Foil Geometry and Design

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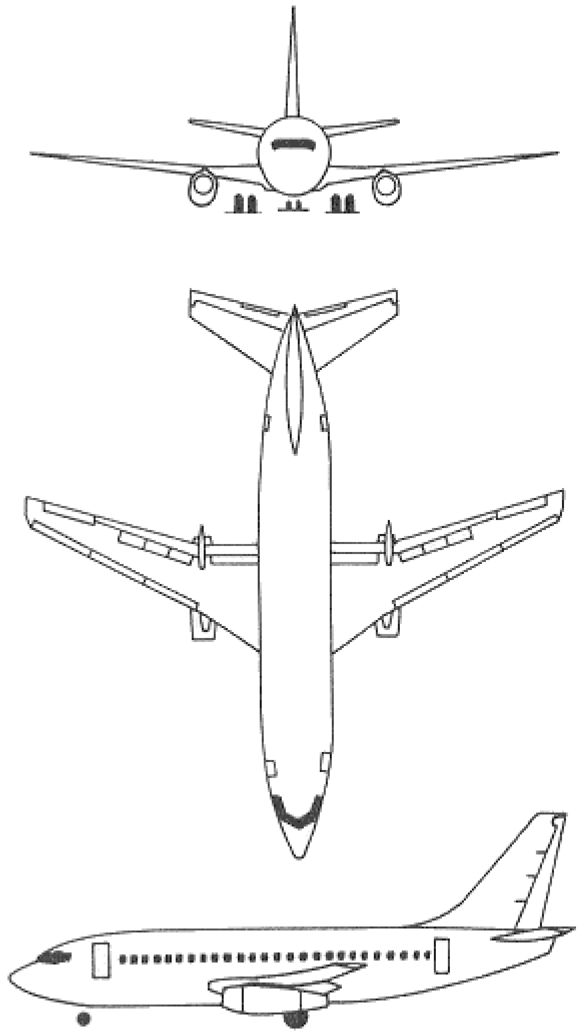
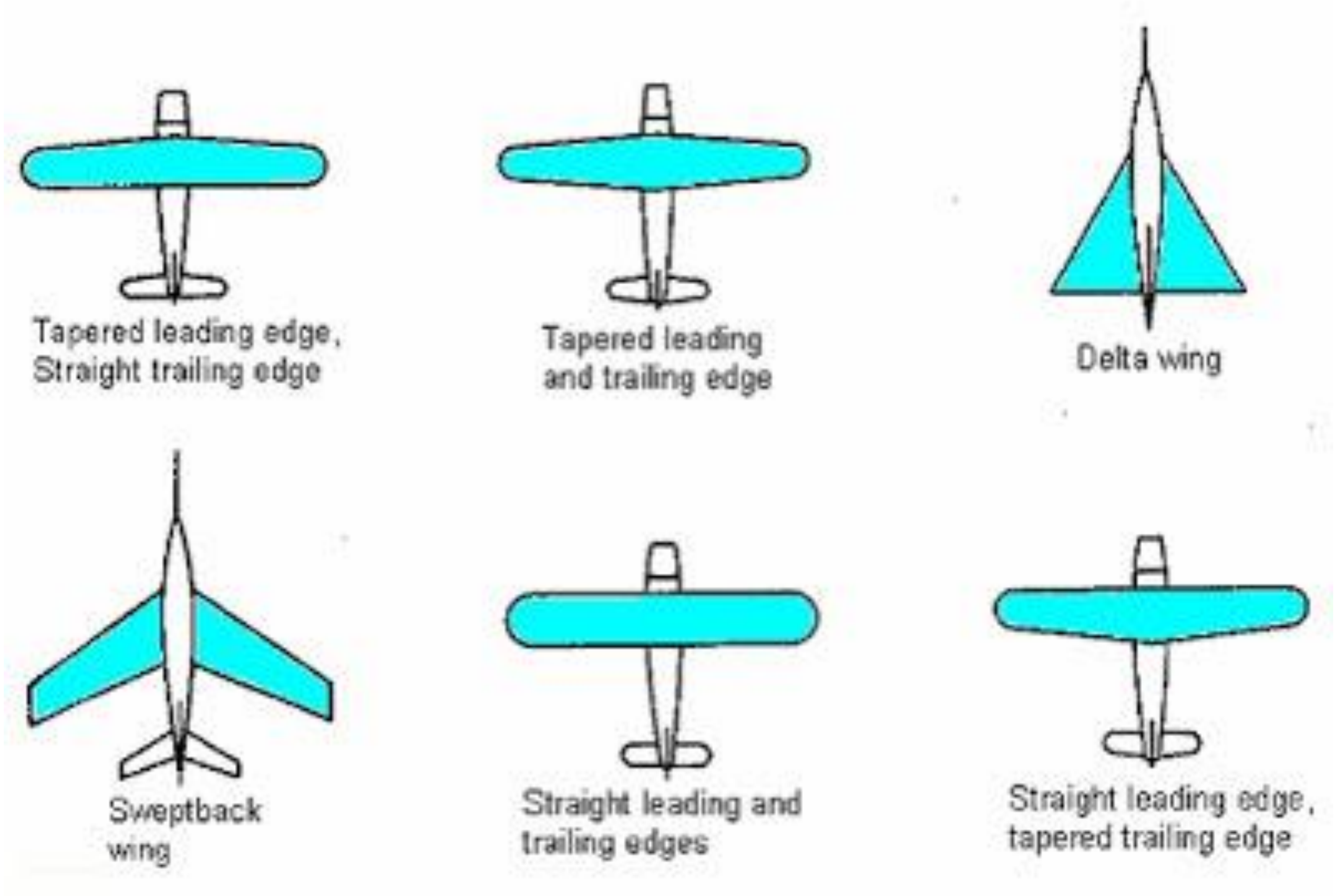
The Views



