at lower values of M_∞. Thus increasing incidence reduces M_{crit}.

The formation of the shock wave will initially have only a small effect on lift, which will be reduced slightly because of the increased pressure over the rear upper surface. There will also be a slight increase in drag due to the loss of directed kinetic energy in the shock.

TO INCREASE CRITICAL MACH NUMBER : SWEEP BACK

The second main way of raising the critical Mach Number (and this applies only to the wings, tail, fin, and control surfaces) is sweepback - not just the few degrees of sweepback that was sometimes used, rather apologetically and for various and sometimes rather doubtful reasons, on subsonic aircraft, but 40^0, 50^0, 70^0 or more.

Sweepback of this magnitude not only delays the shock stall, but reduces its severity when it does occur. The theory behind this is that it is only the component of the velocity across the chord of the wing ($V \cos \alpha$ which is responsible for the pressure distribution and so for causing the shock wave (Fig.10.14); the component $V \sin \alpha$ along the span of the wing causes only frictional drag. This theory is borne out by the fact that when it does appear the shock wave lies parallel to the span of the wing, and only that part of the velocity perpendicular to the shock wave, i.e. across the chord, is reduced by the shock wave to subsonic speeds. As the figure clearly shows, the greater the sweepback the smaller will be the component of the velocity which is affected, and so the higher will be the critical Mach Number, and the less will be the drag at all transonic speeds of a wing of the same *t/c* ratio and at the same angle of attack.

Experiment confirms the theoretical advantages of sweepback, though the improvement is not quite so great as the theory suggests. The dotted line in Fig.10.15 shows how a wing swept back at 45^0 has a higher critical Mach Number than a straight wing at all values of *t/c* ratio, the advantage being greater for the wings with the higher values of *t/c*. Figure10.16 tell us even more; it shows that sweepback not only increases the critical Mach Number, but it reduces the rate at which the drag coefficient rises (the slope of the curve), and it lowers the peak of the drag coefficient - and 45^0 of sweepback does all this better than 30^0. Incidentally this figure also shows that, above about *M*2, sweepback has very little advantage - but that is another story and, in any case, aeroplanes cannot fly at *M*2 without first going through the transonic range.

Figure 10.18 shows various plan forms of swept-back wings.

Of course, as always, there are snags, and the heavily swept-back wing is no exception. There is **tip stalling** - an old problem, but a very important one; in the crescent-shaped wing (Fig.10.18) an attempt has been made - with some success - to alleviate this by gradually reducing the sweepback from root to tip. C_L max is low, and therefore **the stalling speed is high**, and C_L max is obtained **at too large an angle to be suitable for landing** - another old problem, and one that can generally be overcome by special slots, flaps or suction devices. There are also control problems of various kinds, and the designer doesn't like the extra bending and twisting stresses that are inherent in the heavily swept-back wing design. But whatever the problems sweepback seems to have come to stay - at least for aircraft which are designed to fly for any length of time at transonic or low supersonic speeds.

CHARACTERISTICS OF HIGH SPEED AEROFOILS

i High Swept Wings

Consider now a finite wing of zero thickness and incidence, swept back through such an angle that the leading edge lies entirely within the Mach cone generated by the leading edge of the centre section, as shown in Fig.10.14. This condition is satisfied when $\Lambda > 90^0 - \mu$.

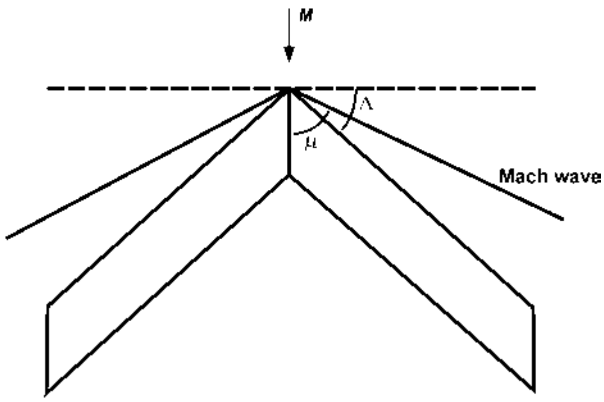


Fig.10.14.

Then

$$\tan \Lambda > \cot \mu \text{ and } \cot \mu = \sqrt{\operatorname{cosec}^2 \mu - 1} = \sqrt{M^2 - 1},$$

so that

$$\tan \Lambda > \sqrt{M^2 - 1}$$

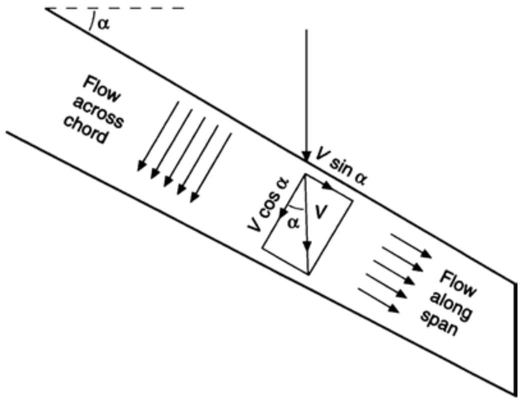


Fig.10.15.

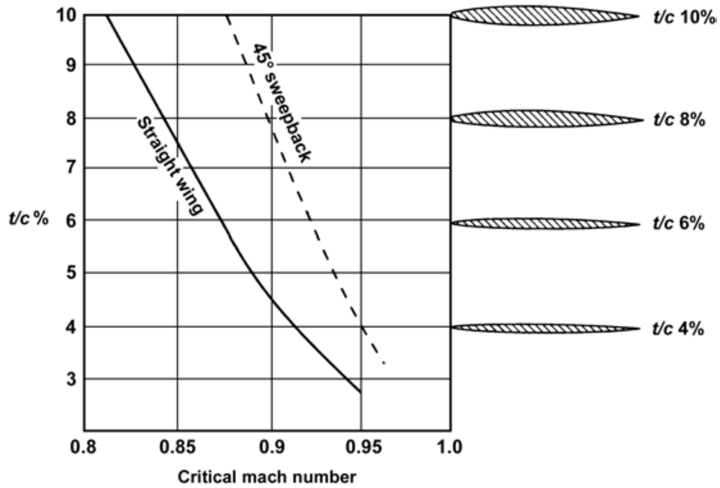


Fig.10.16.

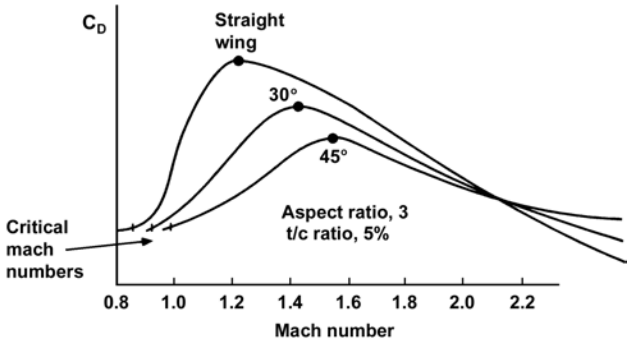


Fig.10.17.

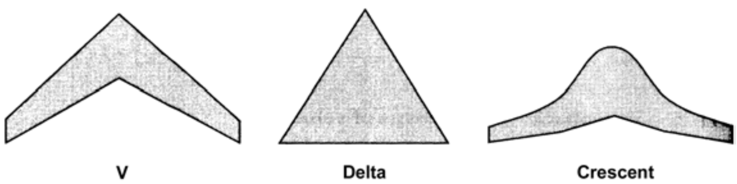


Fig.10.18.

Under these conditions, Mach waves spring from every point of the leading edge of the wing. This implies that the air approaching the leading edges of the outboard sections of the wing is, in effect, warned of the approach of the wing, as in subsonic flow. However, since there is no finite disturbance across a Mach wave, there is no change in Mach number, and all the waves are parallel when viewed in section.

Next consider a swept wing which generates a shock cone of finite strength from the leading edge of the centre section, and suppose that the angle of sweep is such that the leading edge lies within this cone. Further shocks are generated by other points on the leading edge of the wing, and the associated shock angles will tend to increase, since each successive shock causes a reduction in Mach number (Fig.10.19). These shocks progressively slow down the flow, so that, at some section such as *AA*', the flow approaching the leading edge will be subsonic, and there will be a subsonic type flow round the wing at all sections outboard of *AA*'. The section shape used for the outboard parts of the wing should thus be a subsonic section, and supersonic section shapes need only be used over those parts of the wing which are near the root.

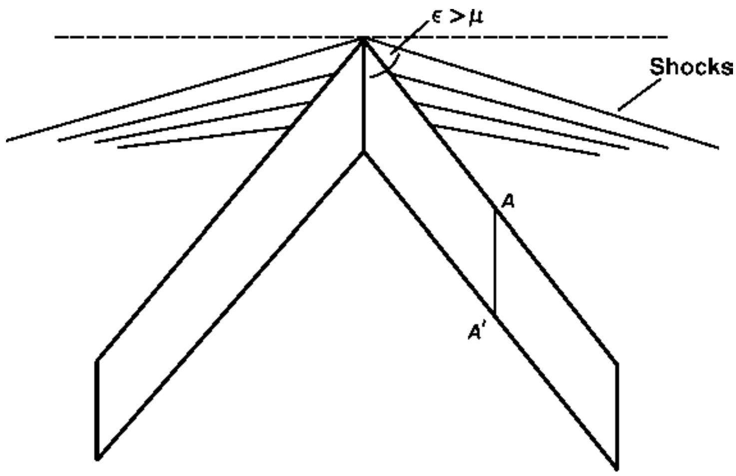


Fig.10.19.

Thus, if the leading edge is swept back within the Mach cone from the leading edge of the centre line, the outboard sections are immersed in a subsonic type flow, with a consequent improvement in performance. A long pointed nose to the fuselage helps further, since the shock it creates reduces the Mach number of the flow approaching the wing; but it must be long enough for the shock not to strike the wing. A wing of this type is said to have a subsonic leading edge, since the normal component of velocity there is subsonic. If M_n is the Mach number associated with the normal component of velocity, then

$$M_n = M \cos \Lambda < M \cos(90^\circ - \mu), \quad \text{since } \Lambda > 90^\circ - \mu,$$

$$M_n < M \sin \mu = 1.$$

Now, for a subsonic leading edge, we require

$$\tan \Lambda > \sqrt{M^2 - 1}$$

and it follows that, to achieve this condition at high Mach numbers, large angles of sweep are necessary. This leads to the concept of using very highly swept wings, e.g., very slender deltas, at very high speeds. For any given angle of sweep, the leading edge is subsonic only when $M \cos \Lambda < 1$, i.e., when $M > \sec \Lambda$. Thus, with increasing Mach number a wing with any given angle of sweep must eventually have a supersonic leading edge, and in this condition its wave drag is higher than that of a straight wing. A wing designed to fly with its leading edge supersonic must have a supersonic, sharp-nosed section over its whole span and a straight leading edge. The concept of variable sweep involves constructing a wing so that it may be swept back through very high angles for high-speed flight, and revert to smaller or zero angles of sweep for lower speed operation, thus avoiding the penalties of high sweep at low speeds.

The nature of the effect of sweepback on wave drag coefficient is illustrated in Fig.10.20, for a wing whose angle of sweep is 45^0. Thus sec $\Lambda = \sqrt{2}$ and the leading edge is subsonic at Mach numbers up to 1.41. Beyond a Mach number a little below this figure, it is better to use a straight wing.

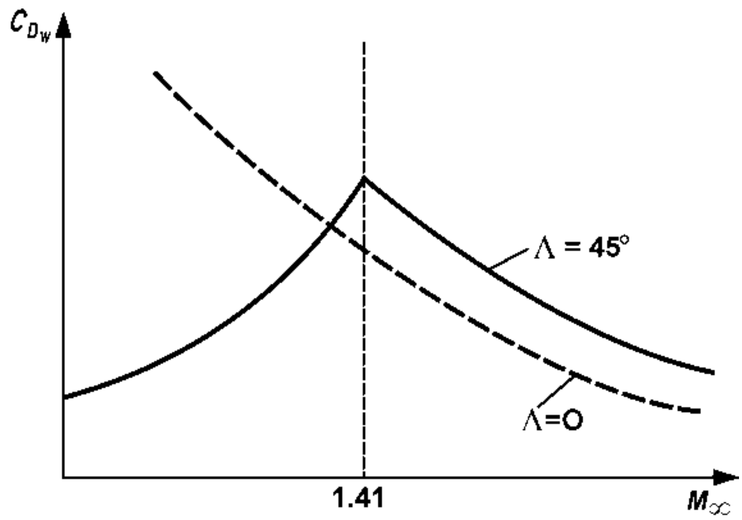


Fig.10.20.

This particular approach to the concept of sweepback replaces, at supersonic speeds, the approach outlined in paragraph 11.13, where sweepback is discussed as a device for raising the Critical Mach Number.

ii Delta Wings
The delta wing has a basically triangular planform, with a straight trailing edge; though the leading edges may be slightly curved. It has a number of advantages for high speed operation.

A low thickness-chord ratio may be employed, yet at the same time, since the root chord is very large, the actual thickness near the root is considerable. This provides a good deal of storage space for fuel, etc., and the engines may be buried, with consequent reduction in drag.

The aspect ration is low. This raises the critical Mach number and, with this planform, tends to reduce the likelihood of tip-stalling. This improvement is effected partly by a modification in the spanwise load distribution, and partly because vortices spring from the leading edge over the upper surface of the wing, and the energy associated with these vortices tends to prevent the boundary layer outflow which contributes to tip-stalling. The disadvantage, inherent in the use of low aspect ratio, of high induced drag, is partly offset by the fact that a tailplane and flaps may normally be dispensed with, giving an appreciable saving in weight and drag. (The functions of tail and flaps are performed by trailing edge control surfaces, though some delta winged aircraft retain a separate tailplane and/or flaps.) Also, because of the large wing area, the wing loading is low.

Aeroelastic effects are very small, because of the great structural strength and stiffness of such a wing. Lateral control at high speeds is improved, because of the straight trailing edge and the reduced tendency; to tip-stalling.

There are, of course, disadvantages in the use of the delta configuration at low speeds, apart from those generally associated with sweepback. They are principally associated with the fact that the lift-curve slope is small, and the stalling angle very large; to achieve high lift, very high incidence is required. This results in a poor range of vision for the pilot in the landing and take-off attitudes, and also in a large region of disturbed air behind the wing which may render ineffective any tail controls.

iii Crescent Wings
The crescent wing employs a graduated amount of sweep across its span, the angle of sweep being high near the root but much less near the tip. Fig.10.21 illustrates this.

At the root, where the angle of sweep is high, the thickness-chord ratio is fairly large; at the tip, where the sweepback is less, the wing is thinner. Different combinations of sweep and thickness may be used to give approximately the same critical Mach number across the span.

The wing loading is kept down by using a large wing area, but with this planform this means a moderate aspect ratio. This helps to minimize the induced drag, but does not help the tip-stalling problem. Tip-stalling tendencies are, however,

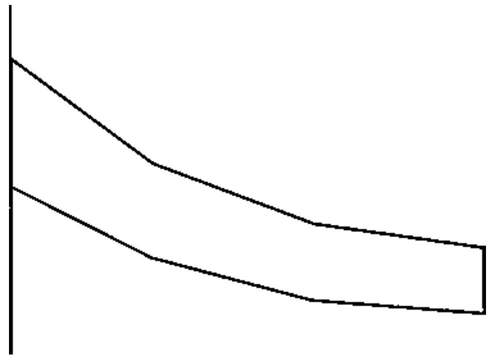


Fig.10.21.

reduced by the fact that the angle of sweep at the tip is quite small. Also, if tip-stalling does occur, the loss of lift is not located so far aft as with ordinary swept wings, so that pitch-up tendencies are reduced.

Lateral control at high speeds is improved, due to the small amount of sweep at the tips.

This construction considerably reduces the tendency of the wing to twist as well as bend under large uploads, because of the changing inclinations of the wing flexural and torsional axes. Thus the aeroelastic effect in a pull-out is reduced. There may still be a slight effect near the tip; but again any nose-up tendency due to this is further reduced because the tip is, relatively, set forward.

However, we must additionally note that the sweep at the outboard sections of the wing may not be reduced too much, or the critical Mach number there will be too low; and further that flaps and a tailplane are necessary with this planform. This tends to offset the reduction in drag due to the higher aspect ratio. Nevertheless, when flying for range the crescent wing appears to have a substantial advantage.

iv M and W Wings

The so-called M and W wings sketched in Fig.10.22 are again compromises, intended to reap the benefits of sweepback without paying too great a penalty. There is no attempt here to avoid tip-stalling; but the planform ensures that the tip is not situated far aft of the aircraft centre of gravity, so that loss of lift at the tips is not necessarily followed by

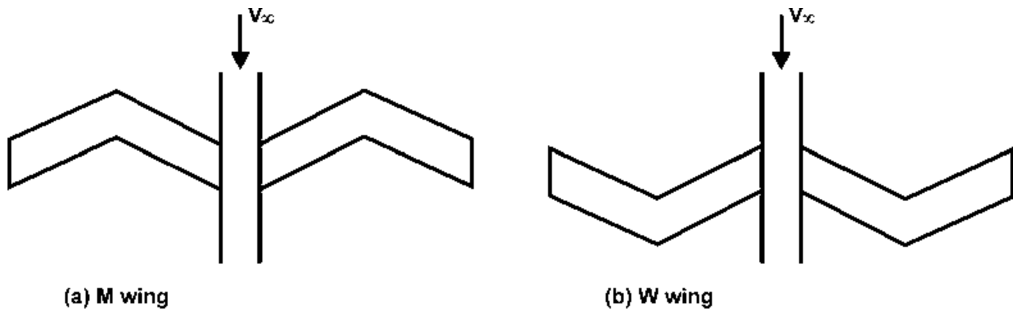


Fig.10.22.

pitch-up. A moderately high aspect ratio could be used, and moderate thickness can be combined with high angles of sweep. Aero-elastic effects need be no worse than on straight wings. Lateral control would probably still be difficult. The major problem might well be the structural one of making a wing with such an angle in it of sufficient strength, without incurring too great a weight penalty.

SECTIONAL FORMS OF HIGH SPEED AEROFOILS

i. Double Wedge

For flow at supersonic speeds, the section shape should have the following properties :

a. *Thinness.* If the aerofoil is too thick, the flow deviations will be large, and so will the consequent shock losses. To minimize these, the wing should be as thin as possible, subject to structural requirements.

Sharp Leading Edge. This is necessary to ensure an attached bow shock wave, and thus to avoid the losses due to a normal, detached shock. Low thickness-chord ratio also implies a small leading edge angle, which is necessary to ensure shock attachment at fairly low supersonic speeds.

Maximum Thickness at Half Chord. This gives expansions behind the maximum thickness point which are similar in magnitude to the compressions ahead of it, and it can be shown that this is conducive to low drag.

Symmetry. In supersonic flow, positive camber results in a positive zero-lift incidence. It follows that at any given incidence the lift is reduced by camber. Also, at low speeds, the effect of camber is to increase the drag. Thus, for both these reasons, basically symmetrical sections are always used for supersonic wings.

The following example illustrates the effect of camber on zero-lift incidence. Consider an aerofoil of the wedge shape indicated in Fig.10.23, at zero incidence. This aerofoil has positive camber. It is easily shown that the increase in pressure due to a given compressive flow deviation is greater in magnitude than the reduction in pressure due to an expansive flow deviation through the same angle. It follows, therefore, that the positive pressure increment, relative to the free stream value, on the front upper surface is greater in magnitude than the negative pressure increment on the rear upper surface. Since there is no pressure increment anywhere on the lower surface, it follows that there will be a net downward force on the aerofoil, i.e., negative lift at zero incidence. Thus the zero-lift incidence must be positive.

The optimum wing section, in theory, for supersonic flow is a simple, infinitely thin, flat plate. This satisfies all the above requirements more fully than any other possible wing. However, it is obviously not a practical solution to the problem, from the structural point of view; some thickness is clearly essential to provide strength. For a given thickness-chord ratio, minimum wave drag is achieved by using the doubly-symmetric double wedge section illustrated in Fig.10.24. Such a section gives virtually the same lift as a flat plate, but more wave drag. Another useful supersonic section is the biconvex section; the upper and lower surfaces are formed by equal circular arcs, as shown in Fig.10.25. Such a section gives rather more wave drag than a double wedge of the same thickness-chord ratio, and approximately the same lift. Also, for the same thickness-chord ratio, the leading edge angle is greater than that of a double wedge, so that the shock attachment Mach number is higher. However, the biconvex aerofoil may be better structurally, and contains more space.

All of these aerofoils are thin, sharp-nosed, and doubly-symmetric.

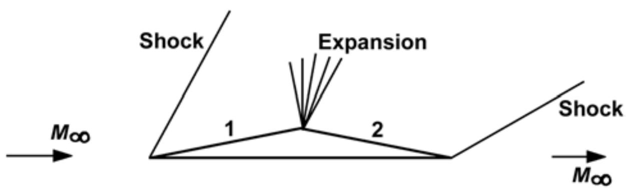


Fig.10.23.

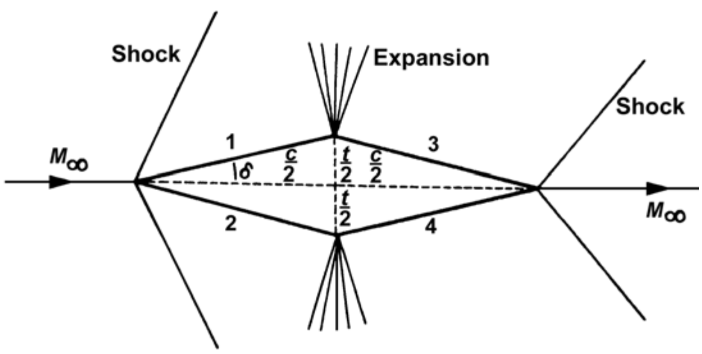


Fig.10.24.

Fig.10.25.

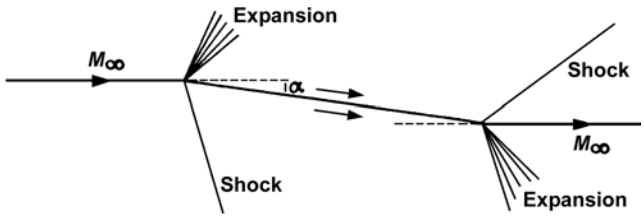


Fig.10.26..

