Aircraft Design

(APRI0004-1)

Lecture 4

Aerodynamics

2015-2016

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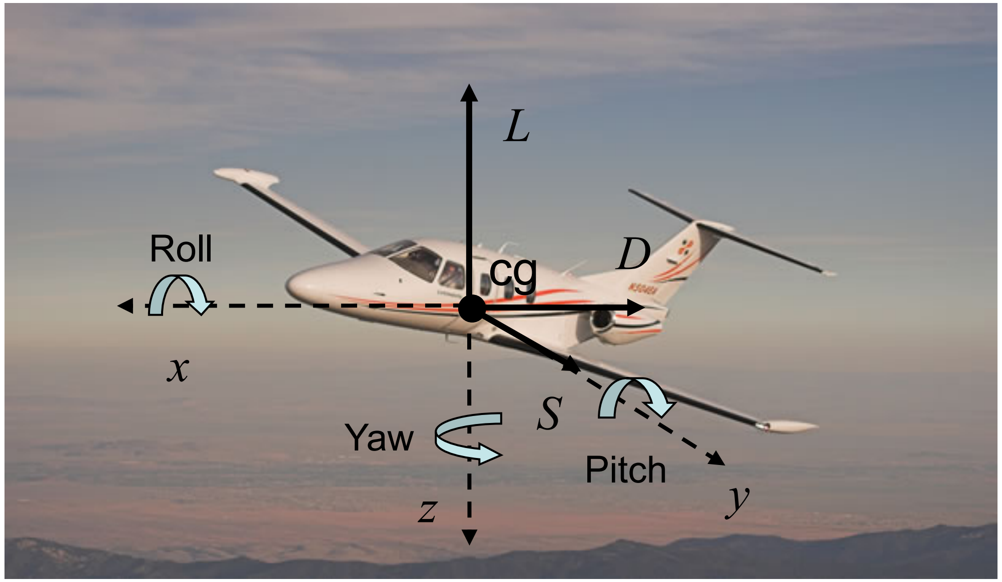
Introduction

- Aerodynamics = study of the loads exerted by the flow of air over a solid structure
- Applications: aircraft, wind turbine, anything that is in air

- Six loads:
  - 3 forces:
    - Lift
    - Drag
    - Sideforce
  - 3moments:
    - Pitch
    - Roll
    - Yaw



Why study aerodynamics?

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Anything can fly, as long as you put a big enough rocket engine under it.

But !

- That's the most expensive and dangerous solution
- There are still stability, control and other problems that can only be resolved through a good aerodynamic study.

→ The objective of this lecture is to give you the big picture of aircraft aerodynamics

→ The course AERO001-1 Aerodynamics (spring 2016) will present the origin of the results presented today

Air vehicles

- Airship
  - A hydrostatic force provides lift
  - Motor(s) provide forward acceleration
- Airplane
  - A lifting surface (wing) provides lift
  - Motor(s) provide forward acceleration
- Helicopter
  - A rotor provides lift
  - The same rotor rotor provides forward acceleration
- Autogyro
  - A rotor provides lift
  - Another rotor provides forward acceleration

More air vehicles

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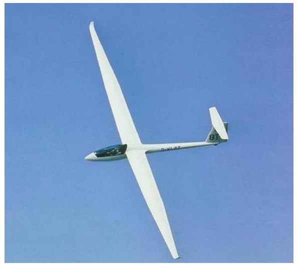
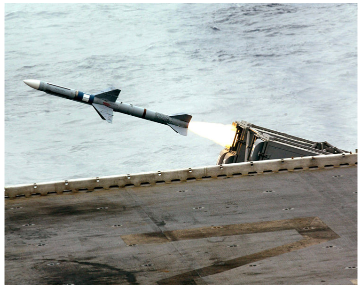
- Glider
  - A lifting surface (wing) provides lift
  - There is no forward acceleration
- Missile
  - Several small lifting surfaces provide lift
  - A motor provides forward acceleration
- Hot air balloon
  - A hydrostatic force provides lift
  - There is no forward acceleration
- Lifting body
  - A lifting body provides lift
  - A motor provides forward acceleration (optional)





Airplane

<u>Classical airplane configuration:</u>

wing + **fuselage** + **tail (**also called ‘Tube and wings’)

→This configuration **mirrors birds:**

The Wright brothers were inspired by bird flight

- Each component has a distinct role:
  - The **wing** provides lift
  - The **fuselage** holds cargo, passengers etc
  - The **tail** provides stability and control

Wings

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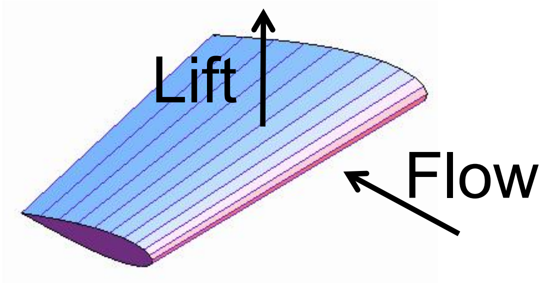
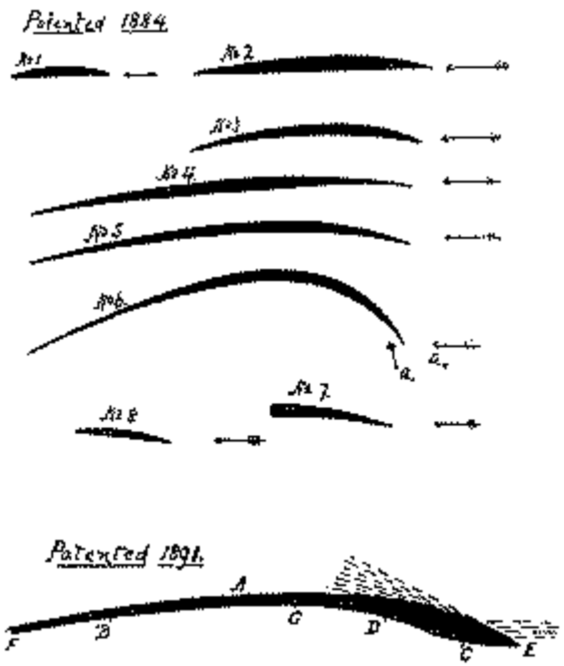
- The role of the wing is to generate lift
- Lift is perpendicular to the incoming flow

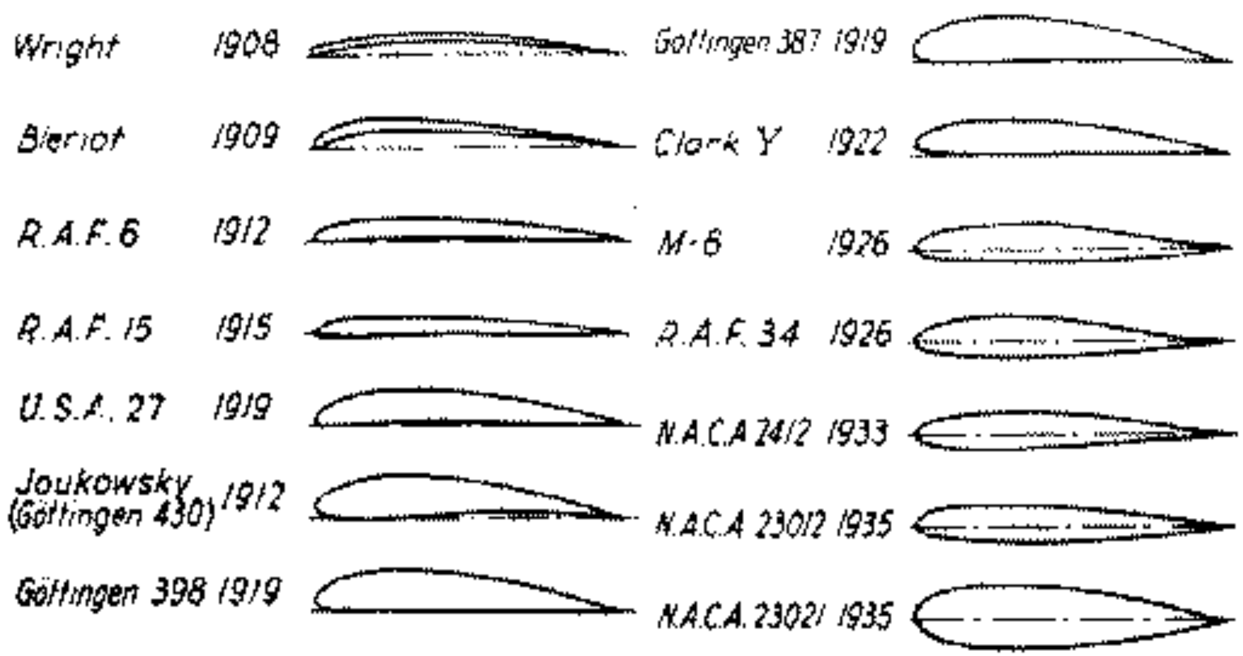


- Lift creation can be described in two ways:
  - **Pressure differential**: The air pressure on the bottom surface of the wing is higher than the air pressure of the top surface. This pressure difference creates a net force upwards
  - **Newton's third law**: The wing pushes air downwards. As a consequence, the air itself pushes the wing upwards.
- Either way, the laws of conservation apply: mass, momentum and energy.

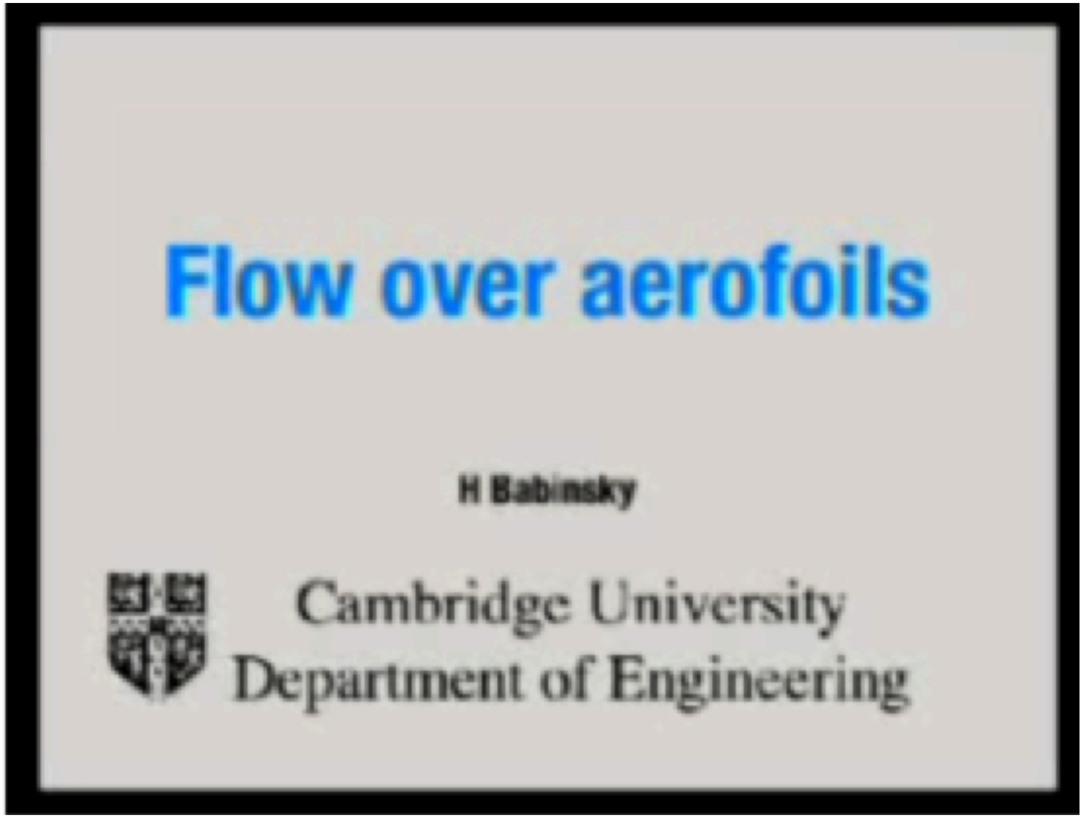
Lift generation

- Lift generation of wings depends on their cross-sectional shape : the airfoil profile
- The Wright brothers were the first to study the effects of different cross sectional shapes
- They determined that the airfoil is the optimum cross-sectional shape for a wing





Flow visualization



Pulsed jets show that the flow moves faster over the top surface.

By Bernouli's principle, faster flow speeds mean lower pressure

Hence the pressure differential causing lift

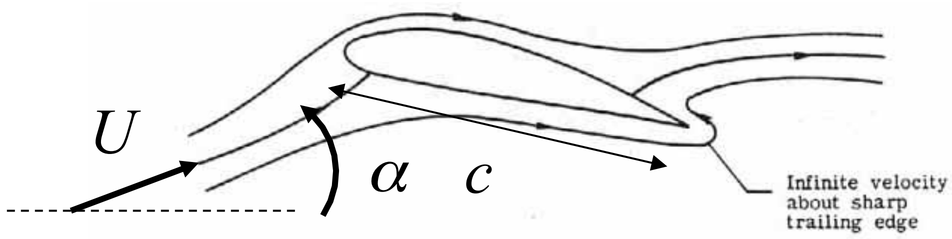
Airfoils

U=airspeed
α=angle of attack
c=chord
Γ=circulation

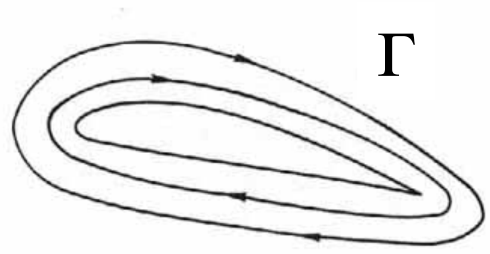
- But why is the flow accelerated on the top surface?

- Because **it must separate at the trailing edge** (physical observation)

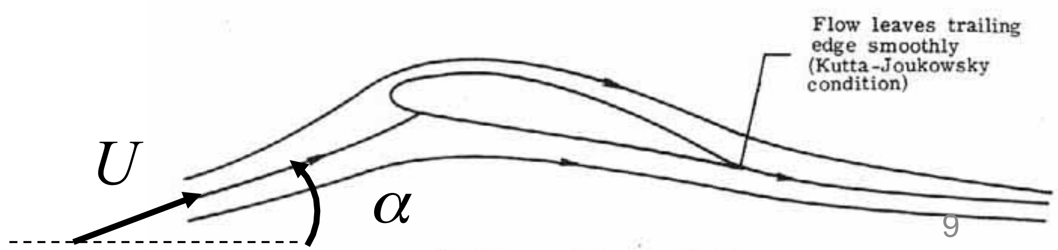
→ Aircraft can fly because of viscosity



(a) Flow with no circulation.

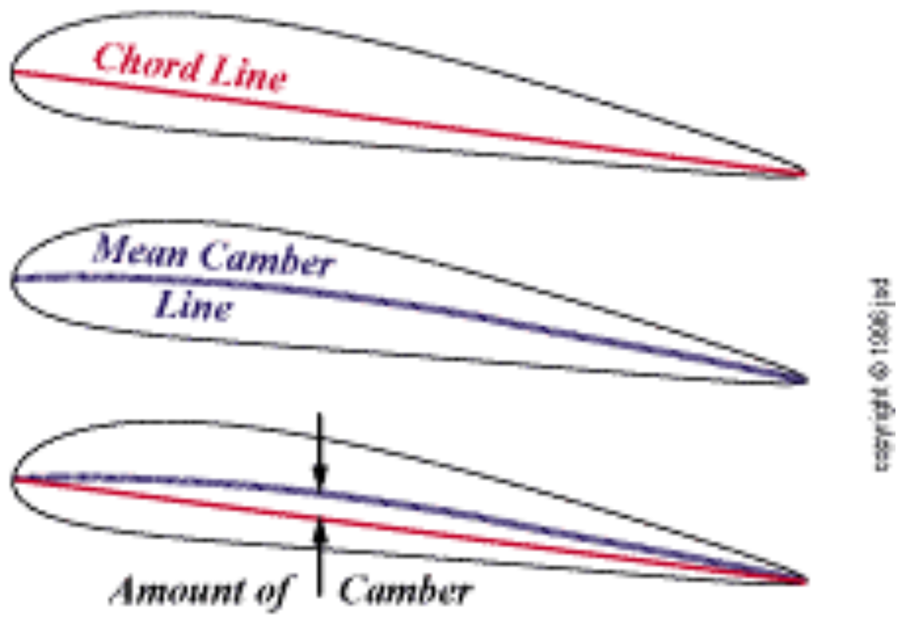


(b) Circulatory flow only.



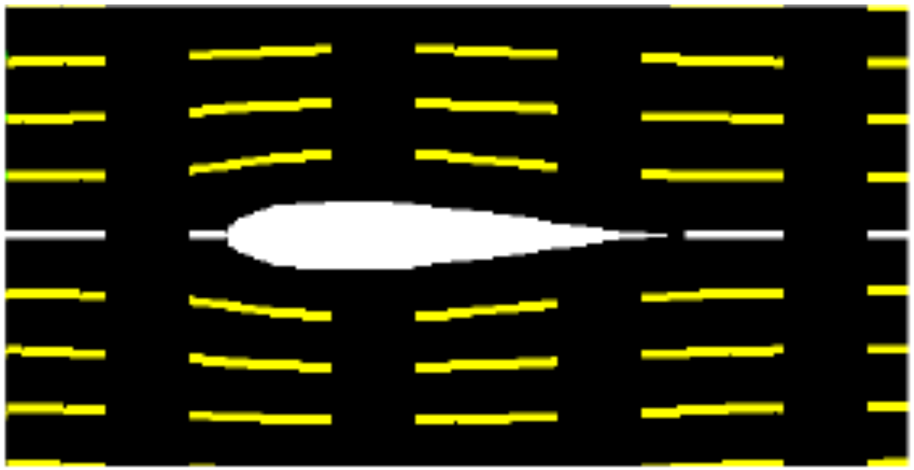
(c) Flow with circulation.

