UNIT 2

BASIC AERODYNAMICS

BERNOULLI'S THEOREM

Bernoulli's theorem or Bernoulli's principle was first postulated by the Swiss mathematician, Daniel Bernoulli, who has been considered as one of the most distinguished mathematician of all times. Bernoulli's theorem is one of the important equations in fluid dynamics, which establishes a very important relation between the velocity, pressure and the gravitational potential energy.

Bernoulli's equation can be considered to be a statement of conservation of energy principle, which is appropriate for flowing fluids. The derivation of Bernoulli's theorem has been derived under severe assumptions:

- The equation relates the states at two points along a single streamline (not conditions on two different streamlines)
- Fluid velocity everywhere on the fluid is less than M=0.3, i.e., the flow is said to be incompressible. The density must either be constant, or a function of the pressure alone
- The flow must be steady, i.e. the fluid velocity at a point in the fluid cannot change with time
- Friction by viscous forces has to be negligible
- No work is done on or by the fluid
- No heat is transferred to or from the fluid. Also, thermal effects, such as natural convection, are ignored
- No change occurs in the internal energy

Bernoulli's principle can also be derived directly from Isaac Newton's Second Law of Motion. If a small volume of fluid is flowing horizontally from a region of high pressure to a region of low pressure, then there is more pressure behind than in front. This gives a net force on the volume, accelerating it along the streamline.

STATEMENT OF BERNOULLI'S PRINCIPLE

In fluid Dynamics, Bernoulli's principle states that "along any given streamline, the total mechanical energy of the fluid comprising of the pressure energy, the gravitational potential energy and the kinetic energy remains constant".

Pressure Energy + Kinetic Energy + Gravitational Potential Energy = Constant

$$P + \frac{1}{2}\rho V^2 + \rho g h = Const$$

OR at two points in a fluid, it can be given as

$$P_1 + \frac{1}{2}\rho V_1{}^2 + \rho g h_1 = P_2 + \frac{1}{2}\rho V_2{}^2 + \rho g h_1$$

V is the fluid velocity at a point on a streamline (m/s)
g is the acceleration due to gravity (m/s^2)
h is the elevation of the point above a reference plane, with the positive h direction pointing upward – so in the direction opposite to the gravitational acceleration (m)
p is the pressure at the chosen point (N/m^2)
ρ is the density of the fluid at all points in the fluid (Kg/m^2)

The constant on the right-hand side of the equation depends only on the streamline chosen, whereas v, z and p depend on the particular point on that streamline. Bernoulli's equation gives a relation between the pressure and flow velocity squared.

LIFT AND DRAG

There are four important forces that act on a powered aircraft in flight. Thrust, Drag, Lift, and Weight Among these forces, the lift and the drag forces are the aerodynamic forces, which primarily depend upon the geometric shape of the wings. The thrust and lift forces are the important and beneficial forces which helps in sustaining flight, whereas, the drag and weight forces are the detrimental forces, which negatively affect the performance of the aircraft. We are concerned only with Lift and Drag at the moment.

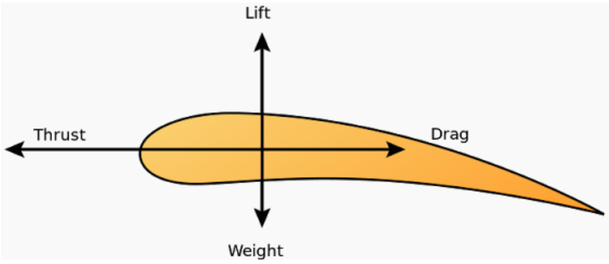


Fig 1: Forces acting on an airfoil

LIFT FORCE

Lift is an aerodynamic force, generated perpendicular to the direction of travel for an object moving through a fluid. Technically, ***lift can be defined as the component of the resultant aerodynamic force perpendicular to the free stream air or free stream velocity*** v_∞**.** Lift is the force that directly opposes the weight of an airplane and holds the airplane in the air. Lift is generated by every part of the airplane, but most of the lift on a normal airliner is generated by the wings. Because lift is a force, it is a vector quantity, having both a magnitude and a direction associated with it. Lift acts through the centre of pressure of the object and is directed perpendicular to the flow direction.

Mathematically, lift is given as

$$L = \frac{1}{2}\rho * v^2 * S * C_l$$

L = Lift, which must equal the airplane's weight in Newtons (N)

ρ = density of the air. This will change due to altitude. These values can be found in a I.C.A.O. Standard Atmosphere Table. (kg/m^3)

v = velocity of an aircraft expressed meters per second (m/s)

S = the wing area of an aircraft in square meters (m^2)

C_l = Coefficient of lift, which is determined by the type of airfoil and angle of attack.

DRAG FORCE

Every physical body that is propelled through the air will experience resistance to the air flow. This resistance is called drag. Technically, drag is defined as ***the component of the resultant aerodynamic force parallel and opposite to the free stream air or free stream velocity*** v_∞**.** Drag acts in a direction that is opposite to the motion of the aircraft. Drag is an unavoidable consequence of an object moving through a fluid. Drag is also a vector quantity which has both direction and magnitude. For drag to be generated, the solid body must be in contact with the fluid. If there is no fluid, there is no drag Mathematically, drag is represented as

$$D = \frac{1}{2}\rho * v^2 * S * C_d$$

D = Drag force generated in Newtons (N)

ρ = Density of the air in (kg/m^3)

v = velocity of an aircraft expressed meters per second (m/s)

S = The wing area of an aircraft in square meters (m^2)

C_d = Coefficient of drag, which is determined by the type of airfoil and angle of attack.

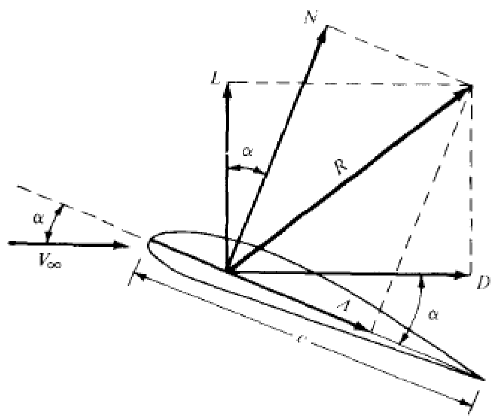


Fig 2 : Components of Aerodynamic Resultant R into Lift (L)and Drag (D)

AERODYNAMIC COEFFICIENTS

- We know that the major objective of Aerodynamics is to determine the Aerodynamic Forces and Aerodynamic Moments, which are Lift, Drag, Pitching Moment, Rolling Moment and Yawing Moment respectively.
- But in Aerodynamics, there are few more quantities which are considered to be even more fundamental and important than the Aerodynamic Forces and Moments and they are termed as Aerodynamic Coefficients.
- Aerodynamic Coefficients are dimensionless numbers which are used to determine the Aerodynamic Characteristics of an aircraft moving through a air. In fact aerodynamic characteristics can be used to describe the aerodynamic characteristics of any object moving through any fluid medium.
- The aerodynamic forces (Lift, Drag, Axial Force, Normal Force) and aerodynamic moments (Pitching Moment, Rolling Moment and Yawing Moment) can both be represented by aerodynamic coefficients respectively.
- **Aerodynamic coefficients for a force can be defined as the ratio of the aerodynamic force to the free stream dynamic pressure q_∞ and the characteristic area S**.

$$\text{Aerodynamic Force Coefficient } C = \frac{Force}{Dynamic pressure * Area}$$

$$Dynamic\ Pressure\ q_\infty = \frac{1}{2}\rho_\infty {v_\infty}^2 \ \mathbf{N/m^2}$$

- The different types of Aerodynamic force coefficients are Lift Coefficient, Drag Coefficient, Normal Force Coefficient and Axial Force Coefficient.
- Therefore Lift coefficient is defined as Lift force to the dynamic force q_∞ and reference area S of the given body.

Mathematically,

$$\textbf{Lift coefficient } C_L = \frac{L}{q_\infty S}$$

$$C_L = \frac{L}{\frac{1}{2}\rho_\infty {v_\infty}^2 S}$$

$$C_L = \frac{2L}{\rho_\infty {v_\infty}^2 S}$$

- Similarly,

$$\textbf{Drag coefficient } C_D = \frac{D}{q_\infty S}$$

$$C_D = \frac{D}{\frac{1}{2}\rho_\infty v_\infty{}^2 S}$$

$$C_D = \frac{2D}{\rho_\infty v_\infty{}^2 S}$$

Normal Force coefficient $C_N = \dfrac{N}{q_\infty S}$

$$C_N = \frac{N}{\frac{1}{2}\rho_\infty v_\infty{}^2 S}$$

$$C_N = \frac{2N}{\rho_\infty v_\infty{}^2 S}$$

Axial Force coefficient $C_A = \dfrac{A}{q_\infty S}$

$$C_A = \frac{A}{\frac{1}{2}\rho_\infty v_\infty{}^2 S}$$

$$C_A = \frac{2A}{\rho_\infty v_\infty{}^2 S}$$

- Furthermore, **Aerodynamic Moment coefficient C_M is defined as Ratio of the Aerodynamic Moment M to the Free stream dynamic pressure q_∞, reference area S and the reference length l.**

Aerodynamic Moment Coefficient $C_M = \dfrac{Moment}{Dynamic pressure * Area * Length}$

$$C_M = \frac{M}{q_\infty * S * l}$$

$$C_M = \frac{M}{\frac{1}{2}\rho_\infty v_\infty{}^2 * S * l}$$

In the above equations,

ρ_∞ = Density of the free stream air far ahead of the body kg/m^3

v_∞ = Velocity of the free stream air far ahead of the body m/s

q_∞ has the units of pressure of N/m^2

- In the above coefficients, the reference area S and reference length l are chosen based on the shape of the body and *S & l* may be different for different bodies.