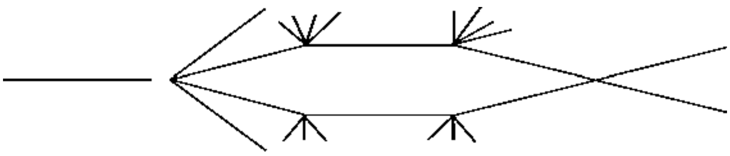


Fig.10.27. Hexagonal

ii DoubleConvex
A bi-convex wing is also quite good (Fig.10.26), and this is better than the others at subsonic speeds. A bi-convex wing has about the same drag as a double-wedge with maximum thickness rather outside the best range, i.e. at about 25 or 75 per cent of the chord.

iii Hexagonal
A variation of the double-wedge is the hexagonal shape (Fig.10.27). This gives greater depth along the chord and so greater strength, and also makes the leading edge rather less sharp, which has advantages both as regards strength and, as will be considered later, aerodynamic heating.

iv Super Critical
The supercritical airfoil, illustrated in Fig.10.28, has a very slight curvature on the upper surface and the maximum thickness is much farther back than normal. The airfoil curves downward at the trailing edge. This design prevents

Fig.10.28. Supercritical wings

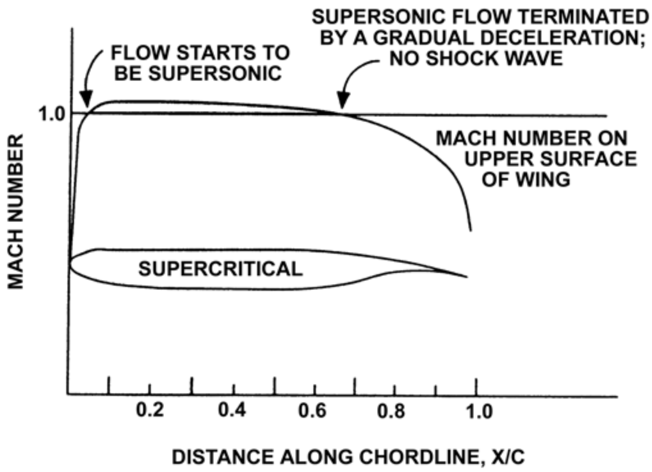


Fig.10.29. Comparison of supersonic airflow pattern across conventional and supercritical airfoils

the rapid pressure rise normally associated with a more cabered airfoil. It also delays and softens the onset of shock waves on the upper surface of a wing. The shock wave is far less severe than on a conventional wing, as is shown in Fig.10.29, and fuel efficiency is substantially improved. This design is being adopted on many transport-category and business-jet aircraft.

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CHAPTER: 11
AUTOROTATION, SPIN, AND LONGITUDINAL OSCILLATIONS

AUTOROTATION

Autorotation may occur with not just one but both wings stalled, in which case the effect is likely to be even more pronounced. It may be initiated as a result of aileron ineffectiveness near the stall, if the aircraft has a tendency to tip-stalling. If tip-stalling occurs asymmetrically, then use of aileron would be ineffective on one side before the other, and aileron deflection would give rise to an unexpected rolling moment which could instigate autorotation. In any case, the ailerons become ineffective once autorotation has begun, so that the motion cannot in general be corrected by use of ailerons. The remedy for the condition is quite simply to reduce the incidence. Neither wing will then be stalled, and the autorotation is eliminated either as a result of the positive damping in roll, or through the deliberate use of the ailerons, which regain their effectiveness once the wing is no longer stalled.

1. Accelerated Autorotation

Consider now an aircraft which is flying straight and level at an incidence very close to the stalling angle. Suppose that its condition corresponds to the point *A* on the lift curve depicted in Fig.11.1. Suppose now that the aircraft is subjected to a disturbance in roll, causing one wing to drop. This wing will now have a downward velocity, the airflow over it will have a relative upwash velocity, and the effective incidence will be increased. This may result in the wing becoming stalled, with a consequent sharp reduction in lift on this downgoing wing. The upgoing wing will, by the same token, experience a reduction in effective incidence by the same amount as the increase on the other side, and this will cause its lift to be reduced also. Since the lift curve is likely to be flatter for incidences just below the stall than for incidences just above it, the loss of lift may be less on the upgoing wing than on the downgoing one. This is illustrated in Fig.11.1, where the points *B* and *C* on the lift curve represent respectively the conditions of the downgoing and upgoing wings. If this is the case, then there will be a rolling moment resulting from the difference between the lift of the two wings, and this moment will be in the same sense as the original roll, and so will constitute, in effect, negative damping in roll. The aircraft will therefore not only continue to roll, but will actually accelerate in roll. This phenomenon is known as Accelerated autorotation.

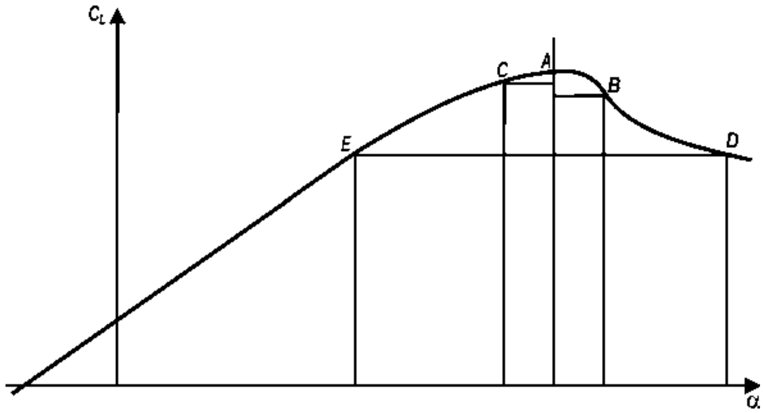

Fig.11.1.

2. Steady Autorotation

It may be that, when the rate of roll reaches a certain value, the corresponding change in incidence of the two wings is such that the lift coefficient is again the same on both sides, as indicated by the points *D* and *E* on the lift curve of Fig.11.1. If so, then at this rate of roll there is no longer any resultant rolling moment, and the aircraft will continue to autorotate at a steady rate.

This phenomenon is known as Steady autorotation. This steady rate of autorotation may easily be estimated at any forward speed if the lift curve is known. The positions *D* and *E* may be noted on the lift curve, and the incidence change to which they correspond, given the initial undisturbed condition, may be measured. This is, of course, a mean incidence change for the wing as a whole, and is then equal to $p\bar{y}/V$, where *p* is the steady rate of autorotation, $\bar{y}$ is the effective distance of the centre of lift of each wing from the rolling axis, and *V* is the forward speed. This gives an approximate value for *p*, which is proportional to the true airspeed.

SPIN (Flat Spin and Steep Spin)

Although it is easy to cause a model to autorotate, the phenomenon does not occur in isolation with an aircraft in flight, since the lateral disturbance which initiates the autorotation also instigates a directional disturbance which complicates the motion. The resulting motion is known as spinning. The directional motion is created as a result of the difference in drag between the two wings. The downgoing wing, as we have seen, is stalled, so that the drag on that side is much greater than on the other side. If the roll is to starboard, say, the drag is higher on the starboard side, there is a yawing moment to starboard, and the aircraft begins to yaw to starboard. There is an interaction between yawing moment due to roll and rolling moment due to sideslip, and ultimately a steady condition is reached which is a steady spin. In the process, the aircraft descends along a helical path. The rate of spin and the steepness of the descent for a given aeroplane are functions of its initial incidence and of its lateral and directional stability characteristics. High incidence, especially incidence well beyond the stall, leads to a so-called flat spin, in which the rate of rotation is high, but the descent not very steep. If the spin occurs at a rather lower incidence, though the incidence must still be high enough for at least one wing to be stalled, then the resulting spin tends to be more stable than the steep spin, so that recovery is more difficult in the former case.

Recovery from the spin is effected, essentially, by reducing the incidence, and so getting away from the stalled condition on both wings. This suggests that recovery may be achieved simply by deflecting the elevator downwards, i.e., by pushing the stick forwards. However, the elevator is generally ineffective in the spin, as a result of the large angle of sideslip. In any case, the sideslip plays a very important part in the spin, and causes it to be a stable condition. If there were no sideslip, autorotation could only occur over a very limited range of incidence, and it would be easy to escape from this condition by use of elevator. The existence of sideslip causes a considerable increase in the range of incidence over which the spin can occur, and also results in a relatively high rate of rotation. It is thus desirable to reduce the sideslip first, by use of the rudder. Because of the descending nature of the motion, the upper part of the rudder is shielded by the elevator and tailplane. If the elevator is deflected downwards, the extent of this shielding is increased, and the rudder effectiveness may be reduced. Thus the use of elevator first is not only ineffective in reducing the incidence, but may also render the rudder ineffective in reducing sideslip. The correct recovery procedure therefore consists in applying full opposite rudder first, in order to reduce the sideslip, and then deflecting the elevator downwards to reduce the incidence. The final stages consist in using the ailerons to eliminate roll, and then pulling out of the dive.

LONGITUDINAL OSCILLATIONS

Phugoid

We turn our attention now to a particular problem in longitudinal dynamic stability. The fundamental nature of the phugoid oscillation has already been described in paragraph 16.8. The oscillation can sometimes take other, slightly different forms, but it is with this basic oscillation that we are concerned here. We suppose that an aircraft is flying straight and level at an airspeed *V*, when it is subjected to a disturbance which changes its speed without changing its incidence. The ensuing oscillation, in which the aircraft successively gains and loses height, with corresponding loss and gain in airspeed, and in which the incidence remains effectively constant, is the phugoid. It is usually only lightly damped, and may even be slightly unstable.

A simple, but approximate, analysis of the motion may be made by using energy considerations. Let *V* be the mean forward speed during the oscillation, i.e., the velocity in the mean, or equilibrium condition of flight before the onset of the disturbance. Suppose that, in falling through a height *h* from the mean altitude, the aircraft increases its speed by an increment υ. Then, the loss in potential energy is *Wh*, and the corresponding gain in kinetic energy is $\left(\frac{1}{2}W/g\right)\left\{(V+\upsilon)^2 - V^2\right\}$. Thus, assuming the conservation of energy,

$$Wh = \frac{W}{2g}(2V\upsilon + \upsilon^2).$$

Since υ is small compared with V, we neglect the υ^2 term and write, approximately,

$$V\upsilon = gh. \tag{i}$$

Now, in the mean condition, lift equals weight, so that $W = \frac{1}{2}\rho V^2 SC_L$. If L is the lift after falling through a height h, then

$$L = \frac{1}{2}\rho(V+\upsilon)^2 SC_L,$$

so that

$$\frac{L}{W} = \frac{(V+\upsilon)^2}{V^2} = 1 + \frac{2\upsilon}{V} + \frac{\upsilon^2}{V^2}$$

$$= 1 + \frac{2\upsilon}{V}, \text{ to the first order.}$$

But, from equation (i), $\upsilon = gh/V$, so that

$$L = W\left(1 + \frac{2gh}{V^2}\right). \qquad \text{(ii)}$$

The resultant vertical force is then (L - W), so the vertical acceleration is (L - W) / (W/g), which is given by

$$\frac{W(1 + 2gh/V^2 - 1)}{W/g} = \frac{2g^2}{V^2}\cdot h$$

and it follows that the equation of motion may be written

$$h = \frac{d^2h}{dt^2} = -\frac{2g^2}{V^2}\cdot h$$

This is the equation of a simple harmonic motion of period T, where $T = 2\pi/\sqrt{2g^2/V^2}$, or,

$$T = \frac{\sqrt{2}\,\pi V}{g}. \qquad \text{(iii)}$$

If υ_{max} is the largest value of during the oscillation, then it corresponds to the lowest point of the flight path, where the value of h is the amplitude of the oscillation. From equation (i), this amplitude is given by

$$\frac{V\upsilon_{max}}{g}$$

and, in any particular case, this clearly depends on the magnitude of the initial disturbance, which determines υ_{max}.

This is, as stated above, only a very simple and approximate analysis of the oscillation. The phugoid will be referred to again briefly when discussing the solution of the longitudinal equations of motion.

Porpoising

As we have seen, the movement of the shock waves with variation in M_∞ causes changes in the position of the centre of pressure. The initial tendency, as the upper surface shock begins to move backwards, is for the centre of pressure to move aft, because of the increase in suction over the rear of the upper surface. When the flow over the lower surface speeds up, this effect is reversed, since the lower surface shock moves backwards more rapidly. By the time the lower surface shock reaches the trailing edge, there is a downward force over the rear of the aerofoil, so that the centre of pressure is now again well forward. This effect is accentuated if separation is induced by the upper surface shock. As the upper surface shock now continues to move back, the tendency is reversed once more, and the centre of pressure moves aft again. It eventually settles down near the half-chord position. Figure11.2. illustrates this movement.

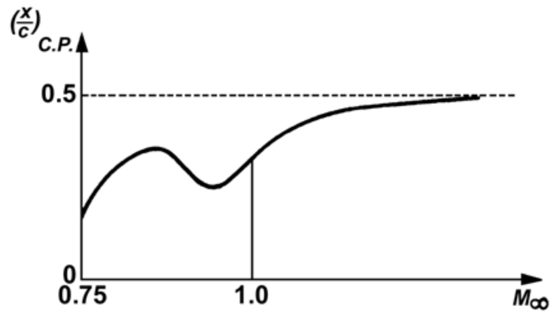

Fig.11.2.

These changes in centre of pressure position give rise to quite sudden changes in longitudinal equilibrium, so that aircraft handling often becomes difficult in the transonic regime. Changes in incidence will give rise to movement of the shock waves similar to those occasioned by increase in M_∞, and will thus cause changes in centre of pressure position. As a result, the stability characteristics of the wing or aircraft will be affected. In the longitudinal sense, there is a possibility of a poorly damped oscillation consisting of a pitching motion combined with a periodic gain and loss in height, which is known as 'porpoising'. This is difficult to explain in detail here, but is due to changes in pitching moment with incidence as a result of both variations in lift and movement of the centre of pressure.

V-n DIAGRAM

The design of an aircraft is dictated by the anticipated use. One of the most important guide lines used by the engineer in defining that use in the diagram relating limit and ultimate load factors to forward speed, the V-N diagram.

A typical V-N Diagram is shown in Fig.11.3. The V-N Diagram in the figure is intended to show the general features of such a diagram and does not necessarily represent the characteristics of any particular air plane. Each aircraft has its own particular V-N Diagram with specific V' and N's. The flight operating strength of an airplane is presented on a graph whose horizontal scale is air speed (V) and whose vertical scale is load factor (N). For the airplane shown, the positive ultimate load factors is 5.7 (3.8 X 1.5). For negative limit load factor conditions, the negative limit load factor is 1.52 and the negative ultimate load factors is 2.28 (1.52 X 15). They never exceed speed, which is the placard red line speed, is 250 knots, and the wing level stall speed is 80 knots. If this air plane is flown at a positive load factor greater than the positive ultimate load factor of 5.7, structural damage will be possible. When the airplane is operated in this region, objectionable permanent deformation of the primary structure may take place and a high rate of fatigue damage is incurred.

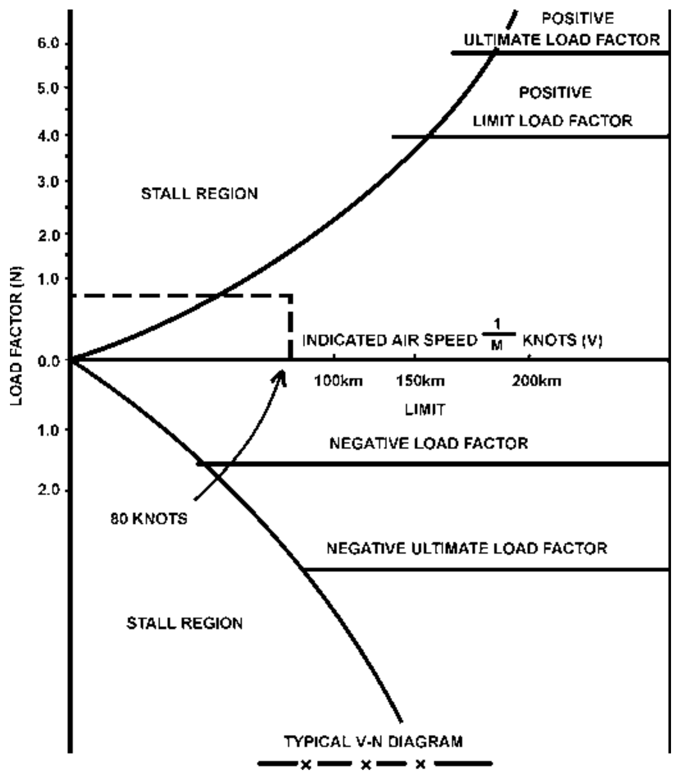

Fig.11.3. V-n Diagram

The same situation exists in negative lift flight, with the exception that the limit and ultimate load factors are of smaller magnitude and the negative limit load factor may not be the same value of all airspeeds.

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CHAPTER: 12
PROPELLERS

GENERAL
A propeller is a means of converting engine power into propulsive force. A rotating propeller imparts rearward motion to a mass of air, and the reaction to this is a forward force on the propeller blades.

Each propeller blade is of aerofoil cross-section. As the blade moves through the air, forces are produced, which are known as thrust and torque, and which may be regarded as roughly equivalent to the forces of lift and drag produced by an aircraft wing. Thrust is the propulsive force, and torque the resistance to rotation, or propeller load. The magnitude of the thrust and torque forces produced will depend on the size, shape and number of blades, the blade angle, the speed of rotation, the air density and the forward speed.

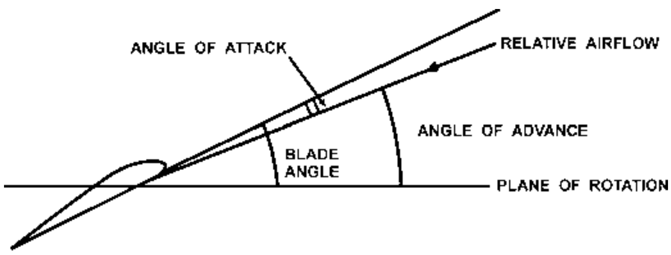


Fig. 12.1. Showing propeller terms.

Since each blade is of aerofoil cross-section, thrust will be produced most efficiently at a particular angle of attack, that is the angle between the chord line at a particular blade section and the relative airflow. This angle varies both with operating conditions and with the design camber of the blade sections, but for a given blade and given in flight condition, it will be found to be relatively constant along the length of the blade. The rotational speed of particular cross-section of a blade will increase with its distance from the axis of rotation, and, since the forward speed of all parts of the blade is the same, the relative airflow will vary along the blade, and it is, therefore, necessary to provide a decreasing blade angle from root to tip. The various terms relating to propeller operation are illustrated in Fig.12.1. This is a simplified diagram omitting inflow angles for clarity, but in practical designs these angles can not be ignored.

The geometric pitch of a propeller is the distance which it should move forward in one revolution without slip; it is equal to $2\pi r \tan\theta$ where r is the radius of the particular cross section and θ is the blade angle at that point. Fixed pitch propellers are usually classified by their diameter and pitch, the pitch being related to the blade angle at 3/4 radius, or other nominated station.

Centrifugal, bending, and twisting forces act on a propeller during flight, and can be very severe at high rotational speeds. Propellers must be both strong enough to resist these forces, and rigid enough to prevent flutter. The main forces experienced are as follows :-

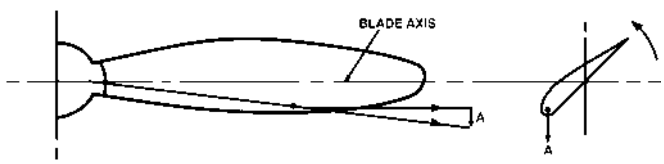


Fig.12.2. Centrifugal twisting moment.

a) Centrifugal forces which induce radial stress in the blades and hub, and, when acting on material which is not on the blade axis, also induce a twisting moment. Centrifugal force can be resolved into two components, in the plane of rotation one is a radial force parallel to the blade axis, and the other a force at 90^{o} to the blade axis; the former produces radial stress, and the latter tends to turn the blade to a finer pitch. The turning effect is referred to as centrifugal twisting moment, and is illustrated in Fig.12.2; the wider the blade, the greater will be the twisting moment.

b) Thrust forces which tend to bend the blades forward in the direction of flight.

c) Torque forces which tend to bend the blades against the direction of rotation.

d) Air loads which normally tend to oppose the centrifugal twisting moment and coarsen blade pitch.

The diameter of a propeller, and the number and shape of its blades, depend on the power it is required to absorb, on the take-off thrust it is necessary to produce, and on the noise-level limits which have to be met. High tip speeds absorb greater power than low tip speeds, but if the tip speed approaches the speed of sound, efficiency will fall, and this consideration limits practical diameter/ rotational speed combinations. High tip speed is also the main source of propeller noise. Large diameters normally result in better performance than small diameters, and blade area is chosen to ensure that blade lift coefficients are kept in the range where the blade sections are efficient. Wide chord blades and/ or large diameters lead to heavy propellers; increase in number of blades increases cost but reduces noise. The design of any propeller is, therefore, a compromise between conflicting requirements, and the features which are given prominence will vary from one application to another. Small two-bladed propellers, of suitable profile, are satisfactory for low-powered piston engines, but for high-powered piston or turbine engines, three, four, or five bladed, or contra-rotating, propellers are used, and are driven through a reduction gear to enable high engine power to be used at efficient propeller speeds.

PROPELLER BALANCE

A propeller is a rotating mass, and if not correctly balanced can produce unacceptable vibration. An unbalanced condition may be caused by uneven weight distribution, or by uneven air loads or centrifugal forces on the blades when the propeller is rotating. Even weight distribution is known as static balance; this is checked by mounting the propeller on a shaft between knife edges; or by use of a single plane precision balancing machine. An unbalanced condition can be corrected by adding weight to the lighter blade (s) and/or removing weight from the heavier blade (s). Material may easily be removed from wooden propellers, but metal propellers are usually balanced by attaching weights to the blade hub or by adding lead wool to the hollow blade roots. If there are significant differences in form or twist between the blades on a propeller, vibration can result because the thrust and or torque produced by the blades is uneven. Procedures for evaluating such differences, and for achieving aerodynamic balance, are often available for large propellers. In their absence, careful checking of the blade profiles and adjustment of any deviations, may often eliminate vibration. It is possible for a propeller to be in perfect static and aerodynamic balance, but still suffer from dynamic unbalance when rotating. The cause of such unbalance is non-symmetrical disposition of mass within the propeller, or non-symmetrical mounting of the propeller. Such unbalance can be corrected by adding balance weights, but this may be a lengthy procedure, involving repeated runs with the propeller installed on the aircraft. Propellers are balanced after manufacture, and whenever repairs, or overhaul, have been carried out, or vibration has been reported.

