Chapter 3

Drag polar

(Lectures 6 to 12)

Keywords: Various types of drags; streamlined body and bluff body; boundary layers; airfoil characteristics and designations; drags of airplane components; drag polars at subsonic, transonic, supersonic and hypersonic speeds; high lift devices

Topics

Chapter 3

Lecture 6

Drag polar – 1

Topics

3.1.1 Introduction- need and definition of drag polar

As mentioned in section 1.9, to obtain the performance of an airplane requires the value of the drag coefficient of the airplane (C_D) when the lift coefficient (C_L) and Mach number (M) are given. The relationship between the drag coefficient and the lift coefficient is called 'Drag polar'. It may be pointed out that aerodynamics generally deals with the drag, lift and pitching moment of individual components like wing, fuselage etc. Whereas, for the estimation of the airplane performance the knowledge of the drag, lift and pitching moment of the entire airplane is required.

Equation (1.6) indicates that the drag coefficient is a function of lift coefficient (C_L), Mach number (M) and Reynolds number (R_e). However, for a given airplane a single drag polar can be used for flights upto critical Mach number (Ref. 1.4 section 10.14); see sections 3.3.3 to 3.3.5 for details of critical

Mach number. For airplanes flying at transonic and supersonic speeds, the drag polar depends on Mach number. Hence, the usual practice is to obtain the drag polar of subsonic airplanes at a suitable flight speed (generally the cruising speed) and for a high speed airplane, the drag polars are obtained at suitable values of Mach numbers spread over the range of operating Mach numbers.

In this chapter the estimation of the drag polar at subsonic, transonic and supersonic speeds is discussed. The topic of drag polar at hypersonic speed is also touched up on.

3.1.1 Contributions to airplane drag

The usual method to estimate the drag of an airplane is to add the drags of the major components of the airplane and then apply correction for the interference effects.

The major components of the airplane which contribute to drag are wing, fuselage, horizontal tail, vertical tail, nacelle(s) and landing gear.

Thus,

$$D = D_{wing} + D_{fuse} + D_{ht} + D_{vt} + D_{nac} + D_{lg} + D_{etc} + D_{int} \tag{3.1}$$

where D_{wing}, D_{fuse}, D_{ht}, D_{vt}, D_{nac} and D_{lg} denote drag due to wing, fuselage, horizontal tail, vertical tail , nacelle(s) and landing gear respectively.

D_{etc} includes the drag of items like external fuel tanks, bombs, struts etc.

D_{int} is the drag due to interference which is described in the next section.

3.1.2 Intereference drag

While presenting the data on the drag of wing or fuselage or any other component of the airplane, the data generally refers to the drag of that component when it is alone in the airstream and free from the influence of any other component. Whereas, in an airplane, the wing, the fuselage and the tails are present in close proximity of each other and the flow past one component is influenced by the others. As a result, the drag of the airplane as a combination of different components is different from the sum of the drags of individual components. To appreciate this, let us consider the case examined in Ref. 3.1. Flow past a rather thick airfoil section, shown in Fig 3.1a, is examined at a

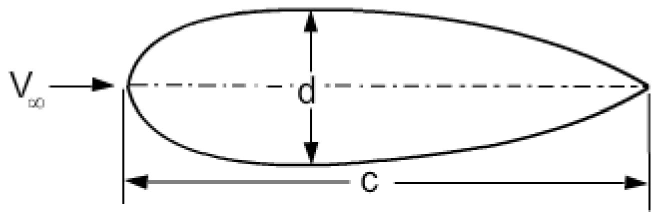
Reynolds number of 420,000. The maximum thickness and the chord of the airfoil are denoted respectively by ‘d’ and ‘c’. The thickness ratio (d/c) for the airfoil in Fig 3.1a is 33.3%.