- For example, for an airplane wing, S is the planform area, and *l* is the mean chord length, for a sphere, S is the cross-sectional area, and *l* is the diameter and for a rocket, S if the surface area of the maximum cross section and *l* is the length of the rocket.

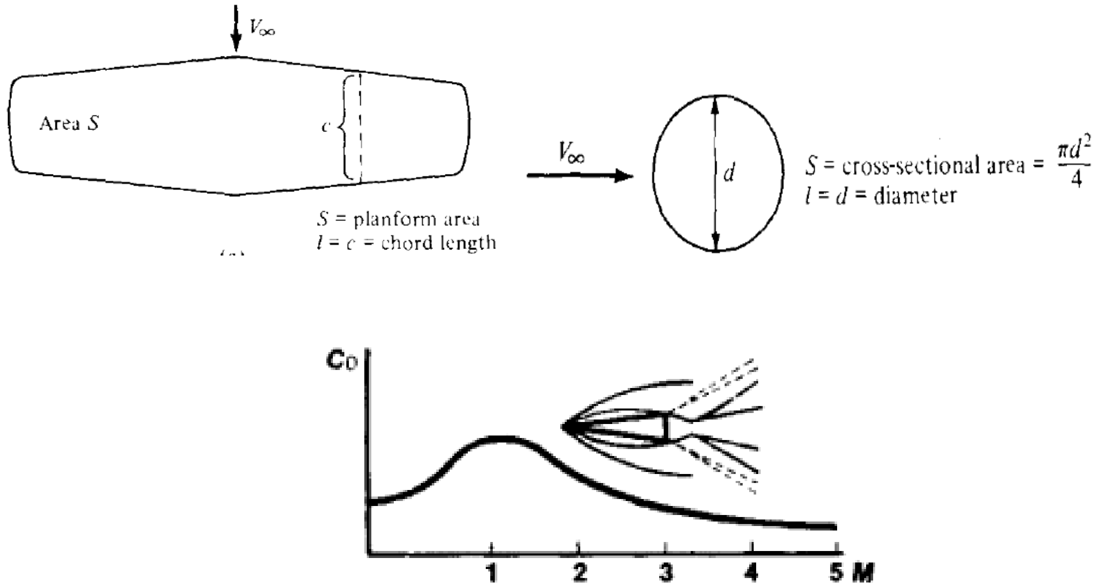


Fig 4: Planform area

- The force and moment coefficients $\mathbf{C_L}$, $\mathbf{C_D}$, $\mathbf{C_N}$, $\mathbf{C_A}$ **and** $\mathbf{C_M}$, are denoted by capital letters for a complete three dimensional body such as an airplane of a wing.
- In contrast for a two dimensional bodies such as airfoils or an infinite wing, the aerodynamic coefficients are denoted by small letters $\mathbf{c_l}$, $\mathbf{c_d}$, $\mathbf{c_n}$, $\mathbf{c_a}$ **and** $\mathbf{c_m}$.
- Aerodynamic coefficients are very important in Aerodynamic studies and calculations, it helps in simplifying the calculations to very great extent.
- Aerodynamic coefficients describe the characteristics of an airplane as ratio of forces rather than just forces. Using aerodynamic coefficients, the aerodynamic efficiencies can quickly be compared.
- For Example, A large plane like Boeing or Airbus A380, will be generating around 50000kN of lift, but a small scale model of the same plane will be generating around 50N in a wind tunnel. This is the indication of the lift force of a particular airplane. But if we indicate the same lift forces in terms of Lift coefficient, then both the planes will result in CL to be around 1.2. So aerodynamic coefficients helps us to describe the aerodynamic characteristics on a same scale and comparison is always easier with aerodynamic coefficients.
- The Aerodynamic Coefficients above are all static and can be found by taking a stationary model in a wind tunnel and measuring the aerodynamic forces and moments. Once these are obtained, then the Aerodynamic Coefficients can be calculated with the above formulas. There is an advantage using the aerodynamic coefficients rather than the actual forces.

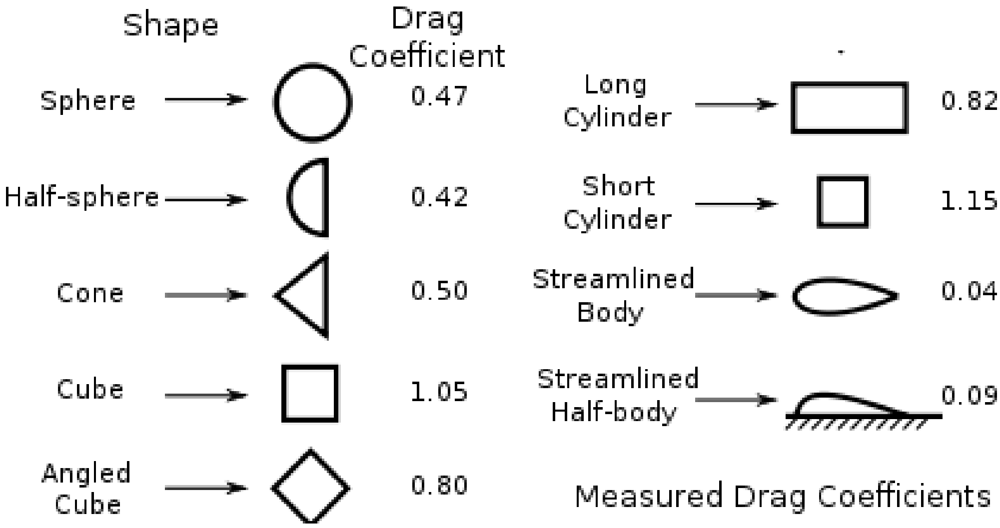


Fig 4: Aerodynamic coefficients for geometrically different bodies

CENTER OF PRESSURE

Any object moving through a fluid will experience pressure forces all over its surface. The concept of centre of pressure is very useful in understanding the effects of these forces. Let's take an airfoil moving in air with subsonic flow attached to the body. As the airfoil moves through air, the velocity of the air varies around the surfaces of the airfoil due to the change in the curvature of the airfoil. This variation in velocity over the airfoil surfaces, produces a variation in the local pressure on the surfaces of the airfoil resulting in the generation of distributed pressure fields all over the surfaces of the airfoil. The distributed pressure fields over the airfoil surface, will be responsible for the generation of Aerodynamic Forces and Aerodynamic Moments on the airfoil surfaces.

- The total effect of distributed pressure fields generated on the airfoil surface can be represented by a single force called as the Resultant Force. The location of the resultant force should be such that, it produces the same force and moment as produced by the distributed pressure field. The location on the chord of an airfoil where the resultant force act is called as the **centre of pressure**.
- In other words, centre of pressure can be defined as the point on the airfoil where the net effect of all the pressure fields act. It usually acts at a point where the resultant force and the plane of the chord of the airfoil intersect.

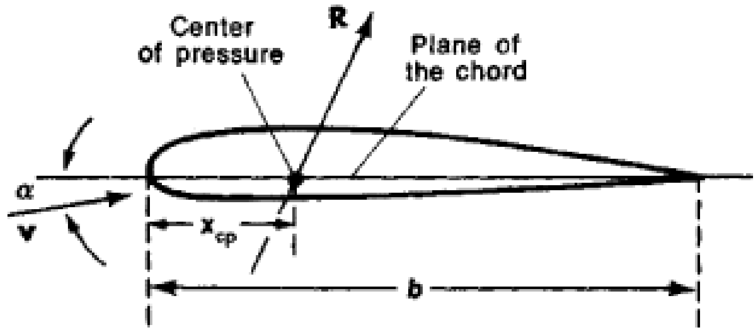


Fig 5: Representation of location of centre of pressure

- The location of centre of pressure is very important in aerodynamics. The resultant force should be located on the body such that it produces the same effect as the distributed loads. For example, the distributed load on a two dimensional body such as an airfoil produces a moment about the leading edge. The resultant force R can be split into two components, parallel and perpendicular to the chord given by Normal N' and Axial forces A'. The axial force A' will be always acting along the chord line, but the normal force will be perpendicular to the chord line. The normal force N' must be located on the chord line in such a way that, it will produce the same moment as produced by the distributed loads. Then N' must be located at a distance x_{cp} downstream of the leading edge such that

$$M' = -(x_{cp})N' \text{ ----------------------- (1)}$$

$$x_{cp} = -\frac{M'_{LE}}{N'} \text{ ------------------------- (2)}$$

- Since N' creates a negative moment (pitch down) which decreases the angle of attack α, the actual moment about the leading edge will be negative. This is consistence with the sign convention used while deriving Aerodynamic moments equation about the leading edge of the airfoil.
- In the above equation x_{cp} is the centre of pressure. If moments were taken about the centre of pressure, the integrated effect of the distributed loads would be zero. Hence, an alternate definition of the centre of pressure is that point on the body about which the aerodynamic moment is zero.
- If the angle of attack of the airfoil is very small, $\sin\alpha \approx 0$ and $\cos\alpha \approx 1$ then, from the equation of Lift

$$L = N\cos\alpha - A\sin\alpha$$

$$L' \approx N' \text{ ----------------------- (3)}$$

$$x_{cp} = -\frac{M'_{LE}}{L'} \text{ ----------------------- (4)}$$

From Eqn. (2) and Eqn. (4), we see that as N' and L' decrease, x_{cp} increases. As the forces approach zero, the centre of pressure moves to infinity. For this reason, the centre of pressure is not always a convenient concept in aerodynamics. However, this is no problem.