TYPES OF PROPELLERS

The various types of propellers are described briefly in this paragraph. The construction and operation of the main types of propellers in common use.

1. Fixed pitch propellers
2. Variable pitch propellers
3. Feathering propellers
4. Reversible pitch propellers **-** On some aircraft, the propeller blades may be turned past the normal fine-pitch setting, to a pitch which will produce thrust in the opposite direction (reverse thrust). On selection of reverse pitch by the pilot, the blades may be turned to a fixed reverse-pitch angle, but on some installations the pilot has control of blade angle, and can select any angle within a given range on each propeller individually, Reversible-pitch propellers provide braking during the landing run, and facilitate aircraft ground manoeuvring.

FROUDE MOMENTUM THEORY

The simplest of the various methods of studying the performance of propulsive systems consists in the application of straight forward momentum consideration. Thrust is obtained by speeding up the airflow through the system, i.e., the thrust is the reaction to the increase in momentum imparted in the rearward direction to the jet of air. This approach can be applied with equal validity to a propeller or to a turbojet engine.

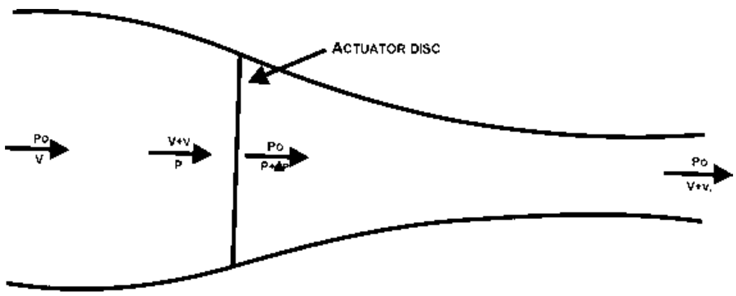


Fig.12.3. Froude Momentum Theory

We introduce first the concept of an actuator disc. This is simply an idealized device which produces a sudden pressure rise in a stream of air passing through it. This pressure rise, integrated over the whole disc, gives the thrust associated with the engine which is so idealized.

Consider therefore a stream of air flowing through an actuator disc, subject to a number of simplifying assumptions, viz.

a. That the pressure increment is constant over the whole disc.
b. That there are no rotational velocities in the slipstream or jet.
c. That the air is inviscid and incompressible.
d. That there is no discontinuity in velocity across the disc.

BLADE ELEMENT THEORY

Simple momentum theory gives an elementary approach to the problem of the performance of a propulsive system. For more detailed work, or to design an airscrew to have any required performance, it is necessary to use a more precise analysis. One such approach consists in investigating the airscrew blade as a lifting aerofoil, by considering first an element of a blade and then integrating to determine the characteristics of the whole propeller. This technique is known as blade element theory.

FIXED PITCH PROPELLERS

Because of its lightness, cheapness and simplicity, a fixed-pitch propeller is often fitted to a single engine aircraft. The pitch selected for any particular engine/airframe combination will always be a compromise, since the angle of attack will vary with changes in engine speed and aircraft attitude. Too coarse a pitch would prevent maximum engine power from being used during take-off and climb, and too fine a pitch would prevent economical cruising, and would lead to over speeding of the engine in a dive.

1. Controlled pitch Propellers

This provides for two positions of pitch. The propeller normally holds the blades at a coarse pitch setting for high speeds, but can be adjusted manually while the engine is running to fine pitch for takeoff or climb. This is accomplished by connecting the blades to the shaft through a mechanism that permits the blades to turn in their hub. The blades are rotated by a cylinder that slides over a piston attached directly to the shaft. Oil pressure under manual control is used to move the cylinder to its outer or fine pitch position. The centrifugal action of counter-weights returning the cylinder to its original position or coarse pitch when the oil pressure is released.

2. Adjustable pitch Propellers

With this type of propeller the blade angle may be varied in flight, so that engine power may be fully utilized. Variable-pitch propellers were originally produced with two blade-angle settings; a fine pitch to enable full engine speed to be used during takeoff and climb, and a coarse pitch to enable an economical engine speed to be used for cruising. The introduction of an engine driven centrifugal governor enabled the blade angle to be altered automatically (within a predetermined range), in order to maintain any engine speed selected by the pilot, regardless of aircraft speed or attitude.

CONSTANT SPEED PROPELLERS

This provides for automatic control of pitch by means of a governor which responds to change of only a few r.p.m. and maintain the pitch at the optimum setting for any forward speed. The governor replaces the manual control of the oil pressure. It is so designed that the r.p.m. are held constant regardless of the throttle setting (within reasonable limits) and regardless of whether the aircraft is climbing, diving or flying level.

FEATHERING OF PROPELLERS

If an engine failure occurs, the wind milling propeller may cause considerable drag, and adversely affect controllability of the aircraft. In order to reduce this drag, the blades of most constant speed propellers fitted to multi-engine aircraft are capable of being turned past the normal maximum coarse-pitch setting into line with the airflow. This is known as the ‘feathered’ position. Feathering the propeller not only reduces drag, but also minimizes engine rotation, thus preventing any additional damage to the engine.

GYROSCOPIC EFFECTS OF PROPELLERS

Early in the takeoffs run of a tailwheel aircraft (e.g. de Havilland Chipmunk) the tail is lifted off the ground to place the aeroplane into a low drag and flying attitude. As the tail is being raised, a torque is applied to the rotating propeller in a nose-down sense. Because a rotating body tends to resist any attempt to change its plane of rotation, when such a change is forced upon it, a gyroscopic precession will be superimposed.

Gyroscopic precession rotates the applied force 90 degrees in the direction of rotation-this phenomenon being known as gyroscopic effect. When nose-down torque is applied to the aircraft to raise the tail on takeoff (which is like a forward force applied to the top of the rotating propeller disc), gyroscopic effect causes a similar force to be applied 90 degrees in the direction of propeller rotation. This will be like a forwards force acting on the right side of the rotating propeller disc, causing the aircraft to yaw. The direction of yaw depends on the direction of propeller rotation.

The amount of gyroscopic effect depends upon the mass of the propeller, how the mass is distributed along the blades and how fast the propeller is rotating (all of this being combined into a physical quantity called the moment of inertia). It will also depend upon how fast you try to change the plane of rotation.

Raising the tail of a high-powered aeroplane like a Mustang on takeoff produces a much greater gyroscopic effect than raising the tail of a Tiger Moth.

Note: The three effects cause a Yaw to the Left in an aeroplane whose propeller rotates clockwise (as seen from the cockpit), which the Pilot will counteract with however much right rudder is required to keep straight. For aeroplanes with anticlockwise rotating propellers (e.g. Mustangs, Chipmunks, Tiger Moths), the yaw on take off will be to the right.

■■■

CHAPTER: 13
WIND TUNNELS

I TYPES OF WIND TUNNELS

1. Low-Speed Wind Tunnels

The basic requirement is for a region of uniform flow with low turbulence, in which a model may be placed and tested. This region is called the working section. Swirl, as well as turbulence, is to be kept as low as possible in the working section. The power required to drive the tunnel is also to be as low as possible for given speed and working section size. The general arrangement in its simplest form consists of a fan with which to blow or suck air through the working section. There is usually a contraction upstream of the working section, and a diffuser downstream. Fig.13.1 illustrates

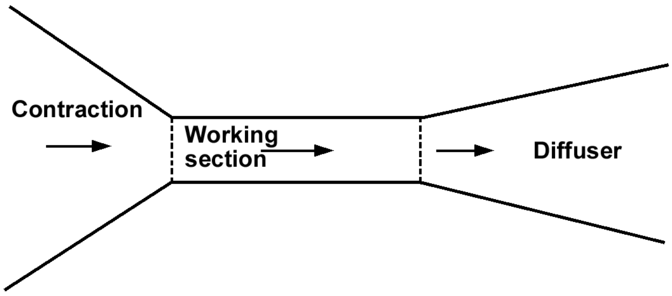


Fig.13.1.

such an arrangement diagrammatically. The main function of the contraction is to speed up the flow so that its speed in the other parts of the tunnel may be much lower than in the working section; in doing so, it causes the streamlines to be squeezed together, and improves the uniformity and turbulence level of the flow. The diffuser serves to retard the flow after it leaves the working section, and so to keep down the power losses.

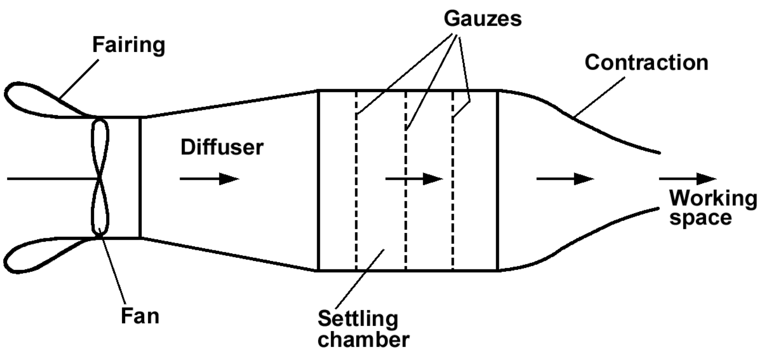


Fig.13.2

Types of Low-speed Wind Tunnel

Straight through tunnels, as they are called, may be of two types, blower or suction. A typical blower tunnel is shown diagrammatically in Fig.13.2. Here the diffuser serves to slow down the air leaving the fan, so that it flows at low speed through a region called the settling chamber. In this region, the slowly moving air settles to a fairly uniform and low turbulence condition, with the help of a series of gauze screens or honeycombs. The flow is then contracted to give a fast jet in the working section. Such a construction is simple and cheep. However, it is suitable only for demonstration purposes, or for very approximate measurements, because the turbulence level is not low enough for any degree of precision. All the kinetic energy of the fast moving air in the working section is wasted, and the consequent high power requirement makes such an arrangement impracticable, except on a fairly small scale.

Figure13.3 is a diagram of a straight through tunnel of the suction type, in which the air is sucked through by a fan downstream of the working section. The tunnel illustrated has an enclosed working section. The entry to the tunnel should be clear of walls and the obstacles in the tunnel room. The settling chamber helps to reduce atmospheric turbulence. There is a large contraction ratio, and a small angle diffuser. Such an arrangement gives reasonably low turbulence, and constitutes a useful general purpose tunnel. There is still considerable power wastage because the kinetic energy of the air leaving the tunnel is not recovered. But it is clearly better in this respect than the blower tunnel, since the speed at exit is less.

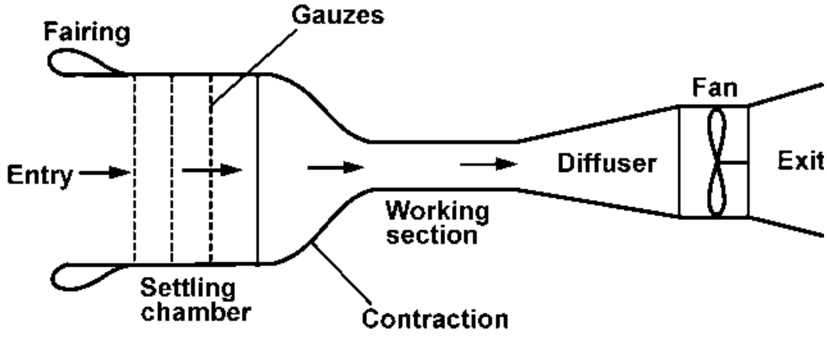


Fig.13.3

In a return circuit tunnel, the air leaving the diffuser is not simply discarded, but collected, and it travels round a closed circuit to be passed through the working section again. In this way, the kinetic energy of the jet is recovered, not wasted, and the power required is reduced. Fig.13.4 depicts a typical return circuit tunnel with an open working section. It has all the features of the straight through tunnel, with the addition of corners. In this case, the diffuser is not immediately

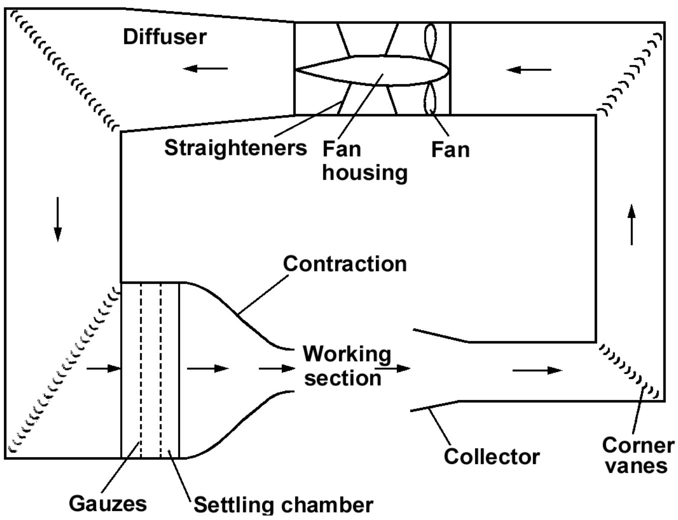


Fig.13.4

downstream of the working section, though it could easily be so. Here it serves to reduce the speed of flow in part of the tunnel circuit, and, in particular, in the settling chamber and at the entry to the contraction. Corner vanes are fitted to enable the air to flow smoothly round the corners. However, with the open jet, it is difficult to obtain uniform, steady flow, and the tunnel is best suited to demonstration work and fairly rough measurement. Its main advantage is the ease of access to the working section.

Alternatively, the working section may be closed, as shown in Fig.13.5, with the arrangement otherwise the same as that of the open jet tunnel in Fig.13.4. There is a breather slot downstream of the working section which serves to maintain the pressure at or near the atmospheric value. The result of enclosing the working section is much improved uniformity and turbulence level. This is the most common type of tunnel for general research and development work.

However, another possible arrangement for a return circuit tunnel is the annular type, illustrated in Fig.13.6, which shows a section through the axis of the tunnel, which is a body of revolution. It is difficult to achieve uniform, steady flow without elaborate arrangements of corner vanes, etc. Also, accesss to the working section is very difficult. This arrangement is therefore only used for variable density tunnels, where, for structural reasons, it is an advantage when pressurization is used, or when the working fluid is other than air, because having few joints reduces leakage problems.

A variable density tunnel generally consists of an annular type tunnel enclosed in a large pressure vessel. By increasing the pressure, and hence the density, it is possible to achieve relatively high Reynolds numbers. Pressurization also

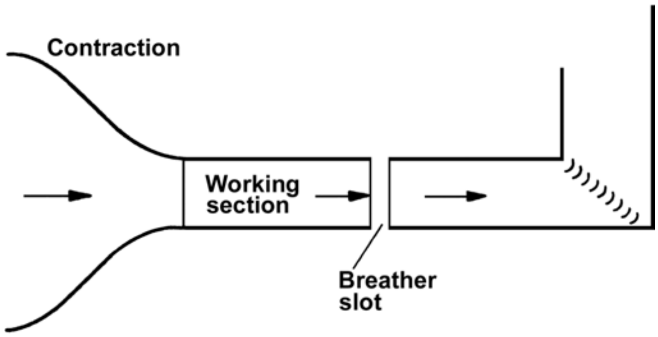


Fig.13.5

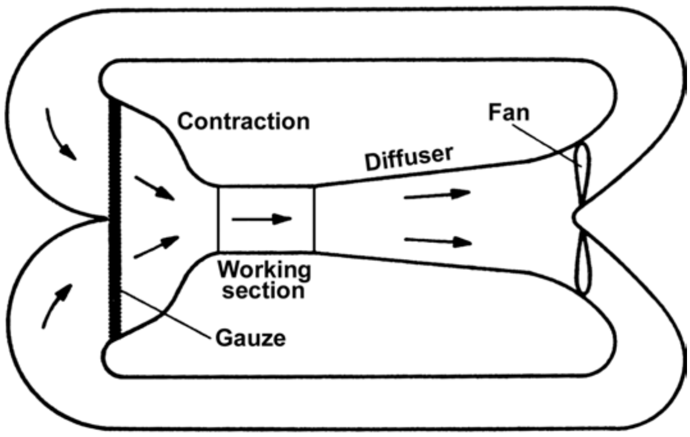


Fig.13.6

results in power economy. Some such tunnels are also designed to operate at much reduced pressures, and hence at low density, and high speeds.

The Working Section

The cross-sectional shape of a wind tunnel is not necessarily always the same right round the circuit. The fan section, for obvious reasons, is circular in section. But other components may be different in section.

Closed working sections are usually rectangular in cross-section, with filleted corners, as shown in Fig.13.7, to prevent the accumulation of dead air in these regions. The fillets may incorporate tunnel lighting arrangements. They sometimes

Fig.13.7

diverge slightly to allow for thickening boundary layers, and thus maintain constant effective area of cross-section. There are usually turn-tables in the roof and floor of the working section, in which models are mounted for easy adjustment of attitude.

Open working sections are usually circular or elliptical. The working section is surrounded by a mixing region which spreads outwards slightly. The collector of the jet must therefore be slightly wider than the exity from the contraction. The static pressure in the working section is atmospheric.

The Difffuser

The function of the diffuser is to slow down the air with the smallest possible loss of energy, while maintaining maximum uniformity of flow. Because of the adverse pressure gradient, it is difficult to avoid flow separation, with the consequent energy loss and increase in turbulence. Experimental results show that small angle diffusers should be used, with an angle of about 5° between opposite walls. If this is done, the diffuser is necessarily long. There may be more than one part of the tunnel where diffusion takes place. It is possible to incorporate devices which help to prevent separation, such as vortex generators, or boundary layer suction devices.

The efficiency of a diffuser is given by the ration of the rise in static pressure to the loss in dynamic pressure. For inviscid flow, Bernoulli's theorem shows that the two are the same, so that there is no loss in total energy, and the efficiency is 100%. Because of viscous effects, the rise in pressure energy is always less than the fall in kinetic energy, and the efficiency is less than unity.

Consider the diffuser depicted in Fig.13.8, where the usual notation is used, and suffixes 1 and 2 represent entry and

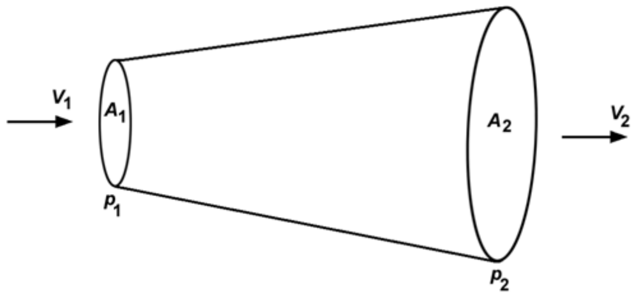


Fig.13.8

exit conditions respectively. By continuity, $A_1V_1 = A_2V_2$, for incompressible flow, so that

$$\frac{V_2}{V_1} = \frac{A_1}{A_2}$$

Thus the diffuser efficiency, denoted by η_D, is given by

$$\eta_D = \frac{p_2 - p_1}{\frac{1}{2}\rho V_1^2 - \frac{1}{2}\rho V_2^2} = \frac{p_2 - p_1}{\frac{1}{2}\rho V_1^2\left(1 - \frac{V_2^2}{V_1^2}\right)}$$

$$= \frac{p_2 - p_1}{\frac{1}{2}\rho V_1^2\left(1 - \frac{A_1^2}{A_2^2}\right)}.$$

Now, if we denote the loss in total head (i.e., total pressure) by ΔH, then

$$\Delta H = \left(\frac{1}{2}\rho V_1^2 - \frac{1}{2}\rho V_2^2\right) - (p_2 - p_1)$$

$$\frac{\Delta H}{\frac{1}{2}\rho V_1^2 - \frac{1}{2}\rho V_2^2} = 1 - \frac{p_2 - p_1}{\frac{1}{2}\rho V_1^2 - \frac{1}{2}\rho V_2^2} = 1 - \eta_D$$

and

$$\eta_D = 1 - \frac{\Delta H}{\frac{1}{2}\rho V_1^2\left(1 - \frac{A_1^2}{A_2^2}\right)}.$$

Gauzes and Honeycombs

Fine mesh gauze screens are inserted to increase uniformity of flow and reduce turbulence. A pressure drop, and hence a loss of energy, occurs across such a screen. A honeycomb performs the same task rather less effectively, but with a smaller pressure drop. Gauzes are most effective in low speed sections, i.e., at speeds of no more than about 10m/s, so the speed is always low in the settling chamber, where the gauzes are located. For a given pressure drop, it is less effective to use one high resistance gauze than to use several low resistance gauzes, with adequate intervals between them. A long settling chamber is therefore required.

The Contraction

The function of the contraction is to speed up the flow into the working section, but in doing so it also squeezes the streamlines together and smooths out some of the velocity fluctuations. For this reason, it is often better to slow down the flow leaving the fan and then speed it up again through a contraction, rather than pass it straight through. The ratio of the areas of cross-section at entry and exit respectively is called the contraction ratio. The higher this is, the better; but it should in any case be arranged so as to give speeds of no more than about 10 m/s in the low speed sections of the tunnel. The shape of the contraction should be such that its area decreases steadily, with no stationary points, so that the velocity increases steadily. The ratio of the velocities at exit and entry respectively is clearly equal to the contraction ratio.

The Corners

Struts, which are of highly cambered aerofoil section, are placed so as to span the tunnel at the corners, and so help the flow to negotiate the corners smoothly. They are called corner vanes. In the absence of such vanes, separation would occur, with consequent non-uniformity, turbulence and power wastage. The vanes require very careful design, especially at corners where the speed is relatively high.

The Fan

The fan should ideally be situated as far as possible from both ends of the working section, so as not to cause more turbulence than necessary. The size of the fan is a matter of compromise. A small fan is inefficient, but a large fan is liable to give blade flutter and some flow pulsations. It is usually placed in a section whose area of cross-section is some one and a half to two times that of the working section. The fan must be of variable speed, and sometimes the blades are of variable pitch, to give good speed control in the tunnel. The tip speed should be as low as possible, and certainly less than half the speed of sound, to avoid compressibility effects.

The flow downstream of the fan is subject to swirl. To offset this effect, vanes are mounted radially behind the fan. These are called straighteners, and they are of aerofoil section, with a degree of camber calculated to produce swirl equal and opposite to that produced by the fan, and so to straighten the flow. They may have adjustable flaps to ensure efficient operation over the whole of the tunnel speed range.

Catchwires

Wires are usually stretched across the tunnel immediately downstream of the working section. If the model or its supports should break, and the model or any part of it is blown downstream, the catchwires are intended to catch it and prevent further damage to the model or the tunnel.

Coolers

If the power is high and the tests are of long duration, the temperature of the air in the tunnel may rise appreciably, and cooling may be needed to prevent over-heating. The cooling may be effected by means of a honeycomb device in the settling chamber, or coolant may be pumped through specially designed corner vanes.

