# Aerodynamics for Airplane Lift and drag

We know that the aerodynamic force on any body moving through the air is due only to basic sources, pressure and shear stress. distribution exerted over the body surface. Lift is created by the pressure distribution, shear stress has only a minor effect on lift. Assumption of inviscid flow is given a good approximation to calculate the lift over an object.

Drag, on the other hand, is created by both pressure and shear stress distribution, analysis just for inviscid flow is not sufficient for prediction of drag.

Lift is produced by the fuselage of an airplane as well as the wing. Wing-body combination is refered a wing plus fuselage. the analysis of the flow field over the body modifies the flow field over the wing and vice-versa. These configuration and interaction must be tested in wind-tunnel or computational fluid dynamic calculation must be made.

Other components of the airplane such as a horizontal tail, canard surfaces, and wing strakes can contribute to lift, either in a positive or negative sense.

For subsonic speeds, however, data obtained using different fuselage thicknesses, d, mounted on wings with different spans, b, show tha the total lift for wing-body combination is constant for $\left(\frac{d}{b}\right)$ varying from 0 (wing only) to 6 (fat fuselage).

Lift of the wing-body combination can be treated as simply the lift on the complete wing by itself, including that portion of the wing that is masked by the fuselage.

For drag. on airplane can not be obtained as the simple sum of drag on each component. For example for wing-body combination, the drag usually is higher than the sum of the separate drag forces on the wing and the body, giving rise to on extra drag componente called interface drag.

We will discussed only for a simple extension of the equation for finite wing.

$$C_D = C_d + \frac{C_L^2}{\pi . e A.R.}$$

For the whole airplane, the equation is rewritten as

$$C_D = C_{De} + \frac{C_L^2}{\pi e . AR.}$$

$C_D$ : total drag coefficient

$C_{De}$ : parasite drag coefficient. These contain not only pressure drag and skin drag but also the friction and pressure drag of the tail fuselage. and any other component of the airplane.

And the lift changes with the angle of attack, we can say that $C_{De}$ is, also, function of $C_L$. A reasonable function is

$$C_{De} = C_{D0} + r C_L^2$$

$r$: is a empirically constant, for $C_L = 0$ then $C_{De} = C_{D0}$

$C_{D0}$ is defined as the parasite drag coefficient at zero lift or zero-lift drag coefficient.

$$C_D = C_{D,0} + \left(r + \frac{1}{\pi e AR}\right) C_L^2$$

Consider $e$, that includes the effect of the variation of parasite drag with lift, so the above equation is defined

$$\left[ C_D = C_{D,0} + \frac{1}{\pi e AR} \cdot C_L^2 \right]$$

where $C_{D,0}$ is the parasite drag coefficient at zero lift and the term $\frac{C_L^2}{e\pi AR}$ is the drag coefficient due to lift including both induced drag and the contribution to parasite drag due to lift.

In the equation, the redefined $e$ is called the Oswald efficiency factor. the Oswald factor for different airplanes typically varies between 0,7 and 0.85 whereas the span efficiency factor varies from 0.9 to 1.0.

the empirical expression for the Oswald factor for straight-wing aircraft. is.

$$\left[ e = 1.78(1-0.045 AR^{0.68}) - 0.64 \right]$$

These equation works for not vary large aspect ratio.

thés equation $\left[ C_D = C_{D,0} + \frac{C_L^2}{\pi e AR} \right]$ is called the drag polar for airplane, representing the variation of $C_D$ with $C_L$. It is the cornerstone for conceptual design of airplane and for predictions of the performance of a given aircraft.

Ex. For the airplane Seversky P-35. this airplane has a wing planform area of $20.4 m^2$ and a wingspan (10.8 m) the break down for drag is 18. for complete airplane configuration. For 18 the total drag is given as $C_D = 0.0275$. when the airplane is at particular angle of attack $C_L = 0.15$.