Typical Planform Geometries



Planform: General Categories

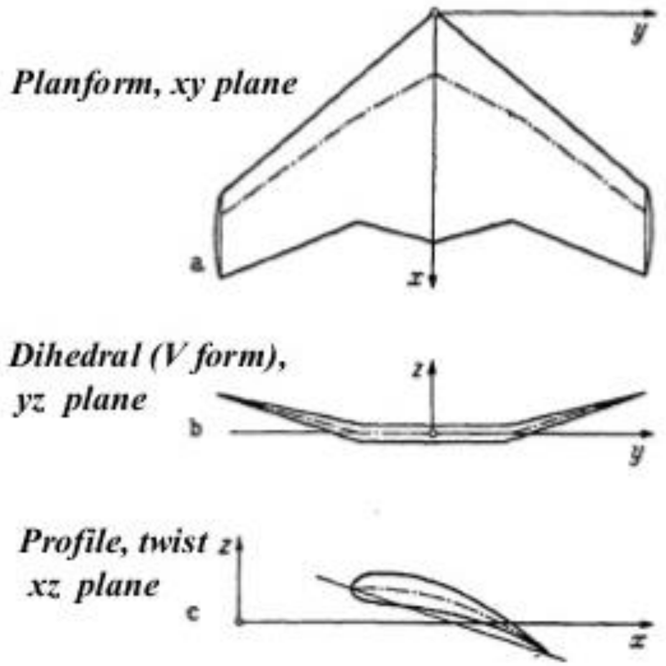
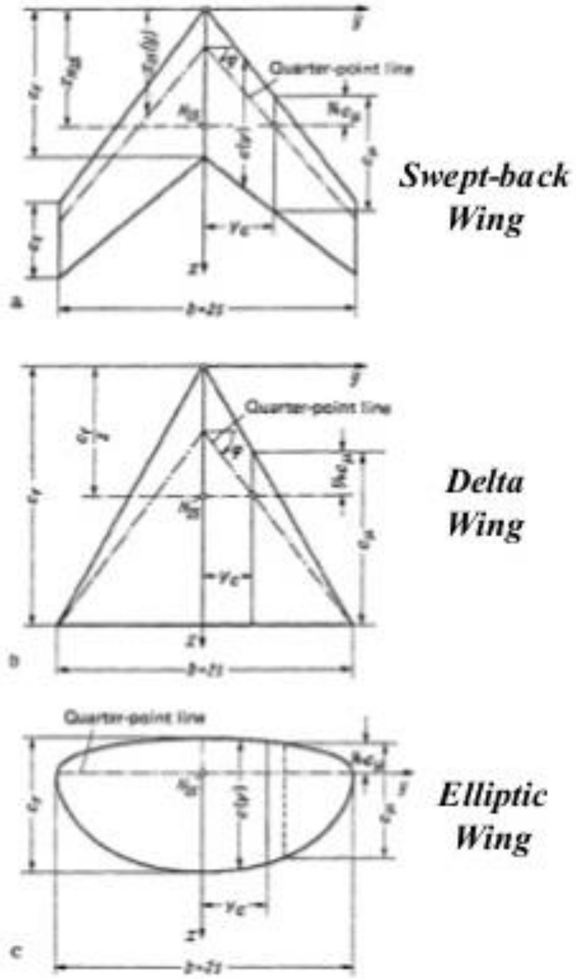
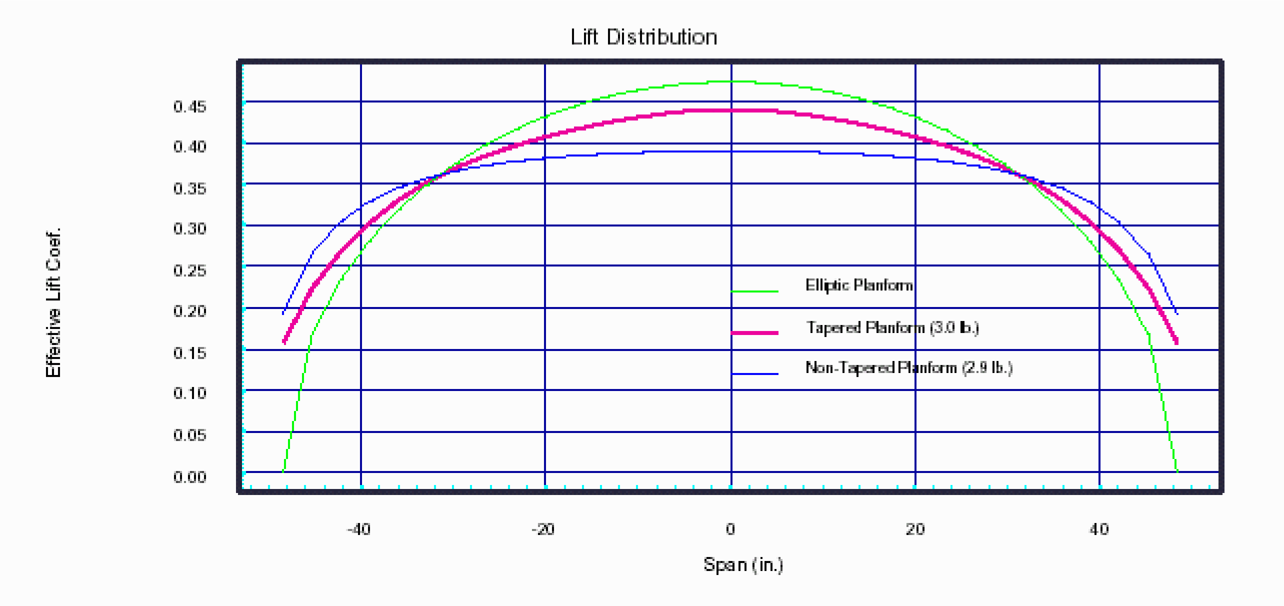


