AE 339 : High speed aerodynamics

(VI) External aerodynamics: Transonic flow past bodies

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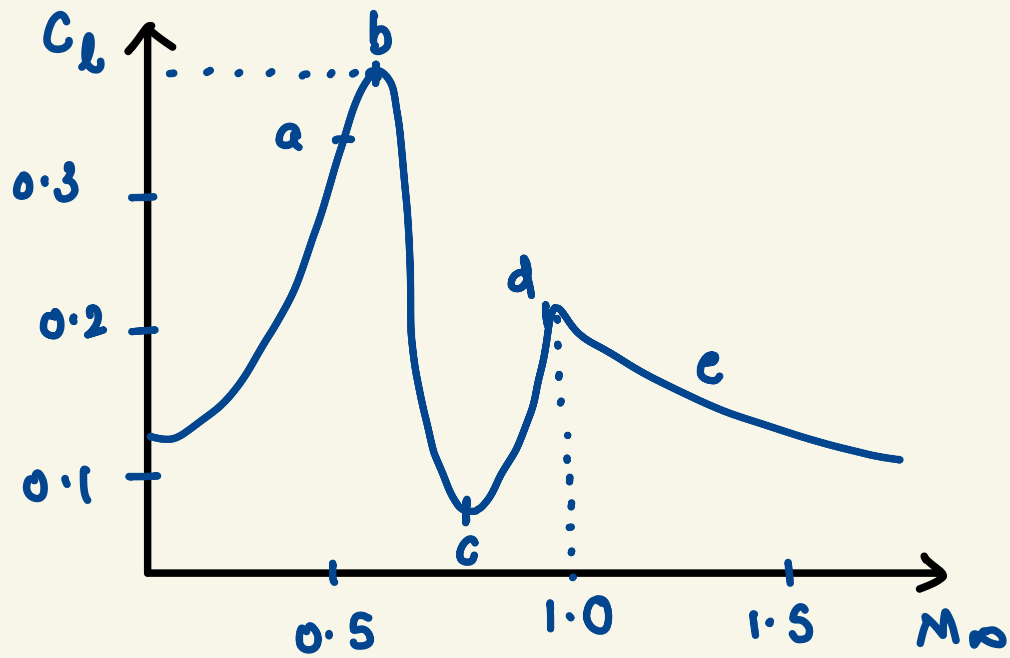
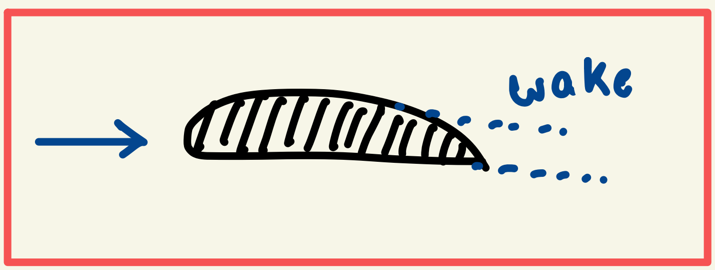
Transonic flow past airfoils

There are complex changes in the flow-field through the transonic speed range.

