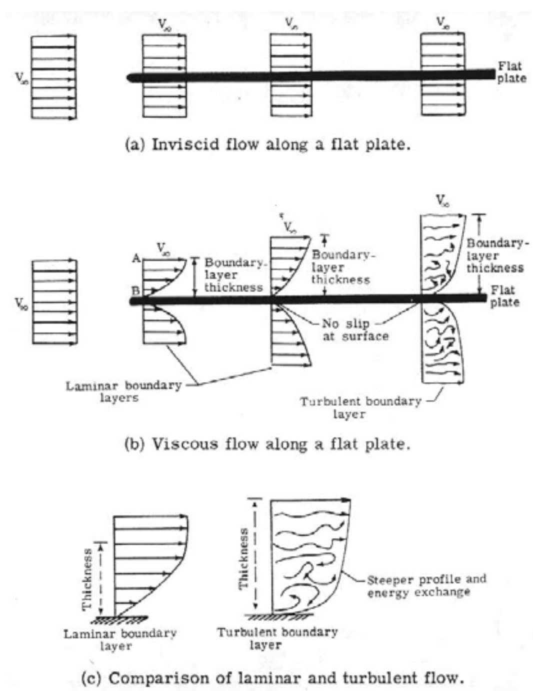
Character of viscous flow

- Laminar flow
  - Streamlines are smooth and regular
    - A fluid element moves smoothly along the streamline
  - Viscous forces are large enough to overcome or smooth oscillations caused by inertia forces
    - We have low Reynolds number in the flow direction
- Turbulent flow
  - Streamlines break up and fluid element moves in an irregular, random, or erratic path
  - Dynamic forces dominate overcome viscous damping
  - The result is cross flow and turbulent (random) mixing
    - We have high Reynolds number flow

Boundary-layer flow in a real fluid

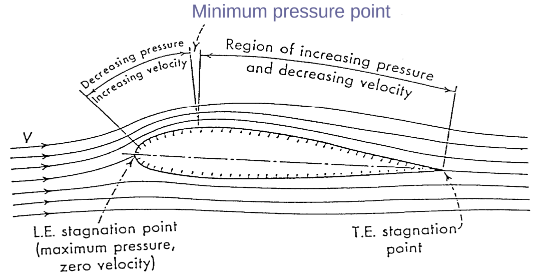
The Reynolds number has an important effect on the boundary layer. As the Reynolds number increases (caused by increasing the flow speed and/or decreasing the viscosity), the boundary layer thickens more slowly.



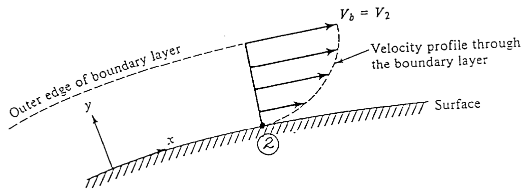
Boundary layers

Laminar and Turbulent

- Air flowing over a wing



- Boundary layer velocity profile

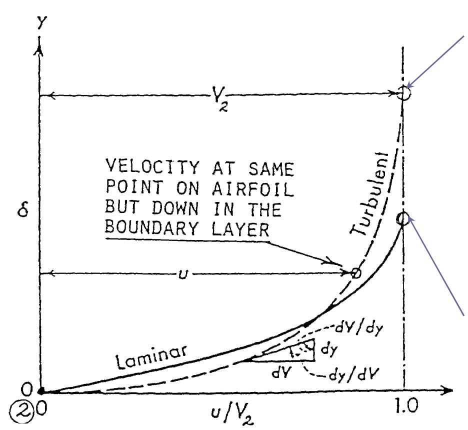


Boundary layers

Laminar and Turbulent Comparison

δ is vertical height above surface of wing at point 2 on the wing

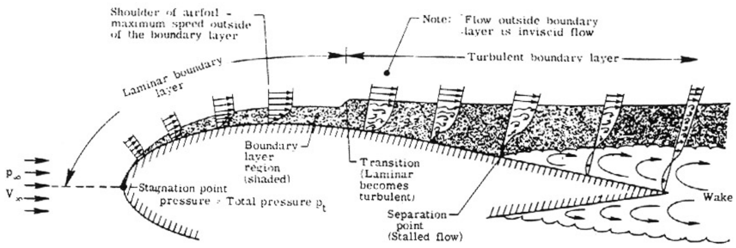
$$\tau_w \equiv \mu \left(\frac{dV}{dy} \right)_{y=0}$$



Velocity at top of boundary layer at point 2

Velocity at top of boundary layer at point 2

- Real fluid flow about an airfoil. Thickness of boundary layers and wake greatly exaggerated. Bottom flow along lower surface is the same as on the upper surface.

