C2 Aircraft Flight and Propulsion

Prof L He 4 lectures

This course is intended to provide engineers with the fundamentals of the methods of predicting aircraft performance from simple modelling of the lift-drag curve. In the C1A Wing Theory course you have studied in detail the vortex theory of wing lift; this course builds upon some of the useful results.

Synopsis

Lecture 1

Review of wing theory & principles;

Lift & drag characteristics

Lecture 2

Steady level flight

Engine characteristics → propeller and jet propulsion

Lecture 3

Climbing flight

Gliding flight

Range and Endurance

Lecture 4

Turning flight

Take off and landing

Reading

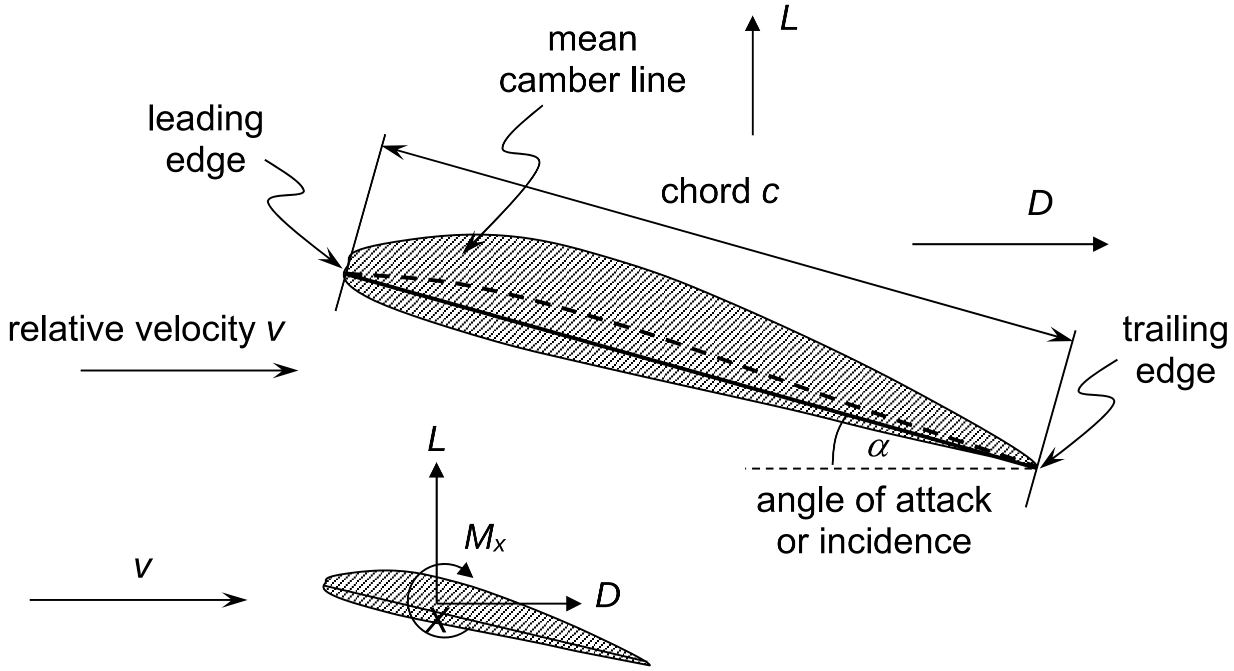
Anderson, J.D., Jr, “Aircraft Performance and Design”, McGraw-Hill, 1999

Clancy, L.J., “Aerodynamics”, Wiley 1975

Anderson, J.D., Jr, “Introduction to flight”, McGraw-Hill, 1978

For a detailed look at design problems:

Stinton, “The Design of the Aeroplane”, 2nd ed., Blackwell, 2001



The force acting on an aerofoil arises due to the pressures and shear stresses over its surface. The resultant force is written in terms of its orthogonal lift L and drag D components. By definition, lift is perpendicular to the free stream velocity, and drag is parallel to the free stream velocity. The resultant force acts through the centre of pressure. The lift and drag can be taken to act at any point X along the chord, provided that the appropriate pitching moment about that point M_X is also applied. Pitching moment is zero at the centre of pressure.

Lift coefficient

The wings of an aircraft provide the lift required for flight. Lift is a function of free stream velocity v, free stream density ρ, wing surface area S, incidence of wings to free stream α, free stream viscosity μ, and free stream sonic velocity a_∞,

$$L = f(\rho, v, S, \alpha, \mu, a_\infty)$$

Applying dimensional analysis: $$\frac{L}{\frac{1}{2}\rho v^2 S} = f\left(\alpha, \frac{v}{a_\infty}, \frac{\rho v \sqrt{S}}{\mu}\right)$$