2. High-Speed Wind Tunnels

The problems of nozzle and diffuser design are common to all supersonic wind tunnels. So also is the problem of providing the necessary pressure ratio, and thus the necessary power, to achieve supersonic flow. This latter problem is solved in very different ways in different tunnels. The different designs include tunnels for intermittent operation, and continuously running tunnels; tunnels in which the flow goes straight through, and return circuit tunnels; tunnels in which the flow is direct, and induced flow tunnels. We shall discuss each type in turn.

Intermittent Operation

Wind tunnels for intermitten operation may be of the blow-through or suction type. In a blow-through tunnel, the arrangement is such as the one illustrated diagramatically in Fig.13.9. Air is compressed to a very high pressure in a reservoir. It is subsequently released through a valve into the nozzle, and is eventually discharged into the atmosphere.

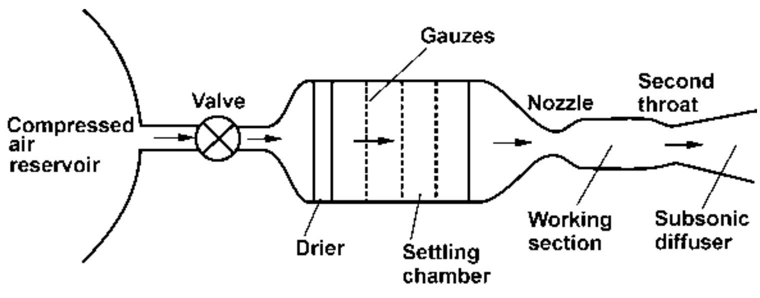


Fig.13.9.

No great power is required, since the reservoir can be charged up relatively slowly by means of a compressor, which does not have to be especially highly powered. However, the running time is generally short, because of the limited amount of air in the reservoir. Once the pressure in the reservoir drops below a certain value, the pressure ratio across the tunnel will no longer be sufficient to give supersonic flow in the working section. It is then necessary to stop the tunnel and re-charge the reservoir, and the charging time between runs may be considerable. A further disadvantage is that the stagnation temperature and pressure will vary during a run, due to the expansion of the air in the reservoir, where the flow originates.

A typical suction type tunnel is illustrated diagrammatically in Fig.13.10. With this tube of tunnel, instead of the air flowing out of a compressed air reservoir, it is sucked into a previously evacuated chamber. Throughout a run, the stagnation conditions remain constant at the atmospheric value, since the flow through the tunnel originates in the

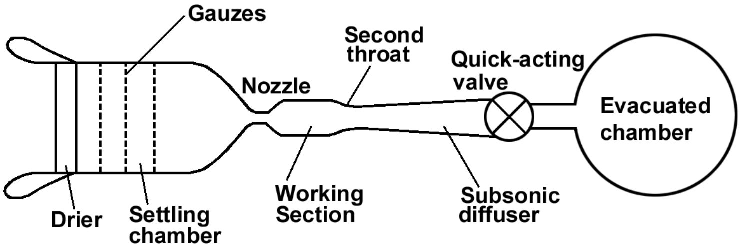


Fig.13.10.

atmosphere outside the tunnel. Also, very high pressure ratios are easily achieved, and this implies that high Mach numbers can be obtained, without the use of a chamber which can withstand high pressurization. The disadvantage is that the running time is likely to be even shorter than that of a blow-through tunnel, especially at high Mach numbers.

Induced Flow Tunnels

In this type of tunnel, a high speed jet is forced through a narrow annular slot downstream of the working section. This jet induces the flow through the working section which, if the blowing pressure is high enough, will be supersonic. The jet through the slots is provided either by means of a compressor or, more usually, from a compressed air reservoir, which, as in the case of a blow-through tunnel, is the effective power supply. Figure 13.11. illustrates the principle involved. To regulate the pressure ratio when a compressed air supply is used, a continuously variable throttle valve is required between the reservoir and the slots.

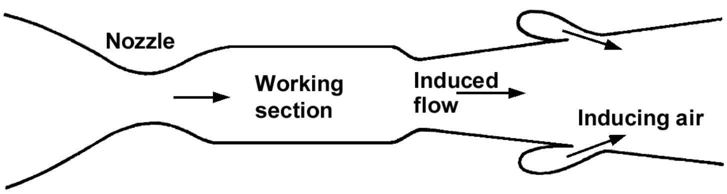


Fig.13.11.

Such tunnels may be operated subsonically or supersonically, but it is generally impossible, because of the difficulties involved in the mixing of the two jets, to achieve Mach numbers much above 1.8. Their main advantage is that, while they have the advantage of the blow-through over the suction type tunnel in terms of running time, they do not suffer from the disadvantage of varying stagnation conditions, since the air which flows through the working section comes, not from the reservoir, but from the atmosphere. The running time depends, of course, on the magnitude of the available compressed air supply - it may even be a continuously running tunnel. In any case, the rate of mass flow from the reservoir is generally less than it is for a blow-through tunnel of similar size and at the same Mach number, so that the running time given a reservoir of similar capacity, is longer.

Continuous Operation

Continuous operation may be achieved in a blow-through or induced flow tunnel if a sufficient compressed air supply is available, i.e., supplied by a compressor whose capacity is such that it can supply air at the required pressure at least at the rate air is consumed by the tunnel. Apart from this, continuous operation may be achieved in effectively the same way as in a low speed tunnel, except that instead of a fan a compressor, usually a multi-stage axial flow compressor, is incorporated in the tunnel circuit, to provide the necessary pressure ratio for the achievement of the design Mach numbers. The other features of such a tunnel, i.e., liner, diffuser, etc., will be the same as in intermittently operating tunnels, except that almost all compressor-driven tunnels incorporate a return circuit.

Return Circuits

A return circuit may be incorporated in a supersonic wind tunnel, just as it may in a low speed tunnel, and since the speed of flow in the return circuit is everywhere quite low, there is no reason for the design to be any different. Some measure of power economy is achieved by using a return circuit. But the principal advantage of such a design is that, since the same air is being re-circulated all the time, there should be no need for continuous drying. One disadvantage is that if the duration of the test is long, the temperature may tend to rise, resulting in varying stagnation conditions, unless some form of cooling is used. If a return circuit is incorporated in an induced flow tunnel, then there must be a break in the circuit somewhere to bleed off the excess air which enters the circuit through the inducing slots.

Choking

The phenomenon of choking occurs when the local Mach number in a high speed tunnel reaches unity at some point in the flow other than the nozzle throat. This may happen at points where the effective area of cross-section of the tunnel is reduced of boundary layer thickening, shock-induced separation, or simply the presence of a model which is too thick, or at too high an incidence. When choking occurs, the supersonic flow in the working section breaks down, and this condition sets a limit to the range of operation of the tunnel-- a limit which may vary according to the model under test, and its attitude.

High-speed Subsonic Tunnels

High-speed subsonic tunnels have parallel-sided liners, and the speed and Mach number are increased simply by increasing the blowing pressure, as in the case of low speed tunnels. However, the effective cross-sectional area of the channel grows less along the length of the working section because of the thickening of the boundary layer, and in consequence the speed of the flow increases. At some subsonic Mach number, the local flow at some point downstream becomes sonic, and the tunnel is choked. If a model is inserted, choking will occur at still lower nominal Mach numbers. Choking my be delayed by using a liner with slightly divergent walls; but, if the walls are solid, sooner or later choking is bound to occur. In any case, it limits subsonic operation of the tunnel to a Mach number well below unity, usually about 0.93 to 0.95 with the tunnel empty, and substantially less with a model present.

Transonic Wind Tunnels

It is usually impossible, then, to achieve flow at a Mach number much above 0.9 with ordinary parallel-sided liners. It is also difficult to achieve supersonic flow at a Mach number below about 1.1 with a convergent-divergent nozzle with solid walls, because of choking difficulties. Special liners are required to cover the transonic range of Mach numbers from, say, 0.85 to 1.15.

The design of such liners is based on a working section whose walls are not solid, but perforated. Slots or holes are made in the walls of the working section, the area of the perforations increasing with distance along the length of the section. The principle is that the excess air associated with boundary layer growth or the presence of the model can escape through these perforations, thus eliminating the effect of a narrowing channel, and preventing choking. The detail positioning, size and shape of the holes or slots is often crucial to the successful operation of the tunnel, and it is largely a matter of trial and error. The perforated walls are surrounded by other, solid walls, and the air which has escaped through the holes into the expansion chamber between, which is known as the plenum chamber, is fed back into the tunnel downstream of the working section. Figure 13.12. illustrates this concept.

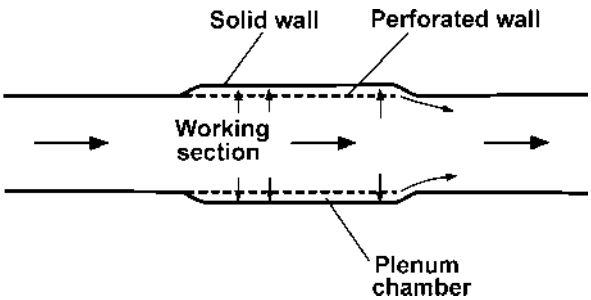


Fig.13.12.

In addition to preventing choking, and so permitting the achievement of Mach numbers up to 1 in the working section, the use of perforated walls, through which the flow can expand, may act in some ways as a divergent channel, so that the flow can expand supersonically, and Mach numbers above 1 may be achieved. In this way, the Mach number range from 1 to about 1.15 can be covered. Sometimes suction may be applied to the plenum chamber, increasing the flow through the holes and improving the efficiency of the device.

Many tunnels have interchangeable liners, parallel-sided, perforated and convergent-divergent, so that the whole range of high speed flows, subsonic, transonic and supersonic my be covered.

The Shock Tube

The shock tube is a device for producing a flow at very high Mach number for a very short time. The principle is illustrated in Fig.13.13. The device consists simply of a long tube, in which a diaphragm separates a region in which the air is compressed to a very high pressure, from an evacuated region. When the pressure ratio across it reaches a certain value,

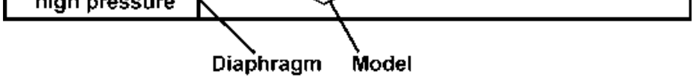


Fig.13.13.

the diaphragm bursts, (or it may be punctured by a device incorporated in the tube) and air rushes at very high speeds into the evacuated region. At the front of this body of air is a mixing region, terminated by an normal shock, behind which is a region of uniform flow at high Mach number. This passes over a model mounted at an appropriate point in the initially evacuated region. When the wave front reaches the end of the tube, it is reflected back again, and once it has reached the model position the flow is spoilt. Thus the time for which uniform flow is achieved is exceedingly short, probably only a few micro-seconds. Measurement of the flow properties is very difficult, and many of the problems created by the use of shock tubes have consisted in the development of special instrumentation for making such measurements on a very small time scale. The tube itself is cheap and easy to construct, and the power requirement is low. More complicated and refined developments of the idea, such as the gun-tunnel, in which a charge of air is fired along the tube, have been developed, and the end of the tube may fork into two branches so as to give increased 'running time', but the above account describes the basic principle of these devices.

Hypersonic Wind Tunnels

The shock tube and gun-tunnel are experimental tools for the investigation of flow at hypersonic speeds. Since hypersonic aerodynamics lies outside the scope of this book, it is not intended to deal further here with experimental facilities in hypersonics, except to say that it is possible to design wind tunnels for use in this speed range, i.e., to operate at very high Mach numbers, on the same basis as the supersonic wind tunnels already described earlier in this chapter. To achieve such high Mach numbers, liners with very narrow throats are required, and this may involve special problems, besides necessitating very high pressure ratios to provide the necessary expansion of the flow. One major problem is the likelihood of liquefaction of the working fluid once the Mach number becomes very large, because of the associated very low temperature. The idea of a heat sink to reduce this likelihood has already been explained, but its usefulness is limited to moderate Mach numbers, not much above 4. One possible solution is to use a working fluid other than air, such as freon gas, with much lower liquefaction temperature, and a lower value of the adiabatic index, γ, which results in reduced temperature and pressure ratios for a given Mach number.

II MODEL TESTING

The most important and widely used tool for such work is the wind tunnel, by means of which we create an airflow in which may be placed and tested a small scale model of the wing, aircraft or other body in which we are interested. It is expected that from the measured behaviour of the model we may infer something of the behaviour of the full scale body. In the interpretation of wind tunnel results, it is always important to take account of scale effect, since the Reynolds number is usually much less than that associated with the full scale body. It is often the aim to provide Reynolds numbers which are as high as possible, in order to minimize scale effects, and this leads to the use of bigger and bigger tunnels, and perhaps to pressurization. In any case, in quoting experimental data, it is always essential to quote the Reynolds number at which the tests were carried out.

■■■

SOME STANDARD VALUE OF VARIOUS PHYSICAL QUANTITIES

It is recognized that, in order to solve some of the examples, assumptions must be made which can hardly be justified in practice, and these assumptions may have appreciable effect on the accuracy of the answers; but the benefit from solving these problems lies not so much in the numerical answers as in the considerations involved in obtaining them.

Unless otherwise specified, the following values should be used—

Density of water	=	1000 kg/m^3
Specific gravity of mercury	=	13.6
specific gravity of methylated spirit	=	0.78
International nautical mile	=	1852 m, or approx 6076 ft
1 knot	=	0.514 m/s
1 ft	=	0.3048 m
Radius of earth	=	6370 km
Diameter of the moon	=	3490 km
Distance of the moon from the earth	=	385000 km

Aerofoil data

C_D for flat plate at right angles	=	1.2
Cylinder	=	0.6
Streamline shape	=	0.06
Pitot tube	=	1.00

Take the maximum length in the direction of motion for the length L in the Reynolds Number formula.

At standard sea-level conditions

Acceleration of gravity	=	9.8 m/s^2
Atmospheric pressure	=	101.3 kN/m^2, or 1013 mb, or 760 mmHg
Density of air	=	1.225 kg/m^3, at 1013 mb and 288%
Speed of sound	=	340m/s = 661 knots = 1225 km/h
Dynamic viscosity of air (u)	=	17.894 x 10^4 kg/ms

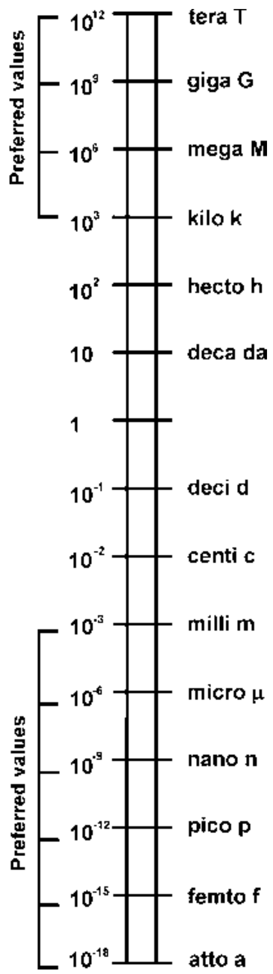
For low altitudes one millibar change in pressure is equivalent to 30 feet change in altitude.

Metrication

Table for Converting Imperial System of Units to S.I. System of Units

Basic S1 Units

Quantity	**Unit**	**Symbol**
Length	meter	m
mass	kilogramme	kg
time	second	s
electric current	ampere	A
temperature	kelvin	K
luminous intensity	candela	cd
amount of substance	mole	mol



Derived units with special names

(Derived SI units are coherent, unit value of a derived quantity is obtained by multiplication and division of unit values of basic quantities)

Quantity	**Unit**	**Symbol**	**Derivation**
force	newton	N	kg. m/s^2
work, energy	joule	J	Nm
power	watt	W	
electric charge	coulomb	C	A s
potential difference	volt	V	W/A
electric resistance	ohm	Ω	V/A
electric capacitance	farad	F	As/V
inductance	henry	H	Vs/A

Some examples of other derived SI units

Quantity	**Symbol**	**Quantity**	**Symbol**
area	m^2	pressure,stress	N/m^2
volume, section	m^3	kinematic viscosity	m^2/s
modulus		dynamic viscosity	Ns/m^2
second moment of	m^4	impulse	Ns
area		momentum	kg m/s
second moment of	kg m^2	impulse of torque	N ms
mass		moment of	kg m^2/s
(Moment of inertia)		momentum	
density	kg/m^3	kinetic energy	kg m^2/s^2
weight/volume	N/m^3	specific heat	J/kg C
angular acceleration	rad/s^2	capacitance	
velocity	m/s	specific entropy	J/k gK

Supplementary units

Quantity	**Unit**	**Symbol**
plane angle	radian	rad
solid angle	steradian	sr

Non-SI units the use of which is expected to continue

Quantity	**Unit**	**Symbol**	**Derivation**
time	minute	min	60s
time	hour	h	3.6 ks
customary temperature	degree Celsius	C	θk-273.15
temperature difference	degree Celsius	C	$(\theta_2 - \theta_1)$K
angle	degree	θ	π/180 rad
volume	litre	1	dm^3
speed	kilometre per hour	km/h	–
angular speed	revolution per second	rev/s	–
angular speed	revolution per minute	rev/min	–
frequency	hertz	Hz	cycle/s
pressure	bar	b	10^2 kN/m^2
stress	hectobar	hb	10 MN/m^2
kinematic viscosity	stoke	St	100 mm^2/s
dynamic viscosity	poise	P	100 mNs / m^2

Conversion Factors

Note that $1mm^3 = 1\ (mm)^3$, not 1m (m^3)

Basic	**1ft=0.3048m (exactly)**	**1lb=0.4536kg (0.453 592 37 exactly)**	**1^o=17.45 milli radian**
Derived			
Length	1in = 25.4 mm 1 mile = 1.609 km	1 'thou' = 25.4 μm 1 nautical mile = 1.853 km	1yd = 0.9144 m
Mass	1 slug = 14.95 kg 1 cwt = 50.80 kg	1 ton = 1.016 Mg 1 stone = 6.350 kg	1 US short ton = 907.2 kg 1 oz = 28.35 g
Time	1 min = 60s	1h = 3.6 ks	1 day = 86.4 ks
Area	1 in^2 = 645.16 mm^2 = 6.452 cm^2 1 yd^2 = 0.8361 m^2	1 ft^2 = 0.092 m^2 = 929.cm^2 1 acre = 4047 m^2	 1 $mile^2$ = 2.590 km^2
Volume (section modulus)	1 in^3 = 16.39 cm^3 1 UK gallon = 4.546L	1 ft^3=0.028 32m^3 = 28.32L 1 US gallon = 3.785L	1yd^3 = 0.7646 m^3 = 764.6L (L= litres)
Second moment of area	1 in^4 = 41.62 cm^4	1 ft^4 = 86.31 dm^4	
Force, weight	1 Ibf = 4.448N 1kgf = 9.807 N (1 tonf = 10kN within the tolerance of a grade A, testing machine)	1pdL = 138.3 mN 1 tonf = 9.964 kN	1 dyne = 10 μN
Density	1 Lb/ft^3 = 16.02 kg / m^3 1 slug/ft^3 =515.4 kg / m^3	1 Lb/in^3 = 27.68 Mg / m^3 = 27.68 kg 1 ton / ft^3 = 35.88 Mg/m^3	
Specific volume	1 ft^3 / Lb = 62.431 / kg		
Weight / volume	1 Lbf / in^3 = 271.4 kN/m^3 1 tonf/ft^3 = 351.9 kN/m^3	1 Lbf/ft^3 = 0.1571 kN/m^3	
Velocity	1 ft/min = 5.08 mm/s 1 mile / h = 0.4470 m/s = 1.609 km/h; 1UK knot = 0.5148 m/s = 1.853 km/h	1 km/h = 0.2778 m/s	1 ft /s = 0.3048 m/s
Pressure, stress	1 Lbf/ft^2 = 47.88 N/m^2 = 0.4788 mb; 1 in H_2O = 249.1 N/m^2 = 2.491 mb 1 ft H_2O = 2.989 kN/m^2 = 29.89 mb; 1 in Hg = 3.386 kN/m^2 = 33.86 mb 1 Lbf / in^2 = 6.895 kN/m^2 = 68.95 mb; 1 tonf/ft^2 = 107.3 kN m^2 = 1.073b 1 tonf/in^2 = 15.44 MN/m^2 = 1.544hb; 1 kgf / mm^2 = 9.807 MN / m^2 = 0.9807 hb		
Power	1 horsepower = 745.7 W		
Moment, torque	1 ft pdL = 42.14 mN m 1 ft tonf = 3.037 kN m	1 in tonf = 253.1 Nm 1 ft Lbf = 1.356 Nm	1 in Lbf = 113.0 mNm
Impulse	1 Lbf s = 4.448 Ns		
Impulse of torque	1 Lbf fts = 1.356 N ms		
Momentum	1 slug ft/s = 4.448 kg m/s	1 Lb ft/s = 138.3 g m/s	
Moment of momentum	1 slug ft^2/s = 1.356 kg m^2/s	1 Lb ft^2/s = 42.14 gm^2/s	
Second moment of mass (moment of intertia)	1 slug ft^2 = 1.356 kg m^2 1 Lb in^2 = 292.6 mg m^2	1Lb ft^2 = 42.14 g m^2	
Rates of flow	1 gal/h = 1.263 ml/s = 4.546 L/h 1 ft^3/h = 7.866 ml/s = 28.32 L/h		

	1 gal/min = 75.77 ml/s = 4.546 L/min 1 ft^3/s = 28.32 L/s; ILb/h = 126.0 mg/s=0.4536 kg/h.
Fuel consumption	1 mile/gal = 0.3540 km/L
Kinematic viscosity	1 ft^2/s = 929.0 cm^2/s = 929.0 St 1 c St = 1 mm^2/s
Dynamic viscosity	1 Lbf s/ft^2 = 47.88 Ns/m^2 = 478.8 P 1 pd Ls/ft^2 = 1.488 Ns/m^2 = 14.88 P 1 cP = 1 mN s/m^2
Energy	1 ft Lbf = 1.356J; 1 horsepower h = 2.685 MJ; 1 cal (IT) = 4.187J 1 kWh = 3.6MJ; 1 Btu = 1.055 KJ; 1 therm = 105.5 MJ
Specific energy	1 Btu/Lb = 2.326 KJ/KJ

Calorific value	1 Btu / ft^3 = 37.26 kJ/m^3
Heat transfer rate	1 Btu/h = 293.1 mW 1 ton of refrigeration = 3.517 W
Specific entropy	1 Btu/1b R = 4.187 kJ/kgK
Specific heat capacitance	1 Btu/1b F = 4.187 kJ/kgC
Mass	1 Ib mol = 0.4536 mol
Intensity of heat transfer rate	1 Btu / ft^2;h = 3.155W/m^2
Specific consumption	1 Ib/horsepower h = 169.0 μg/j = 0.6083 kg/k Wh
Specific impulse	1 Ibfs/Ib = 9.807 N s/kg