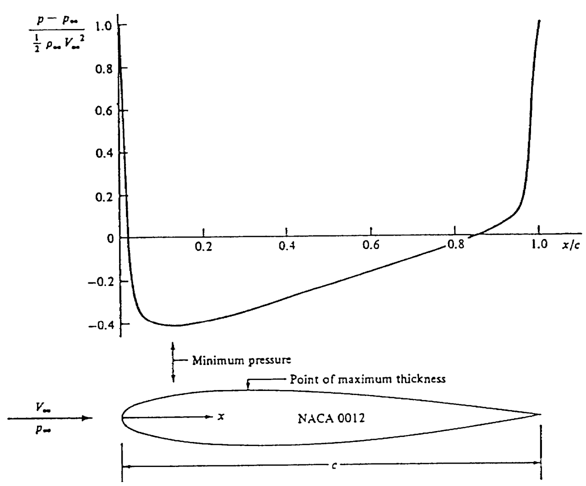


- A boundary layer begins to form because of viscosity. This boundary layer is very thin and outside of it the flow acts very much like that of an ideal fluid. Also, the static pressure acting on the surface of the airfoil is determined by the static pressure outside the boundary layer. This pressure is transmitted through the boundary layer to the surface and thus acts as if the boundary layer were not present at all. But the boundary layer feels this static pressure and will respond to it.

Airfoil design

Classical Symmetrical Airfoil

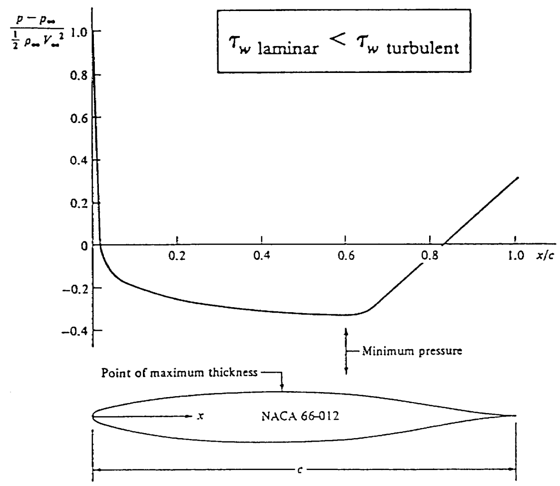
- NACA 0012



Airfoil design

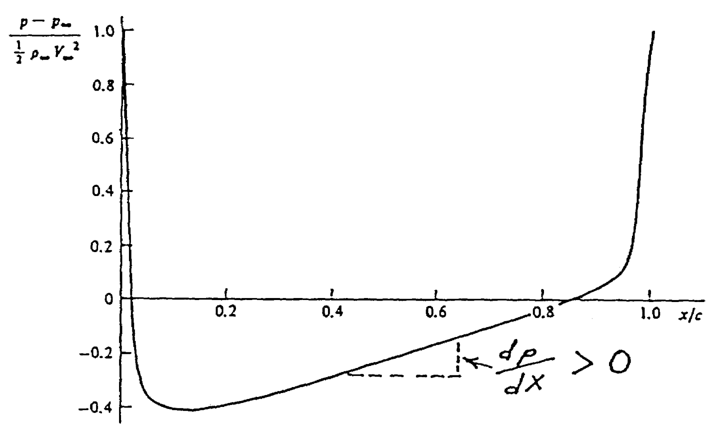
Laminar Flow Airfoil

- NACA 66-012



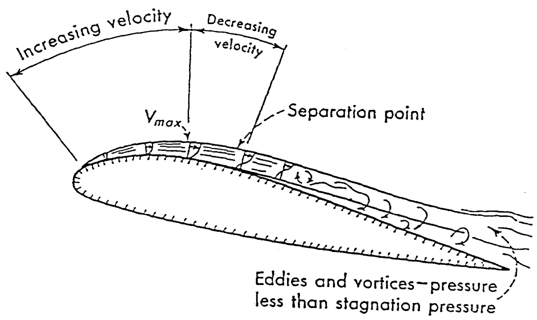
Flow separation

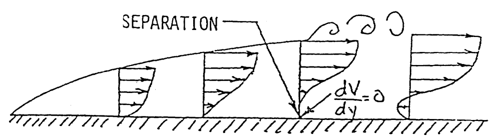
- "Adverse pressure gradient"



Flow separation

- Separation occurs when $\left(\frac{dV}{dy}\right)_{surface} = 0$





Flow separation

- **Major effects of flow separation**
  - Stall or departure
  - Loss of lift
  - Large increase in drag
- **Consequences**
  - Buffet
  - Wing rock
  - Degraded performance
  - Stall
  - Departure
  - Loss of control
  - Crash often ensues