i.e. $$C_L = f(\alpha, M_\infty, \mathrm{Re})$$

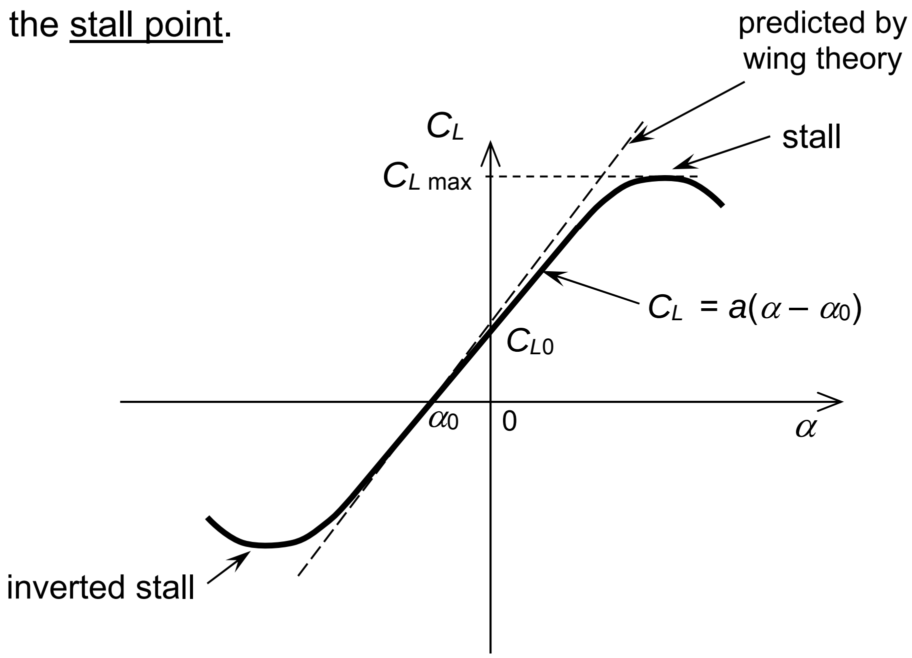
where $C_L = \dfrac{L}{\frac{1}{2}\rho v^2 S}$ is lift coefficient, $M_\infty = \dfrac{v}{a_\infty}$ is the Mach number, and

$\mathrm{Re} = \dfrac{\rho v \sqrt{S}}{\mu}$ is Reynolds number.

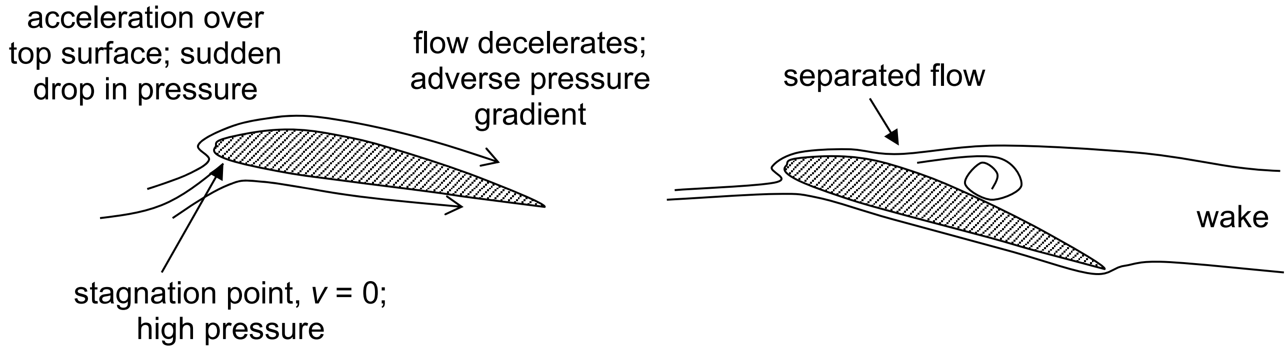
For inviscid flow, shear stresses are neglected and Re effects do not apply, and for incompressible flow (M_∞ < ~0.3) Mach number effects do not apply. C_L is then only a function of incidence, $C_L = f(\alpha)$. Wing theory predicts that for a wing of infinite span:

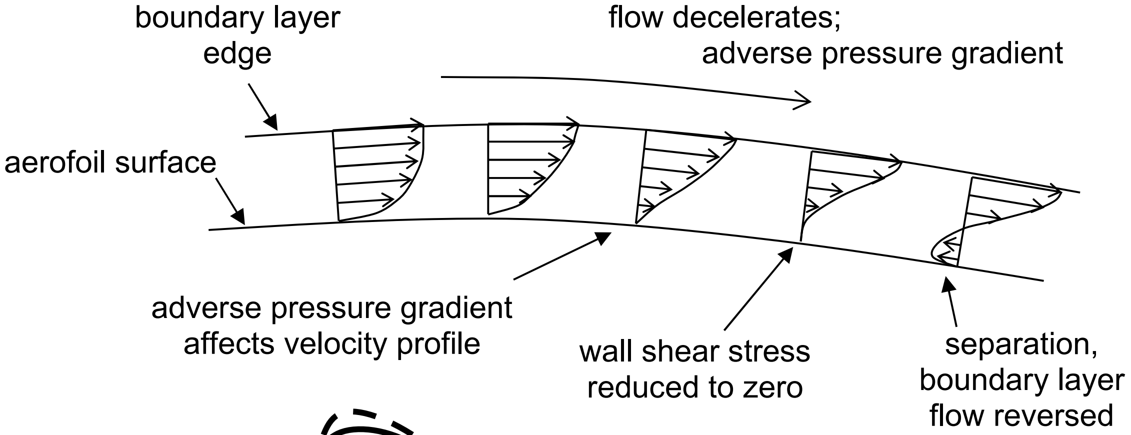
$$C_L = a(\alpha - \alpha_0) \tag{1-1}$$

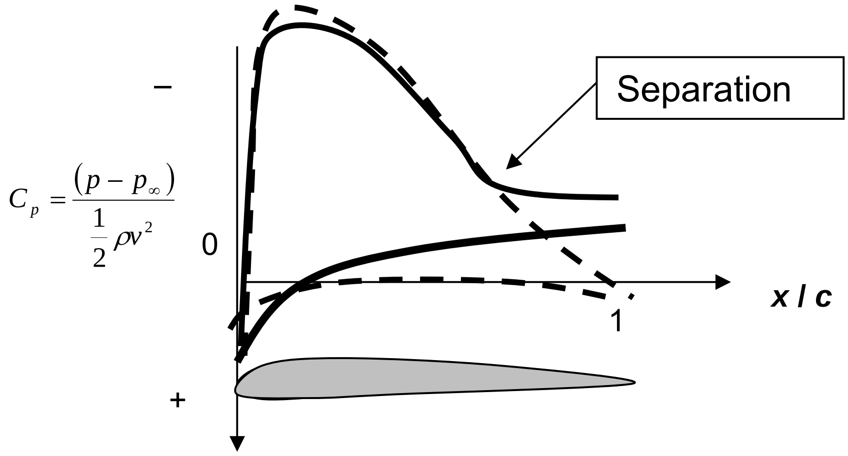
where $a = 2\pi$ and α_0 is the zero lift angle; $\alpha_0 = 0$ for symmetrical aerofoils, and is negative for positive camber. In practice $a \approx 5.7$ /radian, (0.1 /degree) for practical aerofoils, up to the point where the flow separates from the upper surface: the <u>stall point</u>.