Illustration of Wing Geometry



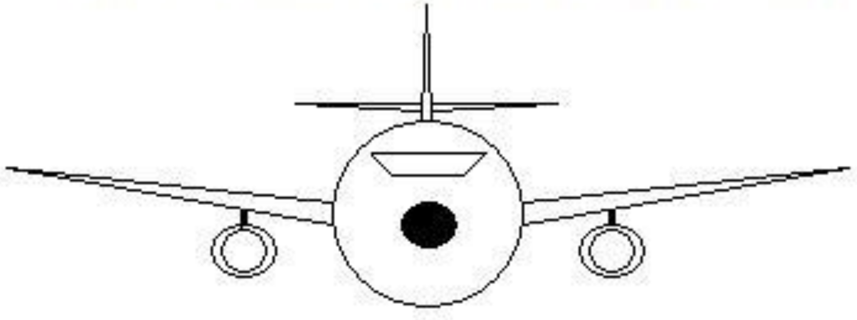
Planform: Force Distribution

$$C_D = C_{D_0} + \frac{(C_L)^2}{\pi e_0 AR}$$

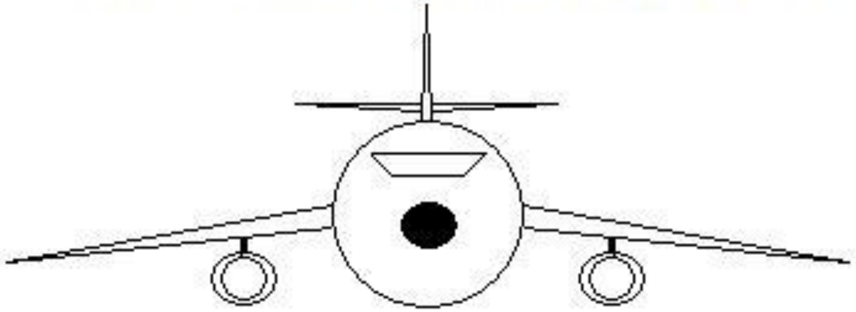