DRAG → PRESSURE D. (due to the net imbalance of surface pressure acting in the drag direction)

DRAG → FRICTION D. (due to the net effect of shear stress acting in the drag direction)

SUBSONIC Viscous drag effects

- Profile drag makeup

C_d = PROFILE DRAG $= c_f + c_{d,p}$

- SKIN FRICTION DRAG — due to shear stress
- PRESSURE DRAG — due to flow separation

- Design compromises
  - Laminar airfoils
    - Lower drag
    - Nasty stall characteristics
  - Boundary layer control
    - Lower stall speeds
    - Mechanical complexity

THIN AIRFOILS

"EXACT" $c_f = \frac{1.328}{\sqrt{Re}}$ Laminar

$c_f = D_f / q_\infty S$

$Re = \rho_\infty V_\infty c / \mu_\infty$

D_f friction drag measured on one side of the flat plate

S planform area plate

AIRFOIL – TURBULENT FLOW

c_f flat plate $\quad c_f = \frac{0.42}{\ln^2(0.056\,Re)}$ $\quad$ flow completely turbulent

$c_f^{-1/2} = 4.13 \log(Re\, c_f)$

laminar – turbulent

↓

$x_{tr} \rightarrow Re_{tr} = \frac{\rho_\infty V_\infty x_{tr}}{\mu_\infty} \approx 350{,}000 \div 1{,}000{,}000$

Since Re high for most conventional airplanes in subsonic wing ⇒ we can assume that the flow is turbulent from the leading edge of the wing

$c_{d,p}$ via experiment (CFD does not provide very good results)

FINITE WINGS

Induced drag coefficient (purely pressure drag) due to wing-tip vortices

- Substituting into the approximation

$$D_i \approx L\alpha_i = L\frac{C_L}{\pi AR}$$

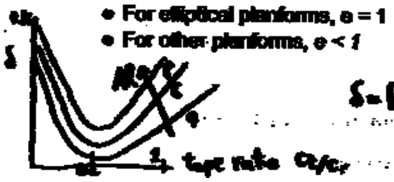
  - But since $L = q_\infty S C_L$ $D_i = q_\infty S \frac{C_L^2}{\pi AR}$ OR $\frac{D_i}{q_\infty S} = \frac{C_L^2}{\pi AR}$
  - Following our usual pattern, we define

$$C_{D,i} \equiv \frac{D_i}{q_\infty S} = \frac{C_L^2}{\pi AR}$$

  - To account for wings which do not produce the ideal elliptical spanwise lift distribution, we introduce a span efficiency factor, e

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

    - For elliptical planforms, e = 1
    - For other planforms, e < 1



$e = \frac{1}{1+\delta}$ $\delta \geq 0$

$\delta = f(AR, \lambda)$

How to reduce $C_{D,i}$?

FUSELAGE

$C_D \neq C_{D_{wing}} + C_{D_{fus}} + C_{D_{interf.}}$

>0

Summary:

Skin friction }
Pressure drag } Profile drag }
due to separation (form drag) }
Interf. drag }

parasite drag: for a complete airplane
portion of total drag { skin frict. + pressure drag due to separat. + interfer. } all airplane

Induced drag

Zero lift drag (for full airplane) $C_L = 0$ ⇒ parasite drag

Drag due to lift (" " ") ⇒ parasite drag $C_L \neq 0$ + induced drag

Transonic Flow Speed Regime

Transonic Area Rule
To reduce the transonic drag

Swept wing and slender body reduce the wave drag

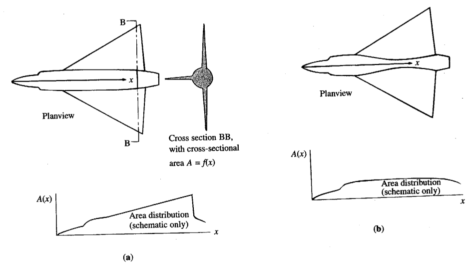


Figure 2.43 Schematics of **(a)** a non-area-ruled aircraft and **(b)** an area-ruled aircraft.

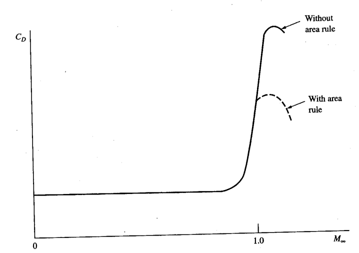


Figure 2.44 A schematic of the drag-rise properties of area-ruled and non-area-ruled aircraft.