The flow over an aerofoil accelerates around the leading edge then decelerates over the top surface; as the flow decelerates, the pressure rises. If the adverse pressure gradient is large enough to reverse the velocity profile, then the boundary layer will separate from the top surface. A low-pressure wake is created, which reduces lift and increases drag.



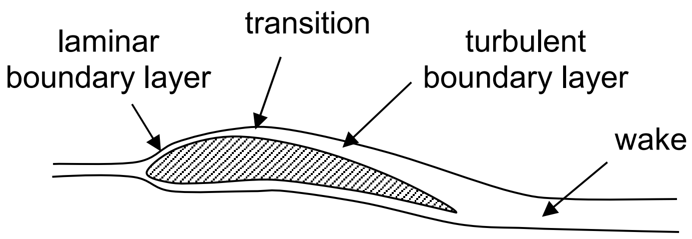


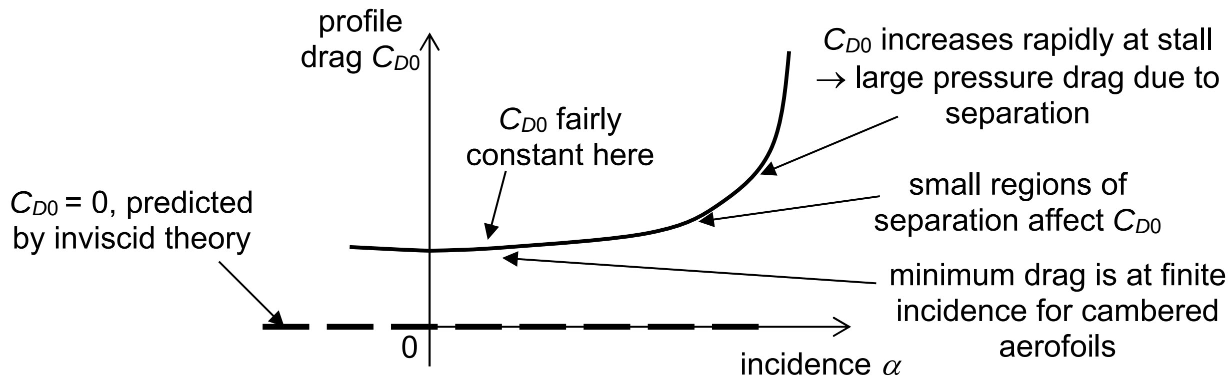


pressure (Cp) distribution over aerofoil surface

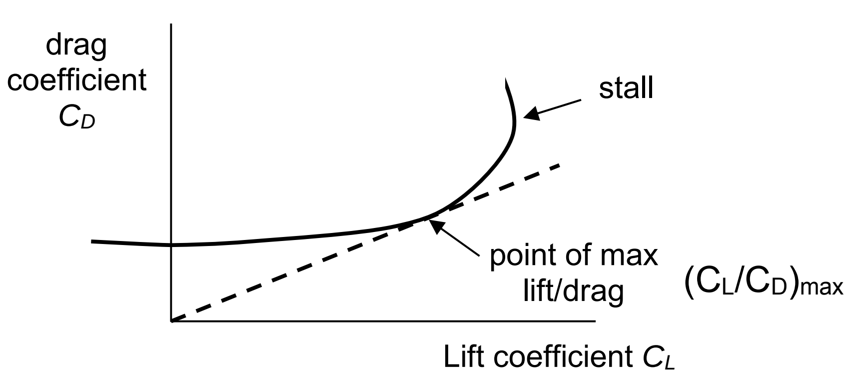
Drag coefficient

The drag coefficient, defined as $C_D = \dfrac{D}{\frac{1}{2}\rho v^2 S}$, is a function of incidence, Reynolds number and Mach number, $C_D = f(\alpha, M_\infty, \mathrm{Re})$. For inviscid and incompressible flow, potential flow theory predicts that C_D = 0 for an infinite span wing. Real wings have drag due to viscous effects: shear stresses in the boundary layer, and momentum deficits in the wake. This is known as profile drag and is denoted by C_{D0} *(also: the component of C_D independent of lift).*