The drag coefficient is defined as :

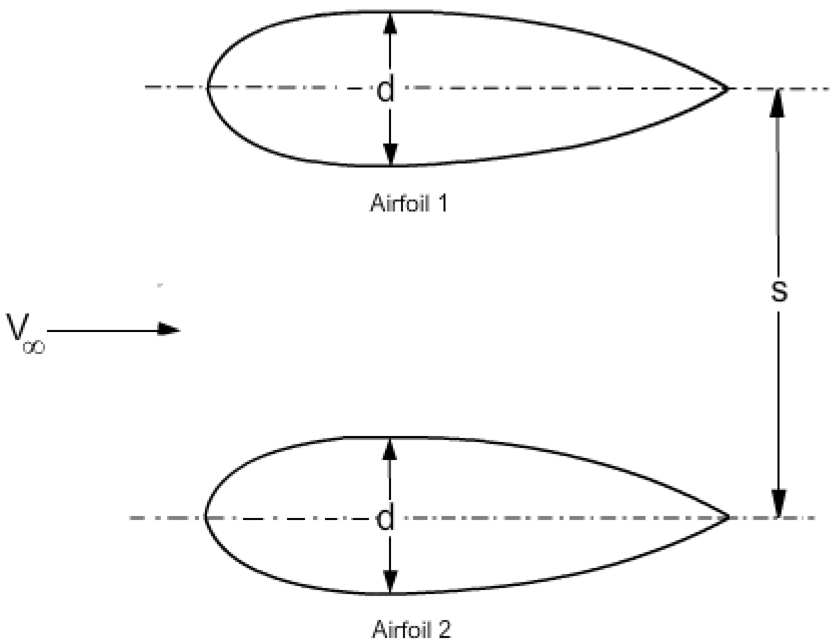
$$C_d = \frac{D}{\frac{1}{2}\rho V_\infty^2 \, c b} \text{ ; b = span of the airfoil model}$$

The drag coefficient (C_d) is found to be 0.0247.

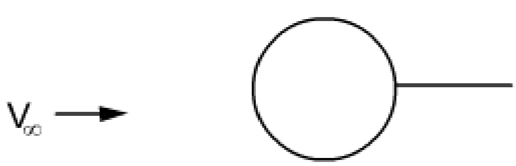
Subsequently, another identical airfoil is placed side by side with a spacing(s) as shown in Fig.3.1b. The tests were carried out for different values of s/d. It is found that for large values of s/d, say s/d > 5, the flows past the two sections do not interfere and the total drag coefficient of the combination is equal to the sum of the drags of each airfoil namely $(C_d)_{combination}$ = 0.0494



(a) Single airfoil



(b) Configuration with airfoils placed side by side as seen in plan view



(c) Circular cylinder with splitter plate at rear

Note : The cylinder is circular in shape. Please adjust the resolution of your monitor so that the cylinder looks circular.

Fig 3.1 Interference effects

However, at closer spacings the results presented in table 3.1 are obtained.

s/d	1.16	1.4	1.8	2.0	2.6	4	5
$(C_d)_{combination}$	0.1727	0.1194	0.0824	0.0761	0.0627	0.0527	0.0494
C_{dint}	0.2233	0.070	0.033	0.0267	0.0133	0.0033	0.0

Table 3.1 Interference drag coefficient for different spacings between two airfoils

Note:

$$(C_d)_{combination} = (C_d)_{airfoil1} + (C_d)_{airfoil2} + C_{dint}$$

It is evident that C_{dint} depends on the relative positions and could be very large.

Remarks:

(i)The drag coefficient of the individual airfoil in this example is large as the airfoil is thick and Reynolds number is rather low. Airfoils used on airplanes would have thickness ratio (t/c) of 12 to 18% and the values of C_d, for Reynolds number of 6×10^6, would be around 0.006.

(ii) Ways to reduce interference drag

A large number of studies have been carried out on interference drag and it is found that D_{int} can be brought down to 5 to 10% of the sum of the drags of all components, by giving proper fillets at the junctions of wing and fuselage and tails and fuselage (Fig 3.2).