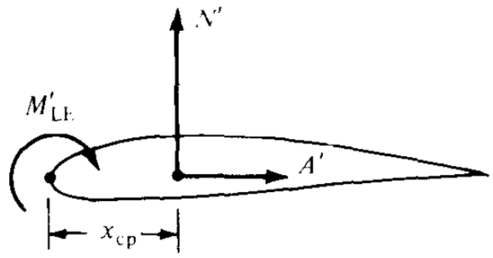


Fig 6: Centre of pressure of an airfoil

- Centre of pressure does not remain at a fixed location on the chord of the airfoil, but keeps varying with angle of attack α . As the angle of attack α changes, the local pressure at every point on the airfoil changes, which in turn changes the pressure distribution on the airfoil. Any change in the pressure distribution will alter the location of the centre of pressure. A change in the location of the centre of pressure has a considerable effect on the dynamic stability of the whole aircraft. In the airplane's normal range of flight attitudes, if the angle of attack is increased, the centre of pressure moves forward and if α is decreased then the centre of pressure moves rearward and this variation in the location of centre of pressure with α is shown in the figure below.
- The location of centre of pressure for a symmetrical airfoil remain unchanged even if the angle of attack α is varied. The centre of pressure on a symmetric airfoil typically lies close to 25% of the chord length behind the leading edge of the airfoil. This is called the "quarter-chord point" and it remains around the quarter chord point for angles of attack below the stalling angle of attack. But on a cambered airfoil the centre of pressure does not occupy a fixed location. For a conventionally cambered airfoil, the centre of pressure lies a little behind the quarter-chord point at maximum angle of attack, but as angle of attack reduces, the centre of pressure moves toward the rear.

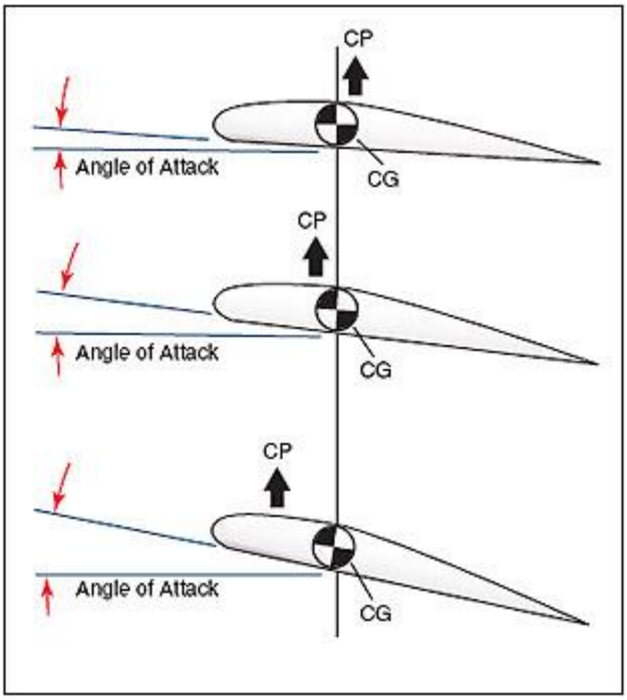


Fig 7: Variation in the location of centre of pressure with α

NOTE: The resultant force R can be defined either in terms of lift and drag or normal and axial forces. One of the significance of splitting the resultant force R into axial and normal force is to determine the centre of pressure x_{cp} of the airfoil or the wing of an aircraft. Because x_{cp} lies on the chord of the airfoil and to determine x_{cp}, the normal force component N' is generally used as it is defined with respect to the chord of the airfoil, where as lift is defined with respect to the free stream velocity v_∞ which will not give the location of x_{cp} on the chord of the airfoil.

WING PLANFORM GEOMETRY

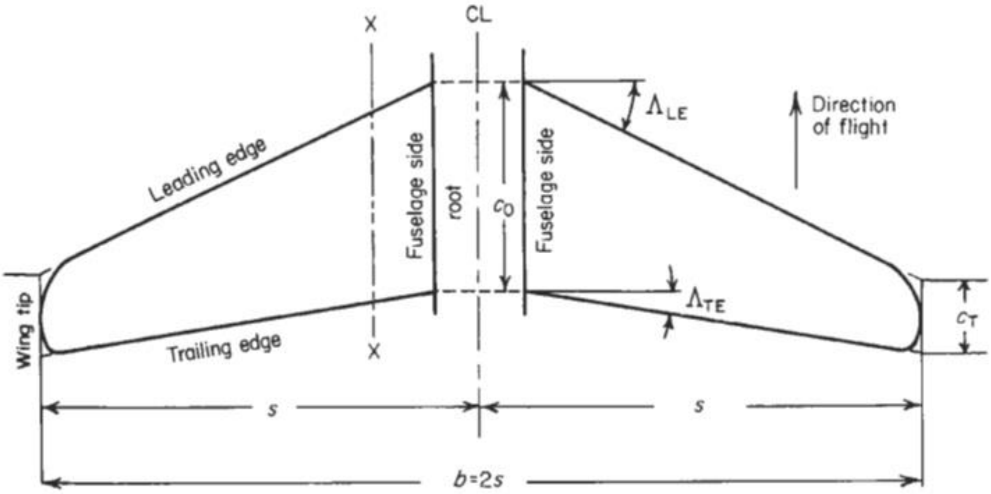


Fig 8: Illustration of various parts of an aircraft wing

The planform geometry of an airplane wing is the shape of the wing seen from the top of the aircraft. The detailed description of the various parts of the wing illustrated in the above figure are as follows:

1. **WING SPAN:** It is the linear distance measured between the extreme wing tips. It is denoted by b. The distance between the centre line of the aircraft and each tip of the wing is called as wing semi span, denoted by s.
2. **WING AREA:** It is the planform area (Top view area) of the wing. If the area of the wing continuing within the fuselage is also considered, then the wing area forms the gross wing area S_G. If the area continuing within the fuselage is not included, then it refers to the net wing area S_N. If unless specified, wing area is considered as gross wing area S_G.
3. **CHORD:** It is the straight line joining the Leading edge and the Trailing edge of the wing. The length of the chord at the intersection of the wing and the fuselage is known as **root chord c_R** and the length of the chord at the tip of the wing is known as tip chord c_T. In a constant chord wing, the length of the chord at the root and the tip will be constant, whereas in a tapered wing the length of the chord at the root and the length of the chord at the tip will not be constant. The degree of wing taper depends on the taper ratio of the wing which is defined as the ratio of tip chord length root chord length.

Taper ratio is denoted by $\lambda = \frac{c_T}{c_R}$

The taper ratio affects the lift distribution and the structural weight of the wing. A typical rectangular wing (constant chord wing) has a taper ratio of 1.0 and a pointed delta has a taper ratio of 0.

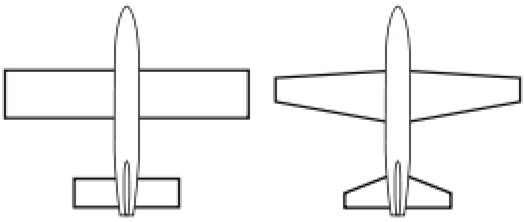


Fig 9: Constant Chord Wing & Tapered wing

4. **ASPECT RATIO:** Aspect ratio is the measure of the narrowness of the wing planform. It is defined as ratio of span of the wing to mean or average chord from root to tip. It is denoted by A, or sometimes AR and is given by

$$AR = \frac{b}{\overline{C}}$$

If both top and bottom of this expression are multiplied by the wing span, by it becomes:

$$AR = \frac{b*b}{\overline{C}*b}$$

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

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

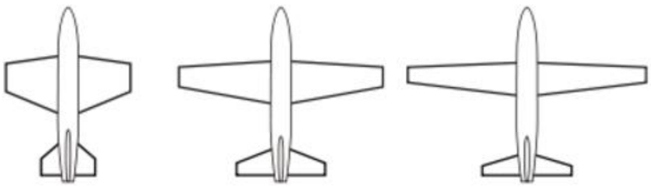


Fig 10: Low, Moderate and High Aspect Ratio Wings

5. **WING SWEEP:** It is usually measured as an angle between the quarter chord line (0.25C) and the perpendicular to the root chord. It is usually denoted by either Λ or ϕ**.** Sweep-back is commonly measured on the leading edge **(**$\Lambda_{\mathbf{LE}}$ or ϕ_{LE}), on the quarter-chord line, i.e. the line of the chord behind the leading edge **(**$\Lambda_{\mathbf{1/4}}$ or $\phi_{1/4}$) & or on the trailing edge **((**$\Lambda_{\mathbf{LE}}$ or ϕ_{TE}).