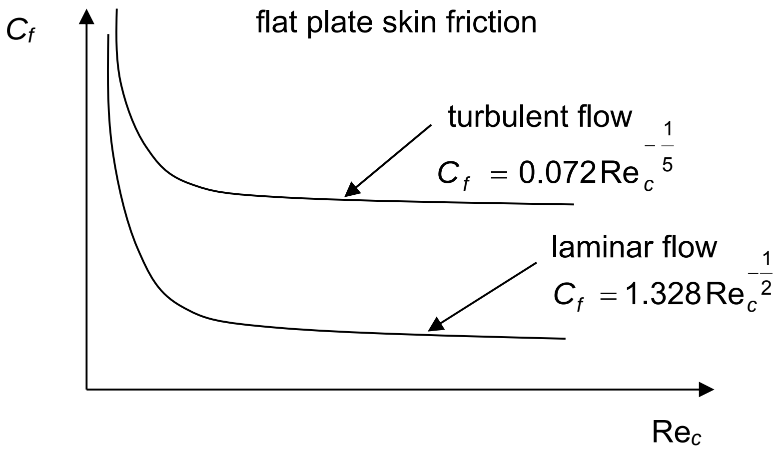


Plot C_D against C_L

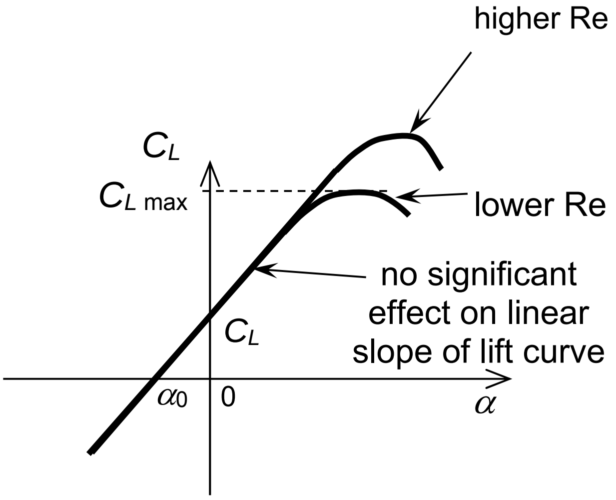


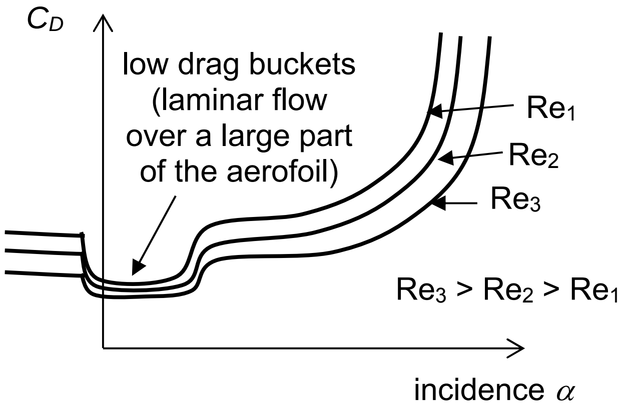
Reynolds number effects

Skin friction affects aerofoil drag but does not have a significant effect on lift. The linear part of the lift coefficient vs incidence curve, slope *a*, does not vary significantly with Reynolds number, however higher values of Re tend to increase $C_{L\max}$ slightly by delaying separation.

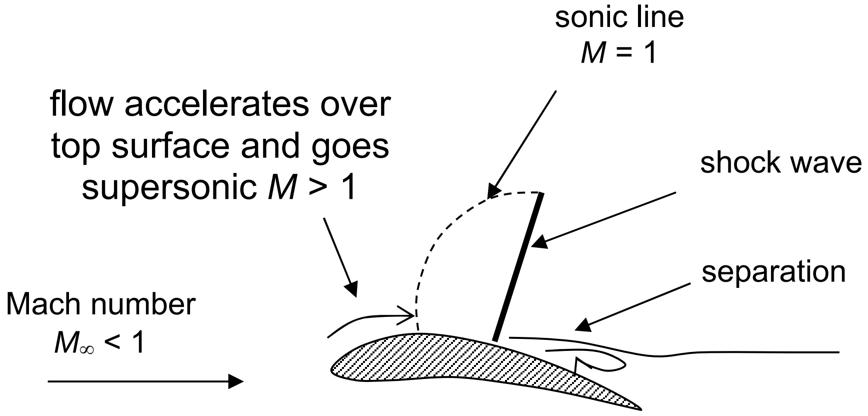


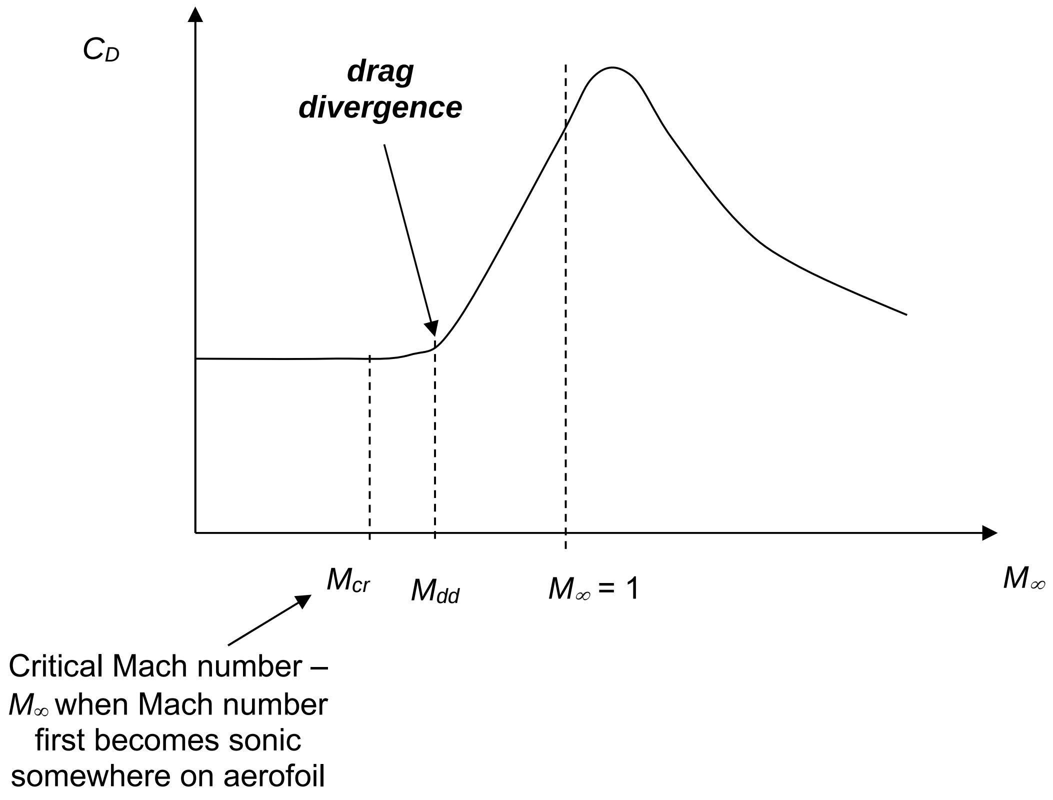
Laminar boundary layers have lower drag than turbulent boundary layers, and skin friction coefficient C_f decreases as Reynolds number increases, (for a flat plate: $C_f = 1.382\,\mathrm{Re}_c^{-\frac{1}{2}}$ for laminar flow, and $C_f = 0.072\,\mathrm{Re}_c^{-\frac{1}{5}}$ for turbulent flow). The drag coefficient therefore decreases slightly at higher Reynolds numbers, and when aerofoils can maintain laminar flow over a large part of the surface, low drag 'buckets' feature at low incidence.