Lift coefficent

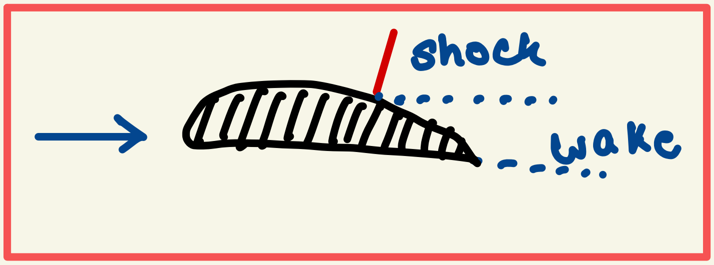


(a) $M_\infty = M_a$



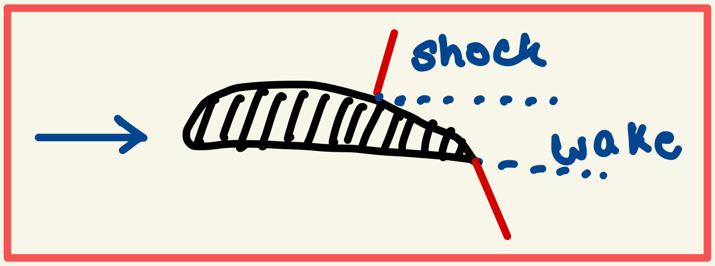
The flow past the upper surface decelerates from local flow velocities which are supersonic without a shock wave. Section lift coefficent is approximately 60% greater than the low speed values at same α.

(b) $M_\infty = M_b$



C_l reaches maximum value, approximately twice the low speed value. Flow is supersonic over 78% of surface, terminating in a shock wave. The flow on the lower surface is subsonic everywhere. As viscous flow separates at the shock, wake is wider.

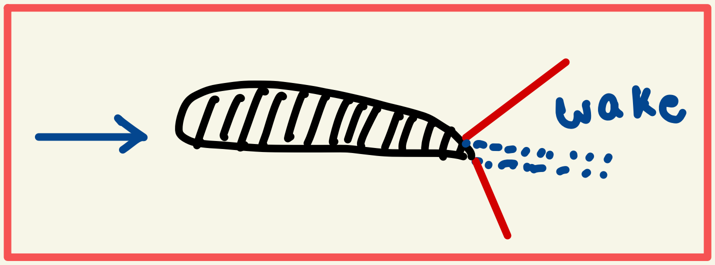
(c) $M_\infty = M_c$



The flow is supersonic over nearly the entire lower surface and deceleration to subsonic speed occurs through a shock wave at the trailing edge ⇒ lower surface pressures less than at $M_\infty = M_b$. Pressure on upper surface greater than lower near trailing edge.

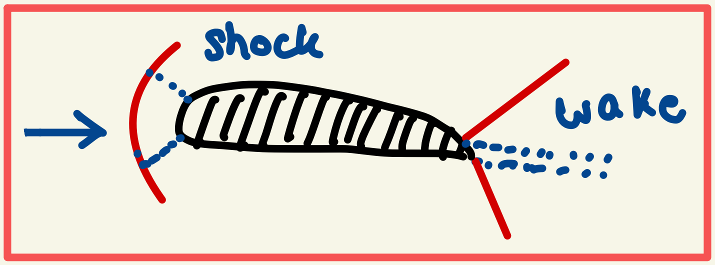
Lift is drastically reduced

(d) $M_\infty = M_d$



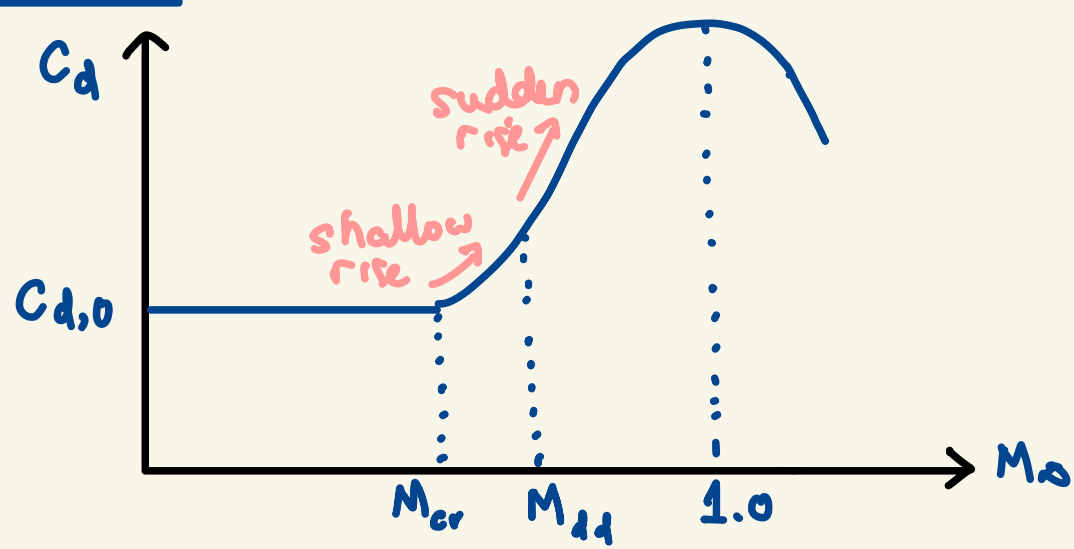
Both shock waves have reached the trailing edge. The local Mach number is supersonic for most of the airfoil.