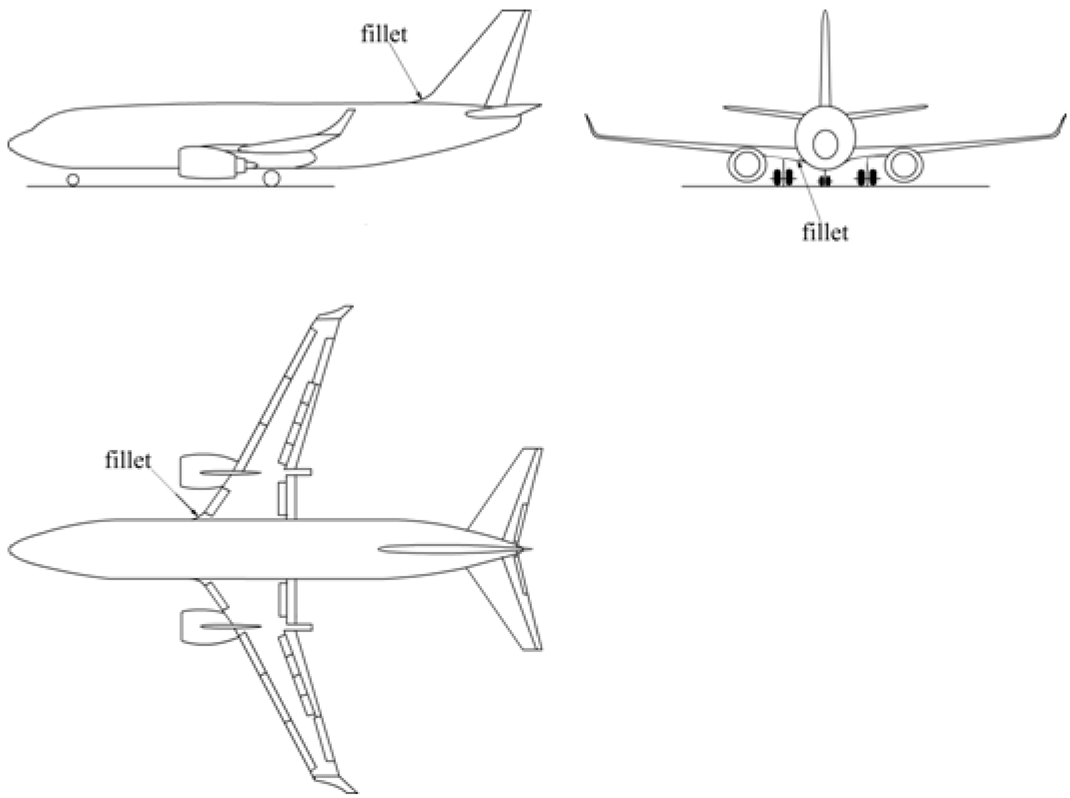


Fig 3.2 Reduction of interference drag using fillets

(iii) Favorable interference effect

The interference effects need not always increase the drag. As an example the drag of a circular cylinder with a splitter plate (Fig 3.1c) is lower than the drag of a cylinder without it at certain Reynolds numbers (Ref 3.2). In an another example, the birds flying in formation flight experience lower drag than when flying individually.

(iv) Chapter VIII of Ref. 3.3 can be consulted for additional information on interference drag.

3.1.3 Contributions to airplane lift

The main contribution to the lift comes from wing-fuselage combination and a small contribution from the horizontal tail i.e.

$$L = L_{wing\ +\ fuselage} + L_{ht} \tag{3.2}$$

For airplanes with wings having aspect ratio greater than six, the lift due to the wing-fuselage combination is roughly equal to the lift produced by the gross wing area. The gross wing area (S) is the planform area of the wing, extended into the fuselage, up to the plane of the symmetry.