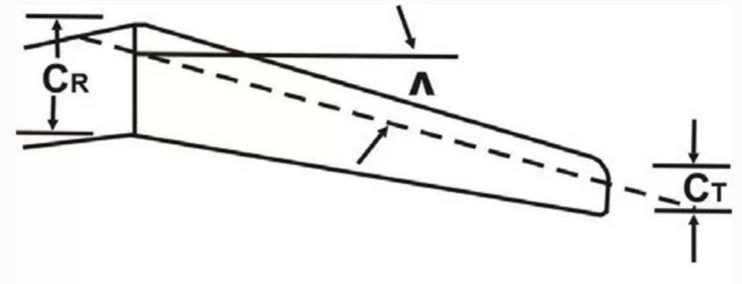


Fig 11: Sweep Angle Λ

Swept back ("swept wing"): The wing sweeps rearwards from the root to the tip.
Forward swept: The wing angles forward from the root

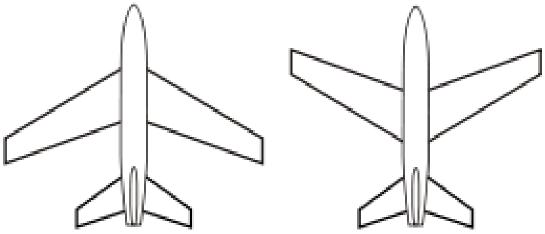


Fig 12: Swept Back & Forward Swept Wing Configurations

6. **DIHEDRAL & ANHEDRAL ANGLE:** The front view of this wing shows that the left and right wing do not lie in the same plane but meet at an angle. The angle that the wing makes with the local horizontal is called the **DIHEDRAL ANGLE**. Dihedral is added to the wings for roll stability; a wing with some dihedral will naturally return to its original position if it encounters a slight roll displacement. You may have noticed that most large airliner wings are designed with **DIHEDRAL.** The wing tips are farther off the ground than the wing root. Highly manoeuvrable fighter planes, on the other hand do not have dihedral. In fact, some fighter aircraft have the wing tips lower than the roots giving the aircraft a high roll rate. A negative dihedral angle is called **ANHEDRAL**.

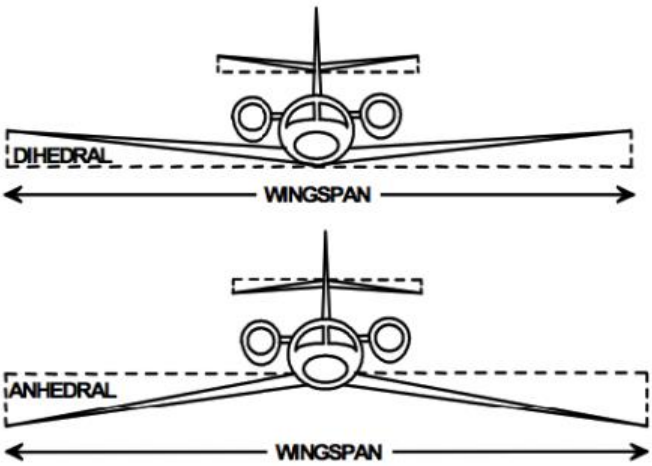
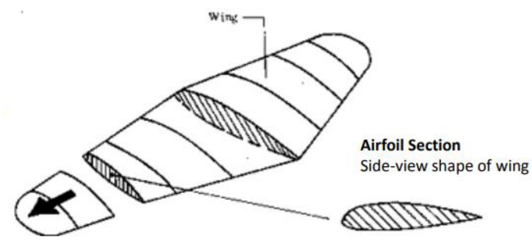


Fig 13: Illustration of Dihedral and Anhedral angles

AIRFOIL GEOMETRY OR AIRFOIL NOMENCLATURE



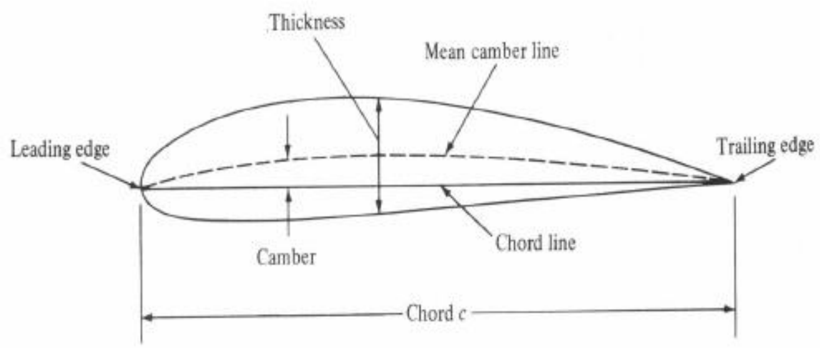


Fig 14: Nomenclature of a typical airfoil

1. **ANGLE OF ATTACK:** Angle of Attack is one of the critical parameter which directly affects the lift generated by the wings or an airfoil. Angle of Attack is defined as the angle between the free stream velocity and the chord line of the airfoil. It is usually denoted by "α". Even for an ascending or descending flight, the angle of attack is defined with respect to the free stream velocity of air only.
2. **LEADING EDGE:** The most forward point of the mean camber line is called as the Leading edge.
3. **TRAILING EDGE:** The most rearward point of the mean camber line is called as the Trailing edge.
4. **CHORD LINE:** The straight line connecting the leading edge and trailing edge is called the chord line.
5. **CHORD "c":** The precise distance from the leading edge to the trailing edge measured along the chord line is called as the chord c.
6. **MEAN CAMBER LINE:** The locus of all the points halfway between the upper and lower surfaces which is measured perpendicular to the mean camber line itself is known as Mean camber line**.**
7. **CAMBER:** The camber is the maximum distance between the mean camber line and the chord line, measured perpendicular to the chord line.
8. **THICKNESS:** The thickness is the distance between the upper and lower surfaces, also measured perpendicular to the chord line.

BASIC AIRFOIL CHARACTERISTICS

However, big or small the aircraft may be, the entire aerodynamic characteristics of the airplane depend on the aerodynamic characteristics of the airfoil. The airfoil characteristics are represented in the form of coefficients of Lift, Drag and Moments. All the aerodynamic characteristics of an airfoil i.e., Cl, Cd and Cm are tested in a wind tunnel at various speeds of flow. Airfoil is also termed as infinite wing (wing without wing tips) which is different from the actual 3dimesnional wing. A wing will exhibit wing tips where the high pressure air from the bottom surface of the wing swirls upwards towards the low pressure region on the top surface of the wing. This phenomenon creates wind tip vortices leading to lift induced drag which will not be encountered on an airfoil. Due to this, the aerodynamic properties of a wing are somewhat different than the infinite wing.

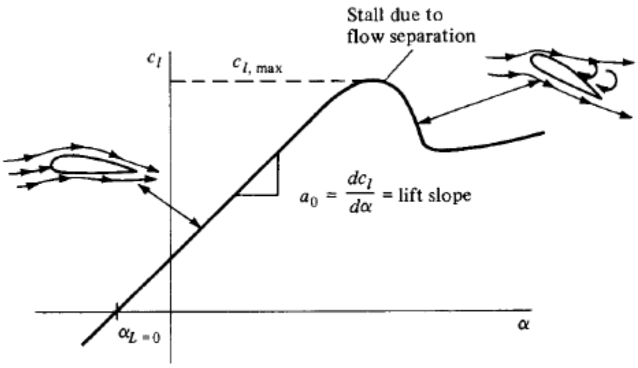


Figure 15: Schematic of Cl vs AoA for a typical airfoil

Let us examine the variation of coefficient of lift c_l with angle of attack α. This variation is sketched in Fig. 15. The experimental data indicate that c_l varies *linearly* with α over a large range of angle of attack. The slope of the linear portion of the lift curve is designated as $\boldsymbol{a_0=dc_l/d_\alpha}$, which is the lift slope. Note that in Fig. 15, when α = 0, there is still a positive value of c_l that is, there is still some lift even when the airfoil is at zero angle of attack to the flow. This is due to the positive camber of the airfoil. All airfoils with such camber have to be pitched to some negative angle o f attack before zerolift is obtained. The value of ***a*** when lift is zero is defined as the *zero-lift angle of attack* $\alpha_l=o$ and is illustrated in Fig. 15. This effect is further demonstrated in Fig. 16, where the lift curve for a cambered airfoil is compared with that for a symmetric (no camber) airfoil. Note that the lift curve for a symmetric airfoil goes through the origin. Refer again to Fig. 15, at the other extreme: For large values of α**,** the linearity of the lift curve breaks down. As α is increased beyond a certain value, c_l peaks at some maximum value $c_{l\max}$ and then drops precipitously as α is further increased. In this situation, where the lift is rapidly decreasing at high α **,** the airfoil is *stalled.* The phenomenon of airfoil stall is of critical importance in airplane design. It is caused by flow separation on the upper surface of the airfoil. This is illustrated in Fig. 17, which again shows the variation of c_l versus α for an airfoil. At point 1 on the linear portion of the lift curve, the flow field over the airfoil is attached to the surface, as pictured in Fig. 17. However, the effect of friction is to slow the airflow near the surface; in the presence of an adverse pressure gradient, there will be a tendency for the boundary layer to separate from the surface. As the angle of attack is increased, the adverse pressure gradient on the top surface of the airfoil will become stronger, and at some value of α —the stalling angle of attack—the flow becomes separated from the top surface. When separation occurs, the lift decreases drastically and the drag increases suddenly. This is the picture associated with point 2 in Fig. 17.