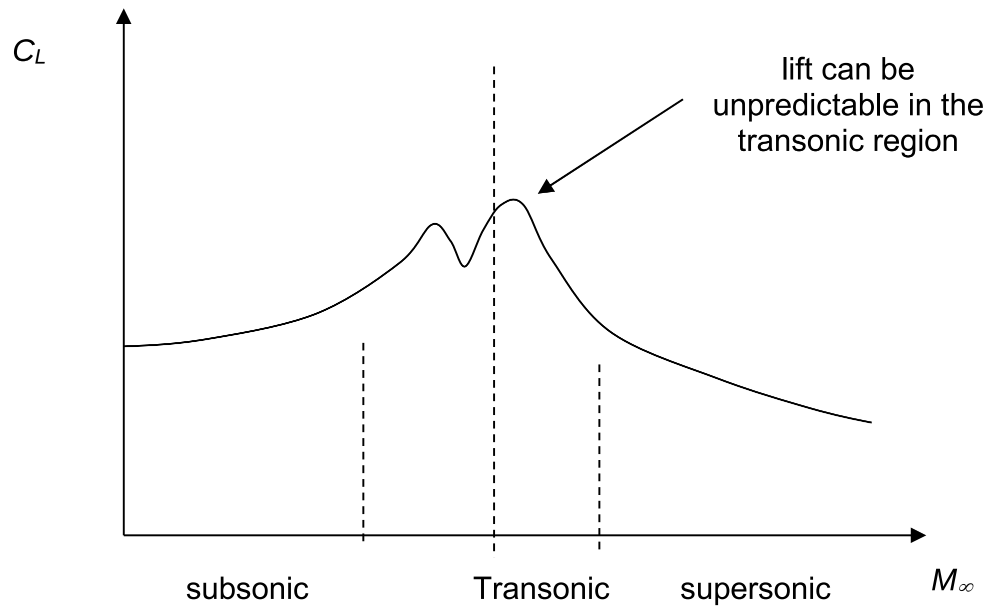




Mach number effects

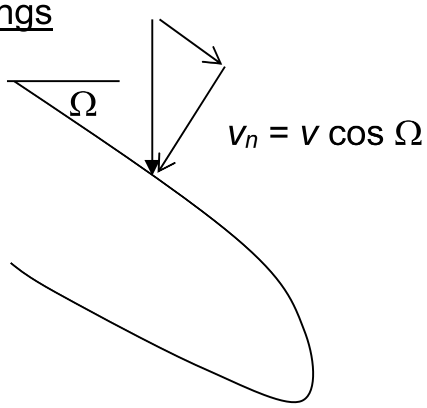






Supercritical aerofoils designed with flat top surfaces to have a nearly constant velocity over the top of the aerofoil, to avoid shockwave formation.