3.1.4 Contributions to airplane pitching moment

The pitching moment of the airplane is taken about its center of gravity and denoted by M_{cg}.

Main contributions to M_{cg} are from wing, fuselage, nacelle(s) and horizontal tail i.e.

$$M_{cg} = M_{wing} + M_{fuselage} + M_{nac} + M_{ht} \tag{3.3}$$

3.1.5 Drag coefficient, lift coefficient and pitching moment coefficient of the airplane

To obtain the non-dimensional quantities namely drag coefficient (C_D), lift coefficient (C_L) and pitching moment coefficient (C_{mcg}) of the airplane, the reference quantities are the free stream dynamic pressure (½ ρV_∞^2), the gross wing area (S) and the mean aerodynamic chord of the wing ($\bar{c}$). Consequently,

$$C_D = \frac{D}{\frac{1}{2}\rho V_\infty^2\ S};\ C_L = \frac{L}{\frac{1}{2}\rho V_\infty^2\ S};\ C_{mcg} = \frac{M_{cg}}{\frac{1}{2}\rho V_\infty^2\ S\bar{c}} \tag{3.4}$$

However, the drag coefficient and lift coefficient of the individual components are based on their own reference areas as given below.

(a) For wing, horizontal tail and vertical tail the reference area is their planform area.

(b)For fuselage, nacelle, fuel tanks, bombs and such other bodies the reference area is either the wetted area or the frontal area. The wetted area is the area of the surface of the body in contact with the fluid. The frontal area is the maximum cross-sectional area of the body.

(c) For other components like landing gear the reference area is given along with the definition of C_D.

Remarks:

(i)The reference area, on which the C_D and C_L of an individual component is based, is also called proper area and denoted by S_π; the drag coefficient based on S_π is denoted by $C_{D\pi}$.

(ii)The reference areas for different components are different for the following reasons. The aim of using non-dimensional quantities like C_D is to be able to predict the characteristics of many similar shapes by carrying out computations or tests on a few models. For this to be effective, the phenomena causing the drag must be taken into account while specifying the reference qualities. In this context the drag of streamline shapes like wing and slender bodies is mainly due to the skin friction and depends on the wetted area. Whereas, the drag of bluff bodies like the fuselage of a piston-engined airplane, is mainly the pressure drag and depends on the frontal area. It may be added that for wings, the usual practice is to take the reference area as the planform area because it (planform area) is proportional to the wetted area.

(iii)At this stage the reader is advised to the revise the background on aerodynamics (see for examples References 1.9, 1.10 and 1.12).

(iv) Following the above remarks, the total drag of the airplane can be expressed as:

$$D = \frac{1}{2}\rho V_\infty^2 S C_{Dwing} + \frac{1}{2}\rho V_\infty^2 S_{fuse} C_{Dfuse} + \frac{1}{2}\rho V_\infty^2 S_{nac} C_{Dnac} + \frac{1}{2}\rho V_\infty^2 S_{ht} C_{Dht}$$
$$+ \frac{1}{2}\rho V_\infty^2 S_{vt} C_{Dvt} + \frac{1}{2}\rho V_\infty^2 S_{lg} C_{Dlg} + \frac{1}{2}\rho V_\infty^2 S_{etc} C_{Detc} + D_{int} \qquad (3.5)$$

It may be recalled that S_{etc} and C_{Detc} refer to areas and drag coefficients of other items like external fuel tanks, bombs, struts etc..

$$\text{Or } C_D = \frac{D}{\frac{1}{2}\rho V_\infty^2 S}$$

$$= C_{Dwing} + C_{Dfuse}\frac{S_{fuse}}{S} + C_{Dht}\frac{S_{ht}}{S} + C_{Dvt}\frac{S_{vt}}{S} + C_{Dnac}\frac{S_{nac}}{S} + C_{Dlg}\frac{S_{lg}}{S} + C_{Detc}\frac{S_{etc}}{S} + C_{Dint} \qquad (3.6)$$

The data on drag, lift and pitching moment, compiled from various sources, is available in references 1.9, 1.10, 1.12 and 3.3 to 3.9.