In the same fashion, the variation of c_d and c_m with angle of attack α is also plotted in figure 18.

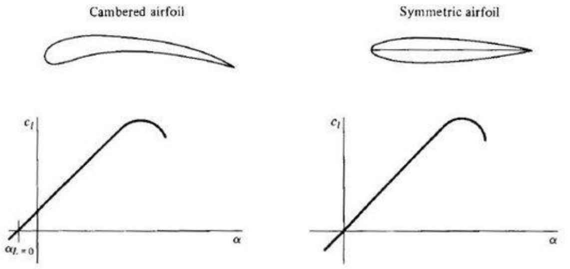


Fig 16: Variation of Cl with angle of attack α

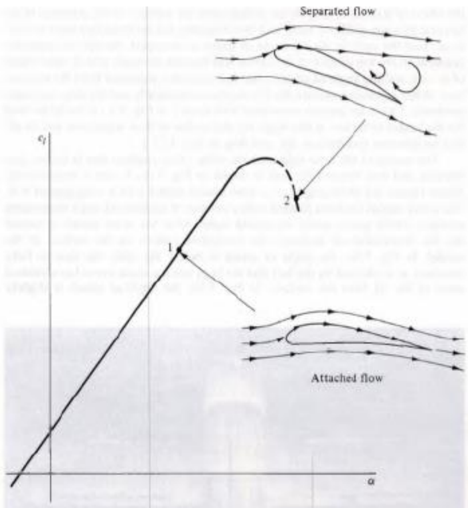


Figure17: Flow separation due to stalling

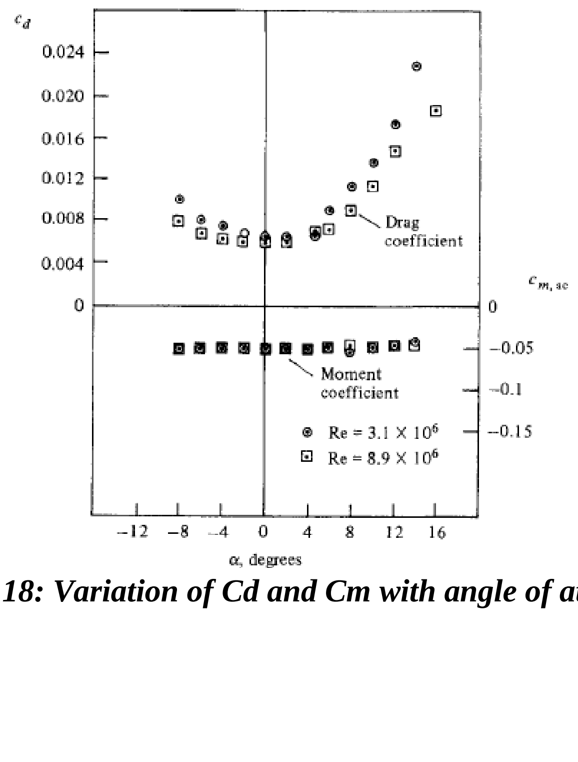
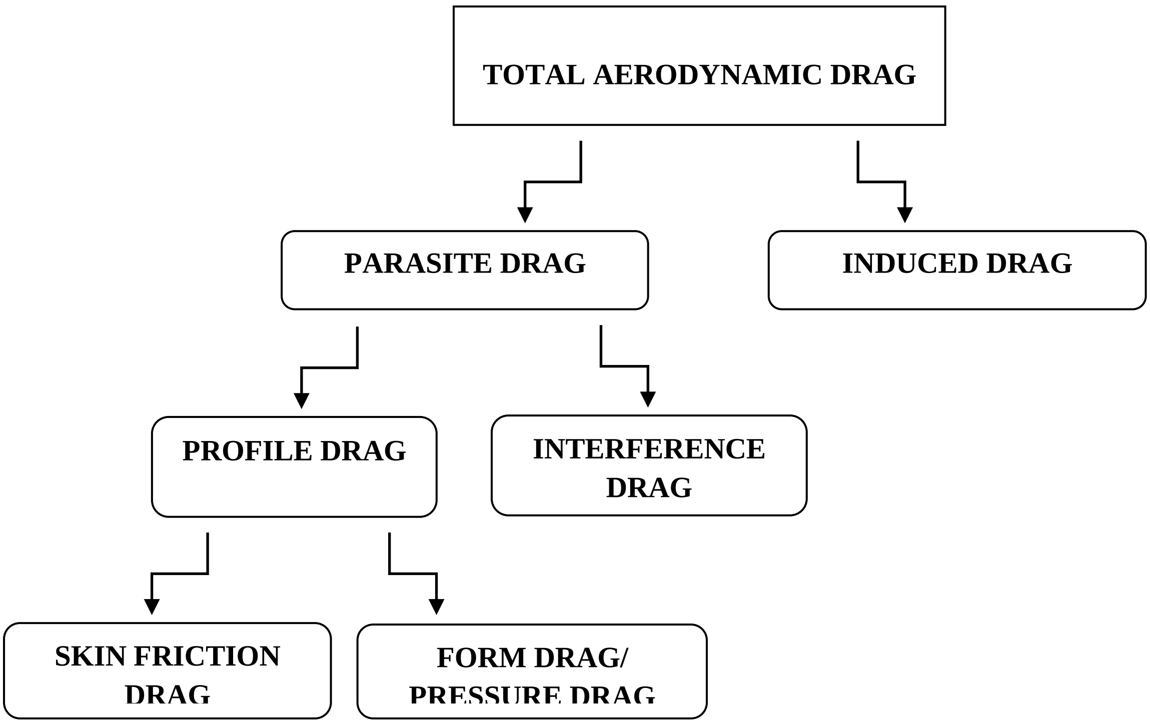


Figure 18: Variation of Cd and Cm with angle of attack α

At low-to-moderate angles of attack, Cl, varies linearly with α. In this region, the flow moves smoothly over the airfoil and is attached over most of the surface, as shown in the Fig. However, as α becomes large, the flow tends to separate from the top surface of the airfoil, creating a large wake of relatively dead air behind the airfoil as shown in the Fig. Inside this separated region, the flow is recirculating, and part of the flow is actually moving in a direction opposite to the freestream-so-called reversed flow. This separated flow is due to viscous effects and results in a precipitous decrease in lift and a large increase in drag; under such conditions the airfoil is said to be stalled. The maximum value of Cl, which occurs just prior to the stall, is denoted by Cl max, it is one of the most important aspects of airfoil performance, because it determines the stalling speed of an airplane. The higher is Cl max, the lower is the stalling speed. With α , Cl, increases linearly until flow separation begins to have an effect. Then the curve becomes nonlinear, Cl, reaches a maximum value, and finally the airfoil stalls.

TYPES OF AERODYNAMIC DRAG

Drag is one of the aerodynamic forces that acts on the airplane moving through the atmospheric air. There are many parameters that affect the generation of drag and in total there are different types of drag generating on an airplane under operation. The detailed classification of drag and its types are



TYPES OF DRAG

- Parasite drag
- Lift induced drag
- Profile drag

- Interference drag
- Form drag or pressure drag
- Skin friction drag
- Wave drag & Ram Drag

Induced Drag/Lift Induced Drag/Drag due to Lift

We know that, the physical mechanism for the generation of lift on the wing is due to the existence of pressure differences i.e., high pressure on the lower surface and low pressure on the upper surface and this imbalance in pressure distribution over the wing helps in the generation of lift. At the wing tips, the high pressure air from the bottom surface will be forced to the low pressure region on the top surface and due to this pressure difference, the flow near the wing tips tend to curl upwards around the wing tips. When the flow curls around the wing tips from the bottom surface to the top surface of the wing, the fluid is set into a circulatory motion, which continues to move downstream of the wing resulting in the generation of trailing edge vortices at each wing tip. Trailing edge wing tip vortices are circulatory pattern of rotating air generated by a wing of a aircraft in motion.

The circulating air will induce a small downward component of velocity in the downward direction at the wing. This downward component is termed as Downwash and tends to reduce the local angle of attack. The greater the angle of attack up to the critical angle (where a stall takes place), the greater the amount of lift developed and the greater the induced drag.

Lift induced drag is an undesirable direct by-product of lift. Induced drag is probably the hardest type of drag to minimize, simply because it’s a consequence of flight itself. In a sense, induced drag is part of the lift; so long as we have lift we must have induced drag, and we can never eliminate it altogether however cleverly the wings are designed. However, modern airliners use winglets/sharklets at the wing tips to reduce the induced drag of the wing. In case of a combat aircraft, incorporating tip tanks at the wing tips reduces the induced drag.