Shape effect (camber)



Camber increases the amount of lift produced by the airfoil

Symmetric airfoil - no lift at 0^{o} aoa

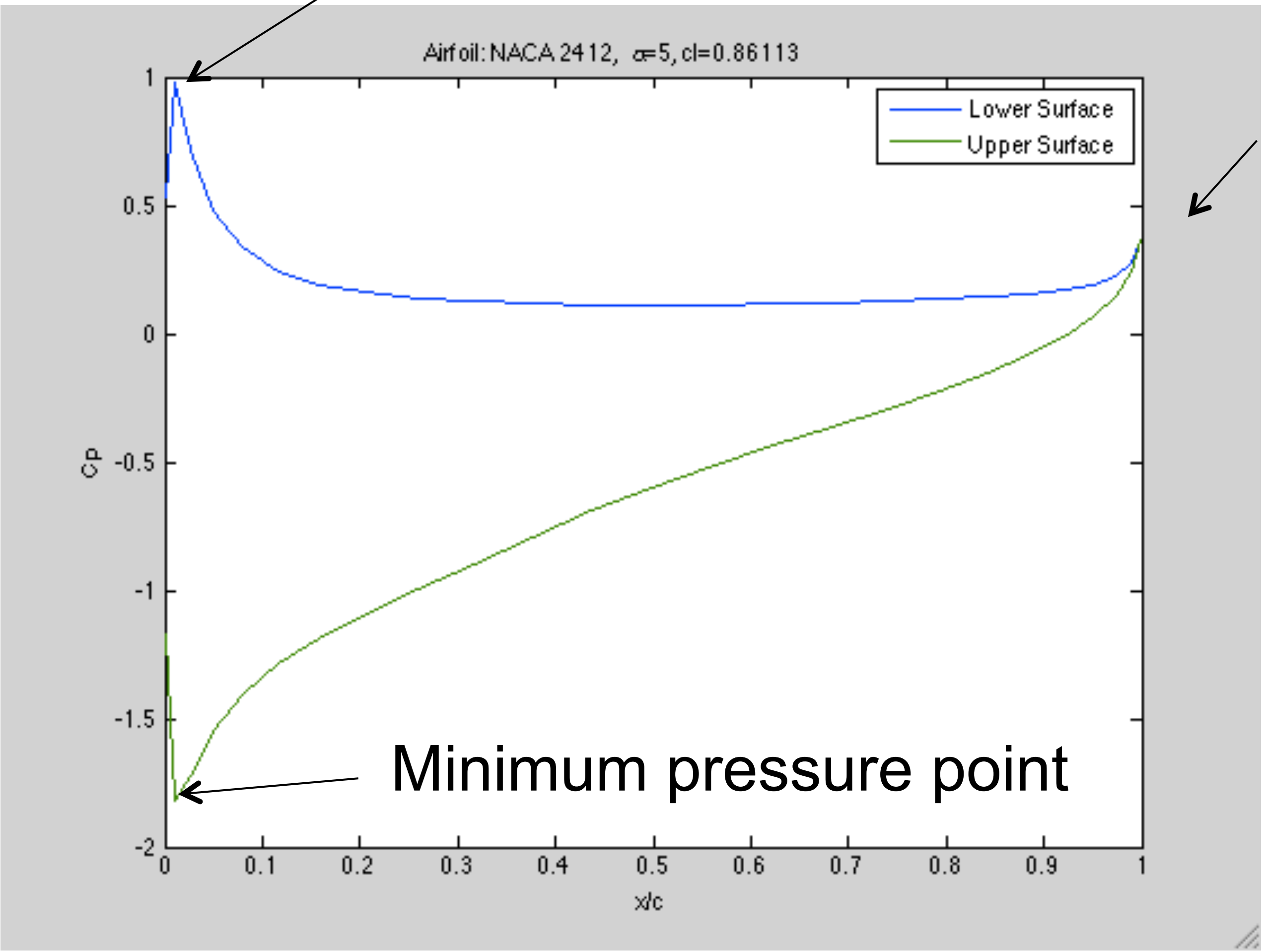


Cambered airfoil - produces lift at 0^{o} aoa

Pressure distribution

Static pressure around a lifting airfoil

L.E. Stagnation point



Minimum pressure point

T.E. Stagnation point

p = pressure
p_{ref} = reference pressure
ρ = air density
L' = lift

Pressure coefficient

$$c_p = \frac{p - p_{ref}}{1/2 \rho U^2}$$

The lift is the integral of the pressure difference over the chord:

$$L' = \int_0^c \left(p_l(x) - p_u(x)\right) \mathrm{d}x$$

Lift

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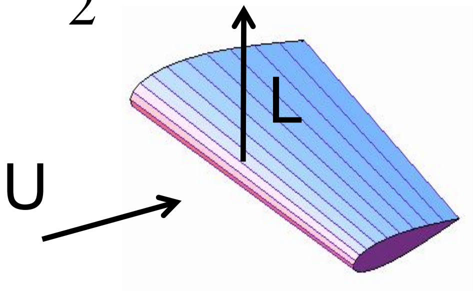
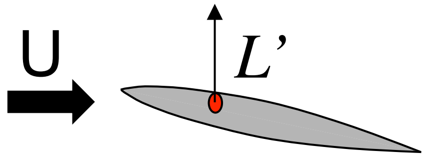
- Any aerodynamic force can be expressed as:

$$F_{AERO}[N] = \frac{1}{2}\rho U^2 S C_{AERO} \quad \text{(3D)} \longrightarrow L'[N/m] = \frac{1}{2}\rho U^2 c\, c_l \quad \text{(2D)}$$





- Kutta Joukowski : $L' = \rho U \Gamma$ (Γ = circulation)

"The amount of lift produced by an airfoil is also proportional to the total circulation required for the flow to separate at the trailing edge"

- For flat plates and small angles of attack this result simplifies to

$$L' = \pi \rho U^2 \alpha c$$

Lift coefficient

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- A lift coefficient is a non-dimensional quantity defined as

$$c_l = \frac{L'}{\frac{1}{2}\rho U^2 c}$$

- For a flat plate: $c_l = 2\pi\alpha$
  - only valid for small values of α
  - α in radian

Real lift

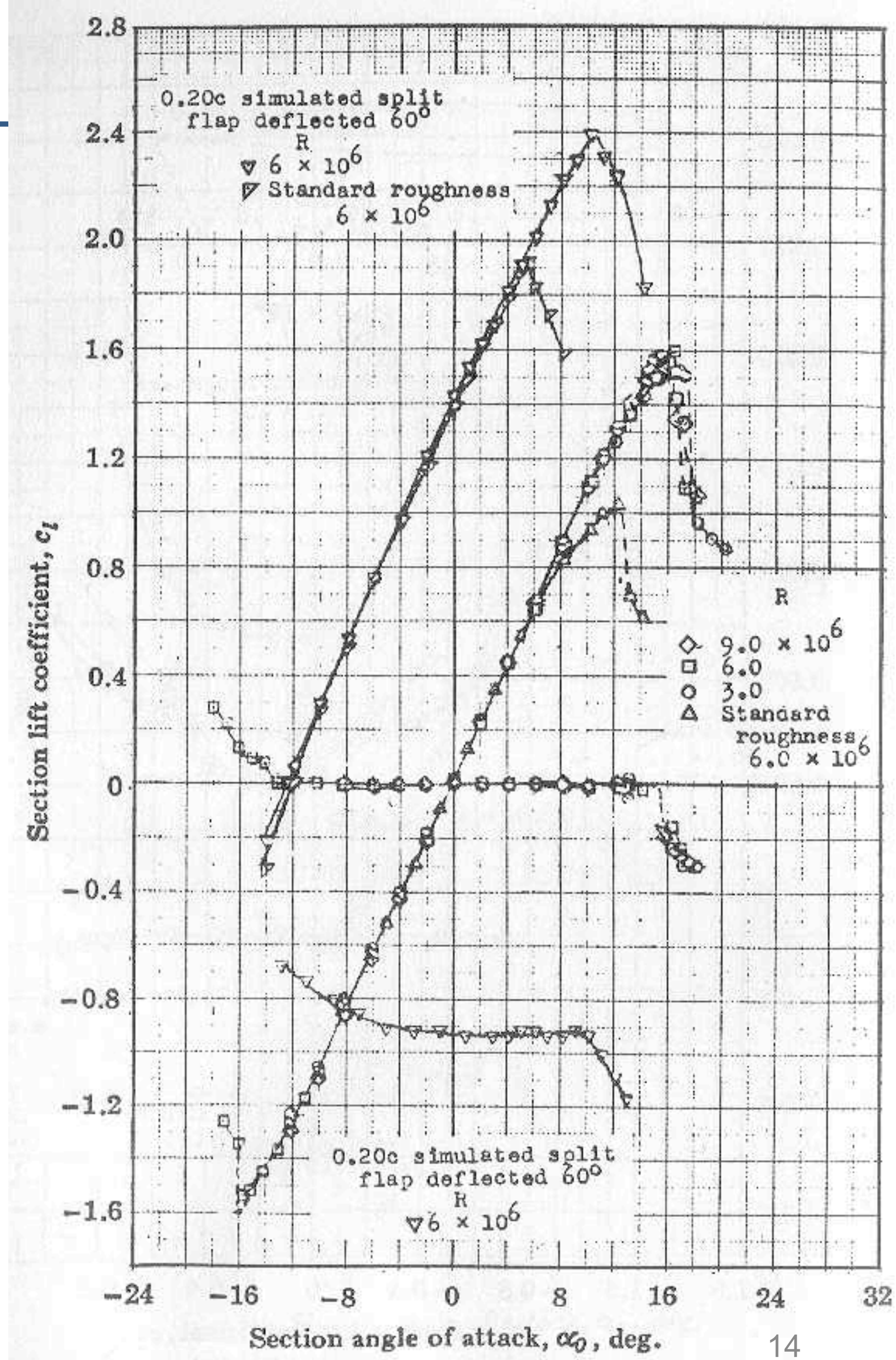
The flow cannot remain attached to the wing's surface at high angles of attack.

→ Flow separation

→ Loss of lift

<u>Important characteristics:</u>

- Maximum lift coefficient
- Maximum lift angle of attack



NACA 0012 Wing Section

Pitching moment

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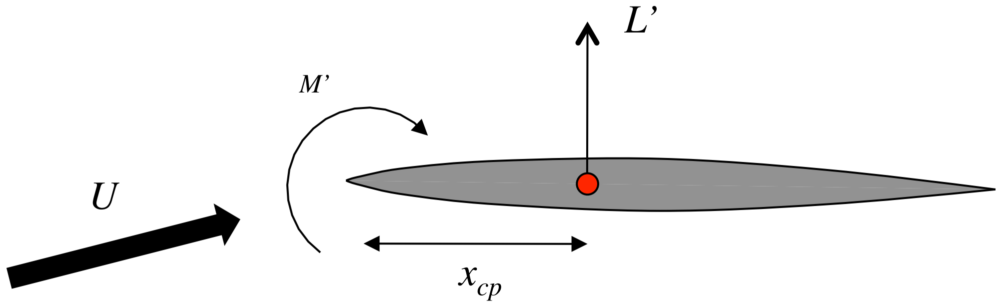


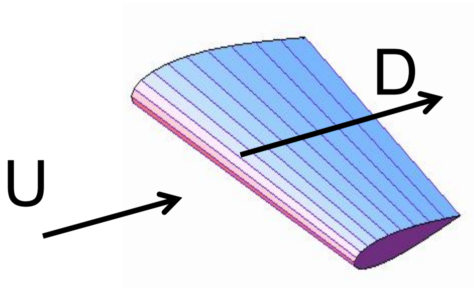
- The lift is a single force but it is caused by a continuous pressure distribution.
- Therefore, it must have a point of application. This point is called the Centre of Pressure (cp).
- The moment caused by the lift acting at the cp around the leading edge is called the **pitching moment M'**
- Pitching moment coefficient:

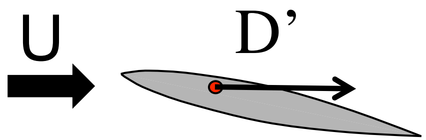
$$c_m = \frac{M'}{\frac{1}{2}\rho U^2 c^2}$$

Drag

All bodies in a real airflow are subjected to a **drag force**

Drag force parallel to the incoming flow stream



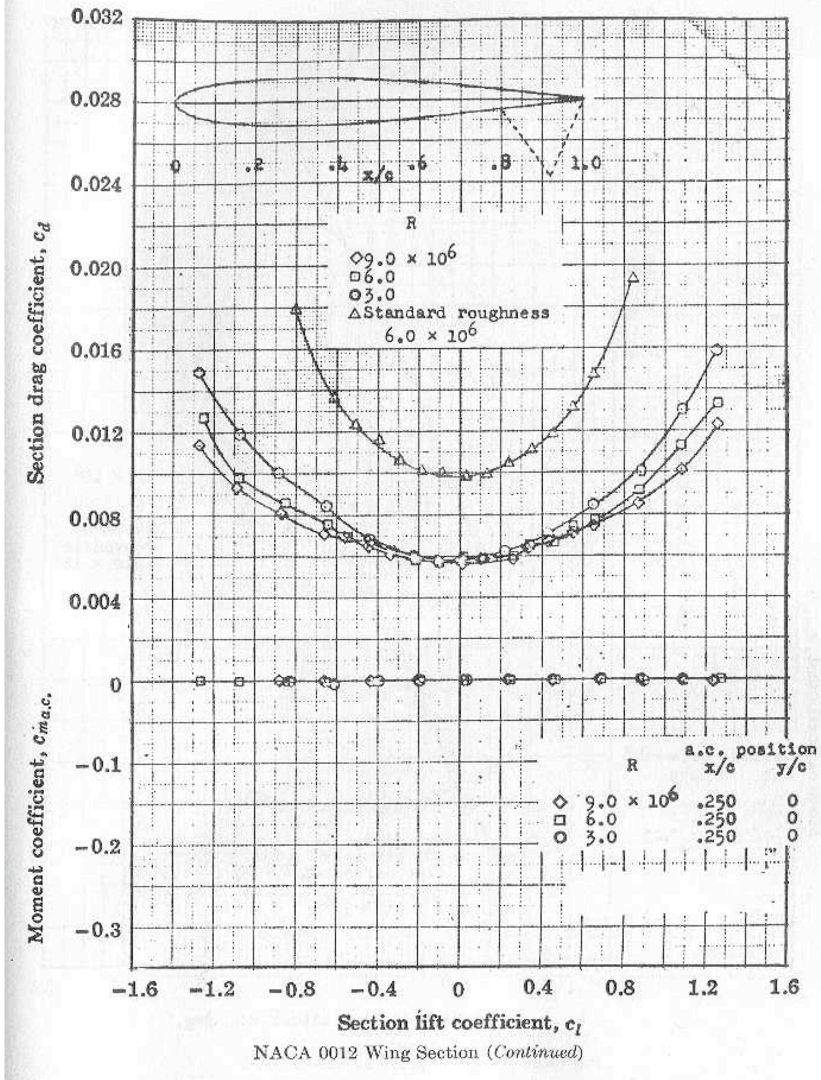


There are many sources of drag and few of them are easily modeled.

Drag: D'

Drag coefficient:

$$c_d = \frac{D'}{\frac{1}{2}\rho U^2 c}$$



Real lift and drag

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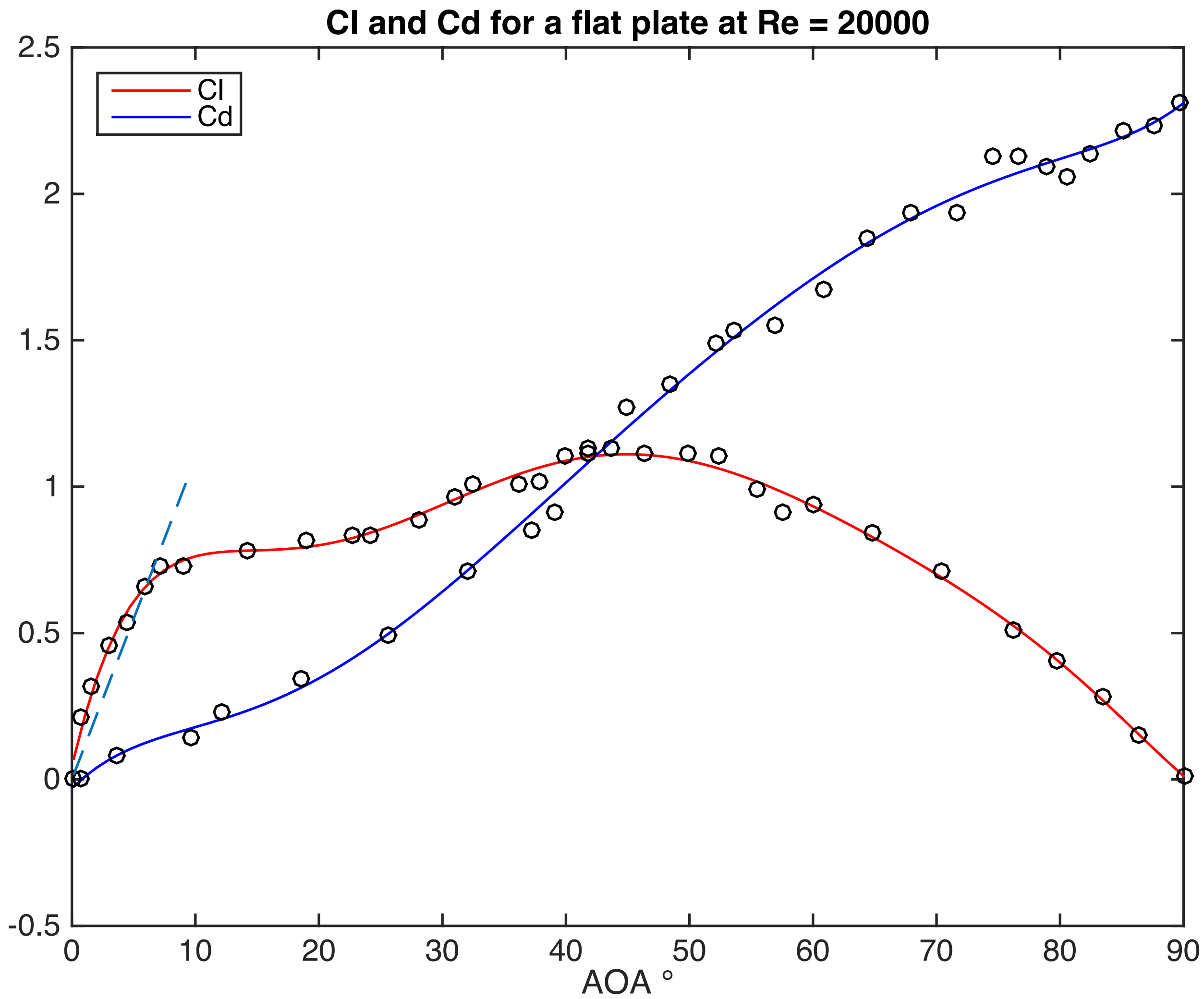
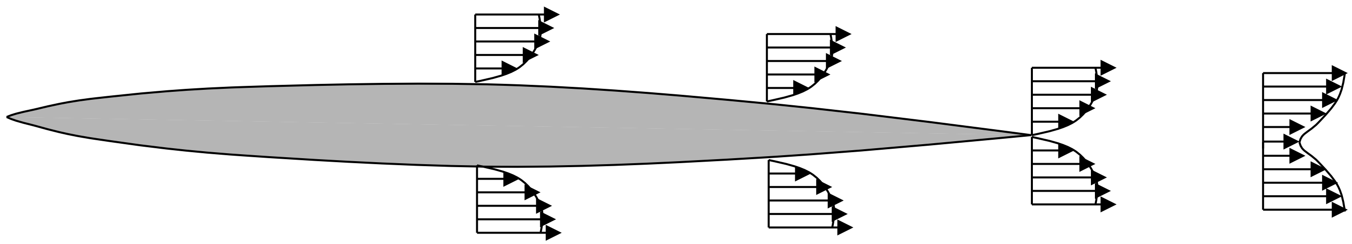
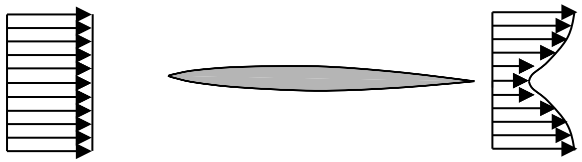
Flat plate @ Re=2 10^4



The boundary layer

- In 2D, **inviscid** theory cannot predict drag forces
- Prandtl was the first to realize that **viscous effects are very important in a thin layer of flow near the wing's surface**
- **Skin friction** and **momentum deficit** within the boundary layer are some of the major sources of drag