Whitcomb, 1955

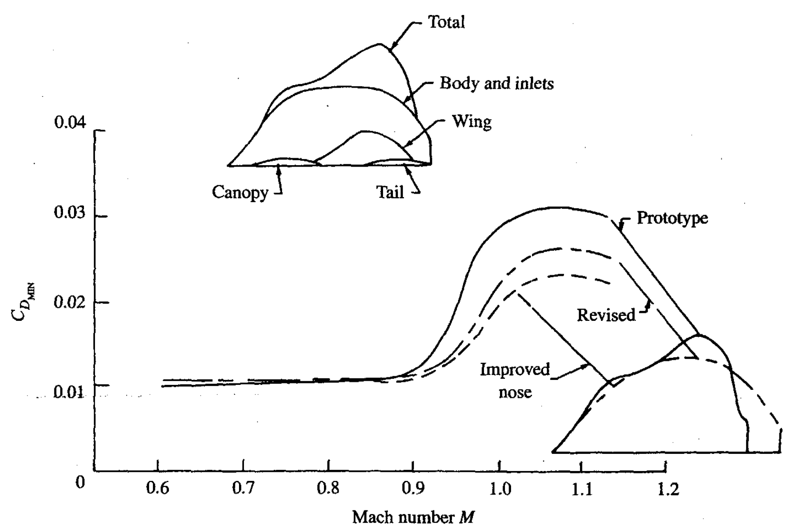


Figure 2.45 Minimum drag coefficient as a function of Mach number for the F-102; comparison of cases with and without area rule. (After Loftin, Ref. 13.)

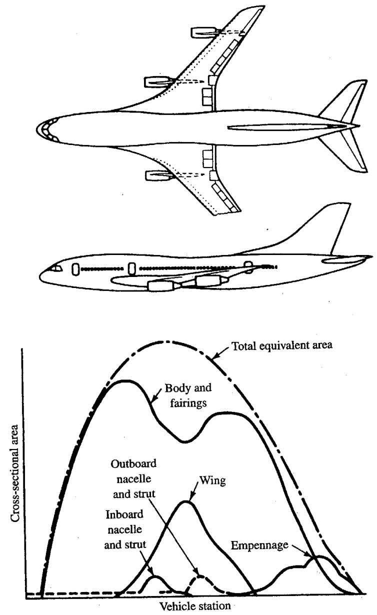
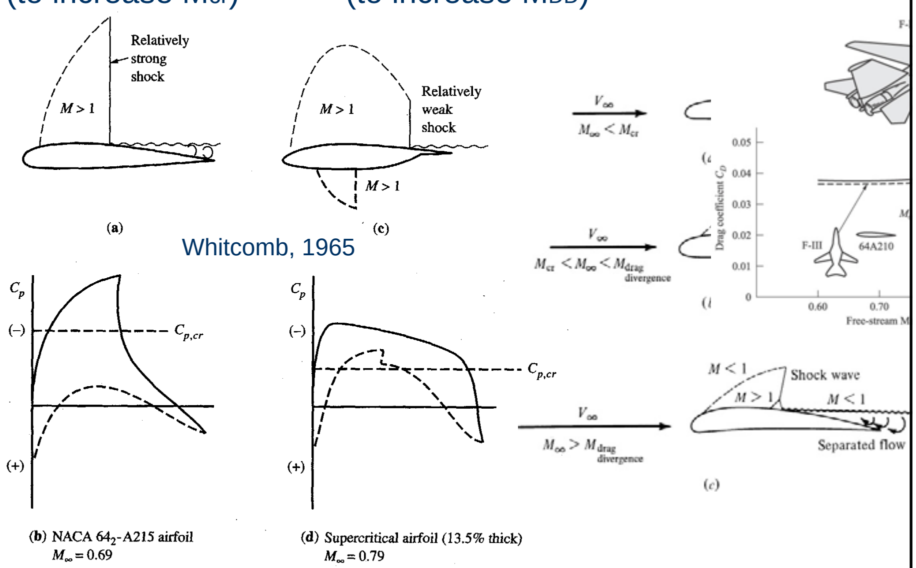


Figure 2.46 Cross-sectional area distribution breakdown for a typical, generic high-speed subsonic transport. (After Goodmanson and Gratzer, Ref. 28.)

Laminar Airfoil vs. Supercritical Airfoil
(to increase M_{cr}) (to increase M_{DD})

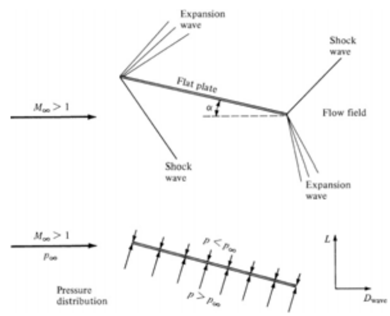


Whitcomb, 1965

Figure 2.47 Standard NACA 64-series airfoil compared with a supercritical airfoil at cruise lift conditions.