3.1.6 Categorization of airplane components

During the discussion in the previous section it was mentioned that (a) for wing, horizontal tail and vertical tail, the planform area is taken as the reference area, (b) for fuselage, the wetted area or the frontal area is taken as the reference area. The reason for these specifications lies in the fact that in aerodynamics the airplane components are categorised as (a) wing type surfaces, (b) bodies and (c) others. This categorisation, described below, is based on common geometrical features of certain airplane components.
Figure 3.3 shows the geometric parameters of a wing. It is observed that the span (b) of the wing is much larger than the chord (c) of the wing section (or the airfoil) and in turn the chord is much larger than the thickness (t) of the airfoil. For wings of subsonic airplanes the ratio (b/c) is between 5 to 12 and the ratio (t/c) for the commonly used profiles is 0.10 to 0.18 or $t/c \approx 0.1$ and $c/b \approx 0.1$. This separation of sizes (or scales in more technical terms) enables the simplification that the flow past a wing can be analysed as a study of flow past an airfoil and then applying correction for the effect of finite wing span. It may be recalled that in aerodynamics an airfoil is treated as a wing of infinite span or a two-dimensional problem.

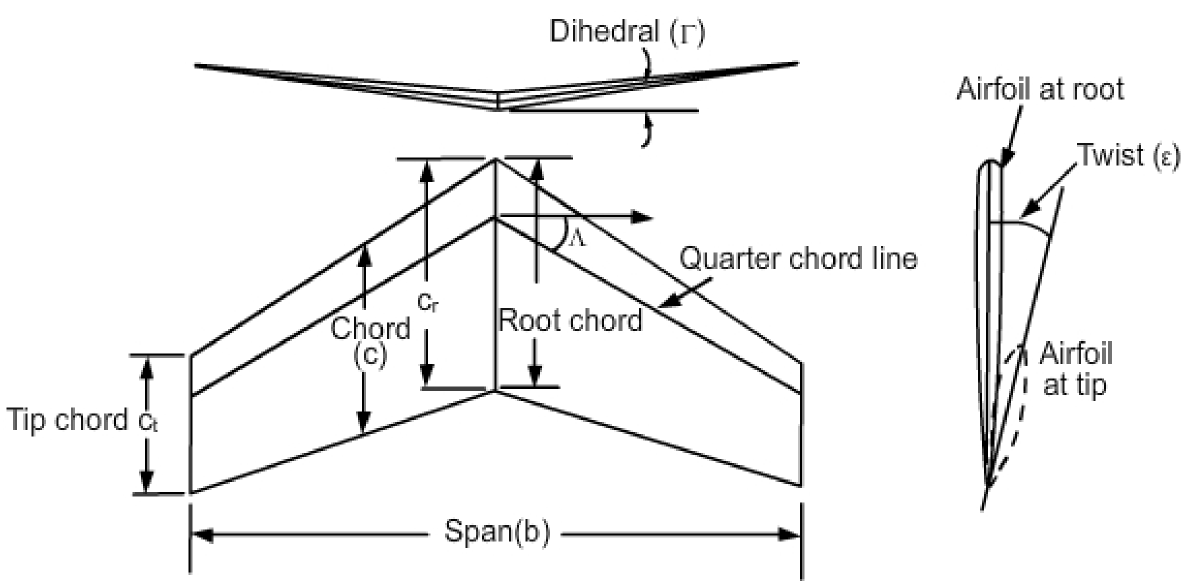


Fig.3.3 Geometric parameters of a wing

Hence, subsections 3.2.2 to 3.2.13 deal with various aspects of flow past airfoils which are relevant to the estimation of drag polar. The subsequent subsection deals with the induced drag which is the result of finite span. It may be added that in aerodynamics, the quantity finite aspect ratio (A) is employed instead of the finite span. The aspect ratio is defined as :

$A = b^2/S$; b = wing span, S = wing planform area

Remarks :

(i)When the aspect ratio is less than about 5, which is characteristic of wings of high speed airplanes, the flow past the wing has to be treated as three-dimensional.