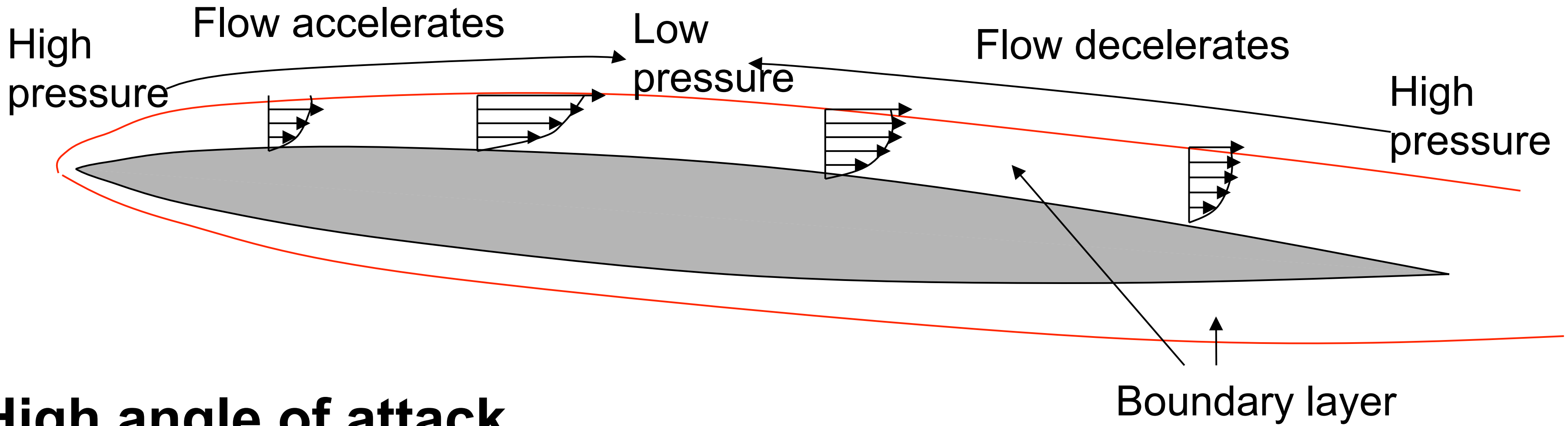




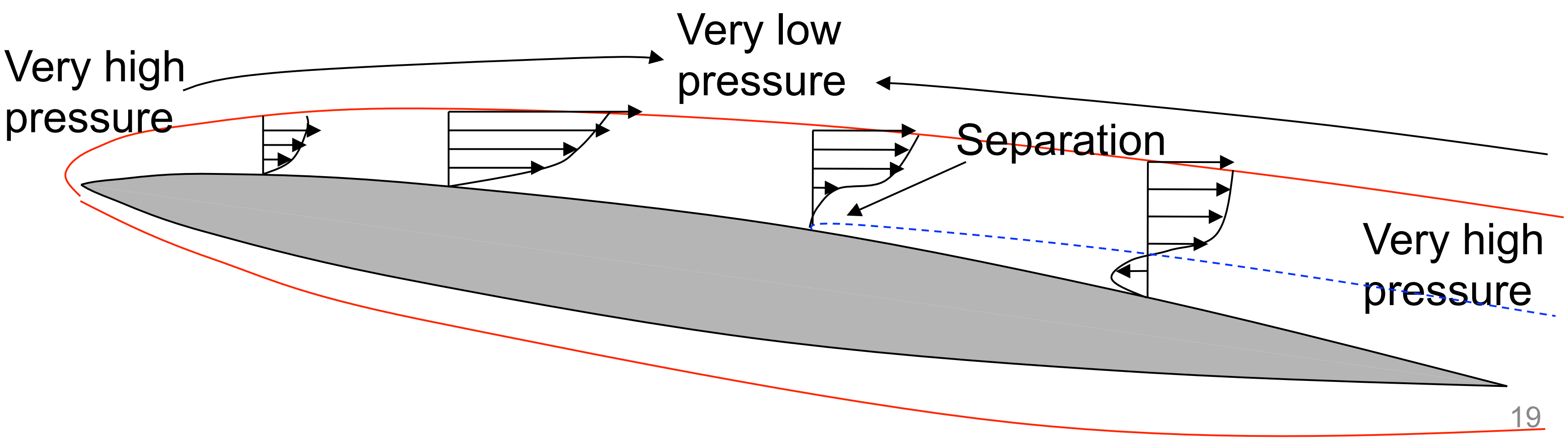
Momentum variation → Force (drag)

Flow separation

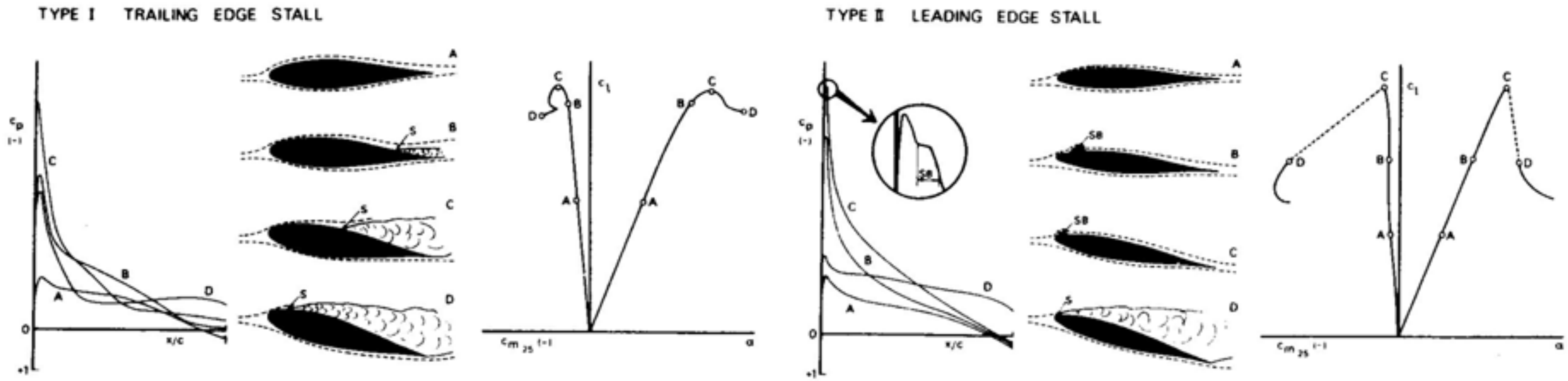
Low angle of attack

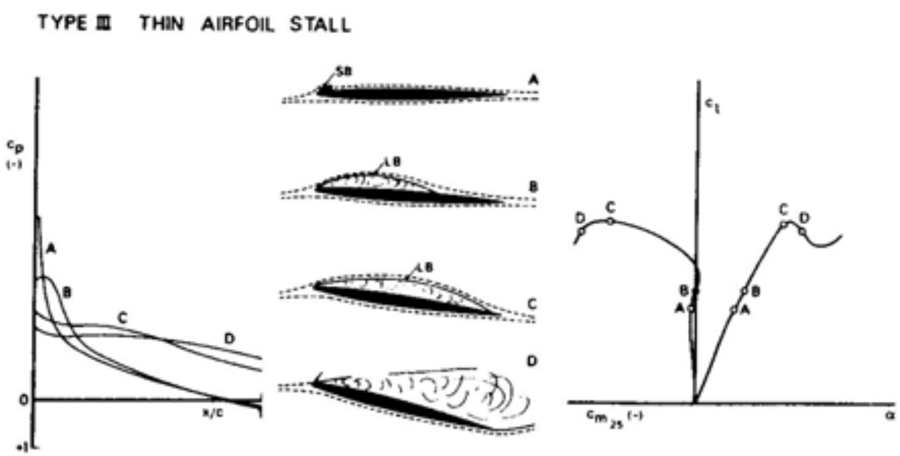


High angle of attack

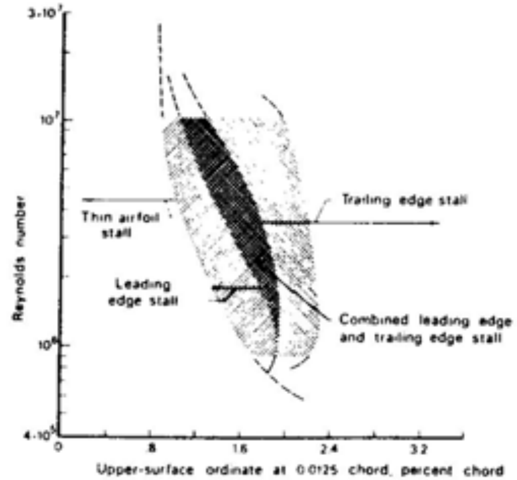


Stall





a. Upper surface pressure distributions, growth of the boundary layer and separation regions and lift and pitching moment curves.



b. Stalling characteristics correlated with Reynolds number and airfoil geometry (Ref. 7-81)

Coordinate system

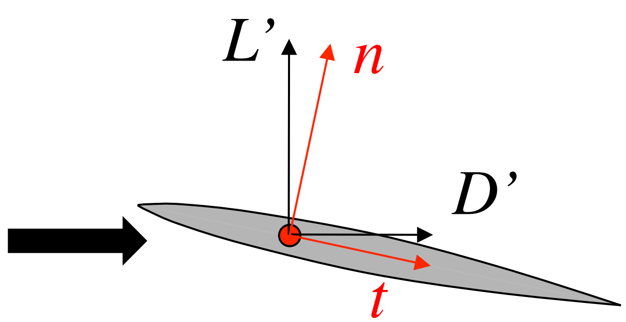
- The values of lift and drag also depend on the coordinate system
- Aeronautic definition of lift and drag:
  - Lift is a force perpendicular to the free stream.
  - Drag is a force parallel to the free stream.

→The integral of the pressure distribution around the airfoil is not the lift but the force acting normal to the surface: $n = \int_0^c \left(p_l(x) - p_u(x) \right) \mathrm{d}x$

And the 2D lift and drag forces are

$$L' = n\cos\alpha - t\sin\alpha$$

$$D' = t\cos\alpha + n\sin\alpha$$



Airfoil characteristics

Design Parameters

- Chord length, (c)
- Thickness, (t)
- Camber, ($\mathrm{d}z/\mathrm{d}x$)
- Shape (e.g. NACA 0012)

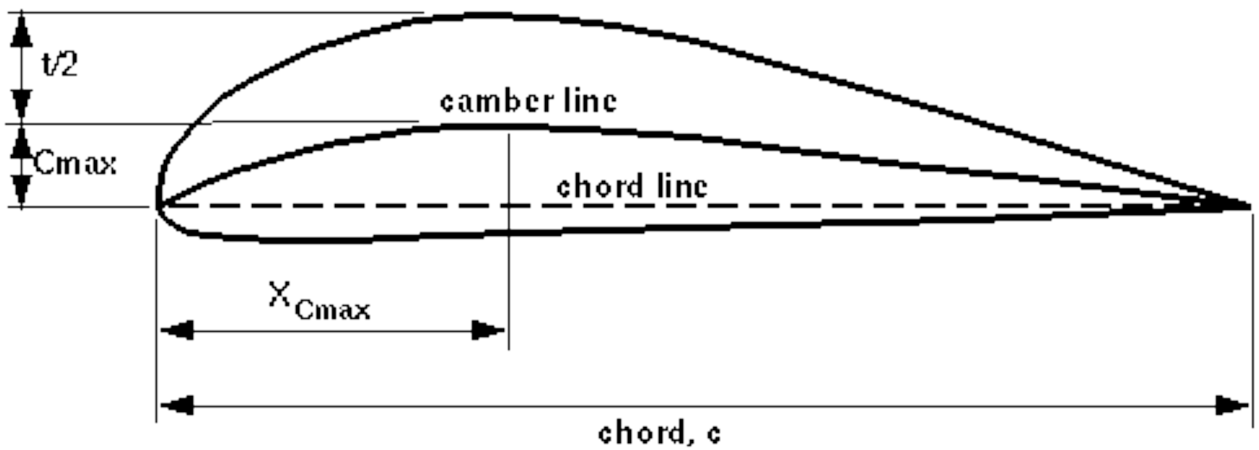
Performance Parameters

- Lift curve slope ($c_{l\alpha}$)
- Zero-lift angle (α_0)
- Max. lift angle ($\alpha_{cl\max}$)
- Max. lift coefficient ($c_{l\max}$)
- Min drag coefficient ($c_{d\min}$)

NACA 4-digit airfoils

National Advisory Committee for Aeronautics

- These airfoils were developed in the 20's and 30's based on earlier Göttingen and Clark Y sections
- They use a very specific terminology



NACA 4-digit airfoils

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NACA 2412

- 12 $= \tau \times 100$ thickness ratio (max. thickness / chord) $\tau = 0.12$
- 4 $= p \times 10$ chordwise position of maximum camber $p = 0.4$
- 2 $= \varepsilon \times 100$ maximum camber ratio (camber / chord) $\varepsilon = 0.02$

$$T(x) = 10\tau c\left[0.2969\sqrt{\frac{x}{c}} - 0.126\frac{x}{c} - 0.3537\left(\frac{x}{c}\right)^2 + 0.2843\left(\frac{x}{c}\right)^3 - 0.1015\left(\frac{x}{c}\right)^4\right]$$

$$\bar{Y}(x) = \begin{cases} \dfrac{\varepsilon x}{p^2}\left(2p - \dfrac{x}{c}\right), & 0 < \dfrac{x}{c} < p \\ \dfrac{\varepsilon(c-x)}{(1-p)^2}\left(1 + \dfrac{x}{c} - 2p\right), & p < \dfrac{x}{c} < 1 \end{cases}$$

$$x_{u,l} = x \mp T(x)/2\sin\theta$$

$$y_{u,l} = \bar{Y}(x) \pm T(x)/2\cos\theta$$

$$\theta = \tan^{-1}\left(\frac{\mathrm{d}\bar{Y}}{\mathrm{d}x}\right)$$

- Camber given by 2 parabolas and thickness by a polynomial
- Thickness perpendicular to camber
- Three parameters: τ, p, ε
- Same thickness for five-digit series but different camber (for far-forward maximum camber)
- Modified version for zero-thickness trailing edge

Wings

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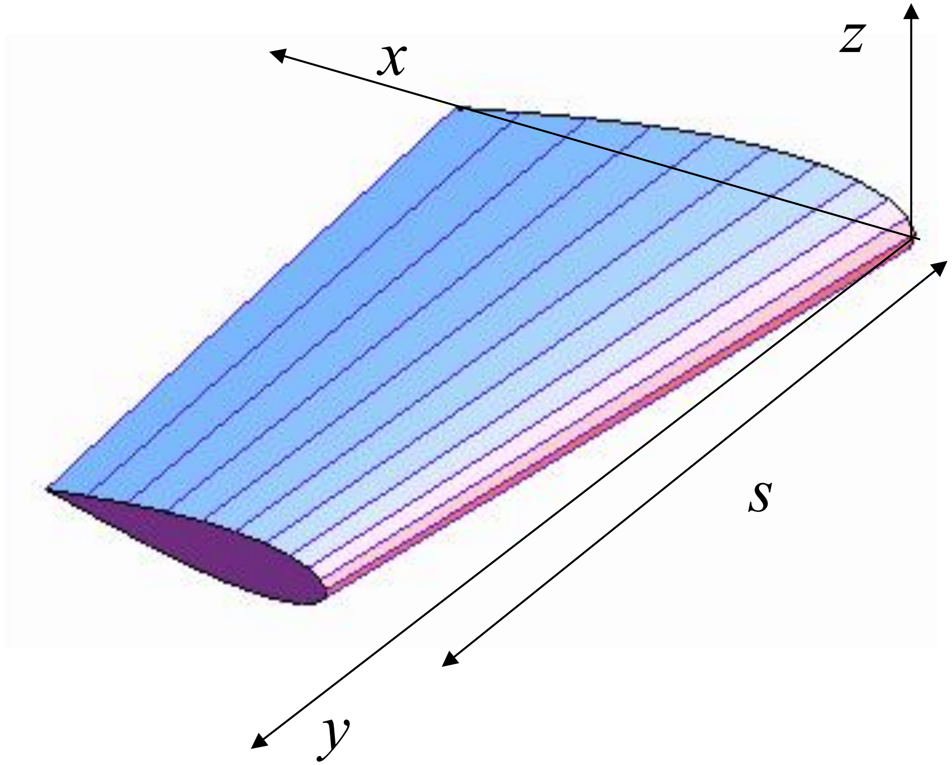
= Extrusion of an airfoil in the y-direction.

Wing span: b

Wing half-span: $s=b/2$

Wing area over full span: S

Aspect ratio: $AR=b^2/S$



2D vs 3D force coefficients

- 2D lift, drag, moment etc. coefficients:
  - Use lowercase letters: c_l, c_d, c_m, etc.
  - Use chord or chord^2 to normalize, e.g.

$$c_d = \frac{D'}{\frac{1}{2}\rho U^2 c}, \quad c_m = \frac{M'}{\frac{1}{2}\rho U^2 c^2}$$

- 3D lift, drag, moment etc. coefficients:
  - Use uppercase letters: C_L, C_D, C_M, etc.
  - Use surface area or surface*chord to normalize, e.g.

$$C_D = \frac{D}{\frac{1}{2}\rho U^2 S}, \quad C_M = \frac{M}{\frac{1}{2}\rho U^2 Sc}$$

Forces on wings

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- In principle, it is simple to calculate the aerodynamic lift acting on a wing: it is simply the integral of the sectional lift over the span:

$$L = \int_{-s}^{s} L'(y)\,\mathrm{d}y, \text{ or, } \mathrm{C}_L = \int_{-s}^{s} c_l(y)\,\mathrm{d}y, \text{ where } \mathrm{C}_L = \frac{L}{1/2\rho U^2 S}$$