Wave Drag

$$c_l = \frac{4\alpha}{\sqrt{M_\infty^2 - 1}}$$

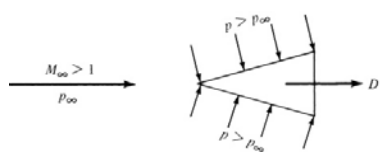


$$c_{d,w} = \frac{4\alpha^2}{\sqrt{M_\infty^2 - 1}}$$

Wave drag due to lift

- "Supersonic Drag"

  = Zero-lift wave drag + wave drag due to lift



Total drag coefficient

- Adding the induced drag to the profile drag gives total drag

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

  - This expression gives the total drag coefficient for a finite (3d) wing at subsonic speeds
  - c_d is the sum of c_{df} (skin friction) and c_{dp} (profile)
  - Also note:
    - If C_D is plotted versus C_L, using the equation above, the resulting curve is a parabola
    - Such a plot is often referred to as the drag polar for an airplane

Lift, drag, and pitching moment coefficients

- Infinite wing (unit span) coefficients

$$c_l \equiv \frac{L}{qS} \quad \text{where} \quad : \quad L = \text{WING LIFT PER UNIT SPAN}$$

$$c_d = \frac{D}{qS} \quad \text{where} \quad : \quad L = \text{WING DRAG PER UNIT SPAN}$$

$$c_m = \frac{M}{qSc} \quad \text{where} \quad : \quad M = \text{WING PITCHING MOMENT ABOUT THE AERO} - \text{DYNAMIC CENTER UNIT PER UNIT SPAN}$$

- Finite wing coefficients

$$C_{L_W} \equiv \frac{L_W}{qS} \quad \text{where} \quad : \quad L_W = \text{WING LIFT FOR A FINITE WING}$$

$$C_{D_W} = \frac{D_W}{qS} \quad \text{where} \quad : \quad D_W = \text{WING DRAG FOR A FINITE WING}$$

$$C_{M_W} = \frac{M_W}{qSc} \quad \text{where} \quad : \quad M_W = \text{PITCHING MOMENT ABOUT THE AERODYNAMIC CENTER}$$

Lift, drag, and pitching moment coefficients

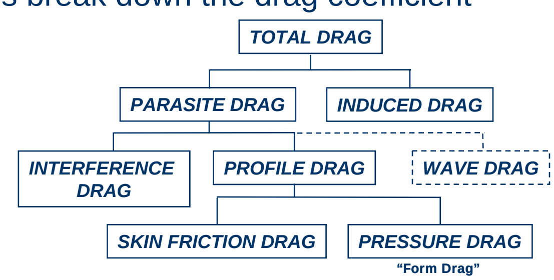
- Complete airplane coefficients

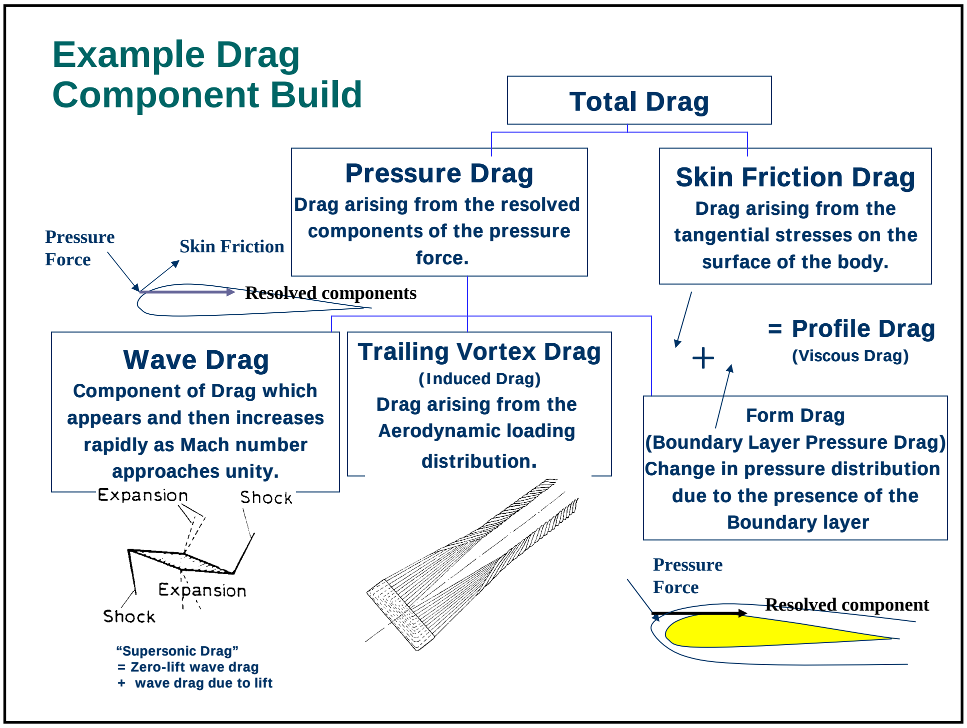
$$C_L \equiv \frac{L}{qS} \quad \text{where} \quad : \quad L = \text{COMPLETE AIRPLANE LIFT}$$