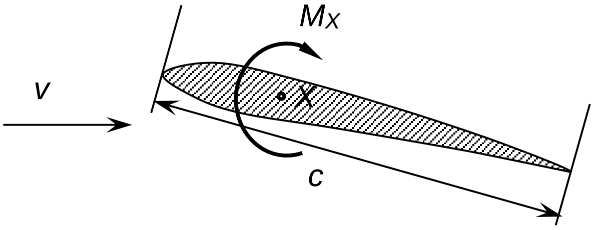
Swept-back wings

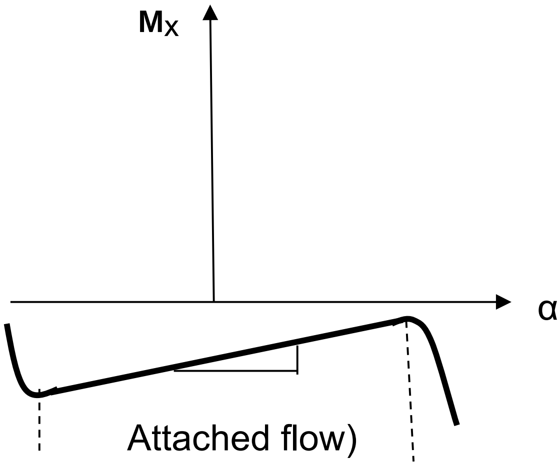


e.g. if M_{cr} = 0.7, and $\Omega = 45^o$

M_{cr} is raised to $0.7\sqrt{2} = 1$

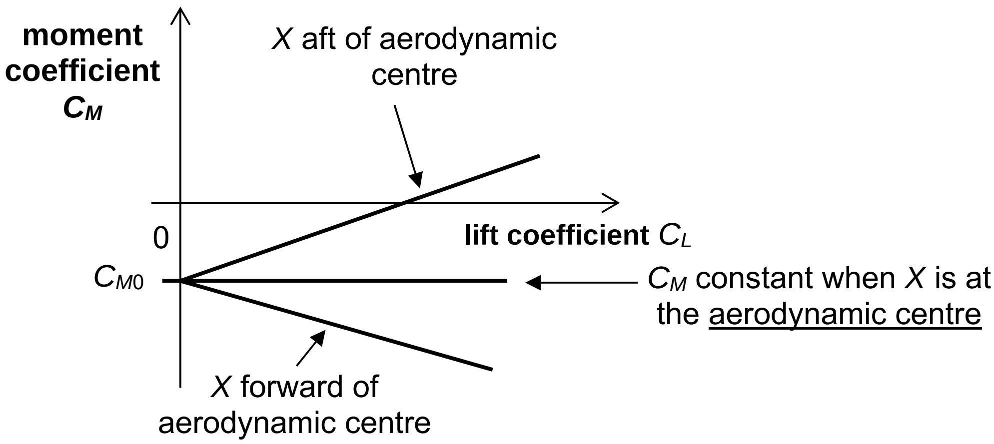
Pitching moment coefficient





The moment coefficient is defined as $C_M = \dfrac{M_X}{\frac{1}{2}\rho v^2 Sc}$, where M_X is the moment about a point X. For inviscid and incompressible flow, C_M is a function of α and the reference point X. Since C_L is proportional to α, we can write

$C_M = C_{M0} + m_0 C_L$, approximately (m_0 — distance of point X measured from a.c.)



For symmetric aerofoils $C_{M0} = 0$, and aerodynamic centre is ¼ chord from the leading edge.

For cambered aerofoils $C_{M0} < 0$ (nose down) and the aerodynamic centre is still ~¼ chord from leading edge.

Choice of aerofoil

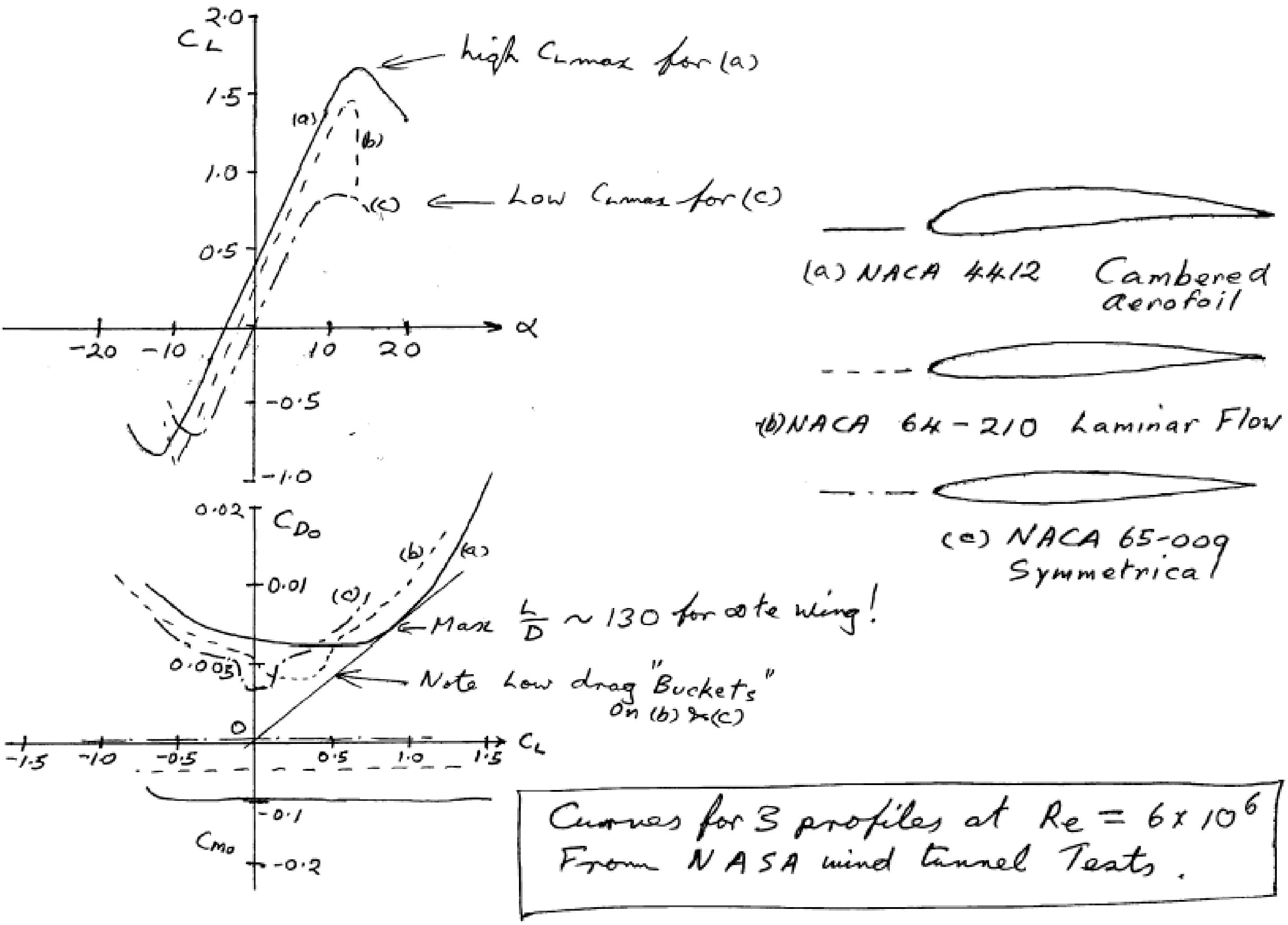
Thickness – C_{Lmax} highest with t/c ≈ 14%, but drag is high at high Mach numbers for thick aerofoils.

Camber – Increases circulation and hence lift. Tends to increase C_{Lmax}.