- Unfortunately, the sectional lift variation is not easy to calculate

Wingtip vortices

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Wingtip vortices

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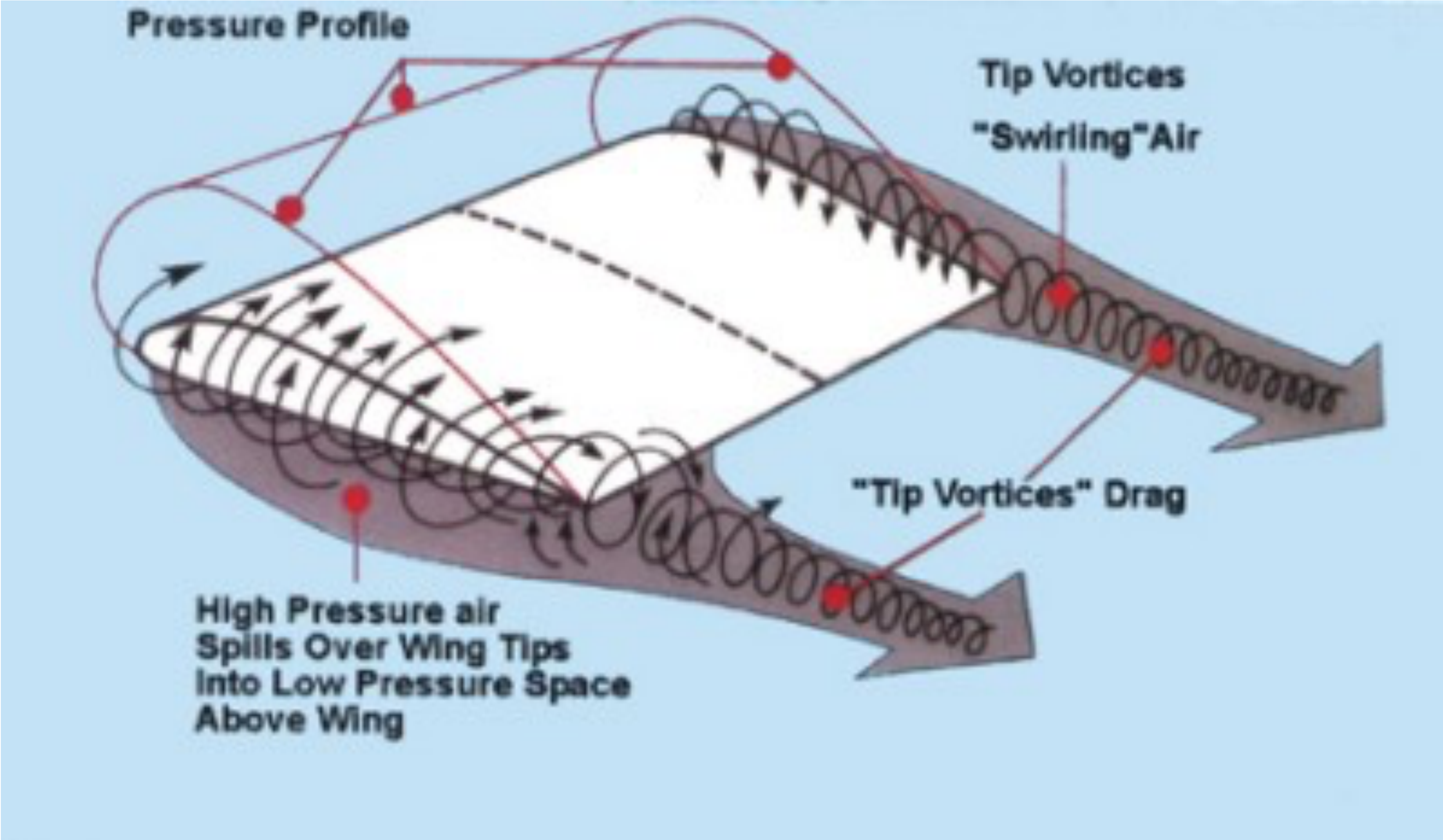
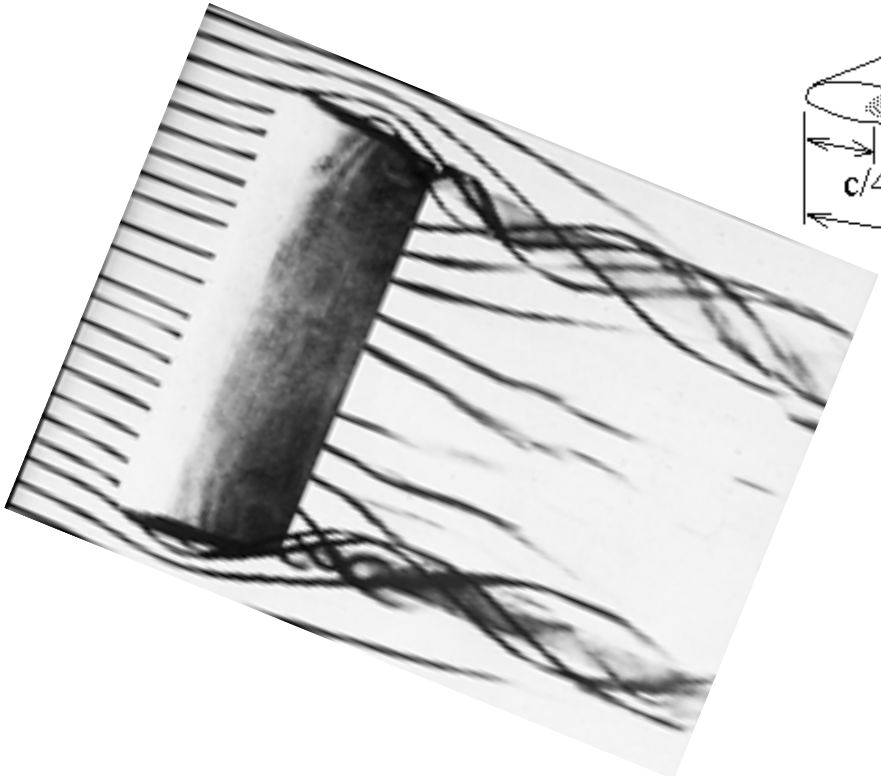
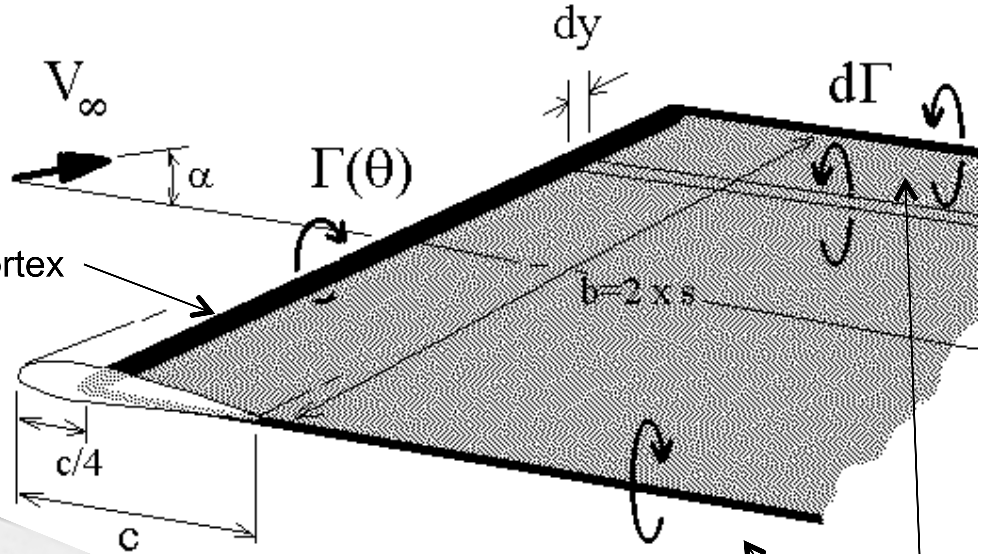


Lifting line theory

Prandt'l : modeling the wing by horseshoe vortices



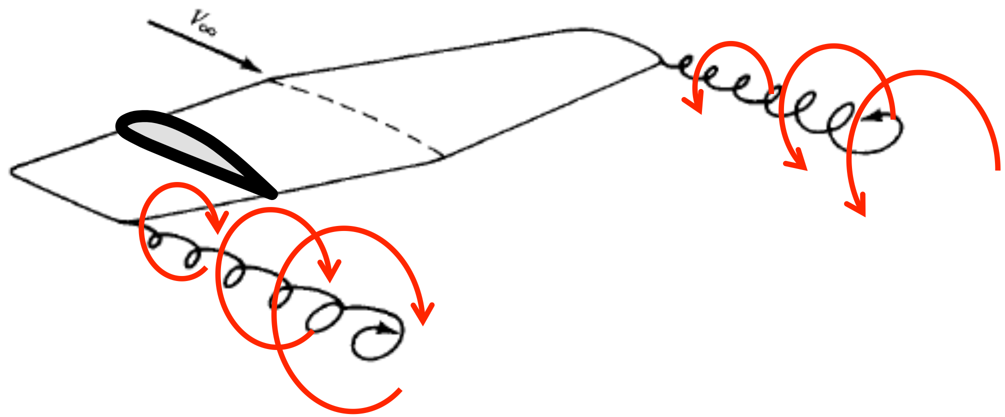


The stronger the vortex,
the higher the lift

Induced drag

- 3D wings feature one very clear and simple to quantify source of drag: **induced drag**.
- In essence, the fact that **a wing produces lift means that it also produces drag**.
- Induced drag coefficient is proportional to the **square** of the lift coefficient
- The source of induced drag is a downwash flow velocity created on the wing by the trailing vortices.

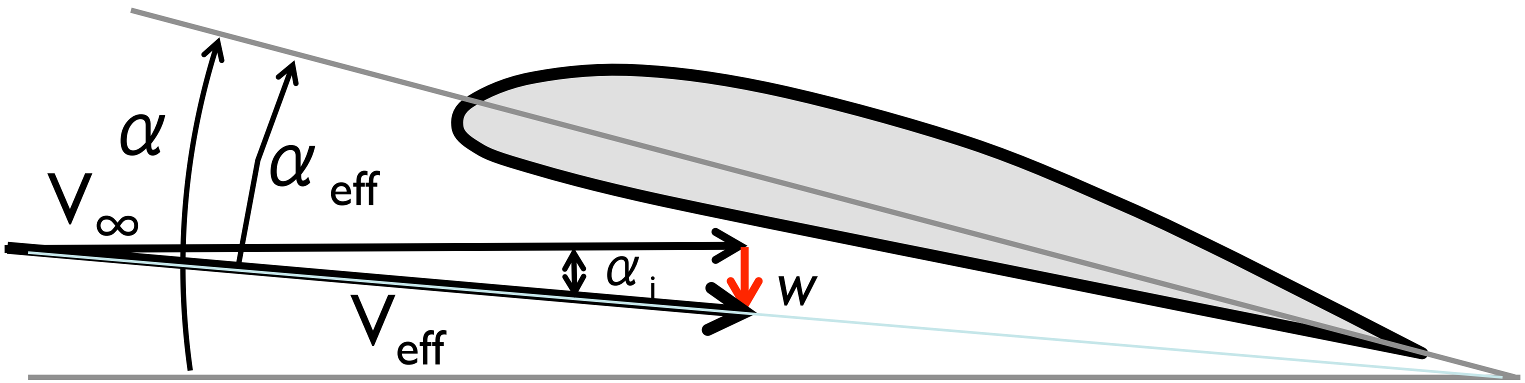
Downwash



Wing-tip vortices → Downward velocity component = Downwash velocity *w*

→ Effective angle of attack $\alpha_{eff} \neq$ geometric angle α

$\alpha_{eff} = \alpha - \alpha_i$ = angle effectively seen by the wing



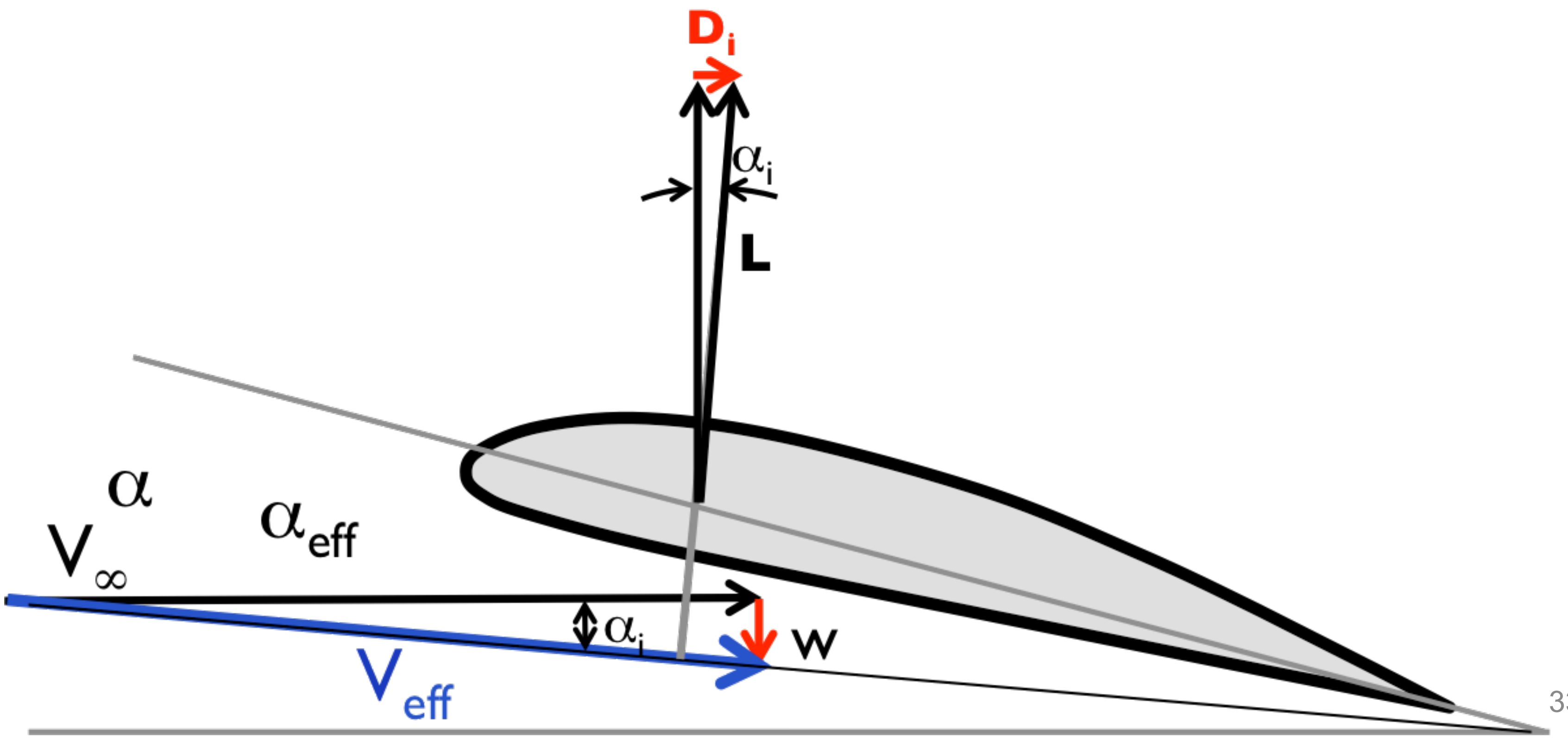
Induced drag

Lift perpendicular to the effective angle of attack

→ Lift inclined of α_i along the vertical

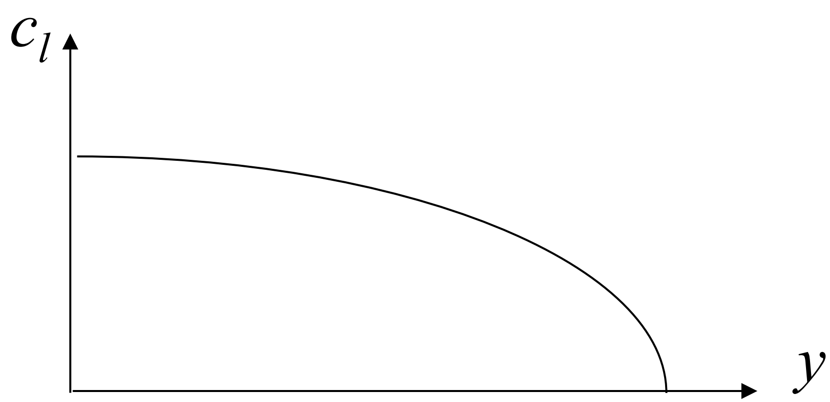
→ Horizontal component of the aerodynamic force = INDUCED Drag

→ 3D inviscid, incompressible flow yields to a drag force !



Elliptical lift distribution

- It can be shown that a wing with an elliptical planform has an elliptical wing distribution



- Furthermore, this wing distribution causes the minimum lift-induced drag
- For such a wing the total lift and induced drag can be easily calculated:

$$C_L = \frac{2\pi AR}{AR+2}\alpha \qquad C_D = \frac{C_L^2}{\pi AR}$$

More on lift distributions

- Few well-known aircraft ever featured an elliptical wing.

- Minimizing lift-induced drag is only one consideration in the design of wings.

- Most wings have non-elliptical lift distributions.

- The lift and induced drag can be calculated for such wings using lifting line theory.

Sources of drag

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- Parasite drag:
  - Skin friction drag (friction)
  - Form drag (also known as pressure or profile drag)
  - Interference drag (vortices created at the intersection of surfaces)
- Induced drag (due to downwash of wings)
- Wave drag (supersonic drag)

Wing characteristics

- **Aspect Ratio** AR = b^2/S
- **Twist**: angle of attack at root is not the same as angle of attack at tip
- **Taper**: chord at tip is smaller than chord at root
- **Thickness ratio**: thickness varies over the span
- **Airfoil**: may change over wingspan
- **Sweep**: wingtip lies behind or in front of wing root

Aspect Ratio

Low aspect ratio wing
F-15 : AR ~ 3

High aspect ratio wing
B-52 : AR ~ 9

Aspect Ratio

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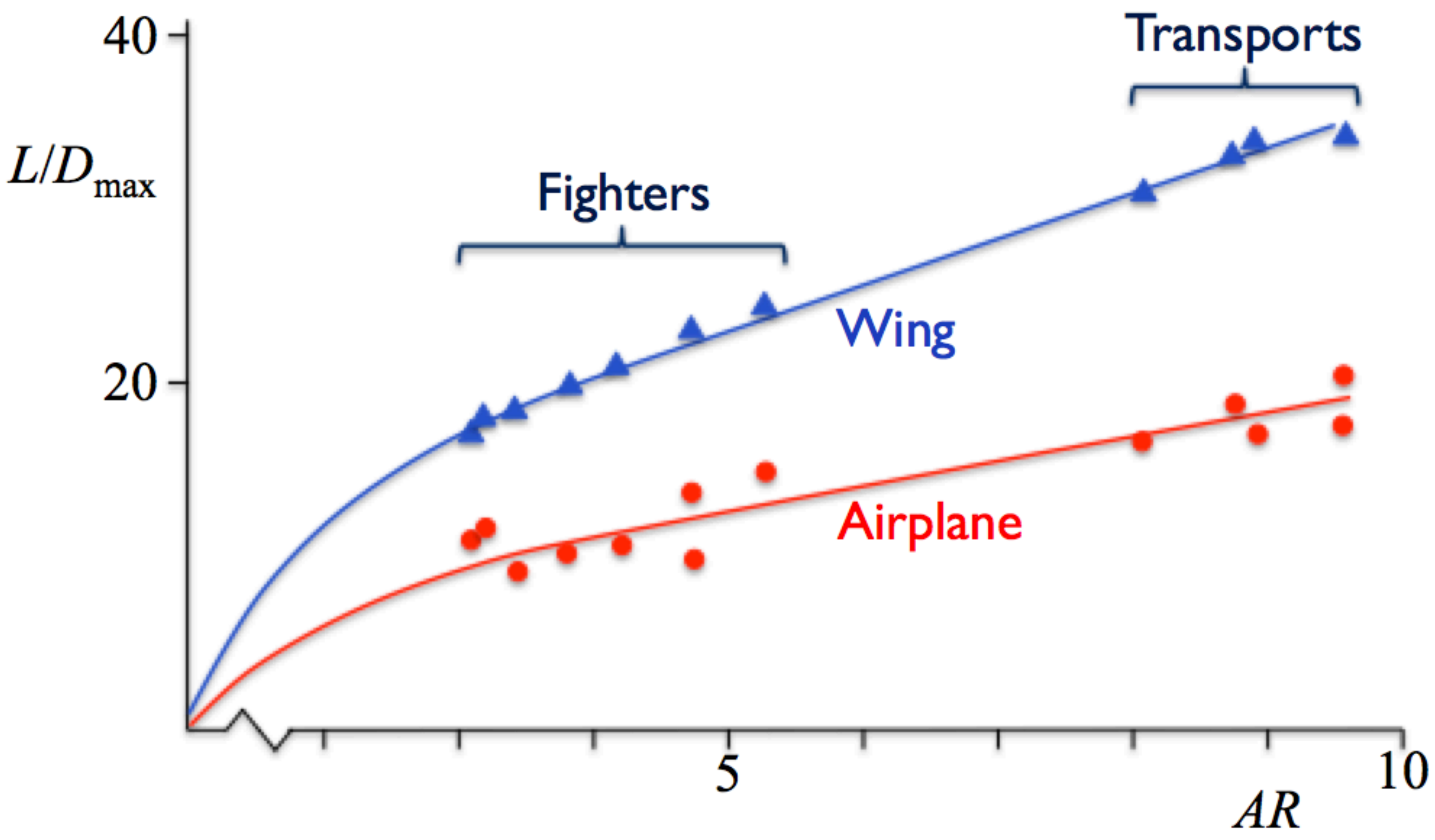
- Aspect Ratio is one of the basic design parameters for transport aircraft.
- It has a significant effect on aircraft performance.
- Large values of AR tend to make the flow around the wind more 2D.
- Remember that 2D flows do not cause induced drag.