(e) $M_\infty = M_e$



A bow shock wave is generated. The flow around the airfoil is supersonic everywhere except very near the rounded nose. Trailing edge shocks become weaker.

Drag coefficient



Let $C_{d,0}$ be the drag coefficient at subsonic speed. We have $C_d \approx C_{d,0}$ for $M \leq M_{cr}$.

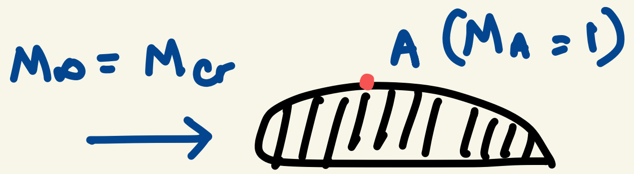
M_{cr} (Critical Mach number) is the lowest freestream Mach number for which the maximum value of the local velocity first becomes sonic

For $M > M_{cr}$, we have transonic flows with mixed subsonic / supersonic flow-fields.

The value of M_∞ at which a sudden rise in drag is observed is defined as the drag divergence Mach number (M_{dd})

The large increase in drag is associated with shock waves that cause severe flow separation

Estimation of M_{cr}



$$\frac{p_A}{p_\infty} = \frac{p_A / p_0}{p_\infty / p_0} = \left[\frac{1 + \frac{\gamma-1}{2} M_\infty^2}{1 + \frac{\gamma-1}{2} M_A^2}\right]^{\frac{\gamma}{\gamma-1}}$$

$$C_{p,A} = \frac{2}{\gamma M_\infty^2}\left[\frac{p_A}{p_\infty} - 1\right]$$

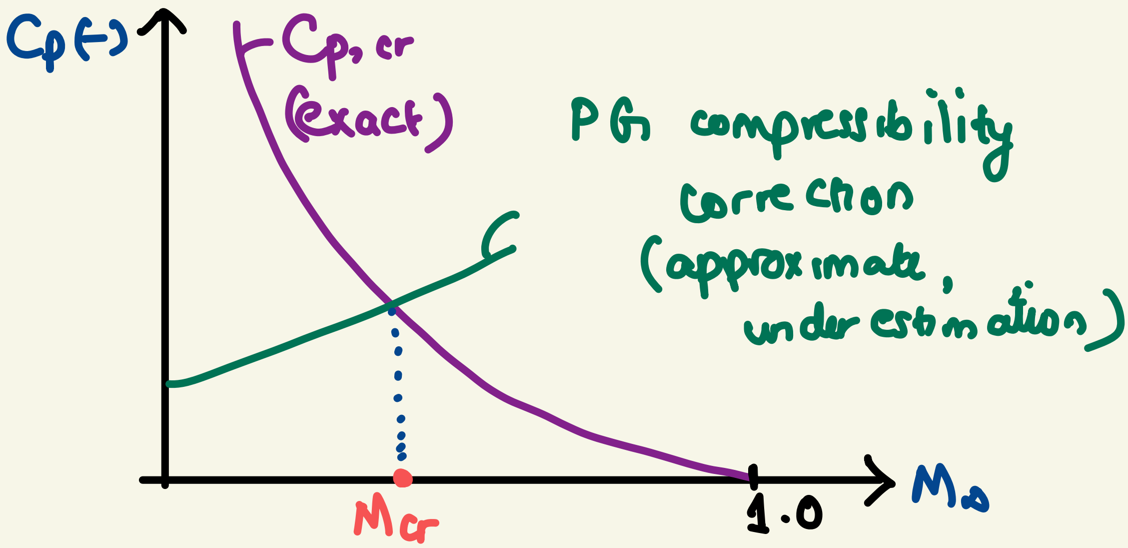
$$= \frac{2}{\gamma M_\infty^2}\left[\left(\frac{1+\frac{\gamma-1}{2}M_\infty^2}{1+\frac{\gamma-1}{2}M_A^2}\right)^{\frac{\gamma}{\gamma-1}} - 1\right]$$