(ii) Horizontal tail, vertical tail and streamlined struts, seen on some low speed airplanes, come under the category of wing type surfaces.

Fig 3.4a shows the fuselage of a jet airplane. Here the length (l_f) is much larger than the height (h) and width (w), but 'h' and 'w' are generally not very different in their dimensions. Hence, the flow past a fuselage cannot be considered as two-dimensional. However, for jet airplanes, l_f/h is around 6 to 10 and the analysis of flow past fuselage can be simplified by assuming the fuselage to be a slender/streamlined body.

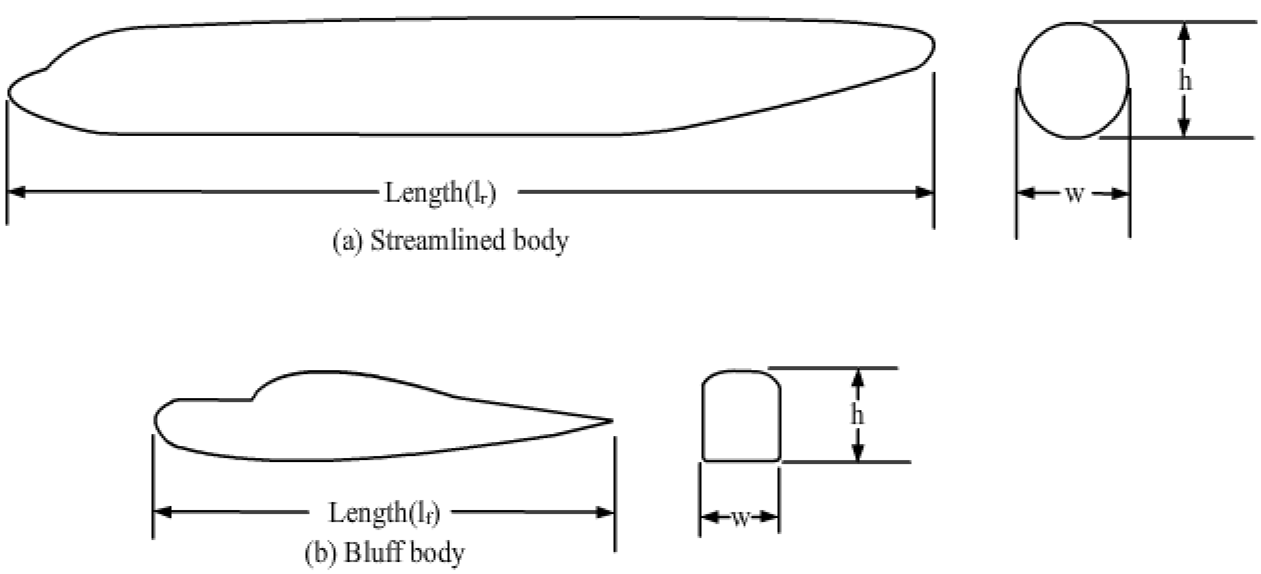


Fig.3.4 Fuselage parameters

Figure 3.4 b shows the fuselage of a low speed airplane. Here l_f/h is rather low and the fuselage is treated as a bluff body.

Precise definitions of the streamlined body and bluff body are given in the subsequent sections.

Remarks:

(i) As regards the analysis of flow is concerned, the fuselage, nacelle, external fuel tanks, bombs, and antenna masts have common geometric features and are categorised as “bodies”.

(ii) Components of airplane like landing gear, which do not fall under the above two categories, are designated as ‘others’.