$$C_D = \frac{C_L^2}{\pi AR} \qquad C_L = \frac{2\pi AR}{AR+2}\alpha$$

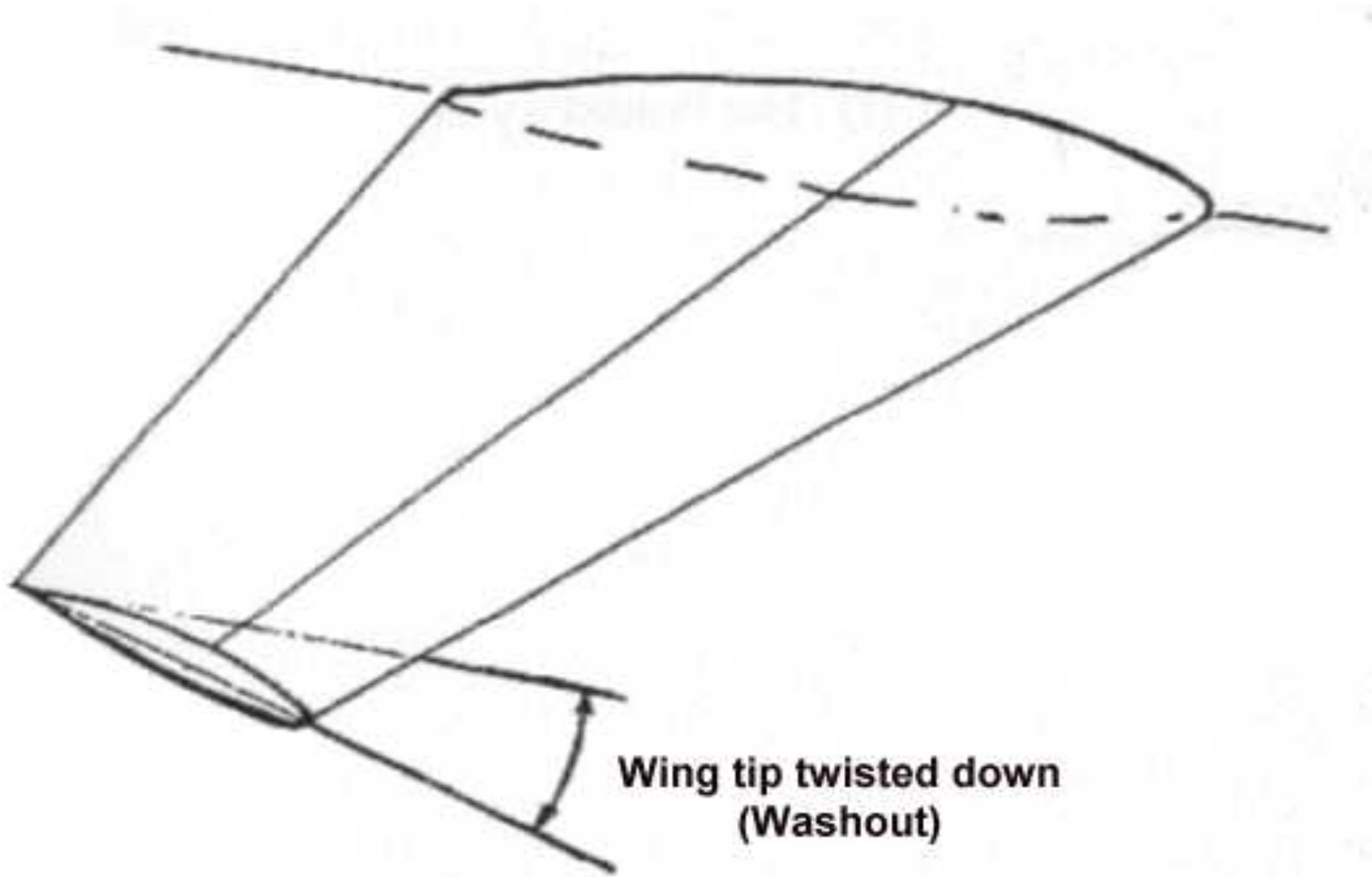
→ High aspect ratio increases the lift coefficient and decreases the drag coefficient.

Lift to Drag ratio



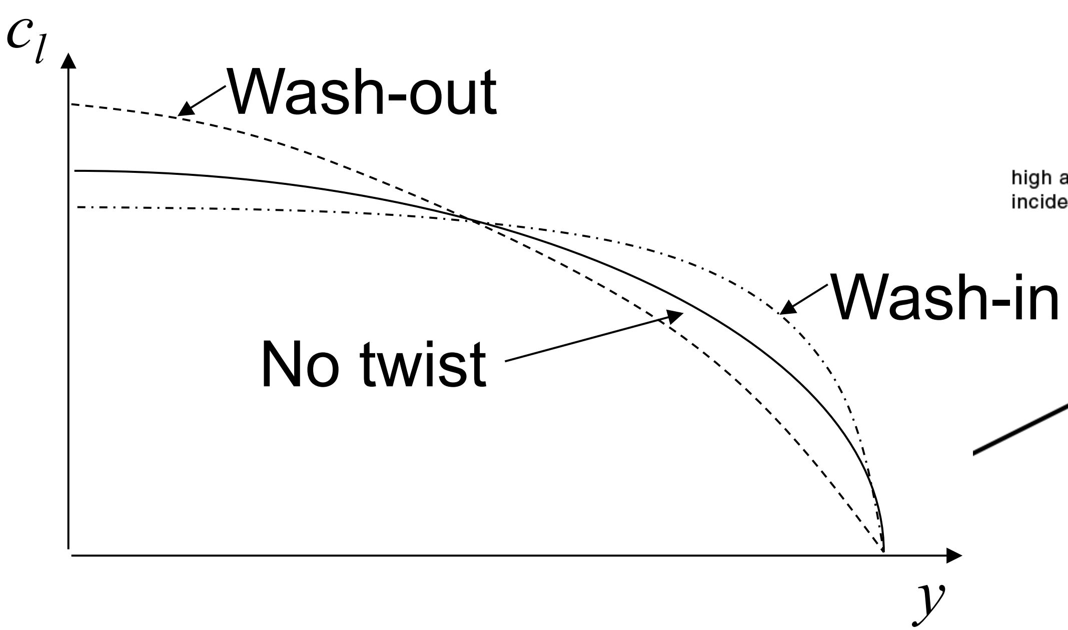
Airplane: ~20
Wing: ~35
Airfoil: ~100

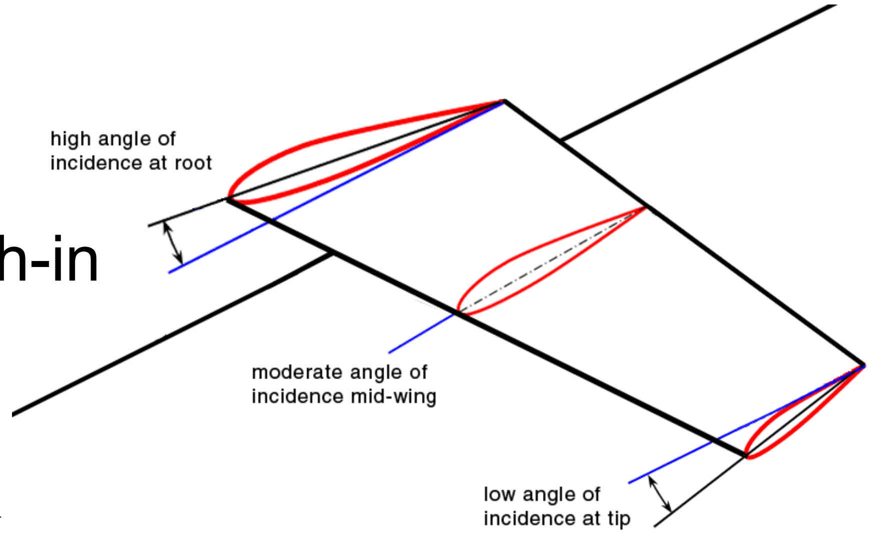
Twisted wings



Wing twist (1)

- Wash-in: $\alpha_{\text{tip}} > \alpha_{\text{root}}$
- Wash-out: $\alpha_{\text{tip}} < \alpha_{\text{root}}$





Wings often have **wash-out** to reduce structural weight and improve stall characteristics

Wing twist (2)

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- The point of initial stalling should be sufficiently inboard, around $0.4s$ from the wing root.

- This can be achieved with suitable twist. If the stall point is too far outboard, a little washout will bring it inboard.

- However, a washout of more than 5^o results in an unacceptable increase in induced drag.

Tapered Wings

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Most WW II fighter aircrafts had tapered wings

Tapered Wings

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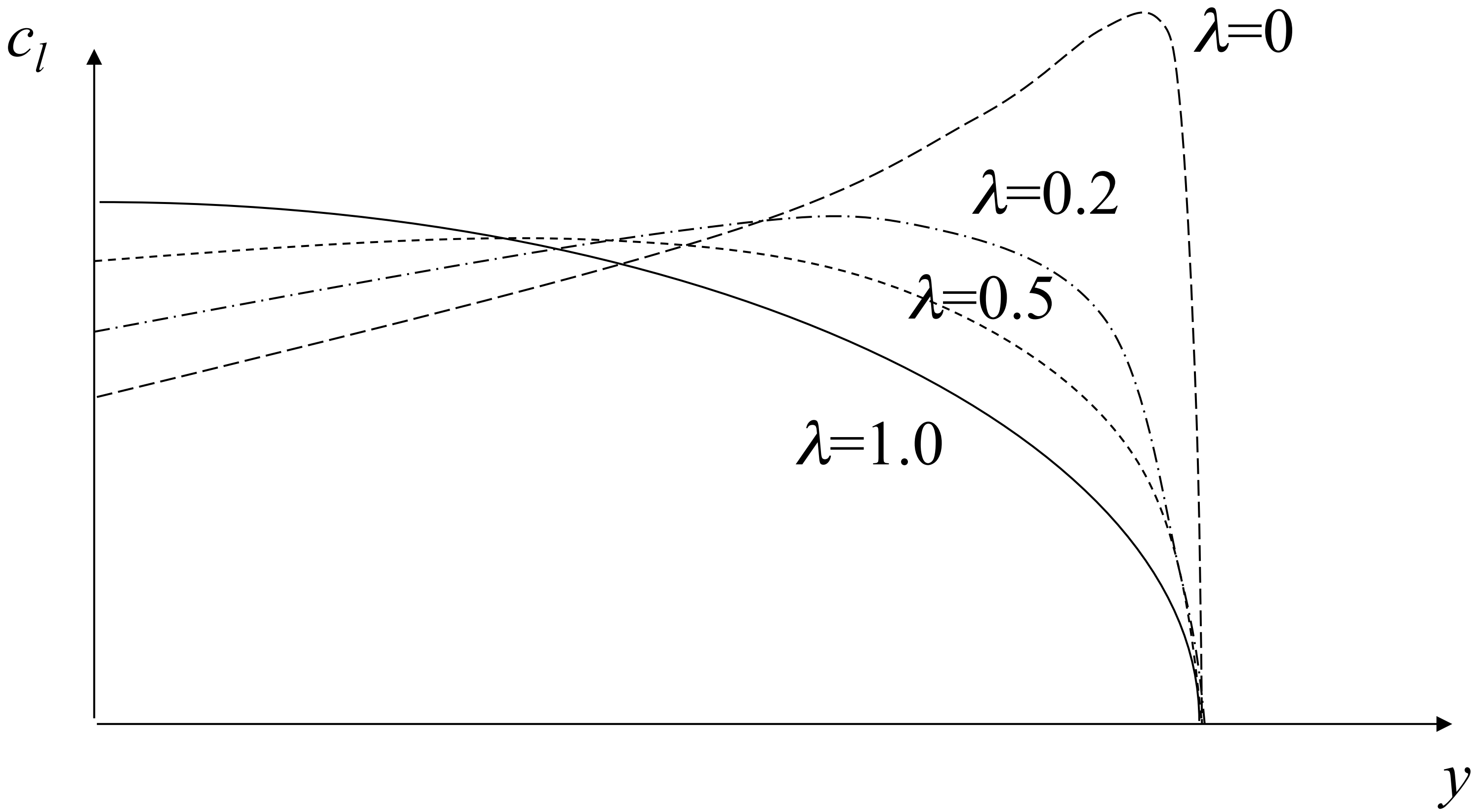
- Taper ratio is defined as $\lambda = c_{\text{tip}} / c_{\text{root}}$

→ Reduction of the amount of lift near the wing-tip.
→ Tip vortex is weaker
→ Induced drag is smaller

- Taper also reduces structural weight
- As the chord at the root is unchanged the maximum lift is not severely affected by taper
- If the taper is not too high, the stalling characteristics are acceptable, even without twist

Effect of wing taper on lift distribution



Taper increases sectional lift coefficient

Non-linear taper

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Taper doesn't have to be linear

P-51 Mustang

Cessna 150

No taper

Untapered wings are easy and cheap to manufacture but aerodynamically inferior to tapered wings

Cessna 120

Cessna 140

Thickness-to-chord ratio

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- **High AR** is interesting for transport aircraft: (decreasing the induced drag)

Structurally → wings are very thick near the root

- Usually achieved by **increasing the thickness ratio near the wing**

- Thickness affects also the **profile drag**

→ Too much increase in thickness can cancel the decrease in induced drag due to high AR.

Thickness-to-chord ratio

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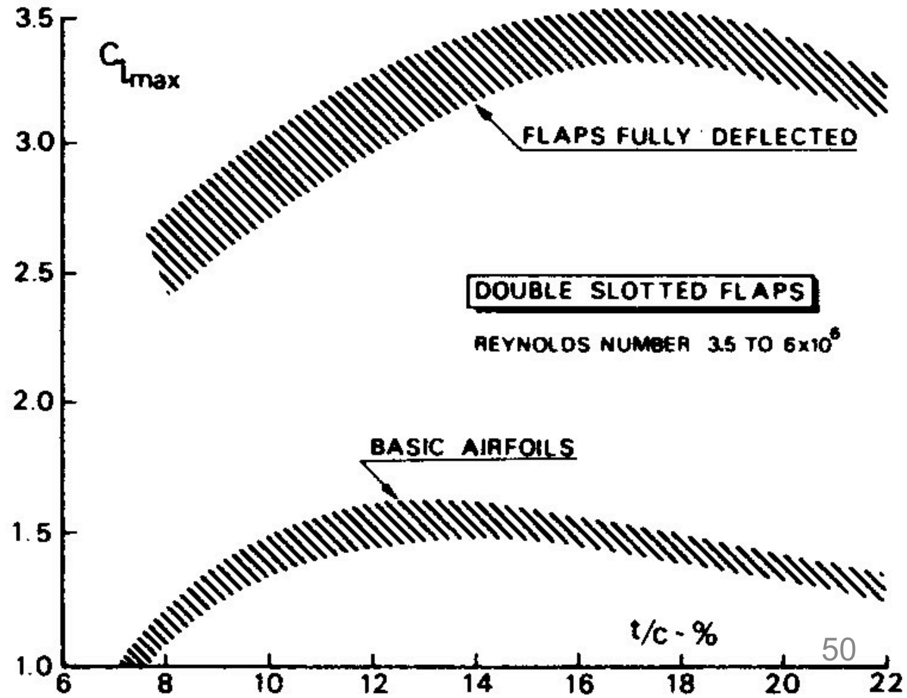
- Thickness ratio also affects the maximum lift
- Optimal thickness ratios:
  - 15-20% near the wing root
  - 10-15% near the wing tip
- Higher than 20% is not good

Maximum c_l for NACA airfoils



Airfoil choices

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- NACA four-digit airfoils:
  - Drag increase with lift is gradual
  - Cambered sections have good maximum lift and docile stall
  - Gradual changes in drag and pitching moment
  - Used in light aircraft (mostly wingtips and tailplanes) and trainers (gradual changes are good for training aircraft).
- NACA five-digit
  - Better maximum lift than 4-digit but very abrupt stall
  - Used inboard, combined with 4-digit airfoils near the wingtip.
- NACA six-series ~ ‘laminar flow’ series
  - Low profile drag at low lift coefficients : ”low drag bucket” region
  - They also have high critical Mach numbers
  - Lower maximum lift than 4- and 5-digit series

Airfoil selection considerations

Airfoil selection should be made using the following considerations:

- An airfoil with low profile drag at the design flight conditions must be chosen
- The airfoil must be capable of giving the desired maximum lift to the wing (with flaps if needed)
- Compressibility issues must be addressed

Compressibility issues

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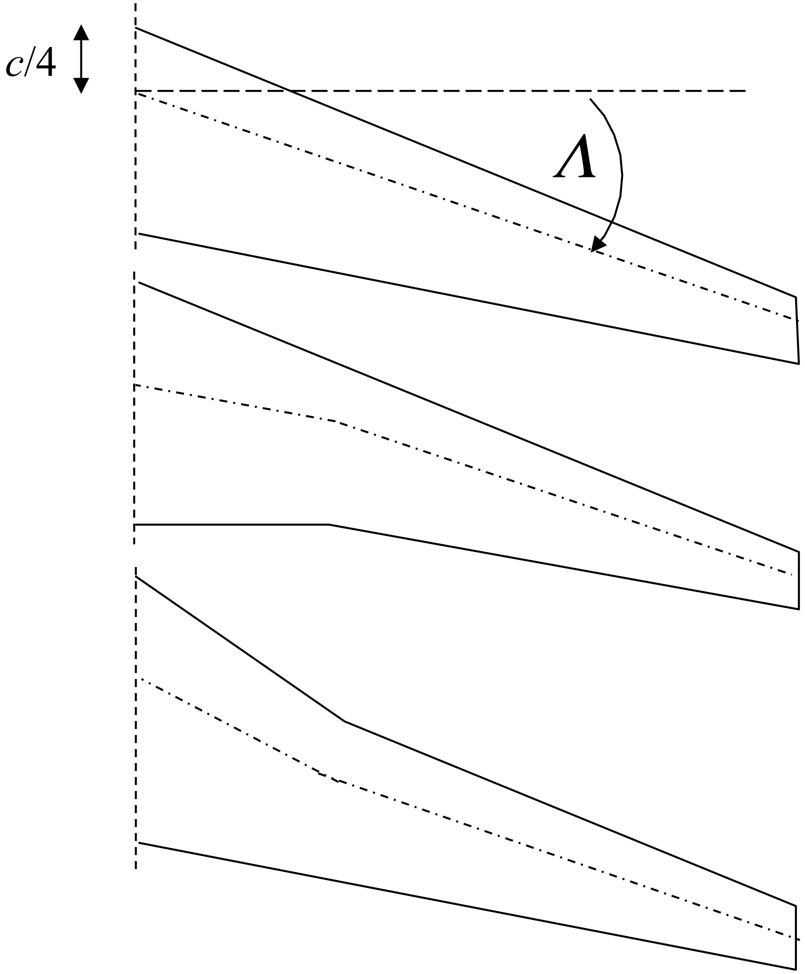
High subsonic cruise Mach numbers can be attained by:

- Using sweepback
- Reducing the thickness-to-chord ratio
- Using improved airfoil sections :

  'supercritical airfoils'
- Optimizing spanwise camber and twist variation

Swept wings



Straight quarter-line chord

Reduced sweep at inboard section

Increased sweep at inboard section

Zero sweep angle

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BAe ATP

- Maximum airspeed of 137m/s at 25000ft, i.e. maximum M=0.44
- Straight tapered wings

High sweep angle

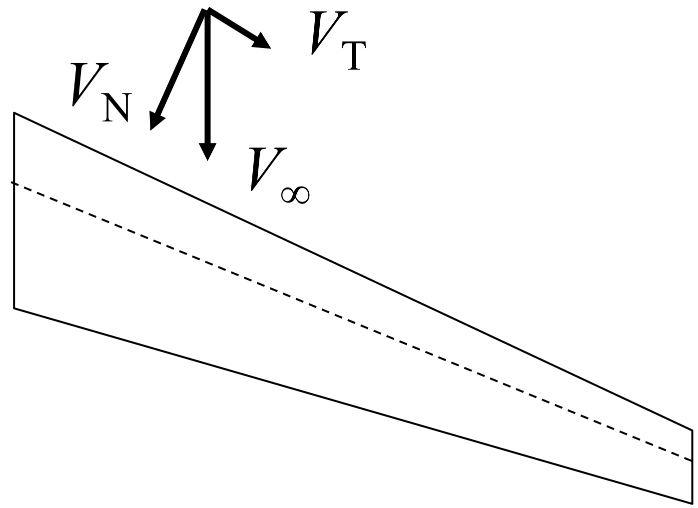
Airbus A380

- Maximum airspeed of 265m/s at 35000ft, i.e. maximum M=0.89
- Highly swept tapered wings

Effect of sweep

A component of the free stream airspeed, V_T is tangent to the wing. Therefore, the airspeed seen by the **airfoil** is only V_N

$$V_N = V_\infty \cos\Lambda$$



The effective Mach number seen by the wing's airfoil is

$$M_{eff} = M_\infty \cos\Lambda$$

→ A higher V_∞ is required to reach sonic conditions

Using sweepback

- Up to M=0.65 or M=0.7

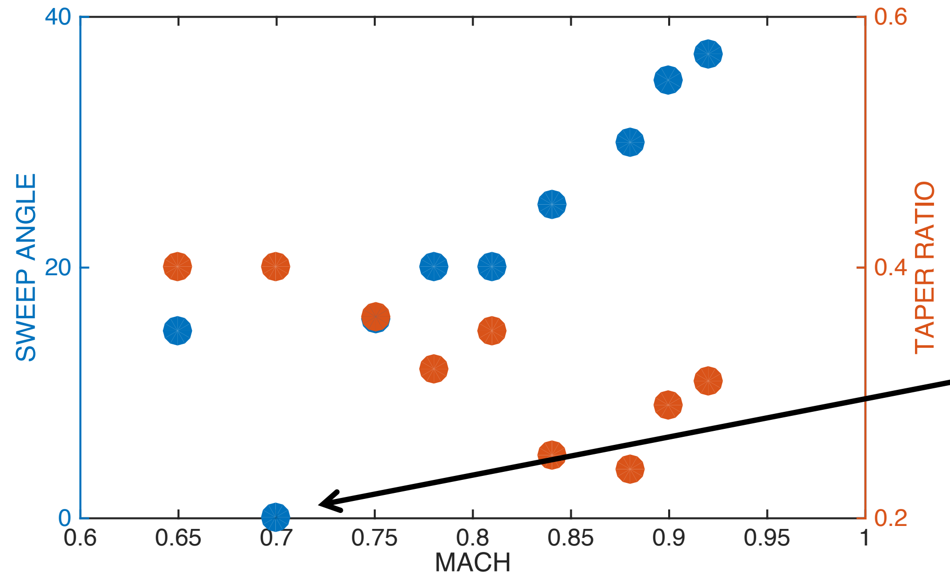
  → straight wings with appropriate thickness ratio are sufficient.
- M > 0.7 → sweepback is required
- Sweep angle of 35° is rarely exceeded

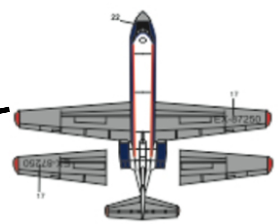
Sweep data

Sweep angle increases with Mach number.