Local value of C_p depends only on local Mach number for given fluid and M_∞

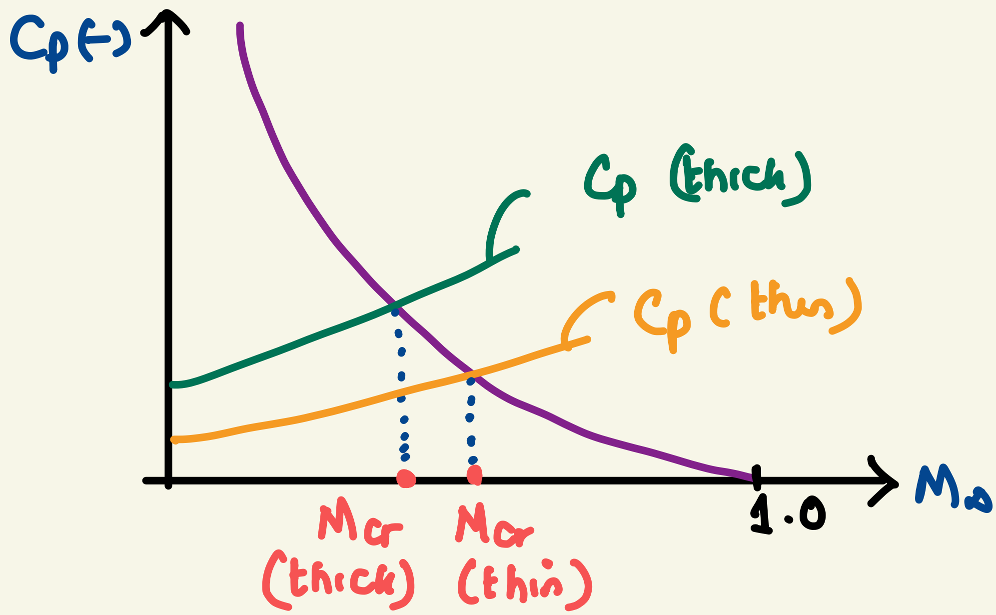
If $M_\infty = M_{cr}$, $M_A = 1$ and $C_p = C_{p,cr}$ (critical pressure coefficient)

$$\boxed{C_{p,cr} = \frac{2}{\gamma M_\infty^2}\left[\left(\frac{1+\frac{\gamma-1}{2}M_\infty^2}{\frac{\gamma+1}{2}}\right)^{\frac{\gamma}{\gamma-1}} - 1\right]} \quad \text{— } (*)$$

This is simply an aerodynamic relation for an isentropic flow and has no connection with the shape of the airfoil.