The induced drag is calculated using the below formula:

$$C_{D,i} = \frac{{C_L}^2}{\pi e AR}$$

$C_{D,i}$=Induced drag coefficient
C_L=Lift Coefficient
e=Efficiency Factor
AR=Aspect Ratio of the Wing

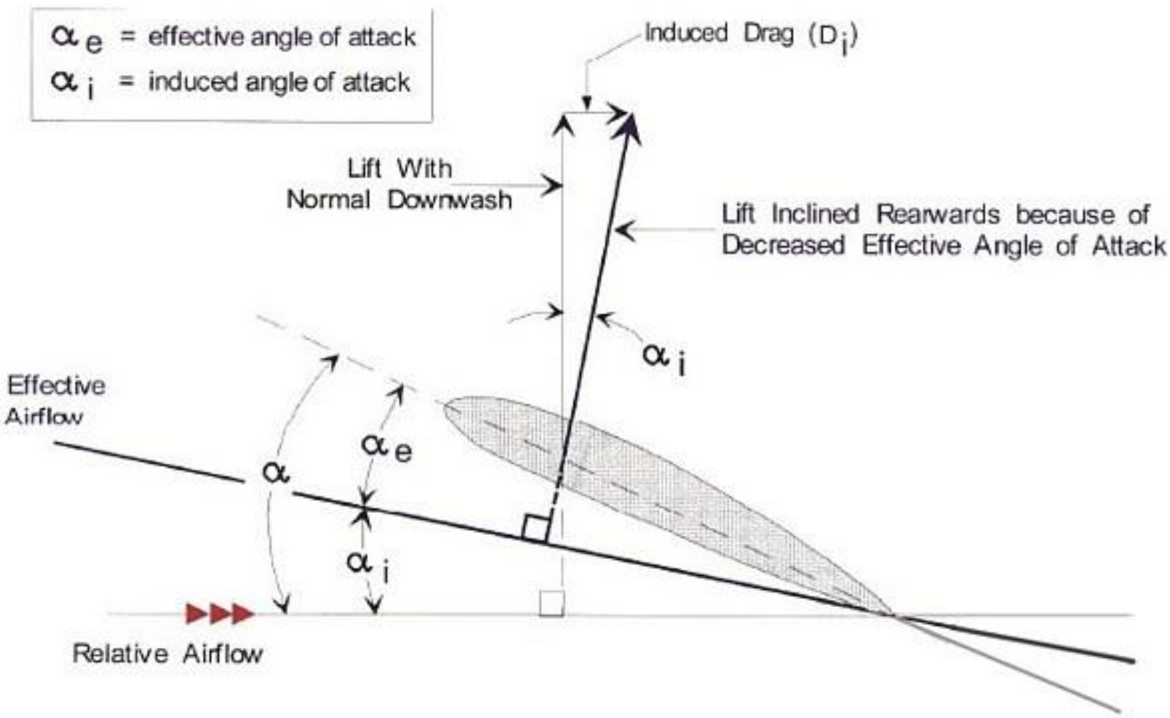


Fig : Induced drag D_i due to the downwash and effective angle of attack

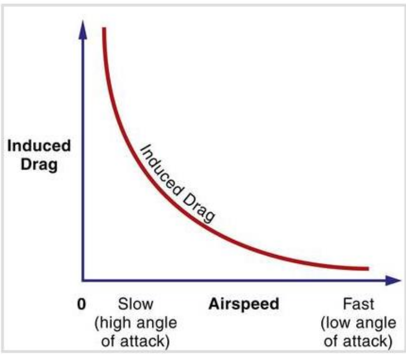


Fig : Variation of Induced drag with airspeed

Induced drag is inversely proportional to the forward speed and directly proportional to the angle of attack α .

Form Drag or Pressure Drag

Form drag and pressure drag are virtually the same type of drag, caused primarily due to the size and shape of the object. Form drag is mainly created when the laminar flow of the air is converted into turbulent flow due to separation of the air flow. Separation refers to the breaking of the smoothly flowing air because of the adverse pressure gradients (hence pressure drag) established over the wing/airfoil surfaces. Once, separation occurs, the fluid will not continue to follow a given surface and the flow will become turbulent, which results in the formation of many turbulent tiny vortex flows called vortices. These vortices causes the flow to be reversed locally, which affects the forward motion of the airplane, eventually resulting in the generation of drag on the aircraft. Since this drag is due to the shape, or form of the aircraft, it is also called form drag.

Bodies with a larger projected cross-section will have a higher drag whereas thin and sleek ("streamlined") objects have lower form drag. Streamlining the aircraft will reduce form drag, and parts of an aircraft that do not lend themselves to streamlining are enclosed in covers called fairings, or a cowling for an engine, that have a streamlined shape. Airplane components that produce form drag include (1) the wing and wing flaps, (2) the fuselage, (3) tail surfaces, (4) nacelles, (5) landing gear, (6) wing tanks and external stores, and (7) engines, (8) wing struts, (9) radio antennae etc. Here in the picture at left we can see a laminar relative air flow approaching the landing gear of an airplane with an exposed landing gear tire. Because the tire is not an aerodynamically clean shape it will cause substantial amounts of form drag as the air flow moves past. In order to avoid this, on small aircrafts, the tires are covered using aerodynamically shaped fairings as shown below.

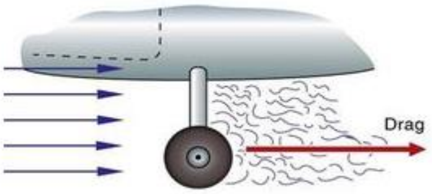


Fig : Generation of form drag from tires and its reduction using faring on the undercarriage

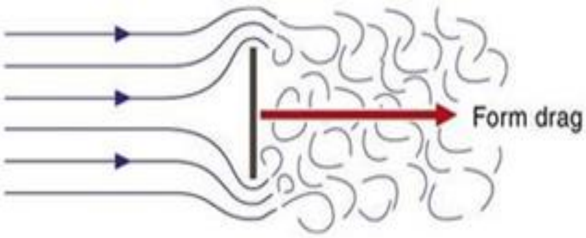


Fig : From drag due to abruptly projected surfaces

But on the other hand, some control surfaces can also be used to create form drag to help the aircraft slow down such as spoilers or speed brakes and even a retracted landing gear can contribute in slowing the aircraft down especially while landing.

Skin Friction Drag/Viscous Drag

Skin friction drag arises from the friction of the fluid against the "skin" of the object that is moving through it. Skin friction arises from the interaction between the fluid and the skin of the body, and is directly related to the wetted surface area i.e., the area of the surface of the body that is in direct contact with the fluid. Larger the wetted surface area, larger is the skin friction drag and vice versa. When there is a relative motion between the airplane and the fluid, the air particles moving over the body of the airplane creates a frictional force which tends to slow down the aircraft.

Since, skin friction drag is generated due to the interaction between a solid (the airplane surface) and a gas (the air), the magnitude of skin friction drag depends on the properties of both the solid and the gas. For the solid airplane, skin fiction drag can be reduced by keeping an aircraft's surface highly polished and clean. For the air, the magnitude of the drag depends on the viscosity of the air. Along the solid surface of the airplane, a boundary layer of low energy flow is generated. The magnitude of the skin friction depends on the conditions in the boundary layer also.

We know that, drag is given by

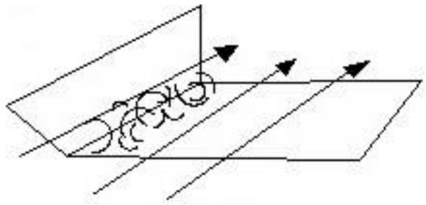
$$D = \frac{1}{2}\rho * v^2 * S * C_D$$

From the above equation, we notice that the drag generated is directly proportional to the square of the velocity and the surface area. Skin friction drag is also dependent on the smoothness of the surface of the skin. Rough surfaces cause more skin friction drag than the smooth surfaces. This is the reason, the skin of an aircraft is well polished to reduce the skin friction drag.

Profile Drag
The sum of skin friction drag and form drag is known as profile drag.

Interference Drag
When different components are combined into a complete aircraft, one component can affect the air flowing around and over the airplane, and hence, the drag of one component can affect the drag associated with another component. These effects are called interference effects. In other words, Interference drag is produced due to the interference of two or more streams of airflow having different speeds arising from different components of the airplane. For example, interaction of airflow from the wings and the fuselage. Besides, interference drag is also Incurred by the non-lifting parts of the aircraft such as the wheels, fuselage, tail fins, engines, nacelles, pylons, empennage, handles, rivets, upsweep, control surface gaps, base areas, and other extraneous items.