Hedral Angles: Self Stabilty

Front view of jet airliner with dihedral wings



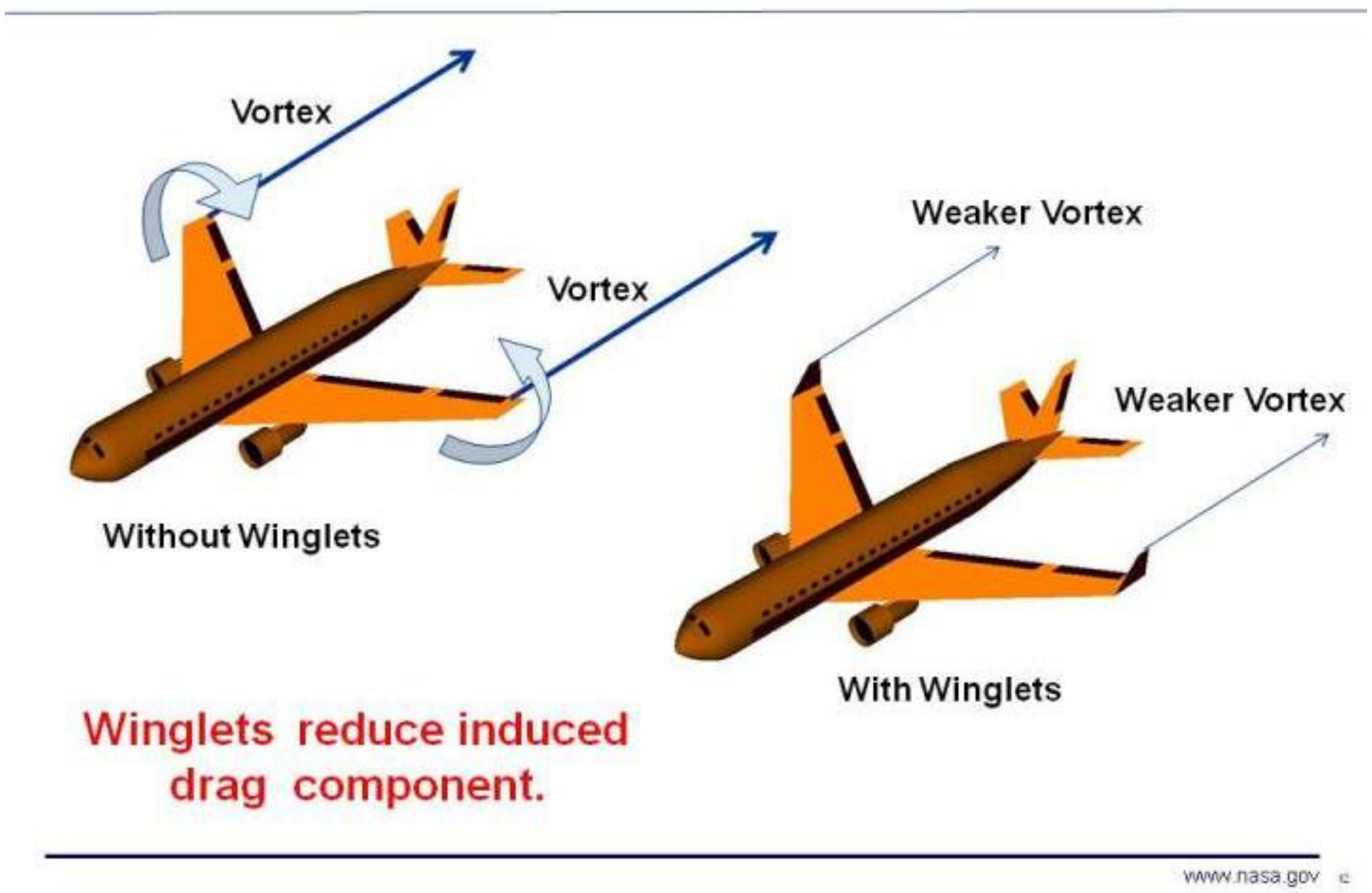
Front view of jet airliner with anhedral wings