Taper ratio also, generally, increases with Mach number.

Aircraft	Aspect Ratio	Taper Ratio	Sweep Angle	Maximum Mach
VFW-Fokker 614	7.22	0.402	15°	0.65
Yakovlev Yak 40	9.00	0.396	0°	0.70
Fokker-VFW F 28	7.27	0.355	16°	0.75
BAC 1-11 200/400	8.00	0.321	20°	0.78
Aerospatialle Caravelle	8.02	0.354	20°	0.81
Boeing 737 100/200	8.83	0.251	25°	0.84
		0.244	30°	0.88
		0.293	35°	0.90
		0.309	37°30’	0.92

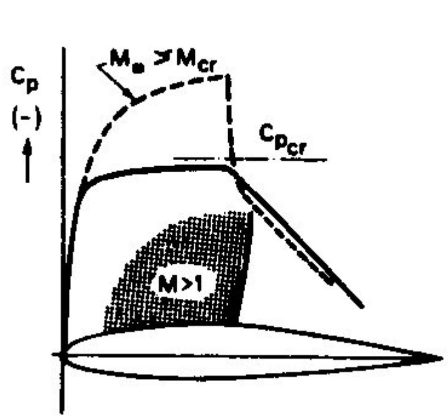




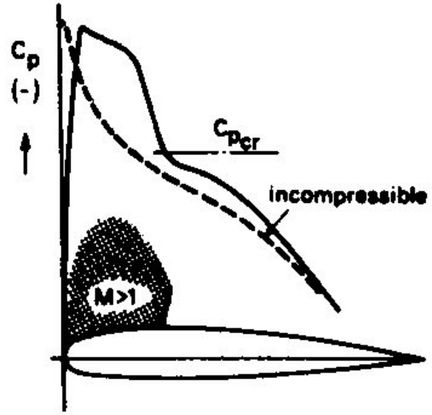
Shock waves

Shock waves can be formed on the upper (and even lower) surfaces of wings travelling in transonic (i.e. M<1) flow.

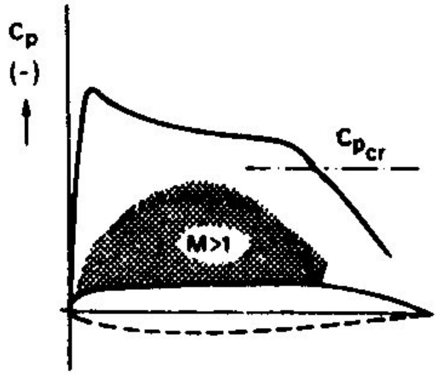
Airfoils for Transonic Conditions



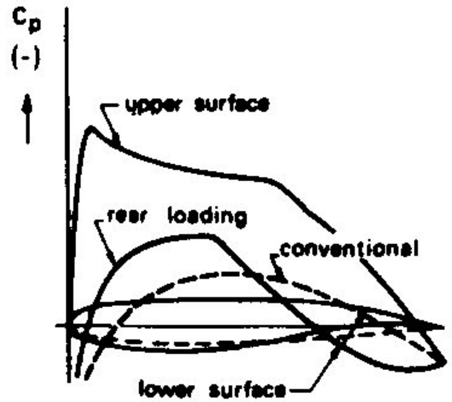
a. Conventional section with roof-top pressure distribution



b. Peaky upper surface pressure distribution

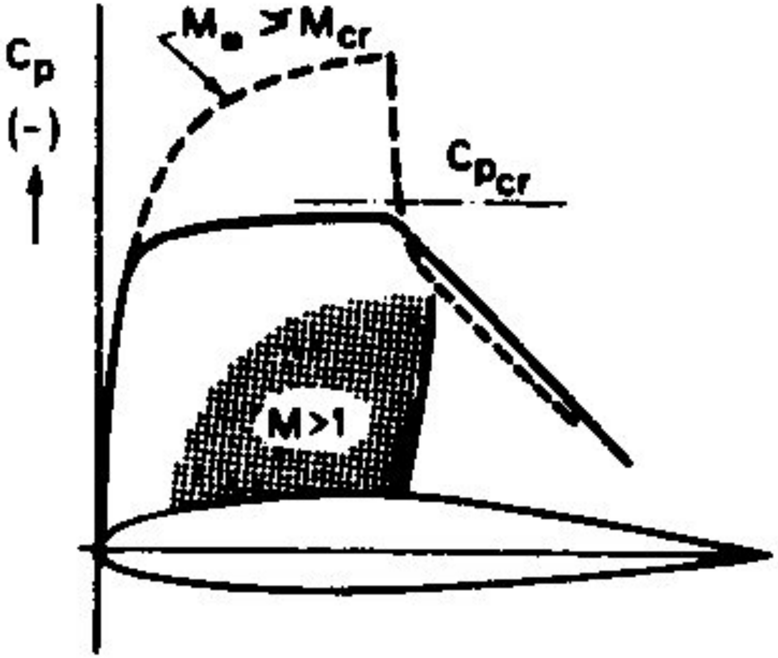


c. Supercritical upper surface pressure distribution



d. Rear loading airfoil compared with conventional airfoil (lower surface)

Airfoils for Transonic Conditions



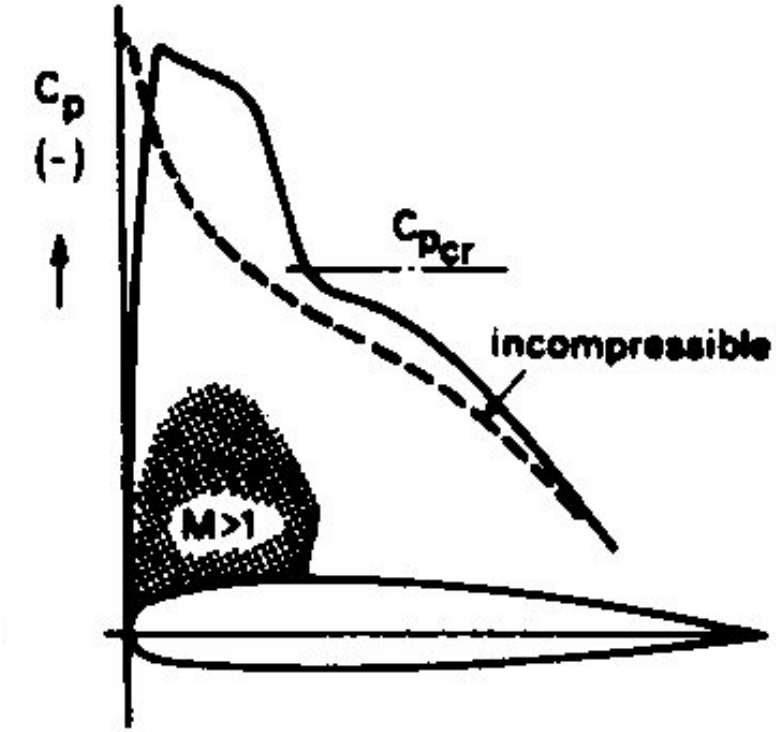
Rooftop

→Relatively flat pressure distribution on the forward upper surface.

→The flow is not accelerated in this region and the advent of M=1 is delayed.

→NACA 6-series

Airfoils for Transonic Conditions



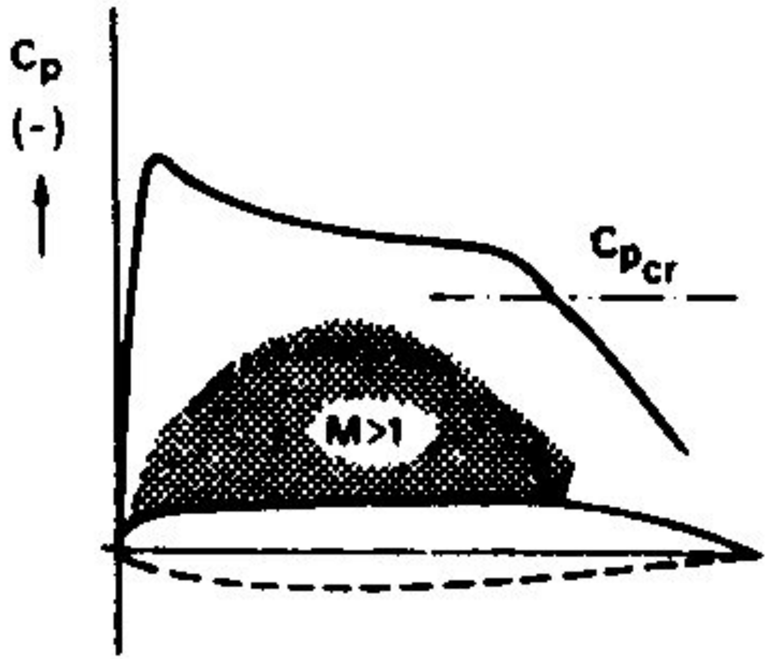
Peaky upper surface

→Supersonic velocities and suction near the leading edge,

→Weak shock wave.

→ Drag rise is postponed to higher airspeeds.

Airfoils for Transonic Conditions

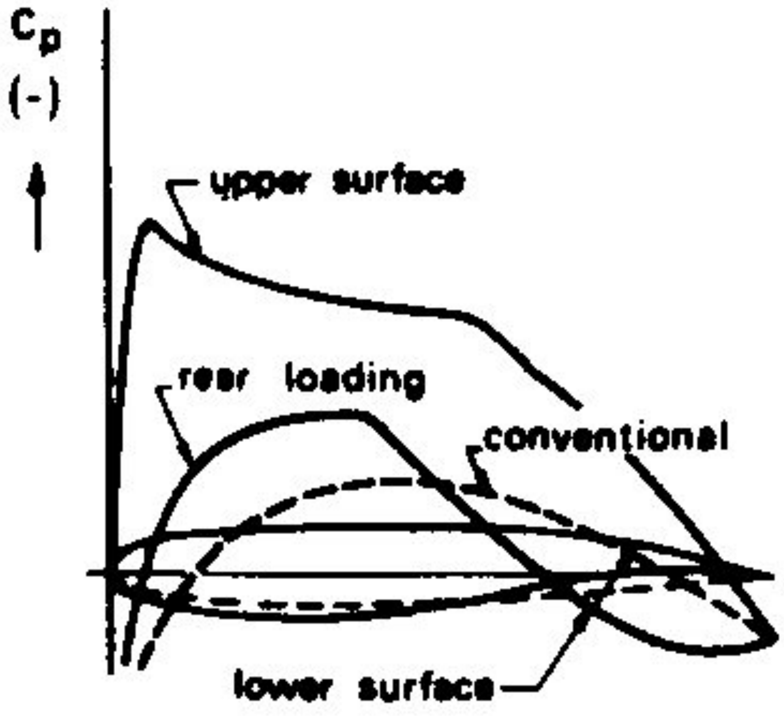


Supercritical airfoils

→Flat upper surface, creating shock-free supersonic flow region.

→This region is much greater than that of the peaky distribution

Airfoils for Transonic Conditions



Rear loading

→Rear lower surface is highly cambered

→Lot of lift is generated near the rear of the airfoil.

→When combined with a flat upper surface (supercritical airfoil) a large decrease in drag for the same Mach number and lift coefficient can be obtained.

This combination has been used by Airbus A300 and all subsequent civil transports

Design thickness ratio

- A value for the most appropriate thickness ratio for a particular Mach number was proposed by Torenbeek

$$t / c = 0.3\left\{\left[1-\left(\frac{5+M^{2}}{5+M^{*2}}\right)^{3.5}\right]\frac{\sqrt{1-M^{2}}}{M^{2}}\right\}^{2/3}$$

where $M^{*}=1$ for conventional airfoils, $M^{*}=1.05$ for peaky airfoils and $M^{*}=1.15$ for supercritical airfoils

- $M\cos\Lambda$ can be used for sweptback wings

Wing lift coefficient (unswept wings)

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- The lift wing coefficient can be estimated by

$$C_L = C_{L_\alpha}\alpha$$

$$C_{L_\alpha} = 0.995\frac{c_{l_\alpha}}{E + c_{l_\alpha}/\pi AR}$$

$$E = 1 + \frac{2\lambda}{AR(1+\lambda)}$$

$C_{L\alpha}$=wing lift curve slope

$c_{l\alpha}$=sectional lift curve slope

E=Jone’s correction

λ=taper ratio

AR=Aspect ratio

Wing lift coefficient (unswept wings)

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- For a sectional lift curve slope of 2π

$$C_{L_\alpha} = \frac{2\pi}{1 + \frac{2}{AR}\frac{1+2\lambda}{1+\lambda}}$$

- Compressibility effects can be taken into account by replacing AR by βAR and $C_{L\alpha}$ by $\beta C_{L\alpha}$, where $\beta = \sqrt{1 - M^2}$

Wing lift coefficient (swept wings)

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- For swept wings, an approximate expression for the wing lift curve slope is

$$\beta C_{L_\alpha} = \frac{2\pi}{\dfrac{2}{\beta AR} + \sqrt{\dfrac{1}{k^2 \cos^2 \Lambda_\beta} + \left(\dfrac{2}{\beta A}\right)^2}}$$

$$\text{where } \tan \Lambda_\beta = \frac{\tan \Lambda_{1/2}}{\beta} \text{ and } k = \frac{\beta c_{l_\alpha}}{2\pi}$$

$\Lambda_{1/2}$ is the sweepback angle at the half-chord

Wing maximum lift coefficient

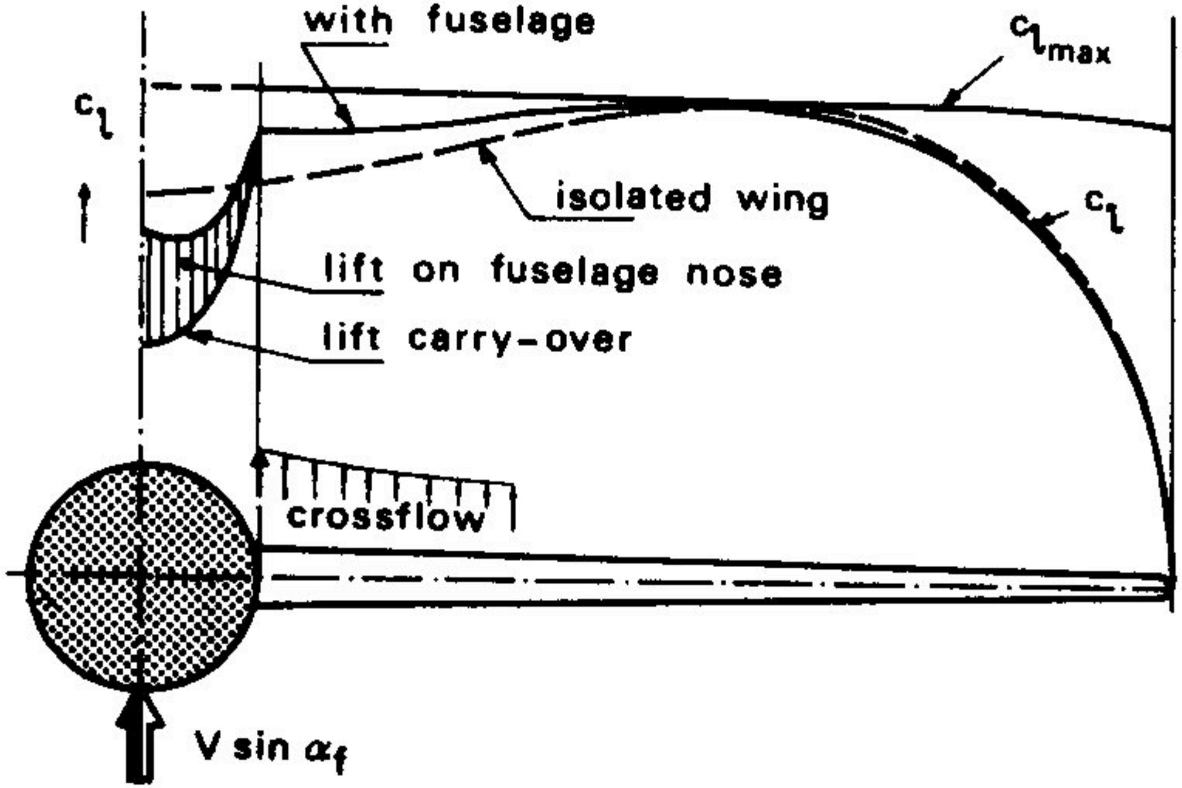
- The maximum lift coefficient of a wing can be approximated by

$$C_{L_{\max}} = k_s \frac{c_{l_{\max,\mathrm{root}}} + c_{l_{\max,\mathrm{tip}}}}{2}$$

where k_s=0.88 for untapered wings
k_s=0.95 for tapered wings

Wing-fuselage interference

The fuselage produces very little (if any) lift. However, some lift is carried over from the wing onto the fuselage.