Interference drag increases with the square of air speed and becomes dominant at higher air speeds. Surfaces at angles to each other as in the figure create turbulence in the region of the joint. This occurs most frequently at the intersection of the fuselage and wing. One of the way to reduce the interference drag is to maintain acute angles between two different parts and also by providing fillets at the junctions as shown in the fig below.



interference drag
C

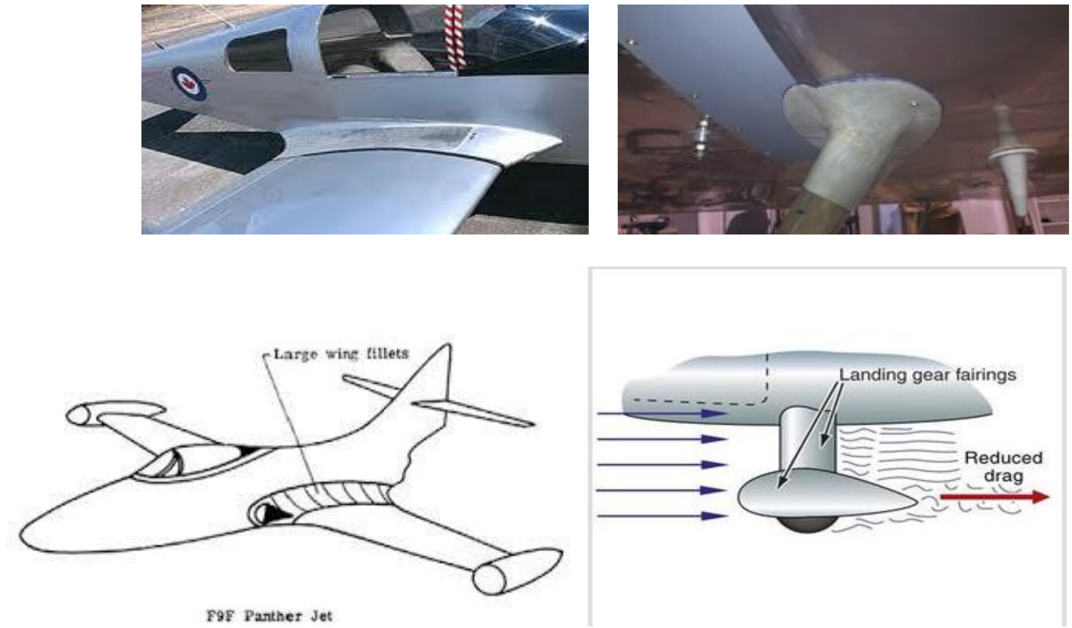


Fig : Providing smooth surfaces at the junction of two different components to reduce interference drag

Parasitic Drag

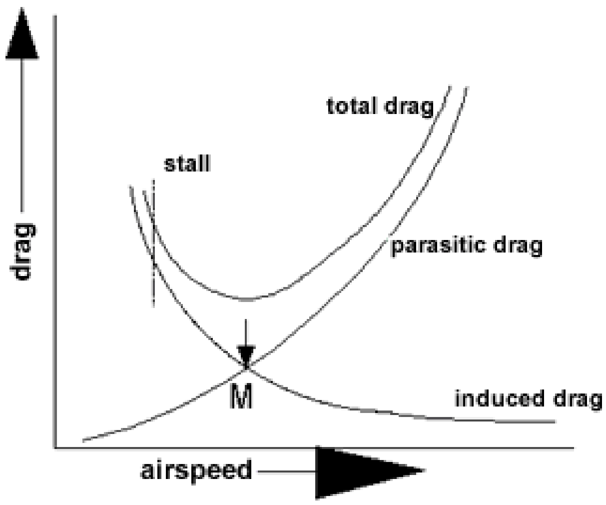
Parasitic drag occurs due to the combination of profile drag (skin friction + from drag) and interference drag. In principle, parasitic drag is simply a combination of Skin friction drag, Form/Pressure Drag and Interference drag. Parasitic drag increases with the increase in the speed of the aircraft.

Wave Drag and Ram Drag

Wave drag is generally produced when the aircraft surpasses the sound barrier which occurs at transonic speed (speed almost equals to speed of sound) & Supersonic speed (speed greater than speed of sound). Due to high speed of airflow, shock waves are produced which are nothing but the disturbance in the air. This disturbance increases drag of the aircraft known as wave drag. The magnitude of wave drag depends on the Mach number of the flow.

Ram drag is produced when free stream air is brought inside the aircraft. Jet engines bring air on board, mix the air with fuel, burn the fuel, then exhausts the combustion products to produce thrust. If we look at the basic thrust equation, there is a mass flow times entrance velocity term that is subtracted from the gross thrust. This "negative thrust" term is the ram drag. Cooling inlets on the aircraft are also sources of ram drag.

Total Aerodynamic Drag = Parasite Drag + Induced Drag



Parasite drag increases with the square of the speed, while Induced drag decreases with speed. The SUM of the two drags (Total Drag curve) shows that there is only one airspeed for a given airplane and load that provides MINIMUM total drag. This is the point M which is the maximum lift over drag ratio (L/D). It is the airspeed at which the aircraft can glide the farthest without power (maximum glide range). This is the airspeed which should immediately be set up in the event of a power failure. This maximum glide airspeed is different for each aircraft design.

NACA AIRFOILS

During the early days of aerodynamic research, little importance was given for the designing of airfoils. Upto world war I, there had been only a little research on studying the geometric effects of airfoils in improving the aerodynamic performance of any aircraft. All the airfoil design were randomly done based on trial & error methods without any proper guidelines & procedure. There was no standard methods for designing & manufacturing of different airfoils for different aircrafts.

This method of vaguely designing the airfoils changed when National ~~[illegible]~~ Advisory Committee for Aeronautics (NACA) was established in 1915. NACA immediately recognised that the aerodynamic performance of an aircraft depends on the geometric design of an airfoil, & hence NACA recognised the need for better & efficient airfoil.

With the development of mathematical equations & in conjunction with extensive experimentation, many aerodynamically efficient airfoils were designed & these airfoils were named as <u>NACA airfoils</u>. The NACA identified different airfoil shapes with a logical numbering system. They are

(1) NACA 4 Series Airfoil
(2) NACA 5 Series Airfoil
(3) NACA modified 4/5 Series airfoils
(4) NACA 6 Series Airfoil
(5) NACA 7 Series airfoils
(6) NACA 8 Series airfoils

① NACA 4 Series airfoil

In a 4 series airfoil,

1^{st} digit → Represents Maximum Camber in hundreths of chord

2^{nd} digit → Represents the location of Maximum camber along the chord length from leading edge in ~~10th~~ tenths of chord.

last 2digits → Maximum thickness in hundreths of chord.

For Example consider NACA 2412

$2 \rightarrow \frac{2}{100} = 0.02C\ C \rightarrow$ Maximum Camber

$4 \rightarrow \frac{4}{10} = 0.4C \rightarrow$ location of maximum camber from leading edge

$12 \rightarrow \frac{12}{100} = 0.12C \rightarrow$ Maximum thickness.

This always beneficial to state the number in percentage of chord i.e 2% of Camber at 40% chord with 12% thickness.

② NACA 5 Series Airfoil

These airfoils are represented by 5 digits. Let us consider NACA 23012

1^{st} digit → multiplied by $\frac{3}{2}$ gives design lift coefficient in tenths of chord.

Next 2 digits → When divided by 2, gives the location of maximum camber in hundreths of chord.

last 2digits → Gives maximum thickness in hundreths of chord from LE.

$2 \times \frac{3}{\cancel{2}} = \frac{3}{10}C = 0.3C \rightarrow$ Design lift co-eff.

$\frac{30}{\cancel{2}} = \frac{15}{100}C = 0.15C \rightarrow$ location of Maximum Camber

$\frac{12}{100} = 0.12C \Longrightarrow$ Maximum thickness.

③ NACA 6 Series airfoils

These are the most commonly used airfoils.

Example NACA 65-218

1st digit → Simply identifies the series

2nd digit → location of minimum pressure in tenths of chord.

3rd digit → Design lift co-efficient in tenths of chord.

last 2 digits → Maximum thickness in hundreths of chord.

6 → Series

$\frac{5}{10} = 0.5c$ → location of minimum pressure

$\frac{2}{10} = 0.2c$ → Design lift coefficient

$\frac{18}{100} = 0.18c$ → Maximum thickness in hundreths of chord.

Apart from these airfoils, there are many other airfoil families like

① Clark Airfoils
② RAF airfoils
③ Jukowsky airfoils
④ Gottingen airfoils
⑤ NREL airfoils etc.

Aerodynamic Forces & Moments

During any fluid-structure interaction (air & airplane), pressure & shear stresses will be distributed all over the body surface. The distribution of the pressure over the body surface results in the generation of Aerodynamic forces like, lift, drag, Axial force & Normal force. Likewise, the distribution of shear stress on the body surface results in the generation of Aerodynamic Moments like the Pitching moment, Rolling moment & Yawing moment.
No matter how complex the body shape may be, the aerodynamic forces & moments on the body are ~~due~~ entirely due to the Pressure & Shear Stresses on the body. Pressure & the Shear Stresses are the only sources by which forces & moments are generated on the body moving through the fluid.

Pressure forces P act normal to the surface & Shear Stress τ act tangential to the surface. Both pressure & shear stresses have same unit N/m^2 or same dimensions.