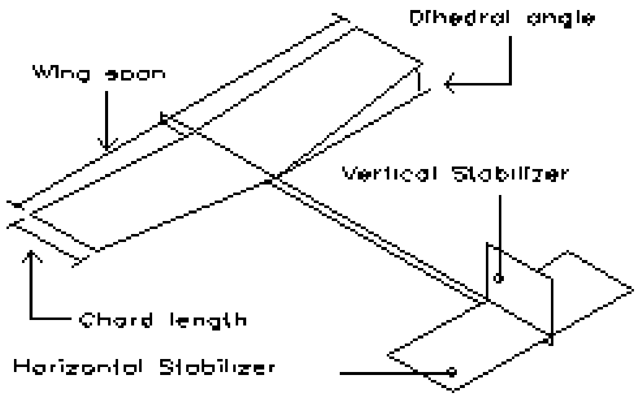
Front view of paper aeroplane with dihedral wings

Front view of paper aeroplane with anhedral wings

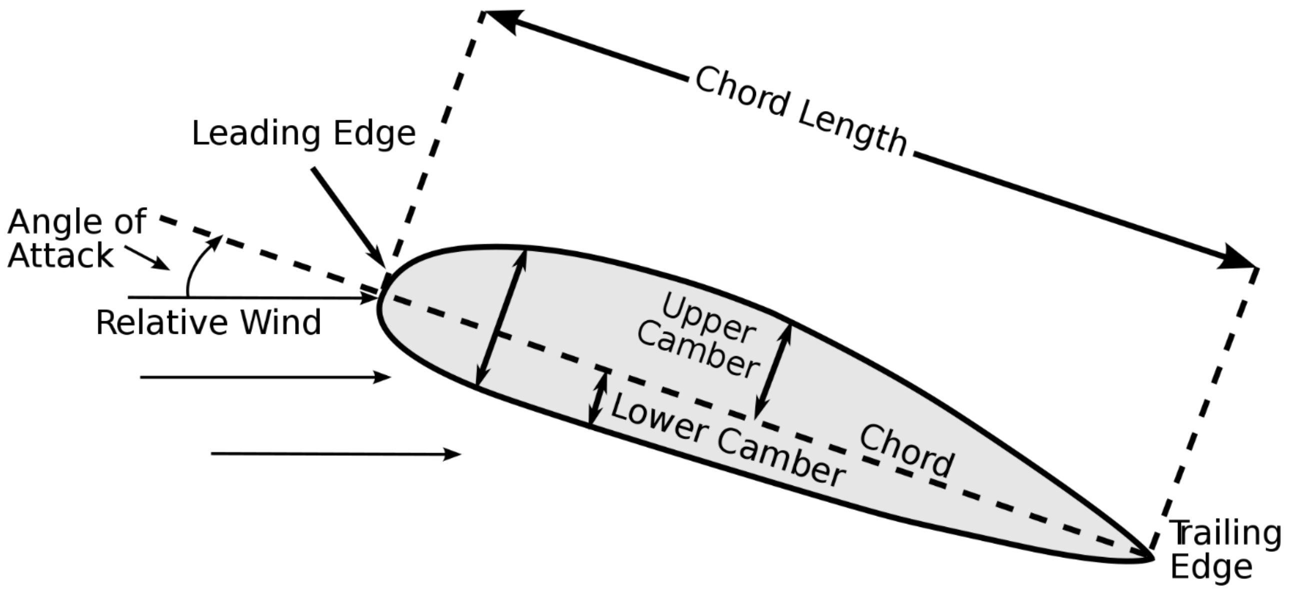
Finite Wings: Wingtip Vortices and Induced Drag