$$C_D = \frac{D}{qS} \quad \text{where} \quad : \quad D = \text{COMPLETE AIRPLANE DRAG}$$

$$C_M = \frac{M}{qSc} \quad \text{where} \quad : \quad M = \text{COMPLETE AIRPLANE PITCHING MOMENT ABOUT THE AIRPLANE'S AERODYNAMIC CENTER}$$

- Now, let's break down the drag coefficient





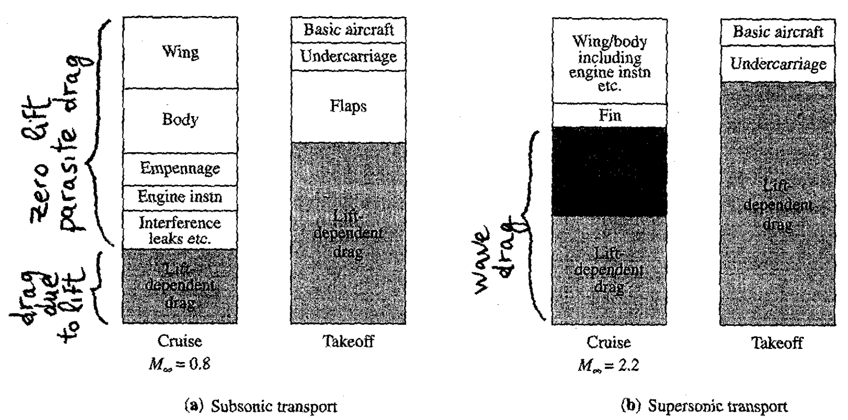


Figure 2.53 Comparison of cruise and takeoff drag breakdowns for (a) a generic subsonic transport and (b) a generic supersonic transport.

Parasite drag definition

- consider parasite drag
  - c_{D_e} parasite drag coefficient for the airplane
    - c_{D_e} is the sum of wing profile drag, tail friction and pressure drag, friction and pressure drag for all other components
    - since c_{D_e} accounts for drag due to separated flow, it will vary with angle of attack
    - at transonic and supersonic speeds, it includes wave drag
  - defining a new term: c_{D0} parasite drag coefficient at zero lift for the complete airplane

$$C_{D_e} = C_{D_0} + rC_L^2$$

    - where r is an empirically determined constant
    - does not always include wave drag (for now, at least, we will keep wave drag separate)

Drag polar

- Total aircraft drag

$$C_D = C_{D_e} + \frac{C_L^2}{\pi e_1 AR}$$

  – substituting for c_{D_e} the equivalent

$$C_{D_e} = C_{D_0} + rC_L^2$$

  – Then, total drag becomes

$$C_D = C_{D_0} + \left(r + \frac{1}{\pi e_1 AR} \right) C_L^2$$

  – Defining a new efficiency factor (Oswald's efficiency factor) such that

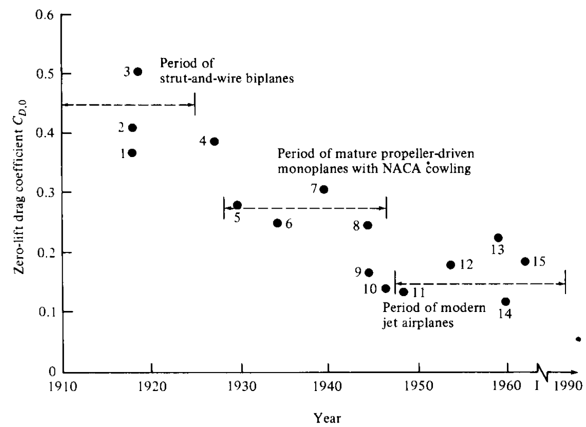
$$\left(r + \frac{1}{\pi e_1 AR} \right) = \frac{1}{\pi e AR}$$