Supersonic flight

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- Supersonic flight requires different wing design because:
  - Lift generation mechanisms are different
  - Much more drag is produced
- Supersonic airfoils are usually **very thin** and **sharp**
- Supersonic wings are either:
  - Trapezoidal
  - Delta-shaped

Trapezoidal wings

Bell X-1

F-104

F-22

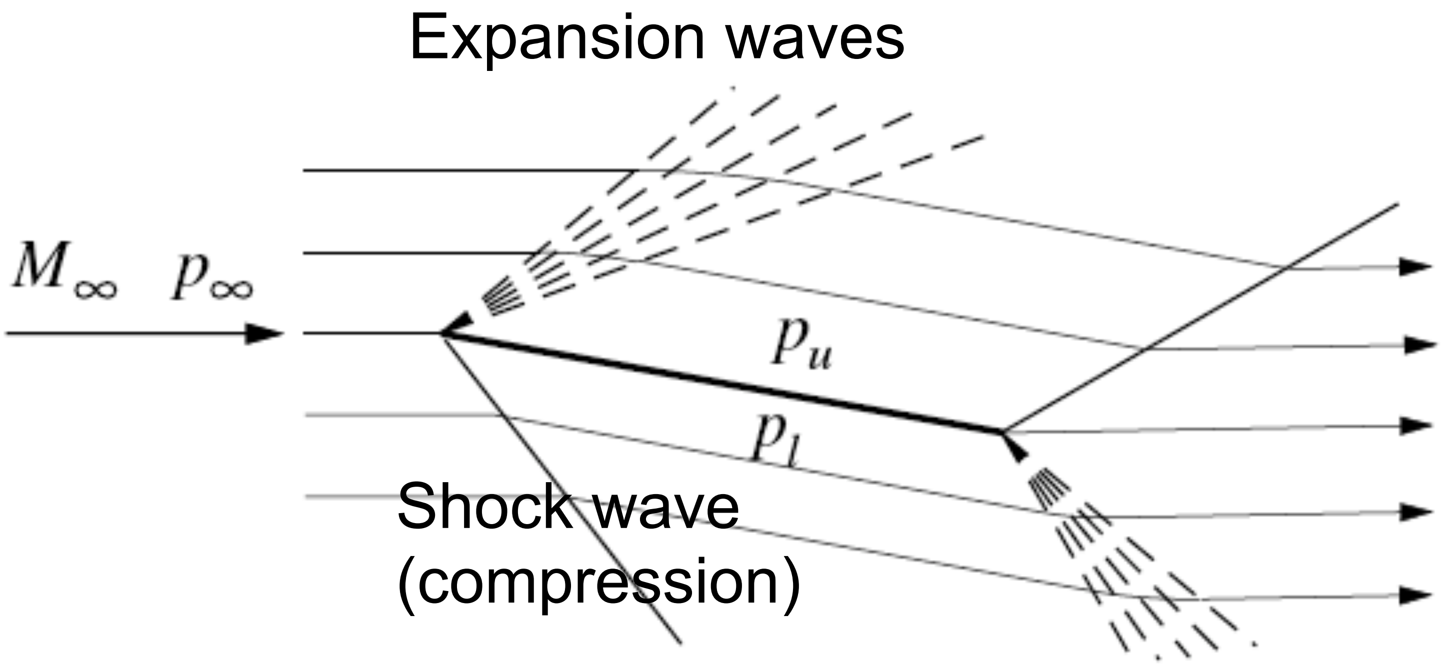
Delta Wings

Concorde

Mirage 2000

MiG-21

Supersonic flow over flat plate



p_u is lower than p_l so that there is a net force upwards : Lift

Lift on supersonic flat plate

- It is easy to calculate the lift on a flat plate airfoil in a supersonic flow
- The compression and expansion cause pressure coefficients of

$$c_{p_l} = \frac{2\alpha}{\sqrt{M^2 - 1}} \qquad c_{p_u} = -\frac{2\alpha}{\sqrt{M^2 - 1}}$$

- The force acting normal to the plate is

$$c_n = \frac{1}{c}\oint c_p \mathrm{d}x = \frac{1}{c}\int_0^c (c_{p_l} - c_{p_u})\mathrm{d}x = \frac{4\alpha}{\sqrt{M^2 - 1}}$$

→ Lift coefficient:

$$c_l = c_n \cos\alpha = \frac{4\alpha}{\sqrt{M^2 - 1}}$$

Drag on supersonic flat plate

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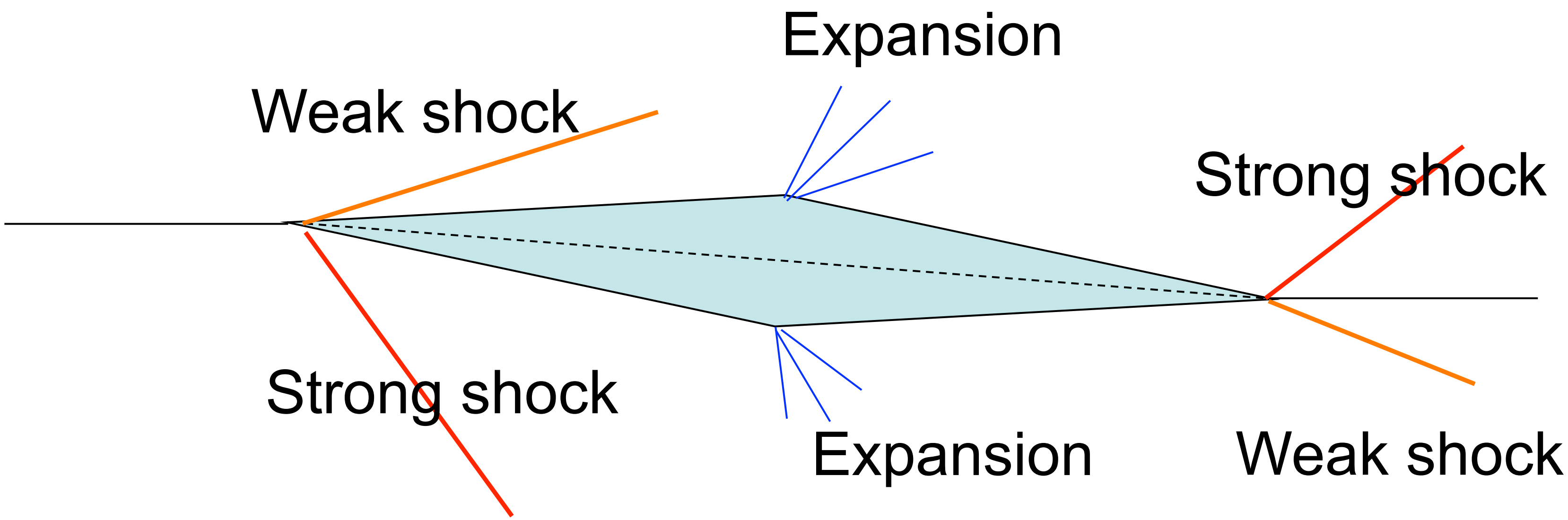
- Unlike incompressible flow, supersonic flow causes drag, known as wave drag.
- The drag force is obtained from the normal force as

$$c_d = c_n \sin\alpha = \frac{4\alpha^2}{\sqrt{M^2 - 1}}$$

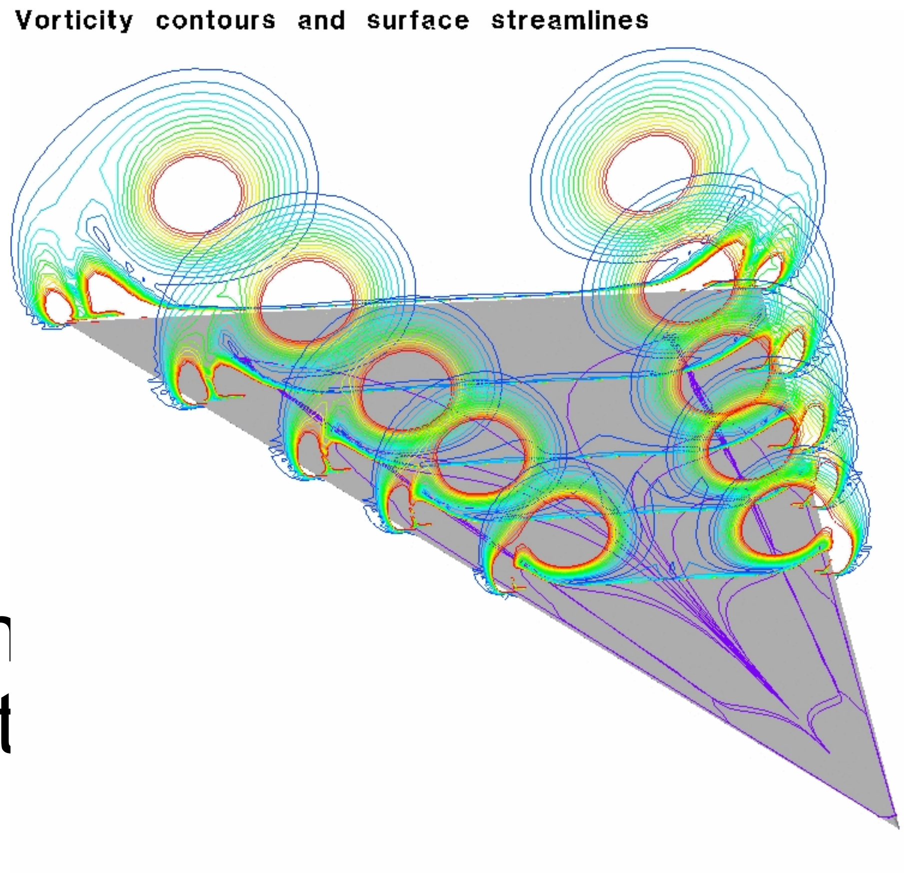
Diamond airfoil



The difference in shock strengths causes the flow on the lower surface to be compressed more than the flow on the upper surface.

Delta Wing

- Lift is generated by creation of conical vortices
- High speed flow under the vortices causes low pressure
- Pressure difference with lower surface causes lift

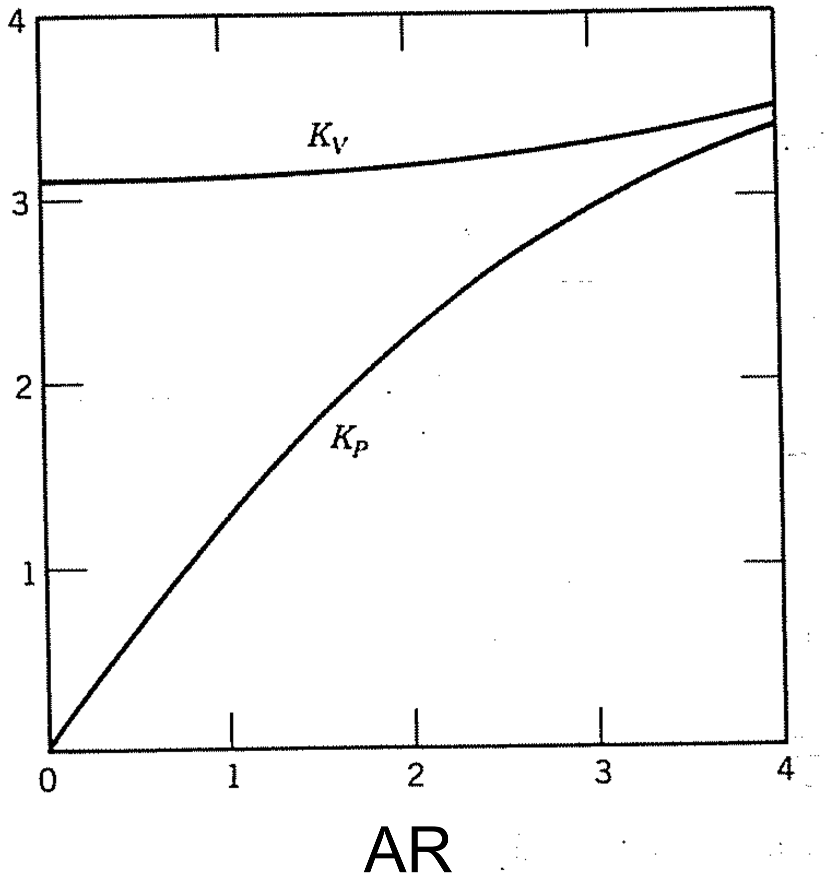


Lift on Delta Wing

2 contributions: potential and vortex

$$C_L = C_{L,P} + C_{L,V} = K_P \sin\alpha\cos^2\alpha + K_V \cos\alpha\sin^2\alpha$$

α = angle of attack

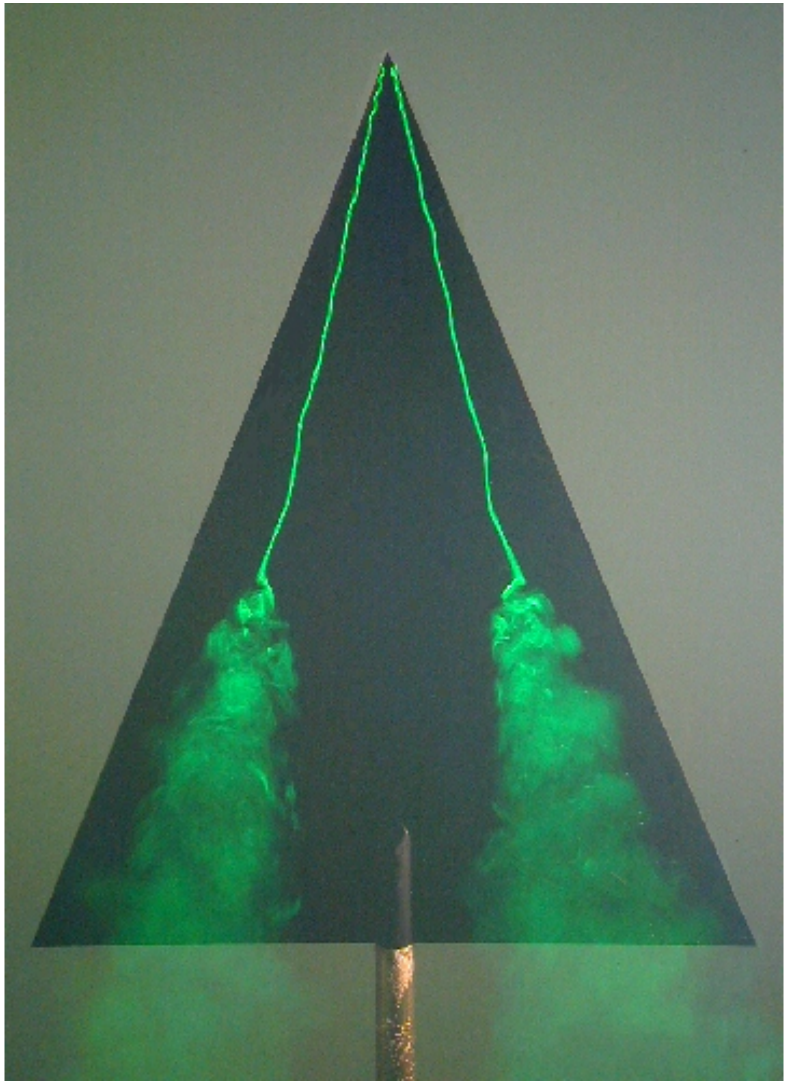


Vortex burst

Very high angles of attack

→ Vortices break down

→ Loss of lift



Delta Wings at low speeds

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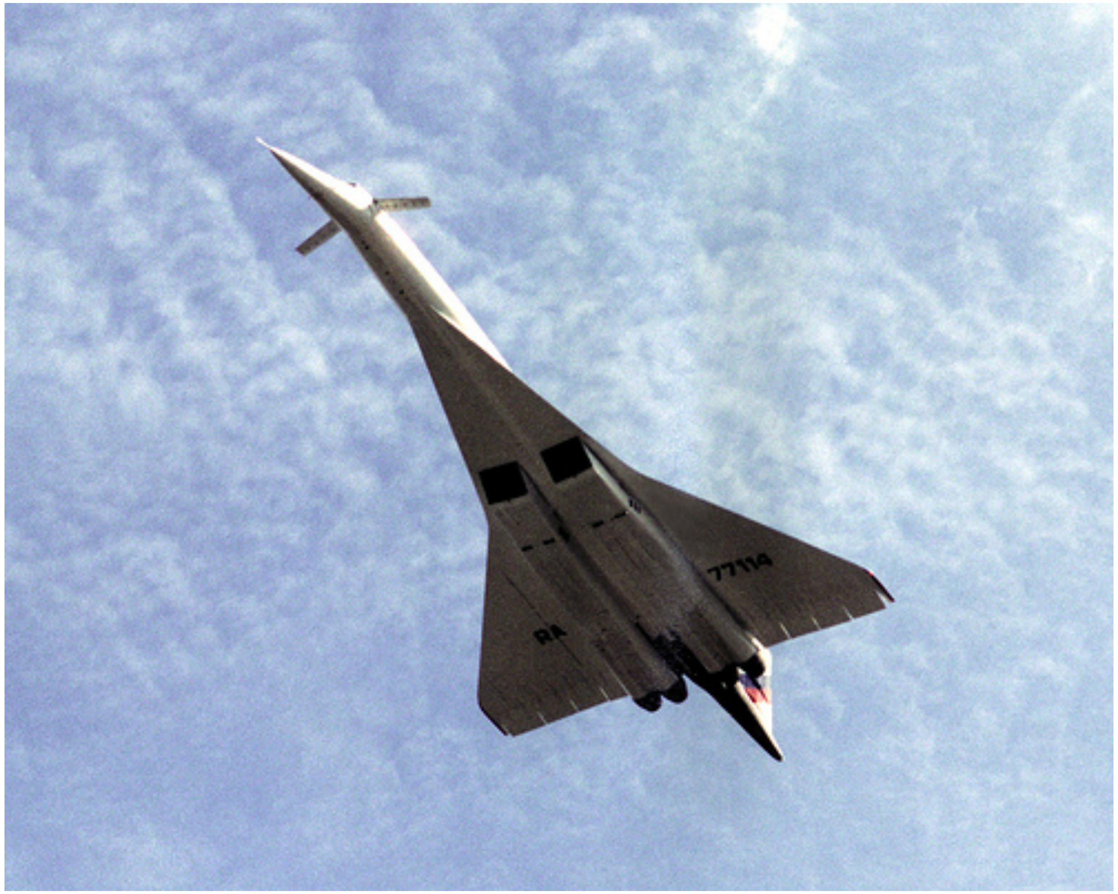
- Quite inefficient at low speeds
- Concorde had to fly at an uncomfortably high angle of attack to take off and landing
- Several solutions have been tried
  - Canards
  - Compound Delta
  - Swing-wing
  - Leading Edge Extensions

Canards

Solution adopted by the Tu-144

→ Control the aircraft at low airspeeds



Compound Delta

2 Delta wings superimposed, with different sweeps

The highly swept section

→ Additional, stronger vortices that increase lift

→ Keep the flow attached at higher angles of attack

Saab-35 Draken

F-16XL

Swing-Wing

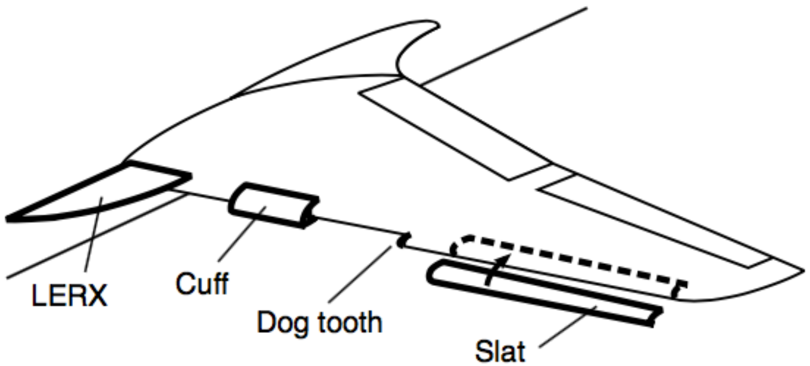
High speeds → Delta wing

Low speeds → High AR trapezoidal wing

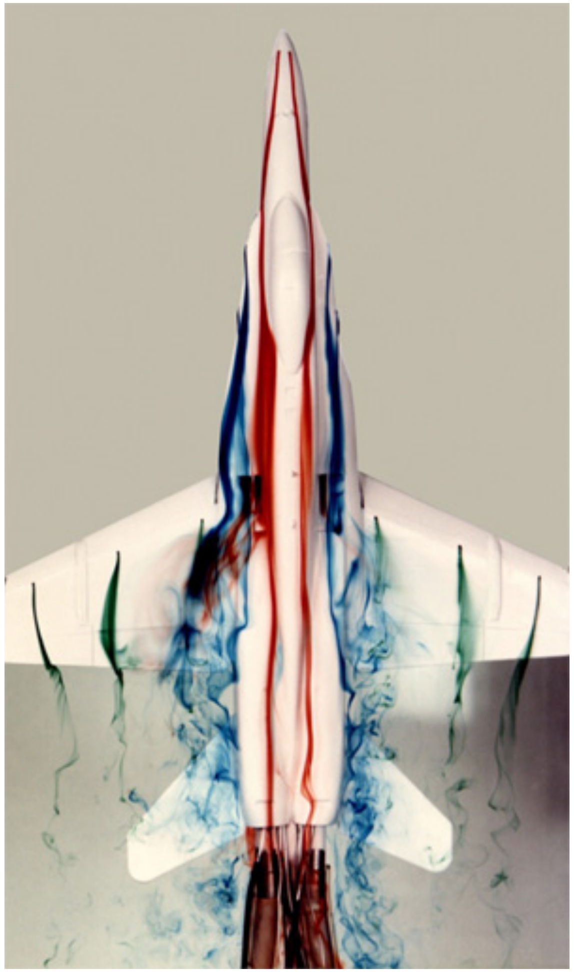
F-14

Leading Edge Extensions

= Combination of the advantages of Delta wings and trapezoidal wings



F18



→Create vortices (Delta Wing)