Selected magnitudes of typical engineering quantities in SI units

The earth	equatorial radius,		3960 mile = 6.37 Mn
	mean solar distance,		92.9 × 10^6 mile = 150 Gm
	angular velocity,		1 rev/day = 72.9 μ rad/s
	standard gravitational acceleration,		32,1740 ft/s^2 = 9.80665 m/s^2
Mankind	height,	5 ft = 1.60m;	6 ft = 1.90 m
	mass,	8 stone = 51 kg;	18 stone = 114 kg
	weight,	112 Lbf = 498 N;	252 Ibf = 1.12 kN
Density	gold,		0.695 Ib/in^3 = 19.2 Mg/m^3
	steel,		0.283 Ib/in^3 = 7.83 Mg/m^3
	aluminium,		0.0925 Ib/in^3= 2.56 Mg/m^3
	concrete,		144 Ib/ft^3 = 2.31 Mg /m^2
	Water		1.94 slug/ft^3 = 1.00 Mg/m^3
	air at sea level,		0.00238 slug/ft^3 = 1.23 kg/m^3
Weight/volume (on earth)	steel,		0.283 Ibf/in^3 = 76.8 kN/m^3
	aluminium,		0.0925 Ibf/in^3 = 25.1kN/m^2
	concrete,		144 Ibf/ft^3 = 22.6 kN/m^3
	water,		62.4 Ibf/ft^3 = 9.81 kN/m^3
Young's modulus	steel,	(29.5 ± 0.5) × 10^6 Ibf/m^2 = 203.3 GN/m^2 (say 200 GN/m^2)	
	aluminium	10 ×10^6 I bf/m^2 = 69.0 GN/m^2 9say 70 GN/m^2)	
	timber,	1.2 to 1.6 ×10^6 I bf/m^2 = 8.3 to 11.0 GN/m^2	
Stresses	soils,	1 ton / ft^2 = 107.3 kN/m^2	
	brickwork,	100 Ibf/m^2 = 689 kN/m^2	
	concrete,	1000 Ibf/in^2 = 6.89 kN/m^2	
	steels,	20000 Ibf/m^2 = 138 MN/m^2 = 13.8 hb	
		10 tons/in^2 = 154 MN/m^2 = 15.4 hb	
		100 tonf/in^2 = 1.54 GN/m^2 = 154 hb	
Breaking load	of 1 in steel wire rope,		3 tonf = 29.9 kN
Loadings, wind, floor,			5 Ibf/ft^2 = 239 N/m^2
Wing			100 Ibf/ft^2 = 4.79 kN/m^2
Diametral pitches	1 DP=25.4 mm module 24 DP = 1.06 mm module 200 DP = 0.127 mm module		
Universal beam properties	Section		
	in × in ×Ib/ft	Ix	Zx
	36 × $16^1/_2$	17230 in^4 = 71.7 dm^4	951 in^3 = 15.6 d m^3
	18 × $7^1/_2$ × 66	1097 in^4 = 4.57 dm^4	119 in^3 = 1.95 d m^3

	$8 \times 5^1/_4 \times 20$	$69.2 in^4 = 2884\ cm^4$ $= 0.288\ dm^4$	$17.0\ in^3 = 279\ cm^3$ $0.279\ dm^3$
Velocities	cutting speed rate of climb, 100 ft/ min = 0.508 m/s, 5000 ft/min = 25.4 m/s		
	Vehicles,	30 miles/h = 48.3 kmh = 13.4 m/s	
		100 mile/h = 161 km/h = 44.7 m/s	
		100 knot = 185 km/h = 51.5 m/s	
		1000 knot = 1.85 Mm/h = 515 m/s	
	Sonic velocity in air at sea level, 762 mile/h = 1.23 Mm/h = 341 m/s		
	escape velocity,		25000 mile/h = 11.2 km/s
	light,		186000 mile/s = 299 Mm/s
Fluid properties at 60^0F	Surface tension of water, 0.005 Ibf/ft = 73.6 mN/m		
	Kinematic viscosity of lubricating oil (SAE 20) $3.0 \times 10^{-3}\ ft^2/s = 279\ mm^2/s$		
		of water,	$12.1 \times 10^{-6}\ ft^2/s = 1.12\ mm^2/s$
		of air,	$160 \times 10^{-6}\ ft^2/s = 14.9\ mm^2/s$
	dynamic viscosity of lubricating oil (SAE 20) 5.7×10^{-3} Ibfs / ft^2 $= 273\ mNs/m^2$		
		of water,	23.4×10^{-6} Ibfs /ft^2 = 1.12 mNs/m^2
		of air,	0.370×10^{-6} Ibfs /ft^2 = 17.7 μ Ns/m^2
	bulk modulus of water, 311000 Ibf/in^2 = 2.14 GN/m^2		
		of air,	2970 Ibf / ft^2 = 142 kN/m^2
Pressure	14.7 Ibf/in^2 = 101 kN/m^2 = 1.01 b		
	1000 Ibf/in^2 = 6.89 MN/m^2 = 68.9 b		
Dynamic pressure	in air at 500 ft/s, 297 Ibf/ft^2 = 14.2 kN/m^2 = 142 mb		
	in water at 15 ft/s, 218 Ibf/ft^2 = 10.4 kN/m^2 = 104 mb		
Velocity	$\sqrt{(2\ gh)}$ for head of 10 ft, 25.4 ft/s = 7.74 m/s		
Head	$u^2/2g$ for velocity of 15 ft / s, 3.50 ft /s = 1.07 m		
Quantity flow rate	at 10 ft/s in 6 in pipe, 1.96 ft^3/s = 55.51/s		
		100 gal/min = 7.58 ℓ /s	
Chezy coefficient	100 $ft^{1/2}$ /s = 55.2 $m^{1/2}$ /s		
Manning coefficient	100 $ft^{1/3}$/s = $67.3^{1/3}$ /s		
Vee notch calibration	if V = 2.5 H^{25} ft^3/s with H in ft then V = 1.38 H^{25} m^3/s with H in m		
Head loss in circular pipe	$h = \dfrac{4\ flu^2}{2\ gd} = \dfrac{flv^2}{3d^5}$ with dimensions in metres		
Brake mean effective pressure	120 Ibf/in^2 = 827 kN/m^2 = 8.27b		
Calorific value	of petrol, 19500 Btu/Ib = 45.4 MJ/kg		
	of coal gas, 500 Btu/ft^3 = 18.6 MJ/m^3		
Specific heat capacitance of air	C_p = 0.240 Btu/Ib F = 1.00 kJ/kgC		
	C_v = 0.171 Btu/Ib F = 0.71 kJ/kgC		
Universal gas constant	1545 ft Ibf/Ib mol °R = 8313 J/mol K		
Gas constant	for air, 53.3 Ibf/Ib °R = 287J/kg K		
	for steam, 85.8 ft Ibf/Ib °R = 462 J/kg K		
Properties of saturated steam	at 100 Ibf/in^2 abs		
	specific volume		v_s = 4.43 ft^3/Ib = 0.277 m^3/kg = 2771/kg
	specific enthalpy of evaporation		h_{fs} = 890 Btu/Ib = 2.07 MJ/kg
	specific entropy of evaporation		s_{fs} = 1.13 Btu/Ib °R = 4.73 kJ/kg K
Specific impulse	250 Ibf s/Ib = 2.45 kN s/kg		

Units and dimensions

Quantity	**Dimension**	**Unit (name and abbreviation)**
Length	L	Metre (m)
Mass	M	Kilogramme (kg)
Time	T	Second (s)
Temperature	θ	Degree Celsius (^{0}C), Kelvin (K)
Area	L^2	Square metre (m^2)
Volume	L^3	Cubic metre (m^3)
Speed	LT^{-1}	Metres per second (ms^{-1})
Acceleration	LT^{-2}	Metres per second per second (ms^{-2})
Angle	1	Radian or degree (^{0}) (The radian is expressed as a ratio and is therefore dimensionless)

Angular velocity	T^{-1}	Radians per second (s^{-1})
Angular acceleration	T^{-2}	Radians per second per second (S^{-2})
Frequency	T^{-1}	Cycles per second hertz (s^{-1}; Hz)
Density	ML^{-3}	Kilograms per cubic metre (kg/m^3)
Force	MLT^{-2}	Newton (N)
Stress	$ML^{-1}\,T^{-2}$	Newtons per square metre or Pascal (N m^2 of Pa)
Strain	1	None (expressed as %)
Pressure	$ML^{-1}\,T^{-2}$	Newtons per square metre or Pascal (Nm^2 or Pa)
Energy work	$ML^2\,T^{-2}$	Joule (J)
Power	$ML^2\,T^{-2}$	Watt (W)
Moment	$ML^2\,T^{-2}$	Newton metre (Nm)
Absolute viscosity	$ML^{-1}\,T^{-1}$	Kilogram per metre second or Poiseuille (kg $m^{-1}s^{-1}$ or PI)
Kinematic viscosity	$L^2\,T^{-1}$	Metre squared per second ($m^2\,s^{-1}$)
Bulk elasticity	$ML^{-1}\,T^{-2}$	Newtons per square metre or Pascal (Nm^{-2} or Pa)

The International Standard Atmosphere

(1) Sea level conditions

$To = +15^0C = 288.16\,K$

$Po = 101325\,N/m^2$

$\rho o = 1.2256\,kg/m^3$

$\mu o = 1.783 \times 10^{-5}$ kg/ms.

$\upsilon o = 1.455 \times 10^{-5}\,m^2/S$.

(2) Relative values

Altitude (m)	Temperature	Pressure	Density		Viscosity	
	$\theta = T/T_0$	$\bar{\omega} = P/P_0$	$\sigma = \rho/\rho_0$	$\sigma^{1/2}$	$\bar{\mu} = \mu/\mu_o$	$\bar{\nu} = v/v_o$
0	1	1	1	1	1	1
250	0.9944	0.9707	0.9762	0.9880	0.9956	1.0198
500	0.9887	0.9421	0.9528	0.9761	0.9911	1.0402
750	0.9831	0.9142	0.9299	0.9643	0.9867	1.0610
1000	0.9774	0.8869	0.9074	0.9526	0.9822	1.0824
1250	0.9718	0.8604	0.8853	0.9409	0.9777	1.1044
1500	0.9661	0.8344	0.8637	0.9293	0.9733	0.1269
1750	0.9605	0.8091	0.8424	0.9178	0.9688	1.1500
2000	0.9549	0.7845	0.8215	0.9064	0.9642	1.1737
2250	0.9492	0.7604	0.8011	0.8950	0.9597	1.1980
2500	0.9436	0.7369	0.7810	0.8837	0.9552	1.2230
2750	0.9379	0.7141	0.7613	0.8725	0.9506	1.2487
3000	0.9323	0.6918	0.7420	0.8614	0.9461	1.2750
3250	0.9266	0.6701	0.7231	0.8503	0.9415	1.3020
3500	0.9210	0.6489	0.7045	0.8394	0.9369	1.3298
3750	0.9154	0.6283	0.6863	0.8285	0.9323	1.3584
4000	0.9097	0.6082	0.6685	0.8176	0.9277	1.3877
4250	0.9041	0.5886	0.6511	0.8069	0.9231	1.4178
4500	0.8984	0.5696	0.6339	0.7962	0.9184	1.4488
4750	0.8928	0.5510	0.6172	0.7856	0.9138	1.4806
5000	0.8872	0.5329	0.6007	0.7751	0.9091	1.5133
5250	0.8815	0.5154	0.5846	0.7646	0.9044	1.5470
5500	0.8759	0.4983	0.5689	0.7542	0.8997	1.5816
5750	0.8702	0.4816	0.5534	0.7439	0.8950	1.6172
6000	0.8646	0.4654	0.5383	0.7337	0.8903	1.6538
6250	0.8589	0.4497	0.5235	0.7236	0.8855	1.6915
6500	0.8533	0.4344	0.5091	0.7135	0.8808	1.7303

6750	0.8477	0.4195	0.4949	0.7035	0.8760	1.7702
7000	0.8420	0.4050	0.4810	0.6936	0.8713	1.8113
7250	0.8364	0.3910	0.4674	0.6837	0.8665	1.8536
7500	0.8307	0.3773	0.4542	0.6739	0.8617	1.8972
7750	0.8251	0.3640	0.4412	0.6642	0.8568	1.9421
8000	0.8194	0.3511	0.4285	0.6546	0.8520	1.9884
8250	0.8138	0.3386	0.4161	0.6450	0.8471	2.0361
8500	0.8082	0.3264	0.4039	0.6356	0.8423	2.0852
8750	0.8025	0.3146	0.3921	0.6262	0.8374	2.1359
9000	0.7969	0.3032	0.3805	0.6168	0.8325	2.1881
9250	0.7912	0.2921	0.3691	0.6076	0.8276	2.2400
9500	0.7856	0.2813	0.3581	0.5984	0.8227	2.2976
9750	0.7799	0.2708	0.3472	0.5893	0.8177	2.3549
10000	0.7743	0.2607	0.3367	0.5802	0.8128	2.4141
10250	0.7687	0.2509	0.3264	0.5713	0.8078	2.4752
10500	0.7630	0.2413	0.3163	0.5624	0.8028	2.5383
10750	0.7574	0.2321	0.3064	0.5536	0.7978	2.6034
11000	0.7517	0.2232	0.2968	0.5448	0.7928	2.6707
11500	Constant	0.2062	0.2743	0.5238	Constant	2.8897
12000	in	0.1906	0.2535	0.5035	in	3.1268
12500	strato-	0.1761	0.2343	0.4841	strato-	3.3833
13000	sphere	0.1628	0.2166	0.4654	sphere	3.6608
13500		0.1505	0.2001	0.4474		3.9611
14000		0.1390	0.1850	0.4301		4.2860
14500		0.1285	0.1709	0.4135		4.6376
15000		0.1188	0.1580	0.3975		5.0180
15500		0.1098	0.1460	0.3821		5.4297
16000		0.1014	0.1349	0.3673		5.8751
16500		0.0937	0.1274	0.3531		6.3570
17000		0.0866	0.1153	0.3395		6.8785
17500		0.0801	0.1065	0.3264		7.4427
18000		0.0740	0.0984	0.3138		8.0532
18500		0.0684	0.0910	0.3016		8.7138
19000		0.0632	0.0841	0.2900		9.4286
19500		0.0584	0.0777	0.2788		10.202
20000	Constant	0.0540	0.0718	0.2680	Constant	11.039
20500	in	0.0499	0.0664	0.2576	in	11.945
21000	strato	0.0461	0.0613	0.2477	strato	12.924
21500	sphere	0.0426	0.0567	0.2381	sphere	13.985
22000		0.0394	0.0524	0.2289		15.132
22500		0.0364	0.0484	0.2200		16.373
23000		0.0336	0.0447	0.2115		17.716
23500		0.0311	0.0414	0.2034		19.169
24000		0.0287	0.0382	0.1955		20.742
24500		0.0266	0.0353	0.1879		22.443
25000		0.0245	0.0326	0.1807		24.284

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SYMBOLS AND NOTATION USED IN AERODYNAMICS

A	Moment of inertia about OX. Aspect ratio [also *(AR)*]. Area. With suffices, coefficients in a Fourier series of sine terms, or a polynomial series in z. Coefficient of stability quartic.
AF	Activity factor of an airscrew.
AR	Aspect ratio (also *A)*.
a	Speed of sound. Axial inflow factor in airscrew theory. Radius of transformed circle. Lift curve slope. $dC_L/d\alpha$ (suffices denote particular values.) Radius of vortex core. Acceleration or deceleration.
B	Number of blades on an airscrew. Coefficient of stability quartic.
b	Rotational interference factor in airscrew theory. Zhukovsk's transformation factor ($\xi = z + b^2/z$). Total wing span (=2s). Hinge moment coefficient slope.
CG	Centre of gravity.
C	Coefficient of stability quartic.
C_D	Total drag coefficient.
C_{Do}	Zero-lift drag coefficient.
C_{D_V}	Trailing vortex drag coefficient.
C_{D_L}	lift-dependent drag coefficient. (Other suffices are used in particular cases.).
C_H	Hinge moment coefficient.
C_I	Rolling moment coefficient.
C_L	Lift coefficient.
C_M	Pitching moment coefficient.
C_N	Yawing moment coefficient.
C_P	Pressure coefficient.
C_P	Power coefficient for airscrews.
C_R	Resultant force coefficient.
c	Wing chord. Specific fuel consumption. A distance.
$\bar{c}$	Standard or geometric mean chord.
$\bar{C}$ or $\bar{C}_A$	Aerodynamic mean chord.
C_0	Root chord.
c_T	Tip chord.
c_C	Elevator mean chord.
c_f	Fin mean chord.
c_P/c_v	Specific heats at constant pressure and constant volume
CP	Centre of pressure.
D	Drag (suffices denote particular values). Airscrew diameter. A length (occasionally), Coefficient of stability quartic.

d	Diameter, occasionally a length.
d_v	spanwise trailing vortex drag grading (=ρωK).
E	Kinetic energy. Coefficient of stability quartic.
e	Eccentricity factor.
F	Fractional flap chord. Force. Mass of fuel.
f ( ), fn ( )	Function of the stated variables.
g	Acceleration due to gravity.
g( )	Function of the stated variables.
H	Hinge moment. Total pressure. Momentum.
h	Franctional camber of a flapped plate aerofoil. Fractional position of CG Distance between plates in Maxwell's definition of viscosity. Vertical eccentricity factor. Altitude, or height generally.
h ( )	Function of the stated variables.
h_o	Fractional position of the aerodynamic centre.
h_n	Fractional position of the neutral point,stick fixed.
$h^1{}_n$	Fractional position of the neutral point, stick free.
I	Momentum of rocket exhuast.
i	The imaginary operator, $\sqrt{-1}$.
J	Advance ratio of an airscrew.
K	Circulation. Modulus of bulk elasticity.
K_O	Circulation at mid-section of a wing.
k	Chordwise variation of vorticity. Lift-dependent drag coefficient factor.
k_{CP}	Centre of pressure coefficient.
k_T'k_Q	Thrust and torque coefficients (airscrews).
L	Lift. Dimension of length.Temperature lapse rate in the atmosphere
LR	Rolling moment about O*x*.
l	Length.
l_{dc}	Effective disc loading of a helicopter
$I_t I_f$	Tailplane and fin moment arms.
l	Rolling moment derivatives.
M	Dimension of mass. Mass of a rocket missile. Mach number. Pitching moment about O*y*.
M_O	Initial mass of a rocket missile.
M_l	All-burnt mass of a rocket missile.
m	Mass. Strength of a source (-sink). Ratio (flight speed/minimum drag speed). An index.
$\dot{m}$	Rate of mass flow. Rate of fuel consumption of rocket missle.
N	Yawing moment about O*z*. rpm of an airscrew.
n	Revolutions per second of an airscrew. Load factor in a turn. Frequency. Ratio (flight speed / minimum power speed). An index.
O	Origin of coordinates.

P	Power. The general point in space.
p	Static pressure in a fluid. Angular velocity in roll about Ox.
Q	Torque, or a genernal moment. Total velocity of a uniform stream.
q	Angular velocity in pitch about Oy. Local resultant velocity. A coefficient in airscrew theory.
q_n, q_t	Radial and tangential velocity components.
Re	Real part of a complex number.
Re	Reynolds' number.
R	Resultant force. Characteristic gas constant. Force between group and undercarriage. Range. Mass ratio (M_0/M_1) of a rocket missile. Radius of turn, or of a circle.
RoC	Rate of climb.
r	Angular velocity in yaw about Oz. Radius vector, or radius generally
S	Wing area. Vortex tube area. Area of actuator disc.
S'	Tailplane area.
S_t	Fin area.
S_e	Elevator area.
s	Semi-span ($=\frac{1}{2}b$). Distance.
s'	Spacing of each trailing vortex centre from aircraft centre-line.
T	Dimension of time. Temperature (suffices denote particular values).
T_c	Static thrust.
t	Time. Aerofoil section thickness. A coefficient in airscrew theory.
f	Unit of aerodynamic time.
U	Velocity, Steady velocity parallel to Ox.
u	Velocity component parallel to Oxz.
V	Velocity, Volume. Steady velocity parallel to Oy.
V_3	Stalling speed.
V_E	Equivalent air speed.
Vn	Resultant speed.
$\overline{V}$	Tail volume ratio.
$\overline{V}_f$	Fin volume ratio.
υ	Velocity component parallel to Oy. Velocity. Velocity of exhaust relative to rocket motor. Rate of climb or descent.
W	Weight. Steady velocity parallel to Oz.
ω	Wing loading. Downwash velocity. Complex potential function ($=\phi + i\Psi$). Velocity parallel to Oz.
X, Y, Z	Components of aerodynamic or external force.
x, y, z	Coordinates of the general point P.
x, X	Distance.
z	The complex variable ($= x + iy$).
α	Angle of incidence or angle of attack. An angle, generally.

α_1	Tail-setting angle.
β	An angle defining vertical shift in Zhukovsky's transformation. A factor in airscrew theory. Tab deflection. An angle generally.
Γ	Half the dihedral angle;the angle between each wing and the Oxy plane.
γ	Ratio of specific heats, C_P/C_V Gliding angle (= arc cot (L/D)).
δ	Boundary layer thickness A factor. Camber of an aerofoil section.
δ_f	Flap deflection (high lift).
ε	Downwash angle. Surface slope.
ζ	Vorticity. Complex variable in transformed plane ($\xi + i\eta$). Rudder deflection.
η	Efficienty. Elevator deflection. Ordinate in ζ–plane.
θ	Dimension of temperature. Angular displacement in pitch. Angle of climb. Polar angular coordinate. Blade helix angle (airscrews).
Λ	Angle of sweepback or sweep-forward.
λ	Taper ratio (C_T/C_O). A constant.
μ	Strength of a doublet. Absolute coefficient of viscosity. Aerofoil parameter in lifting line theory.
v	Kinematic coefficient of viscosity.
ξ	Aileron deflection. Abscissa in ζ–plane.
ρ	Density. Radius of curvature.
Σ	Summation sign.
σ	Relative density. Blade or annular solidity (airscrews). Sidewash angle at fin.
τ	Intensity of viscous traction.
τ_O	Intensity of surface friction.
ϕ	Angular displacement in bank about Ox. Sweepback angle. Velocity potential. A polar coordinate. Angle of relative wind to plane of airscrew disc.
Ψ	Angular displacement in yaw, or sideslip angle. Angle turned through in circling flight. The stream function.
Ω	Angular velocity of airscrew.
ω	Angular velocity in general.
$\bar{\nabla}^2$	Laplace's operator $\bar{\nabla}^2$ $(=\partial^2/\partial x^2 + \partial^2/\partial y^2)$
0	No lift. Standard sea level. Straight and level flight. Undisturbed steam.
1/4	Quarter chord point.
1	Rates of change of aerodynamic characteristics with main surface incidence, or otherwise a particular value.
2	Rates of change of aerodynamic characteristics with flap or control deflection, or otherwise a particular value.
3	Rate of change of aerodynamic characteristics with tab deflection, or otherwise a particular value.
∞	Infinity or two-dimensional conditions.
AC	Aerodynamic centre.
a	available.
CP	Centre of pressure.
FN	Fuselage-nacelle contribution.

fus	Fuselage contribution.
f	Full scale or flight.
G	Gross.
g	Ground.
h	Horizontal.
i	Ideal.
in	Input.
L	Landing. Lower surface.
LE	Leading edge.
l	Local.
m	Model.
max	Maximum.
min	Minimum.
md	Minimum drag.
mp	Minimum power.
mr	Maximum range.
N	Net.
Na	Nacelle contribution.
n	Denotes general term.
opt	Optimum.
out	Output.
p	Propulsive.
r	Required.
s	Stagnation or reservoir conditions. Slipstream. Stratosphere.
TO	Take-off.
TE	Trailing edge.
U	Upper surface.
V	Vertical

Primes and superscripts.