Flaps – Increase C_{Lmax} further at expense of drag. Used for take-off and landing.

Reynolds number – C_{Lmax} increases slightly with Re, and usually C_{D0} decreases, but may increase if transition from laminar flow to turbulent occurs over a section of the aerofoil.

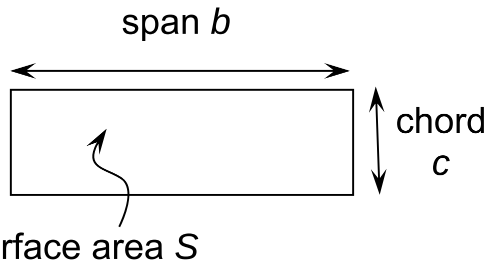
Laminar flow sections – Have low C_D over a region of α ('drag buckets'), achieved by keeping laminar flow over most of the profile – i.e. late transition.



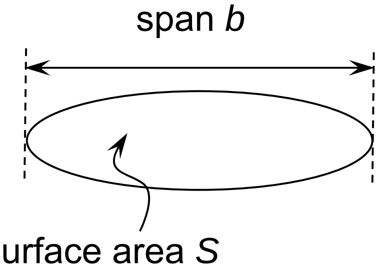
Finite span wings

Aspect ratio: $A = \dfrac{b}{c}$ for rectangular wing

$$A = \frac{b^2}{cb} = \frac{b^2}{S}$$



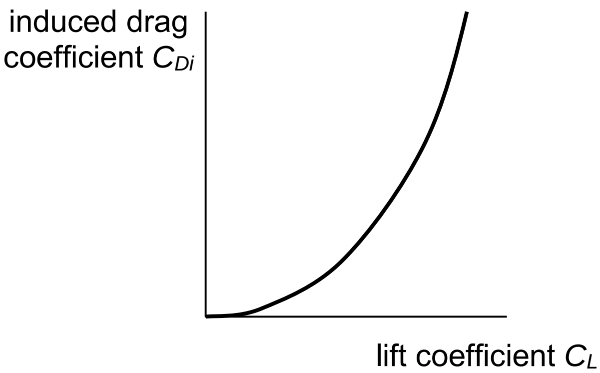
Define aspect ratio $A = \dfrac{b^2}{S}$ for all wings



For elliptic loading, wing theory shows that shed vorticity from finite wing gives an induced drag coefficient, $C_{Di} = \dfrac{C_L^2}{\pi A}$.

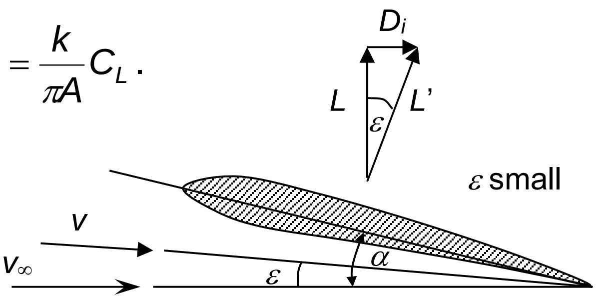
For non-elliptic loading, the effect of greater induced drag is incorporated using the <u>induced drag factor</u> *k*:

$C_{Di} = \dfrac{kC_L^2}{\pi A}$ where *k* may vary from *k* = 1 (elliptical loading) to around *k* = 1.3.



As the induced drag can be considered as being due to the <u>downwash angle ε</u> causing the lift force vector to tilt backwards, we have $D_i \approx L\varepsilon$.

<u>Mean downwash angle</u> $\varepsilon = \dfrac{D_i}{L} = \dfrac{C_{Di}}{C_L} = \dfrac{k}{\pi A} C_L$.



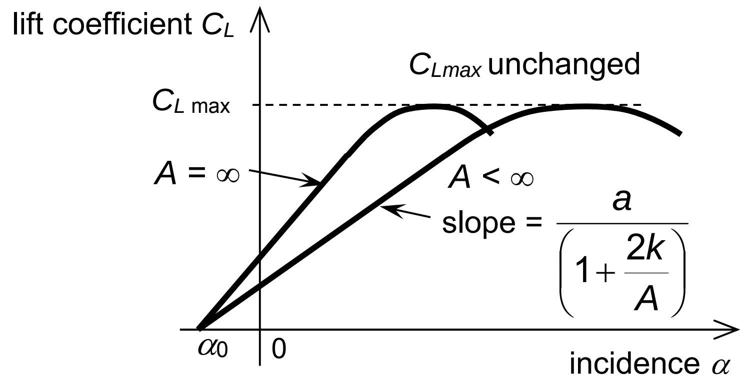
The downwash angle reduces the relative angle of attack to ($\alpha - \varepsilon$). In other words,

$$C_L = a(\alpha - \varepsilon - \alpha_0)$$

$$C_L = a(\alpha - \alpha_0) - \frac{ka}{\pi A} C_L$$

$$C_L = \frac{a(\alpha - \alpha_0)}{\left(1 + \frac{ka}{\pi A}\right)} \qquad (1\text{-}2)$$

$a \approx 2\pi$, $\quad C_L = \dfrac{a(\alpha - \alpha_0)}{\left(1 + \frac{2k}{A}\right)}$



Example

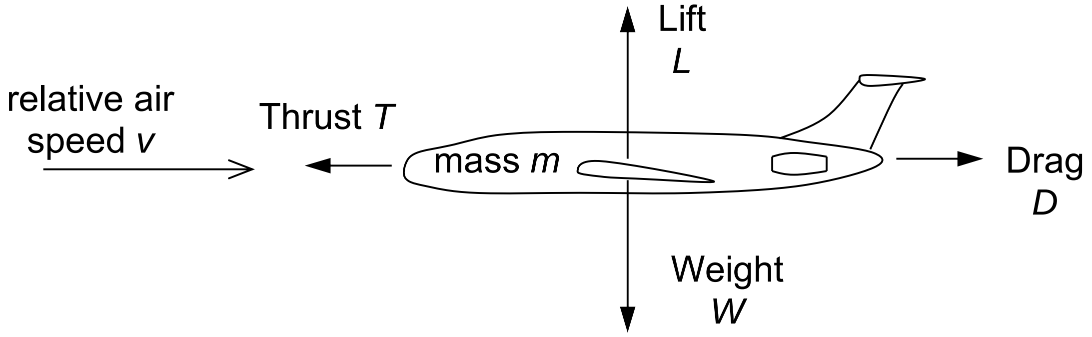
A Harrier Fighter has S = 18.7 m^2, b = 7.7 m, k = 1.2, and a = 0.1 /degree.