Solution

$$AR = \frac{b^2}{S} = \frac{(10.8)^2}{20.4} = 5.72$$

Oswald $e = 1.78\left(1 - 0.045(5.72)^{0.68}\right) - 0.64$

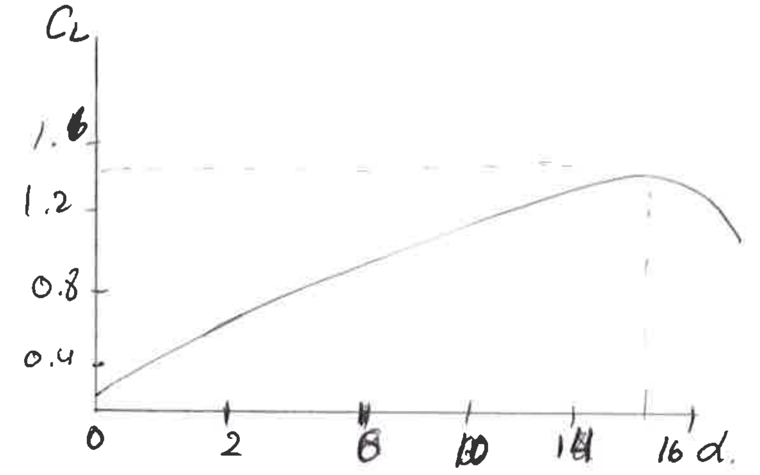
$$e = 0.878$$

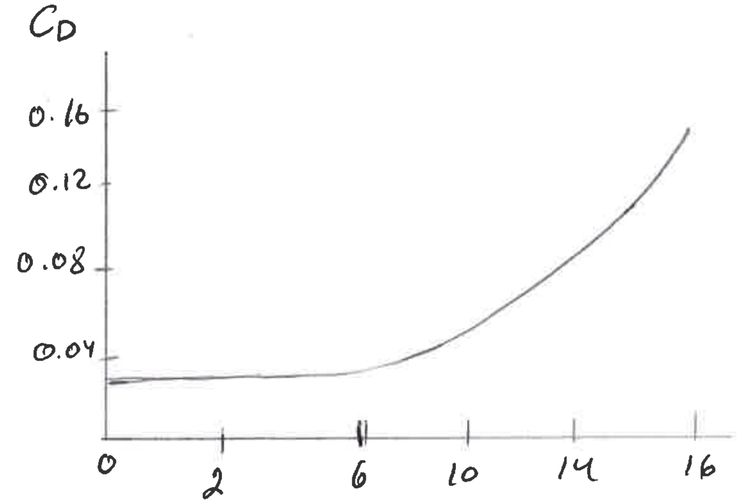
$$C_{D_0} = C_D - \frac{C_L^2}{\pi e AR} = 0.0275 - \frac{(0.15)^2}{\pi \cdot (0.878)(5.72)}$$

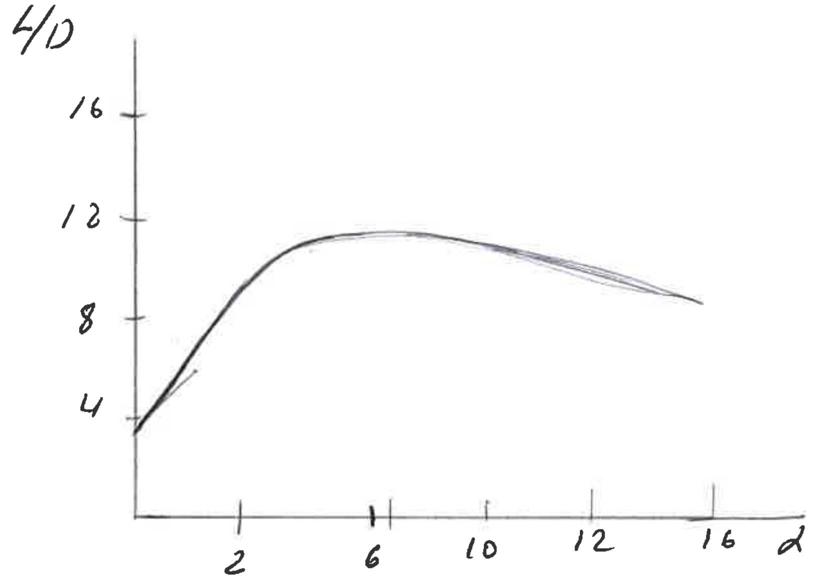
$$\boxed{C_{D_0} = 0.026}$$

Airplane Lift-to-Drag ratio.

The following graph show the variation of $C_L$ vs $\alpha$, $C_D$ vs $\alpha$ and $L/D$ vs $\alpha$ for an airplane.




The $\frac{C_L}{C_D}$ increases with angle of attack, reach a certain maximum value for an $\alpha$. and then decreases

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the maximum lift-to-Drag ratio $(L/D)_{max} = \left(\frac{C_L}{C_D}\right)_{max}$.
is a direct measure of the aerodynamic efficiency of the airplane. and therefore its value is of great importance in airplane design and in prediction of airplane performance.

$$\frac{C_L}{C_D} = \frac{C_L}{C_{D_0} + \frac{C_L^2}{\pi e AR}}$$

For maximum differentiate with respect $C_L$

$$\frac{d(C_L/C_D)}{d(C_L)} = 0 = \frac{C_{D_0} + \frac{C_L^2}{\pi e AR} - C_L[2C_L/\pi e AR)]}{[C_{D_0} + C_L^2/\pi e . AR]^2}$$

$$C_{D_0} + \frac{C_L^2}{\pi e AR} - \frac{C_L^2 \, 2}{\pi e AR} = 0$$

$$\left[C_{D_0} = \frac{\cancel{2} C_L^2}{\pi e AR}\right]$$

When the airplane is flying at the specific angle of attack where lift-drag-ratio is maximum the zero-lift drag and the drag due to lift are precisely equal.

$$\left[C_L = \sqrt{\pi e AR \, C_{D_0}}\right]$$

then $$\left(\frac{C_L}{C_D}\right)_{max} = \frac{(\pi e AR\, C_{D,0})^{1/2}}{C_{D0} + \frac{\pi e AR\, C_{D0}}{\pi e AR}}$$

$$\left[\left(\frac{C_L}{C_D}\right)_{max} = \frac{(\pi e\, AR.\, C_{D,0})^{1/2}}{2\, C_{D,0}}\right]$$

only depends of zero-lift drag coefficient $C_{D,0}$ $e$, $AR$.

Ex. From the above example. Calculate the maximum lift-to-drag ratio for the airplane mentioned above.

We calculated $C_{D,0} = 0.026$

$e = 0.878$

$AR = 5.72$

$$\left(\frac{C_L}{C_D}\right)_{max} = \frac{(\pi e . C_{D0})^{1/2}}{2\, C_{D0}} = \frac{(\pi .\, 0.878 \times .\, 0.026)^{1/2}}{2 \times 0.026}$$

$$\left(\frac{C_L}{C_D}\right)_{max} = 12.32$$

the value tabulated for the airplane is 11.8. Our calculations is about 5%.