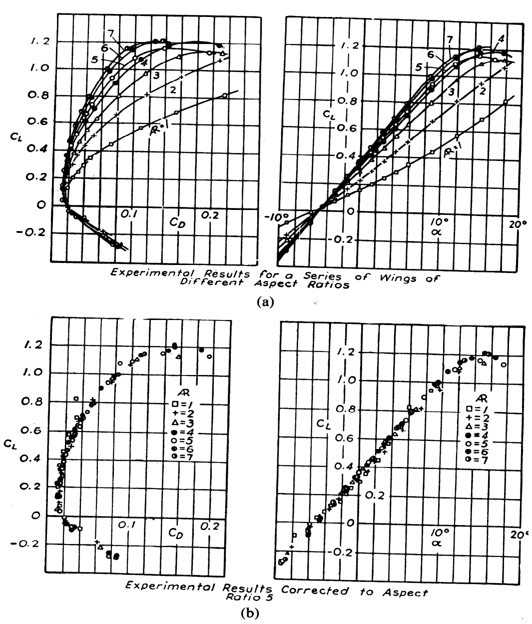
  – Finally, we can write the parabolic form of the total drag equation for a complete airplane

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

Where we have been

- The figure below suggests how drag has been reduced in 65-70 years





(a)

(b)

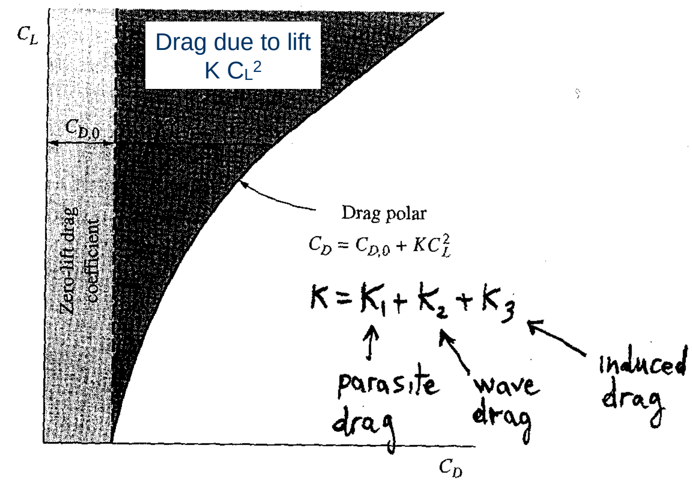
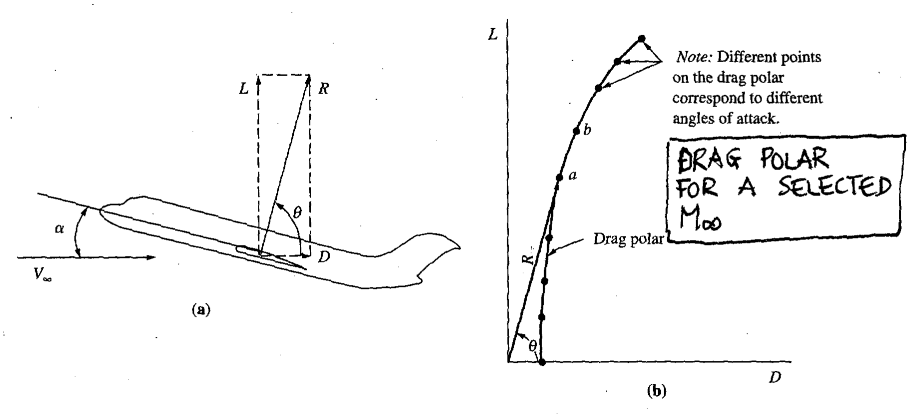


Figure 2.56 Schematic of the components of the drag polar.



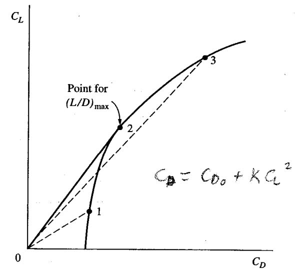
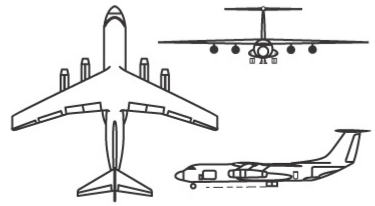
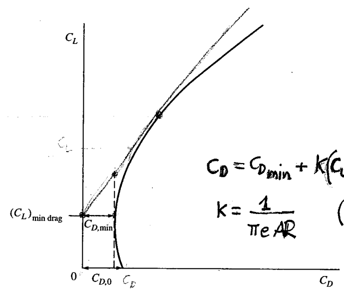