^{1}	Stick free.
’	Tailplane.
”	Fin.
*	Throat (locally sonic) conditions.

The dot notation is frequently used for differentials, e.g.y = dy/dx, the rate of chage of y with x..

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IMPORTANT FORMULAS OF AERODYNAMICS

HYDROSTATICS AND AEROSTATICS

1. $\dfrac{dp}{dh} = -\rho g.$ (Bouyancy equation).

2. $\dfrac{P}{\rho} = RT$ (Aerodynamic form of gas equation).

3. $\dfrac{P_2}{P_1} = \dfrac{\rho_2}{\rho_1} = \exp\left[\dfrac{g(h_1 - h_2)}{RT_s}\right]$ (equation for pressure and density in stratosphere)

4. $T = T_0 - Lh$ $\left[\begin{array}{l}\text{equation for temp at particular height. Where } T_0 = \text{Temp at a sea level or absolute} \\ \text{temp, L : Lapserate} = .0065 \text{ for troposphere and h Height or altitude T : Temp} \\ \text{at desired altitude or height.}\end{array}\right]$

5. $\dfrac{P_2}{P_1} = \left(\dfrac{T_2}{T_1}\right)^{g/LR} = \left(\dfrac{T_2}{T_1}\right)^{5.256}$ (equation for pressure in troposphere)

6. $\dfrac{\rho_2}{\rho_1} = \left(\dfrac{T_2}{T_1}\right)^{\frac{g-LR}{LR}} = \left(\dfrac{T_2}{T_1}\right)^{4.256}$ (equation for density in troposphere)

7. $\dfrac{P_2}{P_1} = \left(\dfrac{\rho_2}{\rho_1}\right)^{\frac{g}{g=LR}}$ (in troposphere)

8. $p = k\,\rho^{\left(\frac{g}{g-LR}\right)} = k(\rho)^{1.256}$ (in troposphere)

9. **Relative Density**

$$(\sigma) = \frac{\rho}{\rho_o} = \frac{\rho}{1.2256} = \frac{\text{Density at relavent Altitude}}{\text{Density at sea level}}$$

also in F.P.S. $\rho_o = 0.002378 \text{ Lb/ft}^3$

10. **Variation of viscocity** $\dfrac{\mu_2}{\mu_1} = \left(\dfrac{T_2}{T_1}\right)^{\frac{3}{4}}$ where T_2 is 273 k.

where μ : absolute coefficient of viscosity.
The value of μ at sea level $= 1.714 \times 10^{-5}$ kg/msec $= \mu_2$

11. **Kinematic viscocity** $(\upsilon) = \dfrac{\mu_o}{\rho}$ (at particular attitudes)

unit of υ is m^2 /S.

12. $n = \dfrac{g}{g - LR}$ if $n = \gamma = 1.4$ which is critical value corresponding to neutral stability, then L = .00975 k/m. If lapse rate exceeds this value then atm is unstable.

WING AND AEROFOIL SECTION GEOMETRY

13. **Wing Span** = b = 2s (i.e. twice of semi span).

14. **Wing Area** = Span × Meancord = b × c.

15. **Mean Cord** = $\overline{C} = \dfrac{S_G \text{ or } S_n}{b}$

or $\overline{C} = \dfrac{\int_{-s}^{+s} C.dy}{\int_{-s}^{+s} dy}$

16. **Aerodynamic Mean Cord** $(\overline{C} \text{ or } \overline{C}_A)$.

$$\overline{C} = \frac{\int_{-s}^{+s} C^2.dy}{\int_{-s}^{+s} C.dy}$$

17. **Aspect Ratio** = $\dfrac{\text{span}}{\text{S.M.C.}} = \dfrac{b}{C}$ if multiplied by 'b' and divided by b

$$AR = \frac{b^2}{\overline{c}b} = \frac{b^2}{\text{area}}$$

18. **Dihedral Angle** = 2 (hydral angle)

$= 2\Gamma$

AIR FLOW

19. $\dfrac{P}{W} + Z + \dfrac{V^2}{2Y}$ = constant (Bernoulli's equation).

20. $\dfrac{P_s}{P} = \left[1 + \dfrac{M^2}{5}\right]^{3.5}$ (where P_s: Pressure at stagnation point).

21. $\dfrac{\rho_s}{\rho} = \left[1 + \dfrac{1}{5}M^2\right]^{2.5}$

22. $\dfrac{T_s}{T} = \left[1 + \dfrac{M^2}{5}\right]$ (where T_s : temp at stagnation point).

23. $V_E = V\sqrt{\sigma}$ where V_E : equivalent air speed.

$= \text{T.A.S.}\sqrt{\sigma}$ V : True Air Speed.

24. T.A.S. = Ma where $a = \sqrt{\gamma RT}$. where $\gamma = 1.4$

25. $P_s - P = \frac{1}{2}\rho_o V_E^2$ where $\rho_o = 1.2256$.

26. **Pressure Co-efficient** $= C_P = \dfrac{P - P_o}{\frac{1}{2}\rho V^2}$

Stagnation Pressure Co-efficient $= C_{Ps} = \dfrac{P_s - P}{\frac{1}{2}\rho V^2}$

$C_P = 1 - \left(\dfrac{q}{v}\right)^2$

where q = Speed of flow at point where p is measured.

$0.7\,pM^2 = \frac{1}{2}\rho V^2$

where ρ = Density at particular ht and V = q

AERODYNAMIC FORCE

27. **Reynold's Number** $= \dfrac{VD\rho}{\mu}$ where V = T.A.S.

D = Span

ρ = Density of particular given condition

μ = Coefficient of absolute visocity.

28. $\dfrac{F}{\rho V^2 D^2} = \text{function}\left[\dfrac{VD}{\upsilon}; M\right]$ is Rayleigh's equation.

29. **For dynamically Similar body or flow R will be equal**

$\therefore V_f \rho_f d_f = V_m \rho_m d_m$

30. **Mach number (M)** $= \dfrac{V}{a} = \dfrac{\text{T.A.S.}}{\text{speed of sound}}$

where $a = \sqrt{\gamma RT}$

$R = 287.3$

$\gamma = 1.4$

31. $\dfrac{\eta_f d_f}{V_f} = \dfrac{\eta_m d_m}{V_m}$ where η = frequency of eddies.

32. $C_F = \dfrac{F}{\frac{1}{2}\rho V^2 S}$ where C_F : aerodynamic force coefficient.

33. $C_L = \dfrac{L}{\frac{1}{2}\rho V^2 S}$ (C_L : Lift coefficient)

34. $C_D = \dfrac{D}{\frac{1}{2}\rho V^2 S}$ (C_D : Drag coefficient)

35. $C_M = \dfrac{M}{\frac{1}{2}\rho V^2 S\bar{\bar{C}}} = \dfrac{M}{\frac{1}{2}\rho V^2 S\bar{C}_A}$

36. $\dfrac{X_{ac}}{C} = \dfrac{a}{C} - \dfrac{d}{dC_L}(C_{Mac})$ (aerodynamic centre)

37. $K_{CP} \cong \dfrac{X_{ac}}{C} - \dfrac{C_{Mac}}{C_L}$ (Centre of pressure)

AIRSCREW AND PROPULSION

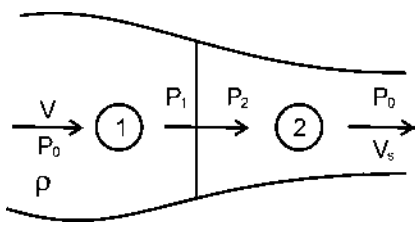
38. $T = \rho S\ V_o\ (V_s - V)$

39. $T = S\ (P_2 - P_1)$

40. $V_o = \frac{1}{2}\ (V_s + V)$

41. $V_o = V(1 + a)$

42. $V_s = V(1 + 2a)$
where a is inflow factor.



43. Rate of increase of energy through the disc.

$$\frac{dE}{dt} = \frac{1}{2}\rho s\ V_o\ (V_S^2 - V^2).$$

44. Efficiency of the disc as a propulsive system

$$\eta_i = \frac{TV}{\frac{1}{2}\rho S V_o\ (V_S^2 - V^2)}$$ where TV : useful work done.

$$= \frac{\rho_S\ V_o\ (V_S - V)\ V}{\frac{1}{2}\rho s\ V_o\ (V_S^2 - V^2)}$$

$$\eta_i = \frac{V}{\frac{1}{2}(V_S + V)}$$

This is ideal propulsive efficiency or froude's efficiency.

$$\boxed{\eta_i = \frac{2}{1 + \frac{V_S}{V}} = \frac{V}{V_o} = \frac{1}{1 + a}}$$

where $V_o = V(1 + a)$

45. Power Supplied $P = \dfrac{TV}{V/V_o} = TV_o$

46. $T = 2\rho SV^2\ a\,(1 + a)$

47. $T = K_T\, \rho n^2 D^4$ where

K_T : Coefficient of Thrust
D : diameter
n : no of revolution per sec

48. $Q = K_Q\, \rho n^2 D^5$ where Q is Torque,

K_Q : Torque coefficient

$$\eta = \frac{TV}{2\pi nQ} = \frac{K_T \rho n^2 D^4 v}{K_Q\, \rho n^2 D^5\, 2\pi n}$$

[where η is airscrew efficiency]

49. $Z = \dfrac{1}{2\pi} \cdot \dfrac{K_T}{K_Q} \cdot J$ $\left[\text{where J is advance ratio} = \dfrac{V}{nD}\right]$

50. Power required to drive airscrew.

$P = 2\pi nQ$

$= 2\pi n\,(K_Q\, \rho n^2\, D^5)$

$P = 2\pi\, n^3\, \rho K_Q\, D^5.$

51. Power coefficient C_P is then defined as

$P = C_P\, \rho n^3\, D^5$

$$C_P = \frac{P}{\rho n^3 D^5} = \frac{2\pi n^3\, \rho K_Q D^5}{\rho n^3 D^5} = 2\pi KQ.$$

52. a. $\eta = \dfrac{1}{2\pi} \cdot \dfrac{K_T}{K_Q} \cdot J$

b. $\eta = J\left(\dfrac{K_T}{C_P}\right)$ where C_P : Coefficient of Power

53. Active factor. (A.F.) $= \dfrac{10^5}{D^5} \displaystyle\int_{0.1D}^{0.5D} cr^3 dr$

54. $P = 4\pi^3 \rho\, C_D B n^3 \left(\dfrac{D}{10}\right)^5 \times (\text{A.F.})$

AIRCRAFT PERFORMANCE IN STEADY FLIGHT

55. $L = W \qquad T = D$

56. $L = W \ = \tfrac{1}{2} \rho\, V^2\, SC_L.$

57. $$V = \sqrt{\frac{W}{\frac{1}{2}\rho SC_L.}}$$

58. $$V_{stalling} = \sqrt{\frac{W}{\frac{1}{2}\rho SC_{L\,max.}}}$$

59. $$V_E = \sqrt{\frac{W}{\frac{1}{2}\rho_O C_L S}}$$ Where V_E is E.A.S. and is independent of Density $\rho \therefore \rho_o = 1.2256$

60. $$\text{Wing loading} = \frac{W}{S} = w$$

61. $$V = \sqrt{\frac{w}{\frac{1}{2}\rho C_L}}$$

62. $$C_D = C_{Do} + KC_L{}^2 \text{ where } (C_{Do} : \text{zero lift drag coefficent})$$
$$= C_{Do} + C_{DL} (\text{where} C_{DL} : \text{lift dependent drag coefficient})$$

63. $$D = \frac{1}{2}C_{Do}\rho V^2 S. + \frac{1}{2}KC_L{}^2 \rho V^2 S.$$ where K : lift dependent drag coefficient.

64. $$D = \frac{1}{2}C_{Do}\rho_o V_E{}^2 S + \frac{KW^2}{\frac{1}{2}\rho_o V_E{}^2 S}$$ [where V_E : equivalent air speed.]

65. $$C_{Do} = KC_L{}^2 = C_{DL} \text{ (for min drag)}$$

66. $$C_{Lmd} = \frac{C_{Do}}{K} \text{ or } C_{Lmd}\sqrt{\frac{C_{Do}}{K}}$$

67. $$V_{md} = \sqrt{\frac{w}{\frac{1}{2}\rho \cdot s\sqrt{\frac{C_{Do}}{K}}}} = \left(\frac{w}{\frac{1}{2}\rho S}\right)^{\frac{1}{2}} \left(\frac{K}{C_{Do}}\right)^{1/4}$$

68. $$C_{D m.d} = 2C_{Do} \because C_{Do} = KC_L{}^2 \text{ (for min drag condition).}$$

69. $$\frac{L}{D} = \frac{C_{L\ md}}{C_{Dm\cdot d}} = \frac{\sqrt{\frac{C_{Do}}{K}}}{2C_{Do}} = \sqrt{\frac{C_{Do}}{K}}\left(\frac{1}{2C_{Do}}\right) = \frac{1}{2\sqrt{KC_{Do}}}$$

70. **L/D Relation at other speed.**

$$W = \frac{1}{2}\rho C_L S(mV_{md})^2 = \frac{1}{2}\rho S V_{md}^2 C_{Lmd} \quad \left[\because m^2 C_L = C_{Lmd}\right]$$

71. $$C_L = \frac{C_{Lmd}}{m^2}$$

72. $$C_D = C_{Do}\left[1 + \frac{1}{m^4}\right]$$ (where m is multiple for min drag speed)

73. $$\frac{C_L}{C_D} \text{ or } = \frac{2m^2}{(1+m^4)} \times \frac{C_{Lmd}}{2C_{Do}} \qquad \text{......(i)}$$

$$\text{but} \left[\frac{L}{D}\right]_{max} = \frac{C_L}{C_D} = \frac{C_L}{C_{Do} + KC_L{}^2} = \frac{C_{Lmd}}{2\,C_{Do}} \qquad \text{.....(ii)}$$

Comparing two equations. i.e. (i) and (ii) we get

$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{2m^2}{(1+m^4)} \times \left[\frac{L}{D}\right]_{max}$$

$$\text{or} \quad \frac{L/D}{(L/D)_{max}} = \frac{2m^2}{(1+m^4)}$$

74. $$\frac{D}{D\,min} = \frac{(L/D)\,max}{(L/D)} = \frac{1}{2}\left[m^2 + m^{-2}\right]$$

75. $$\text{Min thrust required} = \frac{W}{(L/D)_{max}}$$

Power

76. $$P = DV = \left[\frac{1}{2}\rho V^2 S\,C_{Do} + \frac{KW^2}{\frac{1}{2}\rho V^2 S}\right] \times V$$ where V is T.A.S.

77. $$p_{min} = \left(\frac{w^3}{\frac{1}{2}\rho s}\right)^{1/2} \left(\frac{C_D}{C_L{}^{3/2}}\right)$$

78. $KC_{L_{mp}}^2 = 3C_{Do}$ (for min power.)

79. $$C_{L_{mp}} = \sqrt{\frac{3C_{Do}}{K}} \text{ but } C_{L_{md}} = \sqrt{\frac{C_{Do}}{K}}$$

80. $$C_{L_{mp}} = \sqrt{3}\,C_{L_{md}}$$

81. $$V_{mp} = \sqrt{\frac{W}{\frac{1}{2}\rho S C_{L_{mp}}}}$$

82. $$V_{mp} = \frac{V_{md}}{\sqrt{\sqrt{3}}} = V_{md} / 1.36 = 0.76\, V_{md}$$

83. $$\frac{(L/D)m.p}{(L/D)_{max}} = \frac{2m^2}{1+m^4} \qquad \text{where } m = 0.76$$

84. **Total drag coefficient at min power**

$$C_D = 4C_{Do}$$

85. $$\frac{C_{Lmp}}{C_{Dmp}} = \left(\frac{L}{D}\right)_{m.p} = \sqrt{\frac{3C_{Do}}{K}} \times \left(\frac{1}{4C_{Do}}\right) = \sqrt{\frac{3}{16\, KC_{Do}}}$$

86. **Suppose an a/c is flying at 'n' times its mps speed. Then**

$$C_L (n\, V_{mp})^2 = C_{L_{mp}} V^2{}_{mp}$$

$$n^2 C_L = C_{L_{mp}}$$

$$C_L = \frac{C_{L_{mp}}}{n^2}$$

$$C_D = C_{Do} + KC_L{}^2 = C_{Do} + \frac{KC^2_{L_{mp}}}{n^4}$$

$$= C_{Do} + \frac{3C_{Do}}{n^4}$$

$$C_D = C_{Do}\left(1 + \frac{3}{n^4}\right)$$

87. $$\frac{P}{P\,min} = \left(\frac{C_D}{C_{D_{mp}}}\right)\left(\frac{C_{L_{mp}}}{C_L}\right)\left(\frac{V}{V_{mp}}\right) = \frac{3+n^4}{4n}$$

88. **Drag at min power**

$$D = \frac{W}{(L/D)_{mp}}$$

Gliding flight (Shallow angle)

89. $$\sin\gamma = \frac{C_D}{C_L} \; [\text{i.e. when } \gamma \text{ is less than } 10°]$$

$$v = V \sin\gamma = \sqrt{\frac{W}{\frac{1}{2}\rho s C_L}} . \left(\frac{C_D}{C_L}\right)$$

where v : rate of descent or sinking speed.

90. **For steep angle**

$$v = \sqrt{\frac{W\cos\gamma}{\frac{1}{2}\rho s C_L}} = \sqrt{\frac{W}{\frac{1}{2}\rho s C_L}}(\cos\gamma)^{1/2}$$

91. $$\sin\gamma = \frac{C_D}{\sqrt{C_L^2 + C_D^2}}$$

92. $$\cos\gamma = \frac{C_L}{\sqrt{C_L^2 + C_D^2}}$$

93. $$v = V\sin\gamma = \sqrt{\frac{W\cos\gamma}{\frac{1}{2}\rho S C_L}} \times \frac{C_D}{\sqrt{C_L^2 + C_D^2}}$$ (after substitution for sin γ from equation 91)

$$v = \left(\frac{W}{\frac{1}{2}\rho S}\right)^{1/2} \frac{C_D}{\left(C_L^2 + C_D^2\right)^{3/4}}$$ [(i) where v is sinking speed (ii) after substitution for cos γ from equation 92]

94. **Time of descent** (t_d)

$$t_d = t_2 - t_1 = \int_{t_1}^{t_2} dt = \frac{T_o}{V_o m L}\left[\left(\frac{T_2}{T_o}\right)^m - \left(\frac{T_1}{T_o}\right)^m\right]$$

(Where V_o is the rate of descent at sea level)
where :

$$m = \frac{g + LR}{2\,LR} = 3.128$$ (using I.S.A. conditions, and sea level temperature i.e. $T_o = 288\,K$ and $L = .0065$)

T_o : Sea level Temperature

T_2 : Temperature at height h_2

T_1 : Temperature at height h_1

V_0 : rate of descent at sea level

Rate of descent at sea level

$V_0 = (V_E \sin\gamma)$

Rate of descent in case V_E is given

$v = (V_E \sin\gamma)\sigma^{-1/2} = V_o\sigma^{-1/2}$

95. **Climbing Flight**

$$\sin\theta = \frac{T - D}{W}$$

96. $v = V\sin\theta$ where v is the rate of climb.

97. **Max Rate of climb at shallow angle**

$$v = \left(\frac{W}{\frac{1}{2}\rho s}\right)^{\frac{1}{2}} \left(\frac{T}{W} C_L^{-\frac{1}{2}} - \frac{C_{Do} + KC_L^2}{(C_L)^{3/2}}\right)$$

(Where v is the maximum rate of climb at shallow angle of climb, engine performance expressed in terms of thrust.)

98. $KC_L^2 + \frac{T}{W} C_L - 2C_{Do} = 0$ for finding C_L when Max rate of climb is to be found out.

99. Climb at Shallow Angle Engine Performance Expressed in term of Thrust Power :

$\Delta P = Wv$

where W : weight of the aircraft (N)

v : rate of climb. (m/s)

$$v = \frac{\Delta P}{W}$$

where $\Delta P = P_a - P_r$

P_a : Power available

P_r : Power required

100. Time to a Height.

$$t_2 - t_1 = \int_{h_1}^{h_2} \frac{dh}{v}$$

If a curve is drawn then area under the curve gives the time to a ht.

AIRCRAFT PERFORMANCE IN A ACCELERATED FLIGHT

101. $T = T_o - aV^2$

where T_o : thrust at zero forward speed (static)

a : constant

T : Thrust at speed V

Ground Run for take off

102. $X = \frac{-1}{2B} \log e \frac{\left(A - BV_{T_o}^2\right)}{A}$ where $A = g\left[\frac{To}{W} - \mu\right]$ and $B = \frac{g}{w}\left[\frac{1}{2}\rho S\left(C_D - \mu C_{Lg}\right) + a\right]$

Landing Performance

103. **Landing Ground Run**

$$x = \frac{1}{2B} \log e\left(\frac{A - BV_L^2}{A - BV^2}\right)$$ By putting $x = X$ and $V = 0$

$$X = \frac{1}{2B} \log e\left(1 - \frac{B}{A} V_L^2\right)$$ where is V_L is landing speed.

104. **Cruise Range (Breguet Formula for Range)**

$$R = V(t_2 - t_1) = \frac{V}{gc}\frac{L}{D} \log e\left(\frac{w_1}{w_2}\right)$$

Where R is cruise range is Km.