Aspect ratio $A = \dfrac{b^2}{S} = \dfrac{7.7^2}{18.7} = 3.17$, and $\left(1 + \dfrac{2k}{A}\right) = \left(1 + \dfrac{2 \times 1.2}{3.17}\right) = 1.76$.

So to fly at C_L = 0.5, say, would require $\alpha = \dfrac{C_L\left(1 + \frac{2k}{A}\right)}{a} = \dfrac{0.5 \times 1.76}{0.1} = 8.8^\circ$, as opposed to $\dfrac{C_L}{a} = \dfrac{0.5}{0.1} = 5^\circ$ for the infinite span case ($A = \infty$).

Sailplanes have $A \approx 20$, and k = 1.1, so $\left(1 + \dfrac{2k}{A}\right)$ = 1.11, and α is almost unchanged.

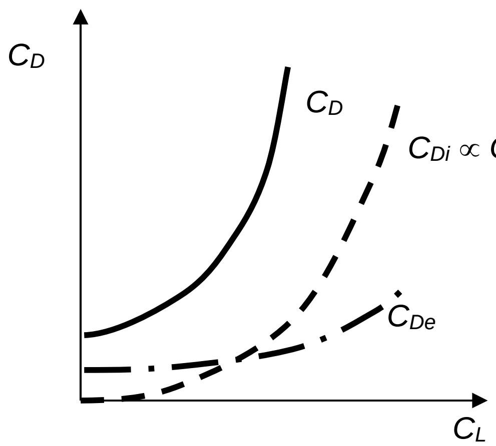
Aircraft drag

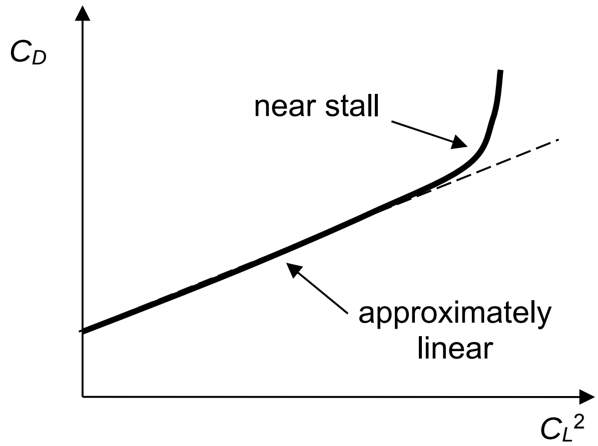


Aircraft drag, D = wing induced drag, D_i
+ wing profile drag, D_0
+ parasitic drag, D_e (from fuselage, engines, tail etc.)

Aircraft drag coefficient: $C_D = C_{Di} + C_{D0} + C_{De}$

The lift-drag polar is an essential part of aircraft design. Wing induced drag varies with C_L^2; the parasitic drag coefficient C_{De} comprises a zero-lift drag component plus the drag due to tail and fuselage lift, which also varies with ~C_L^2. Plotting C_D vs C_L^2, the curve is approximately linear for most of the range:





The drag polar can be written:

$$C_D = C_{D0} + \frac{k}{\pi A} C_L^2 \qquad (1\text{-}3)$$

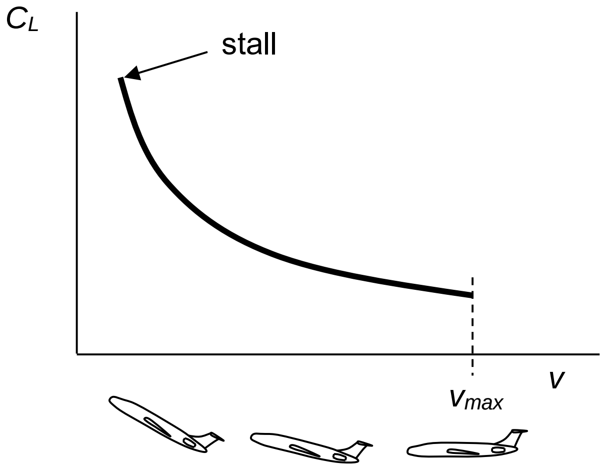
where C_D is the total drag coefficient of the aircraft, and C_L is the total lift coefficient of the aircraft. C_{D0} is the profile and parasitic drag coefficient for the aircraft, and k incorporates the induced drag for the entire aircraft.

Typical C_{D0} values:

Light twin-engined propeller driven	$C_{D0} \approx 0.025$
Jet fighter (clean)	$C_{D0} \approx 0.020$
High performance sailplane	$C_{D0} \approx 0.010$

In level flight, Lift L = Weight W (= mg) so $\frac{1}{2}\rho v^2 S C_L = W$

$$C_L = \frac{2W}{\rho v^2 S} \qquad \rightarrow C_L \text{ changes with the speed of the aircraft.}$$



Flying at low velocity requires higher C_L and hence higher incidence. The incidence required decreases at higher velocities. Flaps or slats are often used to increase C_L at the low take-off and landing velocities.