Thin vs thick airfoils



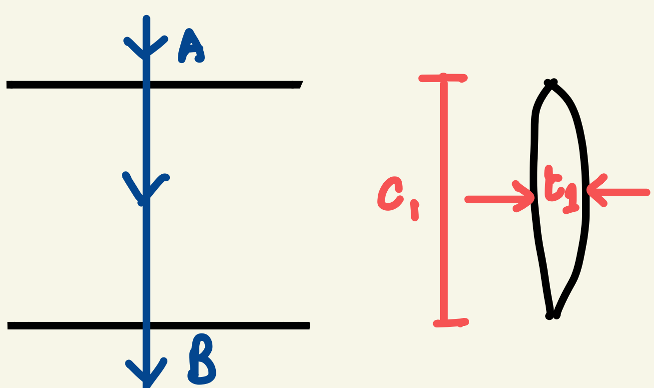
Thin airfoil ⇒ smaller $\hat{u}$ ⇒ smaller C_p

Thick airfoil has a lower M_{cr} than the thin airfoil. Hence, modern high-speed subsonic airplanes have relatively thin airfoils.

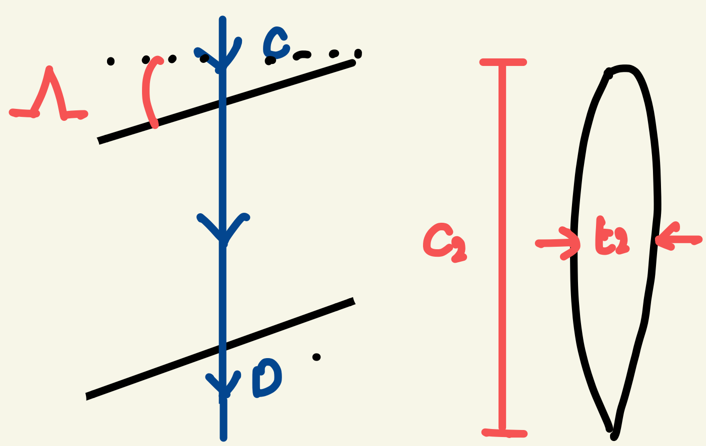
However, airfoil requires a certain thickness for

1. Structural strength
2. storage of fuel.

Swept wings (Busemann 1935)



STRAIGHT WING



SWEPT WING

$$C_2 = \frac{C_1}{\cos\Lambda} > C_1 \quad \text{and} \quad t_2 = t_1$$

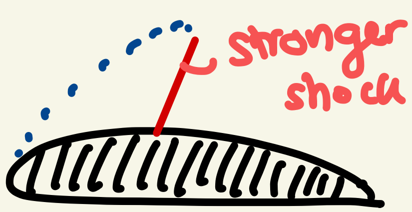
$$\Rightarrow \frac{t_2}{C_2} < \frac{t_1}{C_1}$$ effectively a thinner airfoil

By sweeping the wing, the flow behaves as if the airfoil section is thinner, with a consequent increase in M_{cr}.

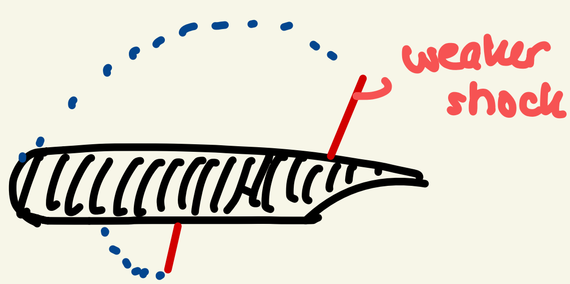
↳ delays the formation of shock waves thus delaying transonic drag rise.

Supercritical airfoil sections (Whitcomb 1965)

These airfoil sections increase the Mach number increment between M_{cr} and M_{dd}.