Derivation for Aerodynamic forces & Moments

Consider an airfoil moving through air. The interaction b/w the airfoil & air will result in the generation of Pressure & Shear stresses on the airfoil as shown in the fig

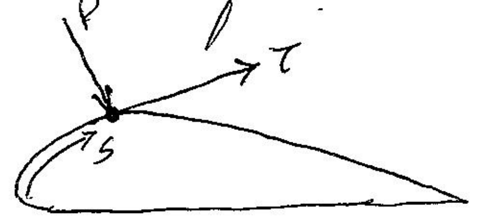
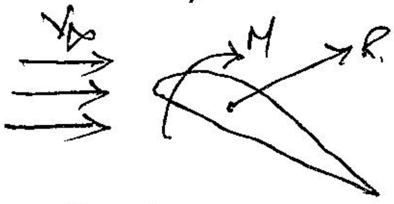


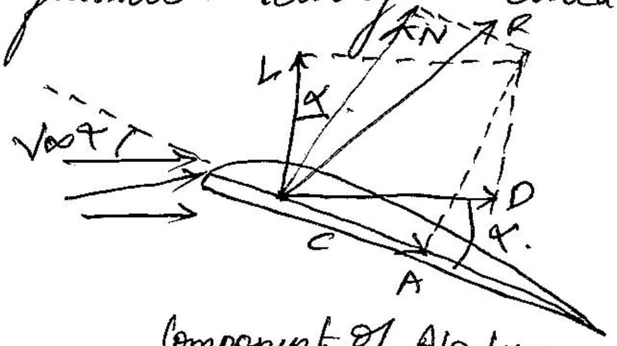
Fig: Illustration of Pressure & Shear stresses on an aerodynamic surface.

The net effect of the pressure p & shear stresses τ distributions integrated over the complete body surface is a resultant aerodynamic force R & moment M on the body. In turn the resultant R can be split into 2 components, one perpendicular to the freestream velocity & other component parallel to the free stream velocity. The flow far away from the body is called as freestream & hence V_∞ is called as freestream velocity. If pressure is considered it is called as freestream pressure.

The component of resultant R which is perpendicular to the freestream velocity is called as Lift & the component of resultant R parallel to the freestream velocity is called as Drag.



Resultant A/D force & moment



Component of A/D force

The chord c is the linear distance from the leading edge to the trailing edge of the body. Sometimes R is split into components perpendicular & parallel to the chord.

N = Normal force = Component of R perpendicular to

A = Axial force = Component of R parallel to c.

The angle of attack α is defined as the angle b/w c & V_∞. Hence α is also the angle b/w L & N & D & A. From fig, the geometrical relation b/w these 2 sets of components is

$$L = N\cos\alpha - A\sin\alpha \quad \text{—} \; (1)$$

$$D = N\sin\alpha + A\cos\alpha \quad \text{—} \; (2)$$

1

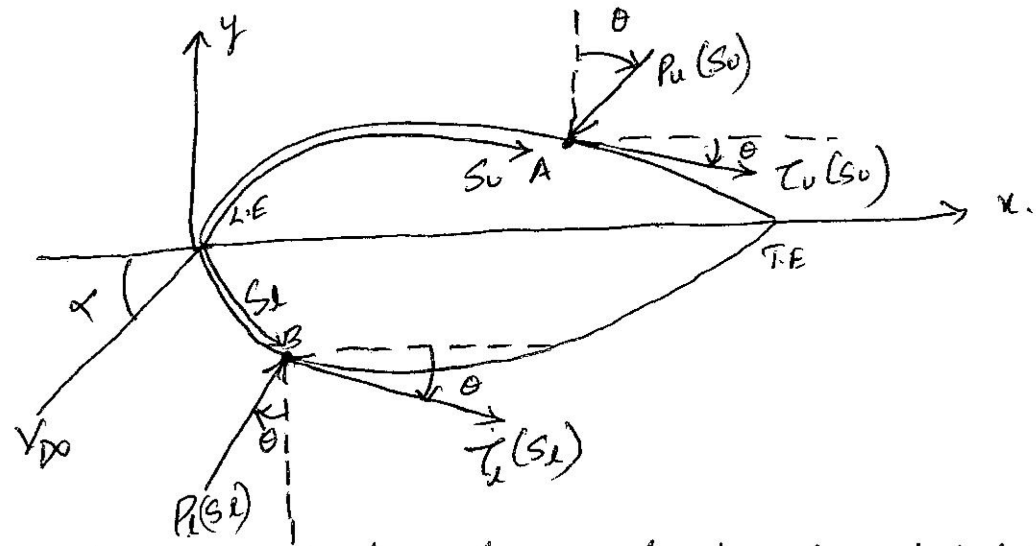


Fig: Nomenclature for integration of pressure & shear stress distribution over a 2-D body surface.

Consider a 2D body as sketched above. Let A be an arbitrary point at a distance S_u from the leading edge & let B be an arbitrary point at the lower surface at a distance S_L from the leading edge. The pressure & shear stress on the upper & lower surfaces are denoted as P_u, S_u & P_l, S_l respectively. At any given point, the pressure is normal to the surface & is oriented at an angle θ with the y axis & shear stress is tangential to the surface & is oriented at the same angle θ with the horizontal x axis. The sign convention for θ is positive when measured clockwise from the vertical line to the direction of P & from the horizontal line to the direction of τ.

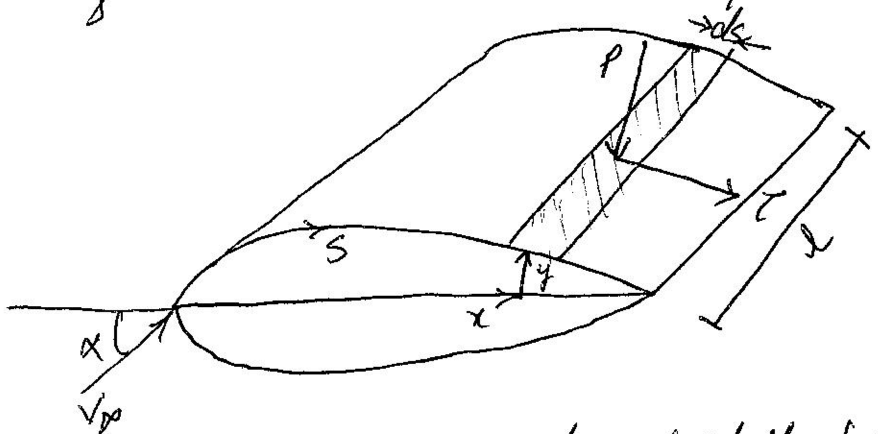


Fig: Aerodynamic force on an element of the body surface.

Now consider the 2D shape as a crosssection of an infinitely long cylinder of uniform section. A unit span of such a cylinder is shown in the above fig. Consider an elemental surface area ds of the cylinder, where $ds = (ds)(1)$ as shown by the shaded area

From the fig, the elemental normal & axial force acting on the elemental surface ds on the upper surface are

$$dN'_u = -P_u ds_u \cos\theta - \tau_u ds_u \sin\theta \quad \text{— (3)}$$

$$dA'_u = -P_u ds_u \sin\theta + \tau_u ds_u \cos\theta \quad \text{— (4)}$$

The prime on N' & A' denote force per unit span.

IIIly on lower body surface are

$$dN'_l = P_l ds_l \cos\theta - \tau_l ds_l \sin\theta \quad \text{— (5)}$$

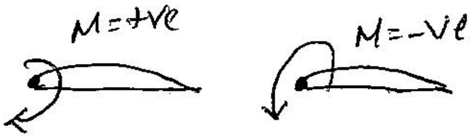
$$dA'_l = P_l ds_l \sin\theta + \tau_l ds_l \cos\theta \quad \text{— (6)}$$

The total normal & axial force per unit span are obtained by integrating Eqn (3) to (6) from leading edge to trailing edge.

$$N' = -\int_{LE}^{TE} (P_u \cos\theta + \tau_u \sin\theta)\, ds_u + \int_{LE}^{TE} (P_l \cos\theta - \tau_l \sin\theta)\, ds_l \quad \text{— (7)}$$

$$A' = \int_{LE}^{TE} (-P_u \sin\theta + \tau_u \cos\theta)\, ds_u + \int_{LE}^{TE} (P_l \sin\theta + \tau_l \cos\theta)\, ds_l \quad \text{— (8)}$$

In turn to get total lift & drag per unit span, eqn (7) & (8) should be substituted in eqn (1) & (2)



To find Aerodynamic Moments

Consider Moments about the leading edge. By convention, moments which tend to increase α (Pitch up) are positive & moments which decrease α (pitch down) are negative

The moments per unit span ~~act~~ on the upper surface on elemental areas

$$dM'_u = (P_u \cos\theta + \tau_u \sin\theta)\, x\, ds_u + (-P_{su} \sin\theta + \tau_u \cos\theta)\, y\, ds_u \quad \text{— (9)}$$

~~on lower surface~~

IIIly moment on bottom surface is

$$dM_L' = (-P_l \cos\theta + \tau_l \sin\theta)x\, ds_L + (P_L \sin\theta + \tau_l \cos\theta)y\, ds_L \quad \text{—(10)}$$

Integrating eqn (9) & (10) from L.E to T.E, we get the eqn of moment about the L.E per unit span

$$\boxed{M_{LE}' = \int_{LE}^{TE} \left[(P_u \cos\theta + \tau_u \sin\theta)x - (P_u \sin\theta - \tau_u \cos\theta)y\right] ds_u + \int_{LE}^{TE} \left[(-P_L \cos\theta + \tau_l \sin\theta)x + (P_L \sin\theta + \tau_l \cos\theta)y\right] ds_L \quad \text{—(11)}}$$

Using Eqn (7) (8) & (11), with the known values of θ, x & y, the integrals can be evaluated.