3D to 2D

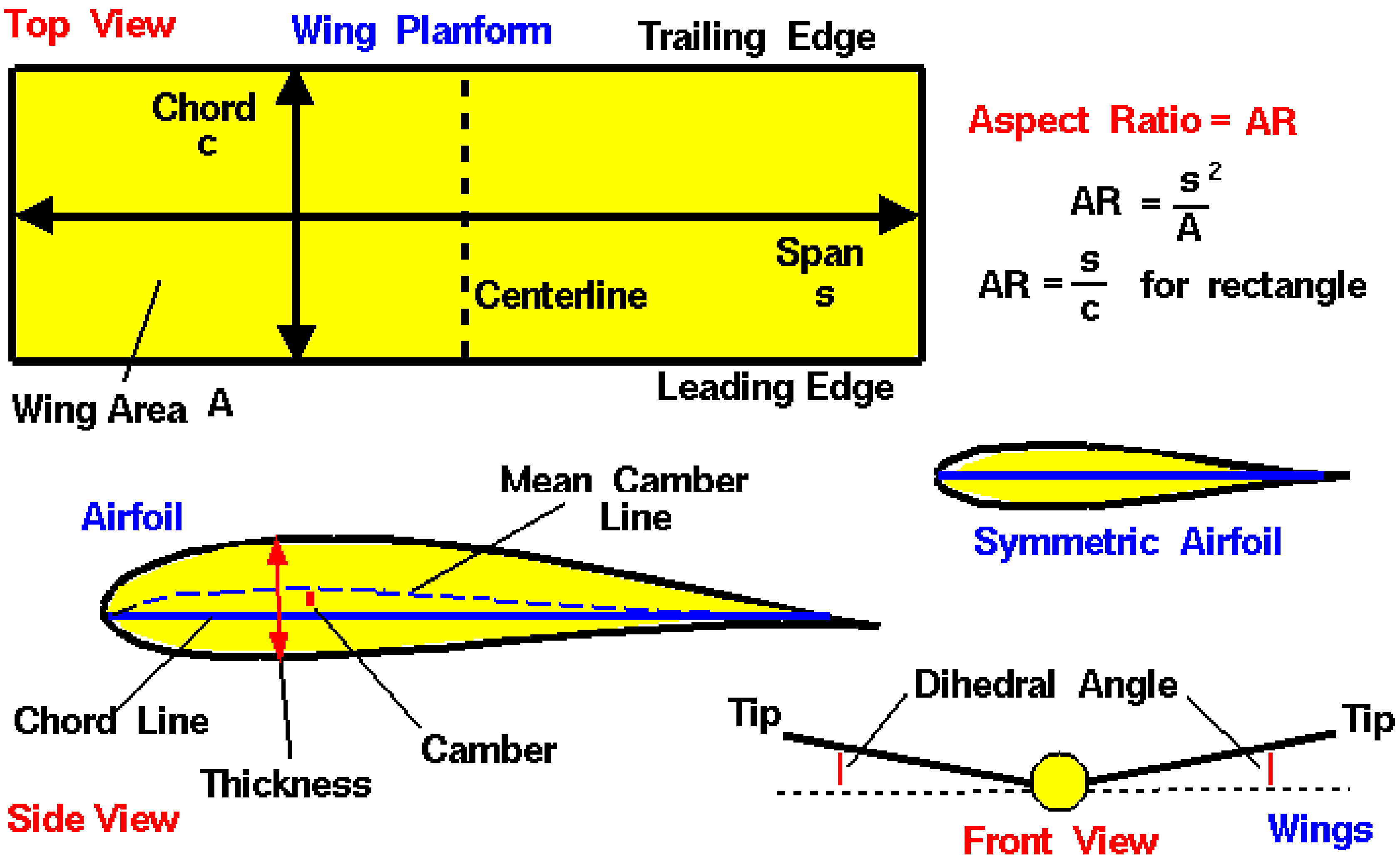


Dihedral angle
Wing span
Vertical Stabilizer
Chord length
Horizontal Stabilizer