105. **Flight for max Duration**

$$C_{Lm.r} = \sqrt{\frac{C_{Do}}{3k}} = \frac{C_{Lmd}}{\sqrt{3}}$$

$$C_{Dm.r} = \frac{4}{3} C_{Do}$$

$$V_{mr} = \sqrt{\sqrt{3}}\ V_{md}\ \underline{\Omega}1.3\ \text{Vmd}$$

106. **Calculation of Range, Engine Performance Expressed in terms of Power**

$$(t_2 - t_1) = \left(\frac{3.6\,\eta}{gc\,V}\right)\left(\frac{L}{D}\right)\log e\,(W_1 / W_2)$$

where η : propulsive (airscrew) efficiency.

c : Specific fuel consumption (Kg/Watt/hr)

$$R = \frac{3.6\,\eta}{gc}\left(\frac{L}{D}\right)\log_e\left(\frac{W_1}{W_2}\right)$$ where R : is range in km

107. $L = nw$ where n : load factor.

108. **Time to Turn Through a given Angle.**

$$t = \frac{V\psi}{g \tan \phi}$$

Where ψ is in radiant

g : acceleration due to gravity.

ϕ : angle of bank.

109. **Min Time to Turn Through a given Angle.**

$$t = \frac{\psi}{g}\sqrt{\frac{w}{\frac{1}{2}S\rho C_L}}.\sqrt{\frac{n}{n^2 - 1}}$$ or

$$t = V_o\left(\frac{\psi}{g}\right)\frac{\sqrt{(\sec \phi)}}{\tan \phi}$$

Where V_o is the speed in st and level flight or the incidence and lift coefficient used in turn.

110. **Force Require to Deflect its Movement by α.**

$F = C \times ½\,\rho V^2 S$

Where C : Coefficient or Force and

F : airstream force

$$\tan \alpha = \frac{\text{Downward force}}{\text{Air Stream Force}}$$

111. **Reynold Number,** $\frac{\rho VC}{\mu}$ where C = wing chord

Velocity = Mach No. × Velocity of sound

μ : absolute coefficient off viscosity.

112. $\frac{HP_2}{HP_1} = \left(\frac{W_2}{W_1}\right)^{3/2}$ or $\frac{V_2}{V_1} = \left(\frac{W_2}{W_1}\right)^{1/2}$ or $\frac{P_2}{P_1} = \left(\frac{V_2}{V_1}\right)^{3}$

113. **Induced drag** $C_{Di} = \frac{C_L^2}{\pi e\, AR_1}\left[C_L = \frac{2W}{\rho V^2 S}\right]$

$C_D = C_{Do} + C_{Di}$, Assume $C_{Do} = .016$ if not given.

GENERAL FORMULAS

114. Centre of Pressure

$$C_P = 0.25 - \frac{C_{Mo}}{C_L}$$

115. Induced Angle = $\frac{C_L}{\pi AR}$

116. Absolute angle

$$\alpha_a = \frac{C_L}{a}$$

117. Effective angle = Absolute angle - Induced angle

118. $BHP = \frac{THP}{\eta} V$

119. Rate of Climb

$$V_C = \frac{(T - D)V}{W}$$

Time of Climb (T) = $\left(\frac{H}{V_o}\right) \log e \frac{H}{(H - H_1)}$

120. Circulation $= \rho V / b$

121. For jet Engine thrust.

$$T = \frac{W_a}{g}(V_j - V_b) + \frac{WF}{g}(V_j) + Ae(P_e - P_n)$$

122. Stalling angle

$$V = V_o(\sec\phi)^{1/2}$$

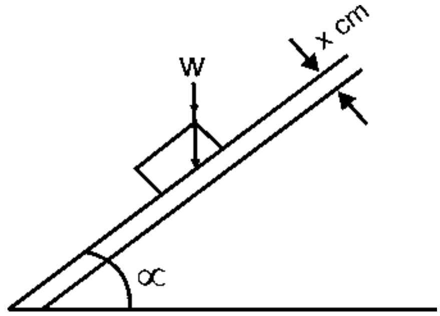
123. Rate of Climb

$$\frac{P_{av} - P_{req}}{W}$$

124. A Jet Propulsive Force

$F = m\Delta V + \Delta A + \Delta P$ (where ΔA : nozzle area)

125. Terminal Speed



y = x cm, Area : A

w = weight, μ : viscocity

α = angle

$$v = \frac{W \sin \alpha y}{\mu A}$$

126. Range : $\frac{V}{gc} \times \frac{L}{D} \log e \frac{W_1}{W_2}$

$$\frac{V}{g} = \frac{W_1}{W_2}\eta \left[\text{if V is not given, then } \frac{V}{g} = \frac{W_1}{W_2}\eta \right]$$

and W_2 = wt. of fuel × Its density in Kg /m^3

127. $$D_{min} = \frac{W}{(L/D)max}$$

128. $$Di = \left(\frac{W}{b}\right)^2 \left(\frac{2}{\pi\rho\, eV^2}\right)$$ (where Di : Induced drag)

129. Minimum Drag condition

$$D = 2D_o = 2D_i$$

$$\frac{L/D}{(L/D)\max} = \frac{2}{m^2 + \frac{1}{m^2}}$$ where $m = \frac{\text{V atitude}}{\text{V min drag}}$

130. Propeller.

$$\frac{1-\eta}{\eta^3} = \frac{550 \times BHP}{2\rho A v^3}$$

* Aerodynamic $\eta = V \times \frac{L}{D}\eta$
* If V = mph = $V\frac{88}{60}$ ft / sec
* V = kmph = $V \times \frac{1}{3.6}$ = m / sec

131. Load Factor in turn.

$$n = \sqrt{1 + \left(\frac{V^2}{Rg}\right)^2}$$

132. Geomatric Angle of Attack.

$$\alpha_1 = \alpha_2 + 57.3\, C_L \left[\frac{1}{e_1 \pi AR_1} - \frac{1}{e_2 \pi AR_2} \right]$$

If α_2 is assumed for infinite AR

$$\alpha_1 = \alpha_o + \frac{57.3\, C_L}{\pi e\, AR_1} \qquad \left[\begin{array}{l} \text{multiplied by 57.3 to convert} \\ \text{to degree i.e. } 180/\pi \end{array} \right]$$

$$\alpha_2 = \alpha_o + \frac{57.3\, C_L}{\pi e\, AR_2}$$

133. Stagnation Pressure-Static Pressure = Dynamic Pressure

$$P_S - P = \frac{1}{2} \rho\, v^2$$

134. Pressure Recorded = Stagnation pressure - static pressure.

135. Induce Angle

$$\alpha_1 = \alpha_2 = \frac{C_L}{\pi AR} \left[\text{where AR} = \frac{\text{Span}}{\text{Chord}} \right]$$

136. Change in Pressure (At Standard Sea Level)

$$P_o + 1/2\, \rho V_o^2 = P + 1/2\, \rho V^2$$

137. Absolute angle of Attack

$$\alpha_a = \frac{C_L}{a}$$

138. Effective Angle

$$\alpha_{eft} = \alpha_a - \alpha_1$$

139. $$C_{DP} = C_{Do} + \frac{C_{DF} \times S_F}{S} + \frac{C_{DOT} \times S_T}{S}$$

where C_{DP} (Total Drag) or Parasite drag.

■■■

PRINCIPLES OF FLIGHT
OBJECTIVE TYPE

Choose the most appropriate answer from the alternative given :

1. The angle of attack which produce the highest L/D ratio
   a. remains constant as weight is changed but decreases as altitude is increased.
   b. increases as height or altitude is increased.
   c. remains constant as altitude is changed but decreased as weight is reduced.
   d. remains constant regardless of weight or altitude.

2. While flying in a steady wind of 25 kts the aeroplane is turned from a direct head wind to a direct tailwind. The IAS would :-
   a. remain the same but ground speed would increase by 50 kts.
   b. increase by 25 kts and ground speed increase by 25 kts.
   c. decrease by 25 kts and ground speed increase by 25 kts.
   d. increase by 50 kts and ground speed increase by 25 kts.

3. What effect does an uphill runway slope have upon takeoff performance :
   a. increase take off distance.
   b. increase take off speed.
   c. decrease take off distance.
   d. decrease take off speed.

4. The use of a slot in the leading edge enables the aeroplane to land at a slower speed because it
   a. changes the camber of the wing.
   b. increases the ground effect.
   c. deaccelerate the upper boundary layer air.
   d. delays the stall to a higher angle of attack.

5. What is the relationship between altitude when the altimeter setting is higher than the standard while flying at 1500 ft indicated altitude ?
   a. indicated altitude is higher than pressure altitude.
   b. indicated altitude is lower than true altitude.
   c. indicated altitude is lower than pressure altitude.
   d. indicated altitude is higher than true altitude.

6. The landing speed in terms of TAS for a particular weight and configuration of the aircraft will
   a. increase as relative humidity increases.
   b. increase as altitude increases.
   c. remains constant regardless of altitude.
   d. decreases as atmospheric pressure decreases.

7. What is the definition of brake horse power.
   a. power delivered at the propeller shaft.
   b. power developed in combustion chamber.
   c. power output corrected for temperature variations.
   d. power converted by the propeller to useful thrust.

8. An irrotational flow over aerofoil turns rotational near the surface because of
   a. surface irregularities.
   b. pressure gradient.
   c. small viscosity of air.
   d. none of above.

9. Dornier DO 228-200 aircraft is powered by
   a. two turboprop engines.
   b. two piston engines.
   c. one turbojet engine.
   d. one turboprop engine.

10. Toughness of a material represents its resistance to
   a. tension. b. crack propagation.
   c. indentation. d. impact.

11. The following Navigational aids require corresponding ground aids :
   a. ADF, DME, VOR.
   b. ADF, ILS, INS.
   c. DME, VOR, INS.
   d. INS, GYRO HORIZON, ILS.

12. The main function of a tail rotor in a helicopter is to
   a. augement lift.
   b. provide pitch control
   c. balance the torque created by main rotor.
   d. increase drag.

13. To determine the mechanical advantage or multiplication of force, that can be accomplished in a hydraulic system, it is necessary to know
   a. system operating pressure.
   b. type of hydraulic fluid being used.
   c. volume of fluid delivered by pumps.
   d. ratio of piston areas of hydraulic cylinder.

14. For an aircraft to have a short landing and takeoff distance, it must have
   a. low stalling speed.
   b. low minimum control speed.
   c. low braking efficiency.
   d. low stalling speed and low minimum control speed.

15. Mass balancing is done to
   a. avoid flutter.
   b. increase bending stiffness.
   c. reduce drag.
   d. distributing mass on control surface.

16. Choose the most appropriate answer from the alternatives given :
   $C_{L_{MAX}}$ of a finite wing is dependent upon.
   a. $C_{L_{MAX}}$ and tail down load required for triming.
   b. $C_{L_{MAX}}$ and spanwise load distribution.
   c. $C_{L_{MAX}}$ spanwise load distribution and tail down.
   d. $C_{L_{MAX}}$ aspect ratio, spanwise load distribution.

17. An airplane with a 200 hp engine has a maximum speed of 120 mph. If the total weight remains unchanged, the maximum speed with 250 hp engine would be
   a. 120 mph.
   b. 125 mph.
   c. 129 mph.
   d. 133 mph.

   Formula to be used $\frac{P_1}{p_2} = \left(\frac{V_1}{V_2}\right)^3$

18. An airplane with 300 ft^2 wing area has a landing speed of 40 mph. If the wing area is reduced to 250 ft^2, the Landing speed is
   a. 40.2 mph.
   b. 43.8 mph.
   c. 45.3 mph.
   d. 39.4 mph.

19. Short columns are sometimes preferred over long columns from
   a. Strength considerations.
   b. Weight considerations.
   c. Buckling considerations.
   d. None of above.

20. The rate of climb on a two engined airplane, after an engine failure will reduce by
   a. 50 %
   b. 75 %
   c. 60 %
   d. None of above.

   Formula used : Rate of Climb $= \frac{\Delta P}{W}$

21. The stiffness of a rod is defined as
   a. Force per unit deflection.
   b. Deflection per unit force.
   c. Strain under loading at high temperature.
   d. Elongation with unit increase in temperature.

22. The very high frequency band ranges from
   a. 30-3000 KHz.
   b. 3000-30000 MHz.
   c. 300-3000 MHz.
   d. 30-300 MHz.

   Hint for Question No. : 20.

   $$\left[\begin{array}{l} \textit{When } \Delta \text{ P has reduced by 50 \%, naturally rate of climb} \\ \text{reduces by 50 \% } C_{L_2} \\ \therefore C_{L_1} \frac{1}{2}\rho_1 V_1^2\, S_1 = C_{L_2} \frac{1}{2}\rho_2 V_2^2\, S_2, \textit{but } C_L \textit{ and } \rho \text{ are} \\ \text{constant} \because V_1^2\, S_1 = V_2^2\, S_2 \end{array}\right]$$

23. For any body to accelerate there must be.
   a. Unbalanced force acting on the body.
   b. Balanced force acting on the body.
   c. Lack of atmosphere surrounding the body.
   d. Three forces acting on the body but meeting at one point.

24. Which of the following materials is isotropic.
   a. Wood
   b. Steel
   c. Composites
   d. Fabric

25. Reynolds number is
   a. Ratio of intertia force to surface tension.
   b. Ratio of pressure force to inertia force.
   c. Ratio of inertia force to frictional force.
   d. Ratio of inertia force to gravity force.

26. ADF is an important navigation equipment. It indicates :
   a. True bearing.
   b. Heading
   c. Relative bearing.
   d. Distance of aeroplane from the airport.

27. The propulsive efficiency of a turbo prop aircraft primarily depends on the characteristics of
   a. Propeller.
   b. Tail cone design.
   c. Burner can configuration.
   d. Inlet design.

28. The shearing stress between two adjacent layers of fluid can be determined if following are known
   a. Fluid viscosity and temperature.
   b. Fluid density and viscosity.
   c. Fluid viscosity and velocity gradient between the layers.
   d. Fluid density and velocity gradient between the layers.

29. Dornier DO 228-200 airplane is powered by
   a. Two turboprop engines.
   b. Two piston engines.
   c. Two turbojet engines.
   d. Two turbofan engines.

30. Mass balancing is done to
    a. Increase bending stiffness.
    b. Reduce drag.
    c. Increase lift.
    d. Avoid flutter.

31. The power developed by an engine at a density altitude of d_1 is related to that at an altitude of d_2 by the following equation :

    a. $P_2 = P_1 \times \frac{d_1}{d_2}$

    b. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^2$

    c. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^3$

    d. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^{\frac{1}{2}}$

32. Toughness of a material represents its resistance to :
    a. tensile loads.
    b. compressive loads.
    c. torsional loads.
    d. crack propagation.

33. An ADF indicates :
    a. relative bearing of an airplane.
    b. true bearing of an airplane.
    c. distance of an airplane from the ADF station.
    d. heading of an airplane.

34. The maximum range of a jet airplane is governed by :
    a. L/D b. 1/D
    c. V (L/D) d. rate of climb

35. A wing having a span of 20 m and lift 1000 N per meter of span moves with a speed of 75m/s. The circulation around the wing would be :
    a. $40/3\rho$ b. $150/\rho$
    c. $2/3\rho$ d. $35/3\rho$

    Formula used $L = \rho VbT$

36. Pratt and Whitney JT 9D-7Q engines are used on
    a. Avro.
    b. Airbus.
    c. Jaguar.
    d. Boeing B-747.

37. A single engined aeroplane is flying at a height 1500 m, when the engine suddenly fails. The L/D required to reach an airport 15 km away would be :
    a. 15 b. 12
    c. 10 d. 1

38. An aeroplane uses symmetrical airfoil for elevators. In order to put the nose up the pilot would :
    a. Pull the stick backward.
    b. Push the stick forward.
    c. Lower the flaps.
    d. Decrease thrust.

39. The deformation at yield stress is :
    a. 100 % elastic and 0 % plastic.
    b. 99.8 % elastic and 0.2 % plastic.
    c. 0.2 % elastic and 99.0 % plastic.
    d. 97.8 % elastic and 2.2 % plastic.

40. The normal system pressure of hydraulic system in a large modern transport aeroplane lies in the range of :
    a. 400 - 470 bar. b. 290 - 360 bar.
    c. 210 - 280 bar. d. 150 - 180 bar.

41. Take-off distance required by an aeroplane at 300 m altitude and -5°C is :
    a. lower than that at SL and 15°C
    b. higher than that at SL and 15°C
    c. higher than that at 3300 m and -3°C
    d. same as that at 3300 m and -3°C

42. An engine rotates shaft at 2000 r.p.m. against torque of 396 N-m. The power supplied to the shaft is :
    a. 2.6 KW b. 264.0 KW
    c. 30.0 KW d. 26.4 KW

    Formula to be used : $\frac{2\pi N}{60} \times 396 = 26400$ watt

    $= 26.4$ k.w.

43. The load factor on an aeroplane in straight and level flight is :
    a. 0 b. 1
    c. 1.5 d. none of the above.

44. Dornier DO 228-200 aeroplane is powered by :
    a. two turboprop engines.
    b. two piston engines.
    c. one turbofan engine.
    d. two turbojet engines.

45. The propulsive efficiency of a propeller to which an engine is delivering 1000 hp and which is furnishing 1500 lb thrust at a forward speed of 295 fps is :
    a. 80.0 % b. 78.5 %
    c. 62.5 % d. 70.0 %

46. In a cambered aerofoil the sectional lift coefficient is directly proportional to :
    a. Absolute angle of attack.
    b. Geometric angle of attack
    c. Induced angle of attack.
    d. Angle for minimum drag.

47. In 'Dead Reckoning' type of navigation, it is necessary to have :
 a. Ground based ADF system.
 b. Ground based ILS and DME system.
 c. Airborne gyros and flight data computer.
 d. Airborne communication system.

48. The turbine inlet temperature in the commonly used turbo fan engines being used on large transport aeroplane is about :
 a. 800ºC
 b. 1100ºC
 c. 1300ºC
 d. 1500ºC

49. When an aeroplane is operating close to the ground the downwash velocities are :
 a. Increased by 1.5 times.
 b. Decreased by 1.2 times.
 c. Unchanged.
 d. Decreased.

50. While flying in a steady wind of 25 kts, the aeroplane is turned from direct headwind to a direct tailwind. The IAS would
 a. Remain the same but ground speed would increase by 50 kts.
 b. Increase by 25 kts and ground speed increases by 25 kts.
 c. Decrease by 25 kts and ground speed increases by 25 kts.
 d. Increase by 50 kts and the ground speed increases by 25 kts.

51. The use of a slot in the leading edge enables the aeroplane to land at a lower speed because it :
 a. Changes the camber of the wing.
 b. Increases the ground speed.
 c. Decelerates the upper boundary layer.
 d. Delays the stall to a higher angle of attack.

52. The stiffness of a rod is defined as :
 a. Force per unit deflection.
 b. Deflection per unit force.
 c. Elongation with unit increase in temperature.
 d. Ultimate stress divided by yield stress.

53. Use of titanium is preferred when higher strength to weight ratio is required at :
 a. Room temperature.
 b. Very low temperature.
 c. High temperature.
 d. All temperature.

54. To determine the mechanical advantage or multiplication of force that can be achieved in a hydraulic system, it is essential to know :
 a. System operating pressure.
 b. Kind of hydraulic fluid being used.
 c. Ratio of piston areas of hydraulic cylinders.
 d. Volume of fluid delivered by the pressure pumps.

55. The very high frequency band ranges from :
 a. 30 - 3000 KHz.
 b. 300 - 3000 MHz.
 c. 30 - 300 MHz.
 d. 3000 - 30000 MHz.

56. The brake horse power is :
 a. Power delivered at propeller shaft.
 b. Power developed in combustion chamber.
 c. Power output corrected for temperature variations.
 d. Power converted by the propeller to useful thrust.

57. An irrotational flow over an airfoil turns rotational near the surface because of :
 a. Surface irregularities.
 b. Pressure gradient.
 c. Small viscosity of air.
 d. None of the above.

58. While designing long members undergoing compressive loading the most important parameter to be considered is :
 a. Fatique strength b. Creep
 c. Buckling strength d. Tensile strength

59. In a streamline flow pattern velocity is :
 a. High when streamline spacing is low.
 b. High when streamline spacing is high.
 c. Independent of streamline spacing.
 d. Low when streamline spacing is low.

60. AN 32 aeroplane is powered by :
 a. Turbojet engines.
 b. Turbo prop engines.
 c. Two turbo fan engines.
 d. Piston engines.

61. Both lift and drag would be increased when which of these devices are extended ?
 a. Flaps b. Spoilers
 c. Dive brakes d. Slots

62. The purpose of an inverter is to
 a. Change d.c. into a.c.
 b. Stabilize generator output.
 c. Monitor the performance of the electrical system.
 d. Stabilize the alternator output.

63. Under which condition is a forward C.G. most critical
 a. On landing.
 b. On takeoff.
 c. During a spin.
 d. When in an unusual attitude.

64. The most desirable types of stability for an aircraft to posses is
 a. Neutral static stability.
 b. Neutral dynamic stability.
 c. Positive static stability.
 d. Positive dynamic stability.

65. A wing with a very high AR will have
   a. Increased drag at high angles of attack.
   b. A low stall speed.
   c. Poor control qualities at low airspeeds.
   d. Decreased wing performance when a climbing attitude.

66. How do deicer boots help remove ice accumulations
   a. By preventing formation of ice.
   b. By melting ice formation.
   c. By breaking up ice formation.
   d. By allowing only a thin layer of ice to build up.

67. What effect will a wheel going into skid have on aircraft braking value.
   a. Greatly reduced.
   b. Slightly reduced.
   c. No effect.
   d. Totally destroyed.

68. The vapour pressure of aviation gasoline is
   a. Lower than that of automotive gasoline.
   b. Higher than that of automotive gasoline.
   c. The same as that of automotive gasoline.
   d. Approx 20 psi at 100°F.

69. An aircraft altimeter is set at 29.92″ Hg on the ground. The altimeter will read
   a. Pressure altitude.
   b. Density altitude.
   c. Field elevation.
   d. True altitude.

70. What is the dielectric of the condenser in a capacitor type fuel indicating system.
   a. Outer shell of the condenser.
   b. Wire coil external to the tank.
   c. Fuel in the tank.
   d. Fuel and air (vapour) above the fuel.

71. If a single rotor helicopter is in horizontal flight, the angle of attack of the advancing blade is
   a. More than the retreating blade.
   b. Equal to the retreating blade.
   c. Less than the retreating blade.
   d. The same at any point around the rotor disc.

72. Changes in the centre of pressure of a wing affect the aircraft's
   a. Aerodynamic balance and controllability.
   b. Centre of gravity location.
   c. Lifting capacity.
   d. Life/drag ratio.