NACA 64-SERIES
($M_\infty = 0.69$)



SUPERCRITICAL AIRFOIL
($M_\infty = 0.79$)

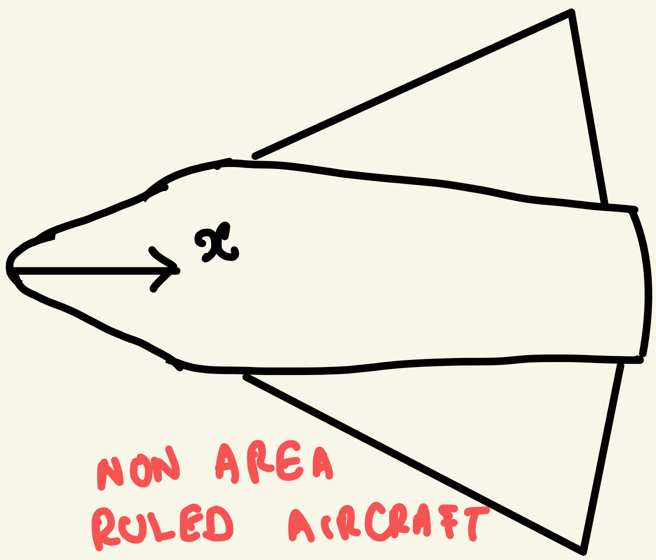
The supercritical airfoil has a relatively flat top, resulting in a weaker terminating shock

$\Rightarrow$ higher M_{dd}

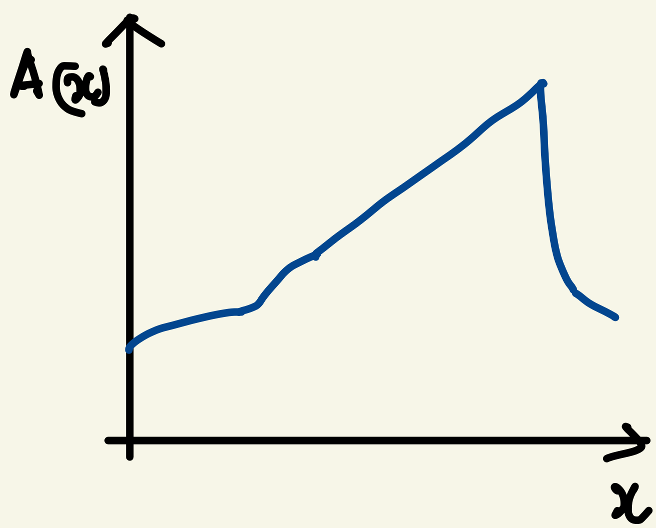
Forward 60% of supercritical airfoils typically have -ve camber which lowers the lift. To compensate, lift is increased by having extreme positive camber on the rearward 30% of the airfoil.

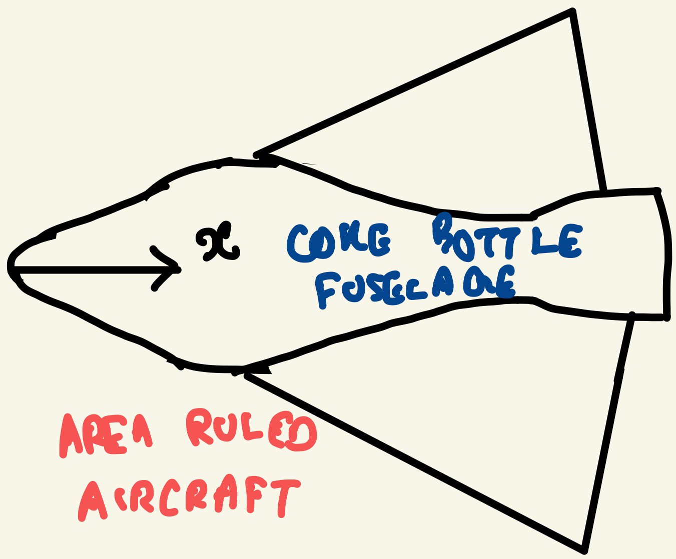
The area rule (Whitcomb)

To reduce peak drag near Mach 1, the variation of cross-sectional area for the airplane should be smooth with no discontinuities



NON AREA
RULED AIRCRAFT





AREA RULED
AIRCRAFT