3.2. Estimation of drag polar at low subsonic speeds

As mentioned in the previous section, the drag polar of an airplane can be obtained by summing-up the drags of individual components and then adding 5 to 10% for the interference drag. As the drag coefficient depends on the angle of attack, this exercise has to be carried-out at different angles of attack. The definition of the angle of attack of the airplane and brief descriptions of the drag coefficients of the airplane components are presented before discussing the drag polar.

3.2.1 Angles of attack of the airplane, wing incidence and tail incidence

For defining the angle of attack of an airplane, the fuselage reference line(FRL) is taken as the airplane reference line (Figs. 1.9 and 3.5).The angle between the free stream velocity and FRL is the angle of attack of the airplane. However, the angles of attack of the wing and tail are not the same as that of the fuselage.

The wing is fixed on the fuselage such that it makes an angle, i_w, to the fuselage reference line (Fig 3.5). This angle is called wing incidence. The angle i_w is generally chosen such that during the cruising flight the wing can produce enough lift when fuselage is at zero angle of attack. This is done because the fuselage produces least drag when it is at zero angle of attack and that is ideal during the cruising flight. In other words, during cruise the wing produces the lift required to balance the weight whereas the fuselage, being at zero angle of attack, produces least drag.

The horizontal tail is set on fuselage at an angle i_t (Fig. 3.5). This angle is called tail incidence. It is generally chosen in a manner that during cruise the lift required from the tail, to make the airplane pitching moment zero, is produced by the tail without elevator deflection. This is because, the drag, at low angles of attack, is least when the required lift is produced without elevator deflection.

Remark :

The angles i_w and i_t are measured clockwise from FRL. The angle i_w is positive but the angle i_t is generally negative.

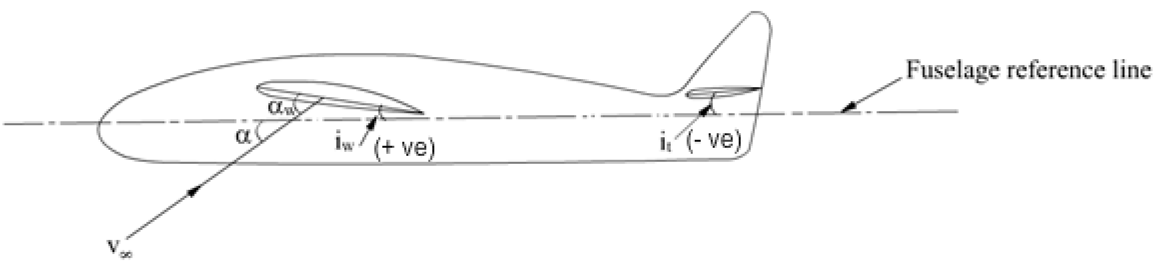


Fig.3.5 Wing incidence (i_w) and tail incidence (i_t)

3.2.2 Skin friction drag, pressure drag and, profile drag of an airfoil

The drag coefficient of a wing consist of the (i) the profile drag due to airfoil (C_d) and (ii) the induced drag due to the finite aspect ratio of the wing (C_{Di}). The symbols C_d and C_l with lower case suffices refer to the drag coefficient and lift coefficient of the airfoil. The profile drag of the airfoil consists of the skin friction drag and the pressure drag. It may be added that an element on the surface of an airfoil, kept in a flow, experiences shear stress(τ) tangential to the surface and pressure (p) normal to it (Fig.3.6). The shear stress multiplied by the area of the element gives the tangential force. The component of this tangential force in the free stream direction when integrated over the profile gives the skin friction drag. Similarly, the pressure distribution results in normal force on the element whose component in the free stream direction, integrated over the profile

Fig.3.6 Shear stress (τ) and pressure(p) on an airfoil

gives the pressure drag. The pressure drag is also called ‘Form drag‘. The sum of the skin friction drag and the pressure drag is called ‘Profile drag’. The profile drag depends on the airfoil shape, Reynolds number, angle of attack and surface roughness.