73. As airspeed decreases in level flight, total drag of aircraft becomes greater than the total drag produced at the maximum L/D speed because of the
   a. Increase in induced drag.
   b. Increase in parasite drag.
   c. Decrease in induced drag.
   d. Decrease in parasite drag.

74. The purpose of a localiser is to
   a. locate lost airplanes.
   b. set the airplane on the proper approach angle of the runway.
   c. Indicate the distance the airplane is from the end of the runway.
   d. Align the airplane with the centre of the runway.

75. The basic air cycle cooling system consists of
   a. Source of compressed air, heat exchanger and a turbine.
   b. Heaters, coolers and compressors.
   c. Ram air source, compressors and engine bleed.
   d. Heat exchangers and evaporators.

76. When the pilot leans the mixture control which of the following is being established.
   a. The volume of air entering the carburettor is being reduced.
   b. The amount of fuel entering the combustion chamber is being increased.
   c. The volume of air entering the carburettor is being increased.
   d. The amount of fuel entering the combustion is being reduced.

77. The stiffness of a rod is defined as
   a. Force per unit deflection.
   b. Deflection per unit force.
   c. Elongation with unit increase in temperature.
   d. Ultimate stress divided by yeild stress.

78. In a streamline flow pattern velocity is
   a. High when streamlines spacing is low.
   b. High when streamlines spacing is high.
   c. Independent of streamline spacing.
   d. None of above.

79. AN 32 aeroplane is powered by
   a. Turbojet engines.
   b. Turboprop engines.
   c. Turbo fan engines.
   d. Piston engines.

80. A relay is a
   a. Magnetically operated switch.
   b. Device which increases voltage.
   c. Device which converts electrical energy to heat energy.
   d. Conductor which receives electric energy and passes it on with little or no resistance.

81. When an aeroplane is operating close to the ground the downwash velocities are
   a. Increased by 1.5 times.
   b. Decreased by 1.2 times.
   c. Unchanged.
   d. Decreased.

82. Pratt and Whitney JT 9D-7Q engines are used on
a. Avro. b. Air Bus.
c. Jaguar. d. Boeing B-747.

83. A single engine aeroplane flies at an altitude of 1500 m when its engine fails. L/D required to reach an airport 15 km away is
a. 15 b. 12
c. 10 d. 1

84. The power developed by an engine at a density altitude of d_1 is related to that at an altitude of d_2 by the following equation

a. $P_2 = P_1 \times \frac{d_1}{d_2}$

b. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^2$

c. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^3$

d. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^{\frac{1}{2}}$

85. An ADF indicates
a. relative bearing of an airplane.
b. true bearing of an airplane.
c. distance of an airplane from the ADF station.
d. heading of an airplane.

86. The maximum range of a jet airplane is governed by
a. L/D b. I/D
c. V (L/D) d. rate of climb.

87. Flap deflection causes
a. change in drag.
b. change in lift.
c. change in lift and drag.
d. change in lift, drag and trim.

88. Load factor of an aeroplane in a straight level and unaccelerated flight is
a. 2 b. 1
c. 0 d. 3

89. The purpose of an inverter is to
a. change d.c. into a.c.
b. stabilize generator output.
c. monitor the performance of the electrical system.
d. stabilize the alternator output.

90. The most desirable type of stability for an aircraft to posses is
a. Neutral static stability.
b. Neutral dynamic stability.
c. Positive static stability.
d. Positive dynamic stability.

91. A wing with a very high AR will have
a. Increased drag at high angles of attack.
b. A low stall speed.
c. Poor control qualities at low airspeed.
d. Decreased wing performance when in climbing attitude.

92. What effect will a wheel going into skid have on aircraft braking value
a. Greatly reduced. b. Slightly reduced.
c. No effect. d. Totally destoryed.

93. An airplane with 300 ft^2 wing area has a landing speed of 40 mph. If the wing area is reduced to 250 ft^2 the landing speed will be
a. 40.2 mph. b. 43.8 mph.
c. 45.3 mph. d. 39.4 mph.

94. The conventional altimeter reads
a. pressure altitude. b. density altitude.
c. true altitude.

95. In a vertical diving flight the normal load factor experienced by the aircraft is
a. 0 b. 1 c. 2

96. The stringers in a wing are subjected mostly to
a. bending streasses. b. shear stresses.
c. uniaxial stresses.

97. In a ramjet engine the air compression is achieved by
a. aerodynamic compression.
b. reciprocating compression.
c. rotary compressor.

98. For a delta wing, if leading edge sweep angle is A the aspect ratio is
a. tan A b. 4 tan A
c. cot A

99. Deflection at the free end of a cantilever beam of span L subjected to a pure moment M at the free end is

a. $\frac{ML^2}{EI}$ b. $\frac{ML^2}{2EI}$

c. $\frac{ML^2}{3EI}$

100. For a boundary layer having a velocity profile varying linearly from zero at the wall to the freestream value at a height the displacement thickness is

a. δ b. $\frac{\delta}{2}$

c. $\frac{\delta}{\sqrt{2}}$

101. In the testing of a high speed under water vehicle model in a water tunnel the parameters to the simulated are
a. Reynolds number and Froudes number.
b. Reynolds number and Mach number.
c. Froudes number and Mach number.

102. For a body moving at a point on the equator with a linear velocity, the coriolis acceleration will be zero if the motion is along.
a. North-South direction.
b. East-West direction.
c. Vertical direction.

103. The maximum shear stress for a solid shaft of square cross-section subjected to a torque occurs at
a. centres of the edges.
b. tips.
c. in between.

104. Entropy across an expansion wave
a. increases.
b. decreases.
c. remains same.

105. For a swept leading edge at a Mach number M to be supersonic, its sweep angle should be

a. less than $\left[90 - \sin^{-1}\frac{1}{M}\right]$

b. greater than $\left[90 - \sin^{-1}\frac{1}{M}\right]$

c. equal to, $\left[90 - \sin^{-1}\frac{1}{M}\right]$

106. The conventional altimeter reads
a. Pressure altitude.
b. Density altitude.
c. True altitude.
d. None of the above.

107. In a Ram Jet Engine the air compression is achieved by
a. Aerodynamic compression.
b. Reciprocating compressor.
c. Rotatory compressor.
d. In stages through axial flow compression.

108. The maximum range of a jet airplane is governed by
a. L/D b. I/D
c. V (L/D) d. Rate of climb

109. The toughness of a material represents its resistance to
a. Tensile loads.
b. Compressive loads.
c. Torsional loads.
d. Crack propagation.

110. Flap deflection causes
a. change in drag.
b. change in lift.
c. change in lift and drag.
d. change in lift, drag and trim.

111. Load factor of an airplane in a straight and level unaccelerated flight is
a. 0 b. 2
c. 1 d. 3

112. While designing long members subjected to compression the most important parameter to be considered is
a. Fatique strength. b. Buckling strength.
c. Creep. d. Shear strength.

113. Pratt 4 Wittney, JT-9D-7Q engines were used on :
a. Avro b. Airbus
c. Jaguar d. Boeing 747

114. A student pushes against the side of a building with a force 6N for 4hrs. The work done is :
a. Zero b. 24
c. 12 d. None of the above

115. An airplane has a thrust excess of 1000N above the drag at a velociety of 150 m/s. The ROC at the airplane wt of 20000 N would be
a. 10 m/s b. 7.5 m/s
c. 0.05 m/s d. 15 m/s

Hint : $V = \frac{TV}{W} = \frac{1000 \times 150}{20000} = 7.5$

116. Time required for a glider, having a sink speed of 3 m/sec and flying at an altitude of 300 m to reach the ground would be.
a. 100 S b. 50 S
c. 15 S d. 10 S

117. A small four cylinder engine has a 10 cm bore, 9 cm stoke, the piston displacement is
a. 3000 cm^3 b. 1000 cm^3
c. 2825 cm^3 d. 3452 cm^3

Hint : Displacement = Vol × no. of cylinders

$$\frac{\pi}{4} d^2 1 \times 4$$

118. The load factor for st and level flight and unaccelerated will be.
a. 0 b. 1
c. 1.5 d. none of the above.

119. The very high frequency band ranges from
a. 30 to 300 MHz.
b. 3000 to 30000 MHz.
c. 300 to 3000 MHz.
d. 30 to 300 Khz.

120. Under ISA condition the temperature at altitude of 300 m is
   a. 300°K b. 279°K
   c. 286°K d. 290°K

   Formula Used $\therefore T = T_o - Lh$
   $= 288 - 0.0065 \times 300$

121. The lift of an aircraft is
   a. Product of air density, angle of attack and surface of wing.
   b. Product of air density, square of airplane velocity and surface area of wing.
   c. Product of air density, velocity of air, circulation of air around wing.
   d. None of the above.

122. The AVRO is a
   a. Subsonic jet airplane.
   b. Piston engined aircraft.
   c. Subsonic turbo prop.
   d. Subsonic aircraft.

123. For any body to accelerate, there must be
   a. Unbalanced force acting on the body.
   b. Lack of atm surrounding that body.
   c. An almost perfect vacuum surrounding that body.
   d. Balanced force acting on the body.

124. A sky diver delays opening in chute for ten sec and therefore free falls in that period of time, the distance he fall will be
   a. 3000 M
   b. 1500 M
   c. 600 M
   d. 490 M

   Hint : $S = Ut + \frac{1}{2}gt^2$ and $V = 0$.

125. To determine the mechanical advantage or multiplication of force that can be accompanied in an hydraulic system it is necessary to know the
   a. System operating pressure.
   b. Kind of hydraulic fluid being used.
   c. Vol of fluid delivered by the pressure pump.
   d. Ratio of piston areas of the hydraulic cylinder.

126. The sp volume fuel consumption of a turbo jet engine
   a. decrease with altitude.
   b. decrease with inlet air temp.
   c. increases with altitude.
   d. increases with inlet air temp.

127. In flight the propulsive efficiency of a turbo prop engine depends on the characteristic of
   a. Propeller.
   b. Inlet opening dimension.
   c. Burner can configuration.
   d. Tail cone design.

128. A hydraulic actuating cylinder lifts a wt. of 11000 lbs, a distance of 2 ft in 5 sec, the power required is
   a. 6 hp b. 8 hp
   c. 10 hp d. 20 hp

129. Low frequancy band ranges from
   a. 30 to 300 MHz. b. 10 to 30 MHz.
   c. 30 to 300 KHz. d. 3 to 30 MHz.

130. The excess power required by an aircraft weighting 60 KN to achive a rate of climb of 6M/S is
   a. 750 kw. b. 420 kw.
   c. 360 kw. d. 1000 kw.

   Hint : $\text{ROC } (\upsilon) = \frac{\Delta p}{W} \therefore P = 6 \times 60 = 360$ kw.

131. A ram jet diff from turbojet because
   a. It has a diffuser, a combustion chamber and a nozzle.
   b. It has no turbine.
   c. It has no compressor.
   d. It has no compressor and turbine.

132. Induce drag is associated with
   a. a wing of finite thickness and span.
   b. a wing with an infinite span.
   c. a wing with finite span.
   d. a swept back wing.

133. Stiffness of rod is
   a. deflection per unit force.
   b. force per unit deflection
   c. stress/strain.

134. Aircraft wings are designed as box beam to
   a. increase the bending strength.
   b. increase the torsional resistance.
   c. none of the above.

135. For short field performance it is ideal to have
   a. Low stalling speed.
   b. low minimum control speed.
   c. both above.

136. Use of titanium is prefered when higher strength/wt. ratio is required at
   a. Room temp. b. Higher temp.
   c. All temp.

137. Range of an aircraft with full fuel starting with max take off wt. would depend on.
   a. L/D
   b. Drag
   c. V (L/D).

138. Centre of pressure of an aerofoil with an increase of angle of attack will
   a. move forward. b. move backwards.
   c. not move.

139. Dihedral on a wing is provided to improve
a. Longitudinal stability.
b. Leteral stability.
c. Directional stability.

140. Yield stress is the stress at which deformation is
a. completely elastic.
b. 98% elastic.
c. 99.8% elastic.

141. Boeing 747 is equiped with
a. 4 engine.
b. 3 engine.
c. 2 engine.

142. If one engine fails on a two engined aircraft, the rate of climb will reduce to by :
a. 50%
b. 30%
c. None of the above

143. Pure Al is not used in aircraft structure as it does not provide
a. adequate strength.
b. corrosion resistance.
c. torsional resistance.

144. Response time of a hydraulic system as compared to that of a pneumatic system is
a. Shorter.
b. Larger.
c. 50% longer.

145. On a positively chambered airfoil C_L is zero, when
a. $\alpha = 0$.
b. $\alpha < 0$.
c. $\alpha > 0$.

146. If the velocity of an airplane is doubled, profile drag will increase by
a. two times.
b. Four times.
c. Sixteen times.

147. An I.L.S. indicate to the pilot a/p's position wrt
a. localizer beam.
b. glide slope path.
c. both above.

148. An aeroplane uses symmetrical airfoil for elevators. In order to put the nose up the pilots
a. Pull the stick backwards.
b. Lowers the flap.
c. pushes the stick forward.
d. decrease thrust.

149. An isentropic state means that the
a. entropy is constant.
b. enthalpy constant.
c. Temp. and pressure constant.
d. entropy and enthalpy constan.

150. The efficiency of an inflight turboprop would largely depend on
a. inlet opening dimention.
b. propeller efficiency.
c. Burner Can configuration.
d. Tail cone design.

151. The take off distance of an airplane will increase if
a. tail wind exist.
b. head wind exist.
c. no wind exist.
d. cross wind exist.

152. A limit load is one that can be supported by a structure
a. with 12% yielding.
b. with 10% yielding.
c. with 8% yielding.
d. without yielding.

153. For an aircraft to have a short landing distance, it must have
a. low stalling speed.
b. low min control speed.
c. low braking.
d. low stalling speed and low min control speed.

154. Take off distance of an aircraft will decrease if
a. tail wind exists.
b. head wind exists.
c. cross wind exists.

155. Which of the following instruments works on the principle of reflection of electromagnetic radiations
a. instrument landing system.
b. Inertial navigation system.
c. distance measuring system.
d. Radar.

156. Which of the material are non isotropic
a. steel and wood.
b. wood and fibre reinforced plastic.
c. chromium, fibre reinforced plastic and Al.
d. steel and Al.

Note : Isotropic material are those which have the same properties in all direction.

157. The max temp experienced by a turbine blade in the present day turbofan is of the order of
a. $50^{o}C$
b. $500^{o}C$
c. $1000^{o}C$
d. $2500^{o}C$

158. What effect does an uphill runway slope have upon take off performance
a. increases take off distance.
b. increases take off speed.
c. decreases take off speed.

159. The use of a slot in the leading edge enables the airplane to land at a lower speed because it
   a. changes the chamber of wing.
   b. increases the ground effect.
   c. decelerates the upper boundary layer air.
   d. delays the stall to a higher angle of attack.

160. What is the relationship between attitude when the altimeter setting is higher than the standard. While flying at 1500 ft indicated attitude?
   a. indicated attitude is higher than pressure attitude.
   b. indicated attitude is lower than true attitude.
   c. indicated attitude is lower than pressure attitude.
   d. indicated attitude is higher than true attitude.

161. The landing speed in terms of TAS for particular ht. and configuration of the aircraft will.
   a. increase as realtime humidity increases.
   b. increase as attitude increase.
   c. remain constant as attitude decreases.
   d. decreases as atm.

162. What is the definition of Brake Horse Power
   a. power delivered at propeller shaft.
   b. power developed in combustion chamber.
   c. power output corrected for temperature.
   d. power converted by the propeller to useful thrust.

163. The following navigational aids require corresponding ground aids

a.	ADF	DMF	VOR.
b.	ADF	ILS	INS.
c.	DME	VOR	INS.
d.	INS	gyro horizon	ILS.

164. For an aircraft to have a short landing and take off distance it must have
   a. low starting speed.
   b. low min control speed.
   c. low braking.
   d. low stalling and min control speed.

165. Massbalancing is done to
   a. avoid flutter.
   b. increase bending stiffness.
   c. reduce drag.
   d. distributing mass on control surface.

166. The power developed by an engine at a density altitudes d_1 is related to that at an altitude of d_2 by the equations
   a. $P_2 = P_1 \times \frac{d_1}{d_2}$
   b. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^2$
   c. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^3$
   d. $P_2 = P_1 \times \left(\frac{d_1}{d_2}\right)^{\frac{1}{2}}$

167. The deformation at yield stress is
   a. 100% elastic and 0% elastic.
   b. 99.8% elastic and 20% elastic.
   c. 2% elastic and 99.8% plastic.
   d. 97.8% elastic 2.2% plastic.

168. The normal system pressure of hydraulic system in a larger modern transport airplane lies in range of
   a. 400 - 470 bar.
   b. 290 - 360 bar.
   c. 210 - 280 bar.
   d. 150 - 180 bar.

169. Take off distance required by an airplane at 3000m altitude and -5^{o}C is
   a. lower than that a sea level and 15^{o}C.
   b. higher than that at sea level and 15^{o}C.
   c. higher than that at 3300m and 3^{o}C.
   d. same as that at 3300m and 3^{o}C.

170. An engine rotate the shaft at 2000 rpm against torque of 396 N-m. The power supplied to the shaft is
   a. 2.6 kw.
   b. 264 kw.
   c. 30 kw.
   d. 26.4 kw.

Hint : $\frac{2\pi N}{60} \times 396 = 26400 \text{ watt} = 26.4 \text{ kw}.$

171. If 'm' is the number of members of a statically determinate truss with 'J' joint then
   a. m = 2J–3.
   b. m = 3J–2.
   c. m = 2J–3.
   d. m = 3J+2.

172. The relation between load factor 'n' acceleration 'a' of air plane and 'g' is given by
   a. n = 1–a/g.
   b. n = 1+a/g.
   c. n = 1+g/a.
   d. n = 1–g/a.

173. A simple beam of length 'L' loaded with an upward uniform load of 'W' per unit length, as maximum bending moments at its midspan given by
   a. $w L^2/8$.
   b. $-w L^2/8$.
   c. $w L^2/4$.
   d. $-wL^2/4$.

174. As an airplane nears stalling, its lift to drag ratio
   a. Falls rapidly.
   b. Falls gradually.
   c. Falls marginally.
   d. Does not change.

175. As an airplane approaches ground in the process of landing its lift force from the wings
    a. Increase gradually.
    b. Decreases grafually.
    c. Increase suddenly.
    d. Decrease suddenly.

176. As an airoplane while flying with Nose-up attitude, the horizon bar of the artificial horizon would be
    a. above the miniature airplane.
    b. Below the miniature airplane.

177. The maximum deflection at the free end of a cantilever beam of length 'L' having a concentrated load 'W' at the free end is given by (E and I have usual meanings)
    a. $Wl^3/3EI$.
    b. $Wl^3/4EI$.
    c. $2WL^3/3EI$.
    d. $WL^3/8EI$.

178. Air Bus A-300 is powered by
    a. Two turbo-jets hanging below its high wings.
    b. Two turbo-jets hanging below its low wings.
    c. Two turbo-fans hanging below its high wings.
    d. Two turbo-fans hanging below its low wings.

179. Actual altitude of an aircraft is given by
    a. Conventional altimeter.
    b. Radio Altimeter.

180. The static Longitudinal stability of an airplane is provided by
    a. The lift from the wings and horizontal tail.
    b. Product of the 'tail arm L' and tail lift Lt.
    c. Product of the 'tail arm L' and wing lift Lw.

181. Ratio of 'Strength to weight' in case of mild steel is of the order of
    a. 125.
    b. 150.
    c. 80.

182. The main frame of x-15 Hypersonic research air plane of USA was made up of
    a. Titanium alloys.
    b. Advanced composite materials.
    c. High Tensile Steels.
    d. Nimonic Alloys.

183. An aircraft climb using
    a. Up elevators.
    b. High lift devices.
    c. Excess Engine power.
    d. Inter-change of kinetic & potential energy

184. The flight instrument which assists the pilot in determining best throttle setting for most efficiency flight speed is
    a. Altimeter.
    b. Vertical speed indicator.
    c. Airspeed indicator.
    d. Accelerometer.

185. Compressibility effects become pronounced at the flight Mach Number
    a. approaching unity.
    b. exceeds unity.
    c. exceeding 0.3.
    d. exceeding 0.85.

■■■

Answers

1. d.
2. a.
3. a.
4. d.
5. d.
6. b.
7. a.
8. c.
9. a.
10. b.
11. c.
12. c.
13. d.
14. d.
15. a.
16. d.
17. c.
18. b.
19. c.
20. a.
21. a.
22. d.
23. a.
24. b.
25. c.
26. c.
27. a.
28. c.
29. a.
30. d.
31. d.
32. d.
33. a.
34. c.
35. c.
36. d.
37. c.
38. a.
39. b.
40. c.
41. b.
42. d.
43. b.
44. a.
45. a.
46. a.
47. c.
48. a.
49. c.
50. a.
51. d.
52. a.
53. c.
54. c.
55. c.
56. a.
57. c.
58. c.
59. a.
60. b.
61. d.
62. a.
63. a.
64. a.
65. b.
66. c.
67. d.
68. b.
69. a.
70. c.
71. c.
72. a.
73. c.
74. d.
75. a.
76. d.
77. a.
78. a.
79. b.
80. a.
81. c.
82. d.
83. c.
84. d.
85. a.
86. c.
87. d.
88. b.
89. a.
90. a.
91. b.
92. d.
93. b.
94. a.
95. a.
96. a.
97. a.
98. c.
99. b.
100. a.
101. b.
102. b.
103. b.
104. c.
105. c.
106. a.
107. a.
108. c.
109. d.
110. d.
111. c.
112. b.
113. d.
114. a.
115. c.
116. a.
117. c.
118. b.
119. a.
120. c.
121. b.
122. c.
123. a.
124. d.
125. d.
126. b.
127. a.
128. b.
129. c.
130. c.
131. d.
132. c.
133. b.
134. b.
135. a.
136. b.
137. c.
138. a.
139. b.
140. c.
141. c.
142. b.
143. a.
144. a.
145. b.
146. b.
147. a.
148. a.
149. a.
150. b.
151. b.
152. d.
153. d.
154. a.
155. c.
156. b.
157. c.
158. a.
159. a.
160. d.
161. b.
162. a.
163. c.
164. d.
165. a.
166. c.
167. b.
168. c.
169. b.
170. d.
171. b.
172. b.
173. c.
174. a.
175. a.
176. b.
177. a.
178. d.
179. a.
180. c.
181. c.
182. b.
183. a.
184. c.
185. a.

INDEX

K

L

M

N

O

P

R

S

T