Figure 2.58 Construction for maximum lift-to-drag point on a drag polar.

$$C_D = C_{D,e} + C_{D,w} + \frac{C_L^2}{\pi e AR}$$

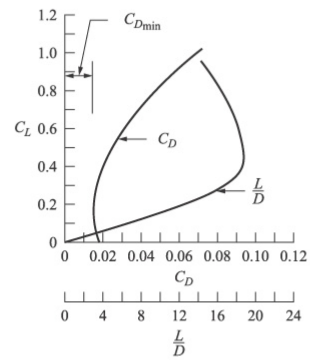




$$C_D = C_{D_{min}} + K\left(C_L - C_{L_{min\,drag}}\right)^2$$

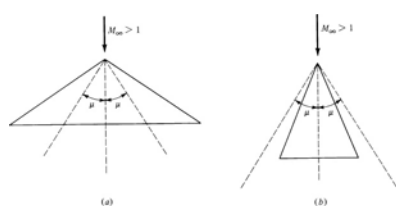
$$K = \frac{1}{\pi e AR} \qquad \text{if: } (K_1 = K_2 = 0)$$

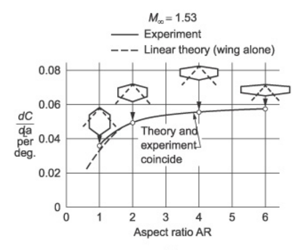
Figure 2.59 Illustration of minimum drag and drag at zero lift.

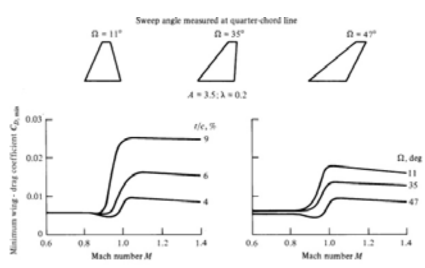


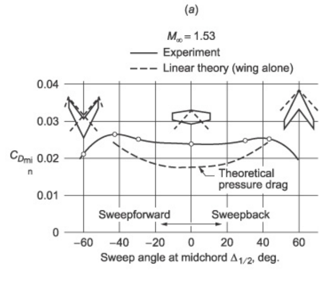
M = 0.4

Effect of the Sweep Angle on Drag

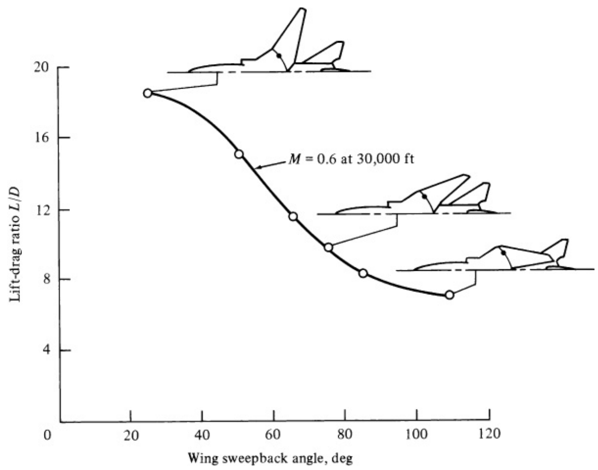








L/D: Effect of sweep angle



Viscous drag effects

- Profile drag makeup

PROFILE DRAG

SKIN FRICTION DRAG | **PRESSURE DRAG**

- Design compromises
  - Laminar airfoils
    - Lower drag
    - Nasty stall characteristics
  - Boundary layer control
    - Lower stall speeds
    - Mechanical complexity

Induced drag coefficient

- Substituting into the approximation

$$D_i \approx L\alpha_i = L\frac{C_L}{\pi AR}$$

  - But since $L = q_\infty S C_L$ $\quad D_i = q_\infty S\frac{C_L^2}{\pi AR}$ $\quad OR \quad \frac{D_i}{q_\infty S} = \frac{C_L^2}{\pi AR}$
  - Following our usual pattern, we define

$$C_{D,i} \equiv \frac{D_i}{q_\infty S} = \frac{C_L^2}{\pi AR}$$

  - To account for wings which do not produce the ideal elliptical spanwise lift distribution, we introduce a span efficiency factor, e

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

    - For elliptical planforms, e = 1
    - For other planforms, $e < 1$

Finite Wing Corrections

- Drag coefficient due to angle of attack is corrected
  - *AR* is the aspect ratio of the wing
  - *e* is the Oswald Efficiency Factor
  - $C_{d,i}$ is the induced drag coefficient

$$C_D = C_d + C_{d,i}$$

$$C_D = C_d + \frac{C_L^2}{\pi eAR}$$