→Vortices can keep the flow over the wing attached at very high angles of attack

Blended Wing Body (BWB)

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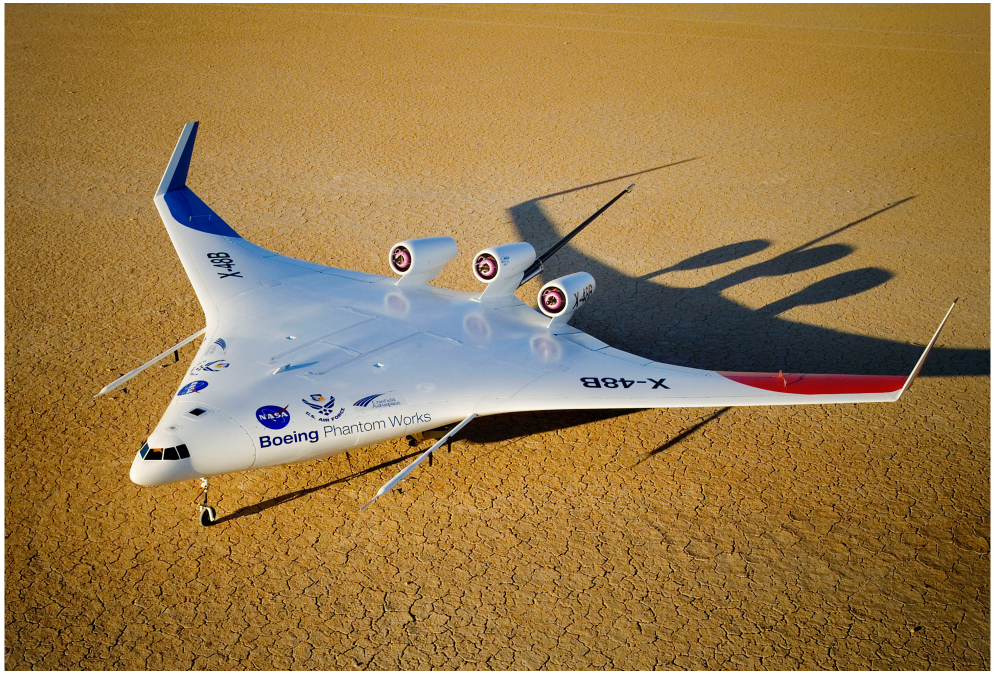
BWB = non-conventional aircraft
in opposition to 'Tube-Wing' aircraft
considered as the next generation of aircraft

BOEING + NASA → Phantom Works → X48 program



Flights with scaled models (6-8 m span)

<u>Future specifications:</u>

- 80m span
  (airport constraint)
- Mach 0.85
- L/D ~ 20

Blended Wing Body (BWB)

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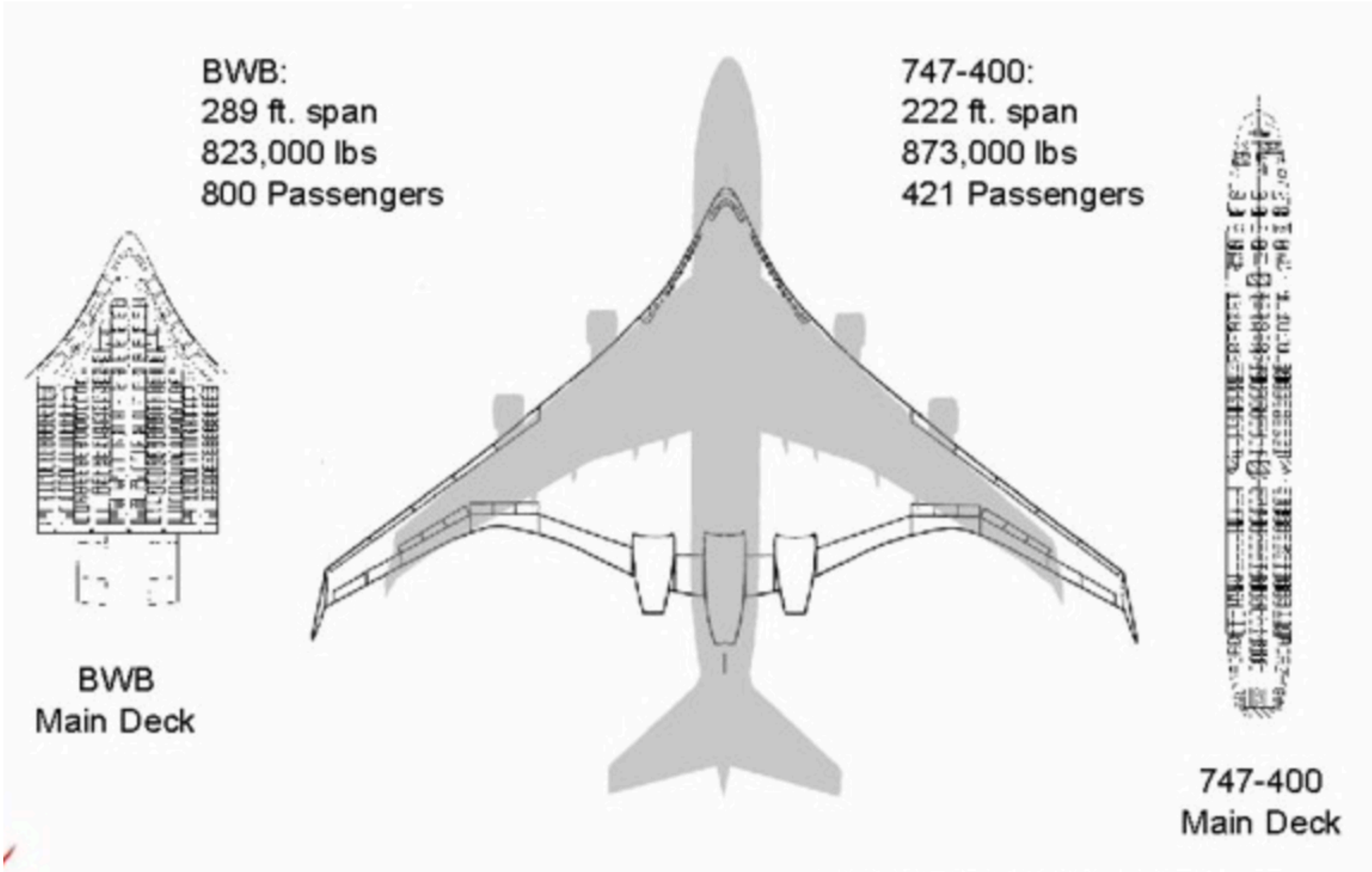
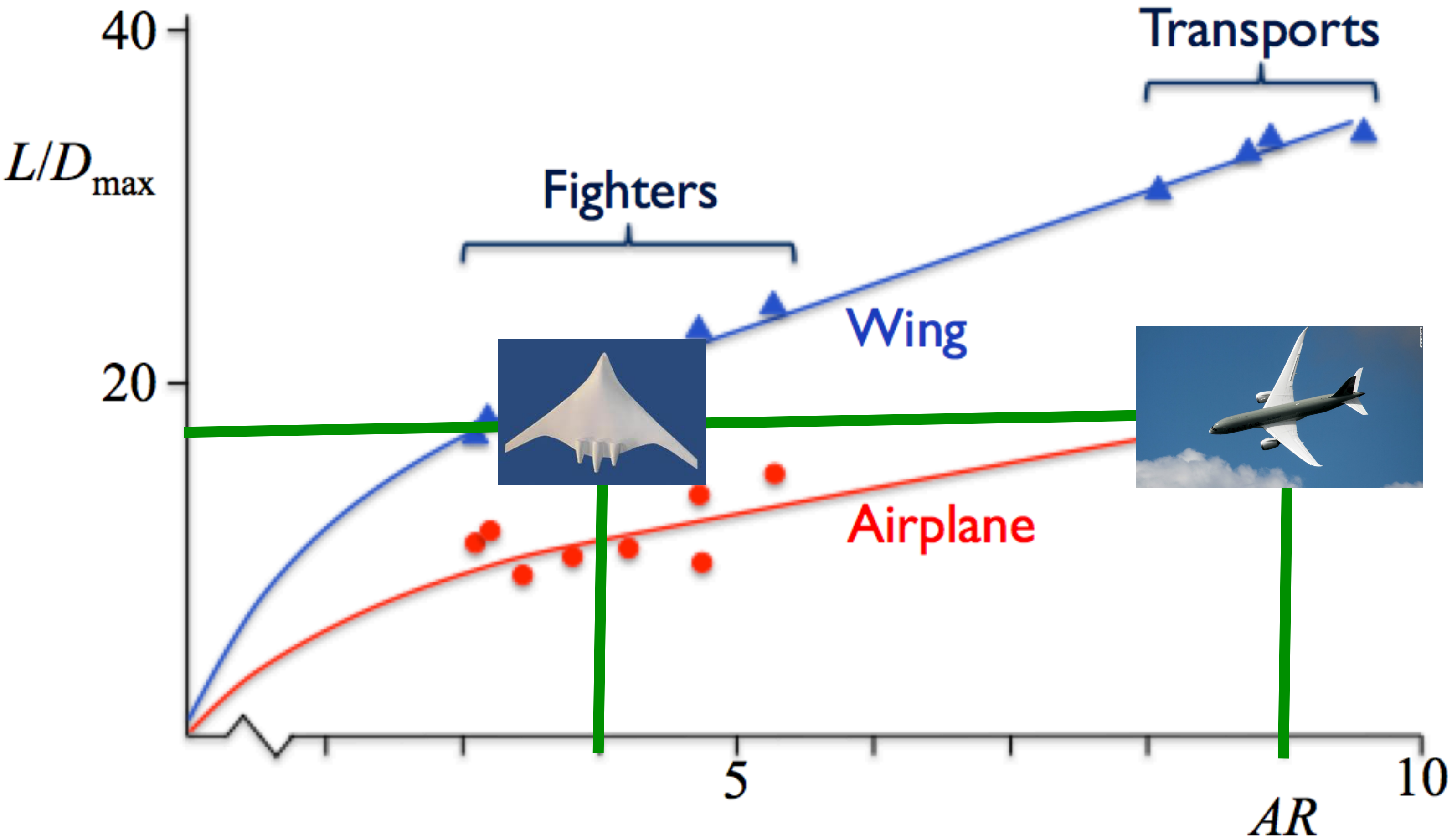


Lift to Drag ratio



Conventional Airplane: ~20 with HIGH AR
BWB : ~ 19 with LOW AR

Blended Wing Body (BWB)

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B2 ~ Delta wing ~ BWB

Blended Wing Body (BWB)

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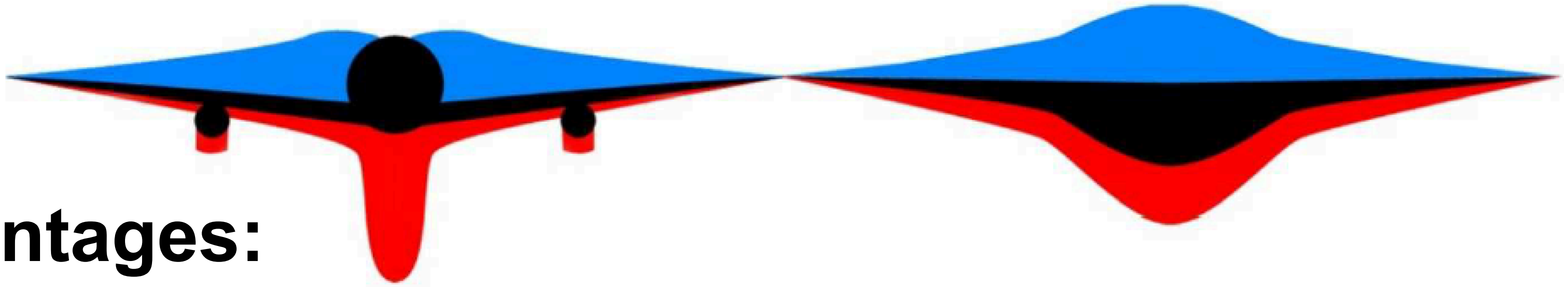
Characteristics:

- Tailless
- Low AR (typically around 4)



Advantages:

- Maximization of the volume / wing span ratio
- Minimization of the wetted area to volume ratio
- Reduction of the interference drag

$$\left(\frac{L}{D}\right) = b\sqrt{\frac{\pi}{kS_{DO}}}$$ → (L/D) 20% higher than 'Tube-Wing'

→ Thick airfoil leading to better structural efficiency

Blended Wing Body (BWB)

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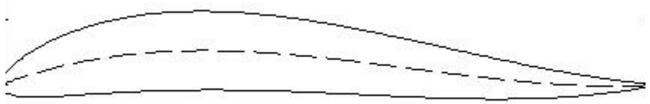
Drawbacks:

- Regulatory compliance : passenger evacuation
- Psychological : no windows
- **Stability** !

No tail → pitch and yaw stability problems



Solutions:

→ Custom airfoils : low camber and reflexed camber line

→ Twist : aerodynamic or physical

→ CG placement, in front of the center of pressure (CP)

→ Large surface of control : elevator, winglets with rudders

Roll stability → solved using swept wings and dihedral

Summary

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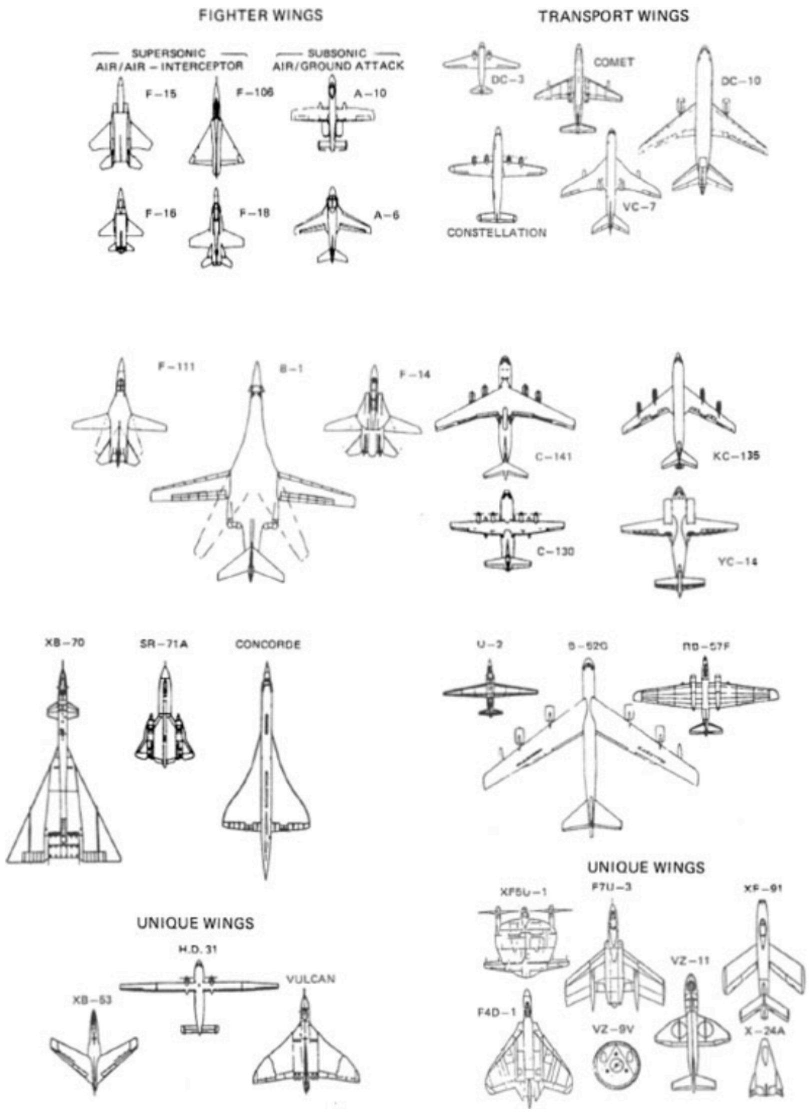
- Several types of aircraft design exist

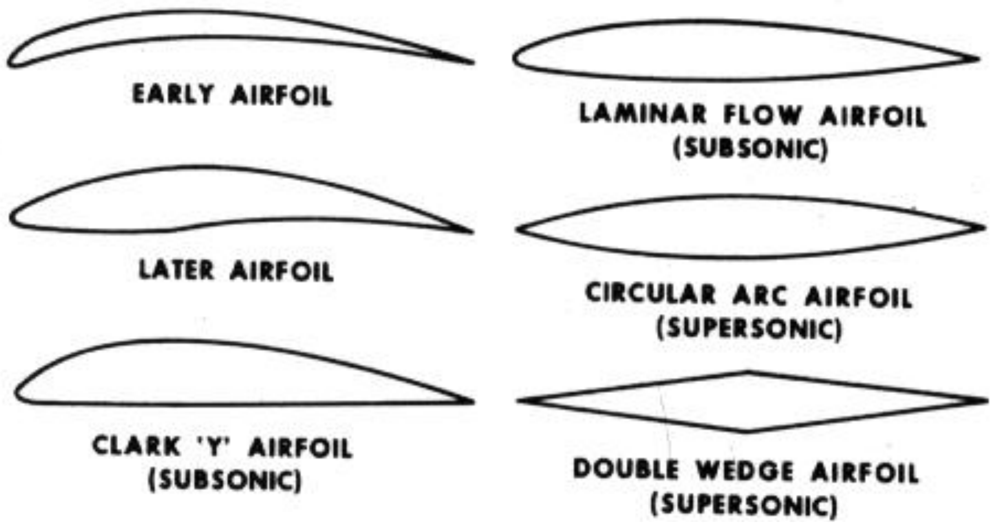
Mach ↗



Summary

- **Starting point** : choosing an AIRFOIL profile (2D)

→ L' et D' or better c_l, $c_{d,}$ c_m



- **From 2D to 3D**

→ L et D or better C_L, C_D, C_M + C_S, C_{Mroll}, C_{Myaw}

Summary

How to find airfoils and wings characteristics ?

2D Aerodynamics

→ Airfoil tables (e.g. NACA tables)

→ Inviscid theories (thin airfoil theory)

→ Viscous

→ Panel codes (Xfoil)

→ CFD models

3D Aerodynamics

→ Inviscid theories (Prantl lifting line)

→ Panel codes (ADS, PAN-AIR, …)

→ CFD models

Summary

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- This is only the AERODYNAMIC part !
- All other aspects must be taken into account !
  - Structure
  - Performances
  - Stability
  - Aeroelasticity
  - Propulsion

→ AIRCRAFT DESIGN = compromise between all these aspects