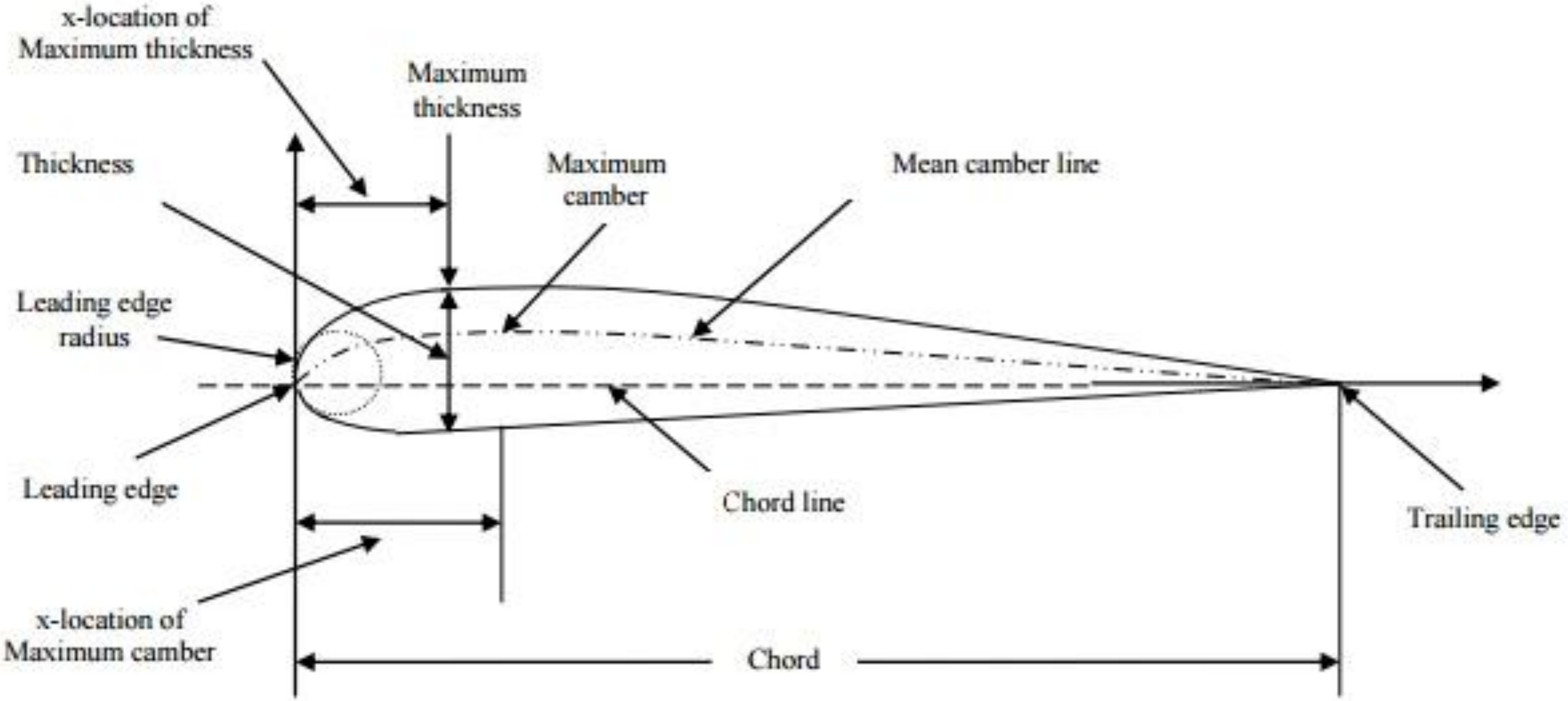
Chord Length
Leading Edge
Angle of Attack
Relative Wind
Upper Camber
Lower Camber
Chord
Trailing Edge

3D to 2D

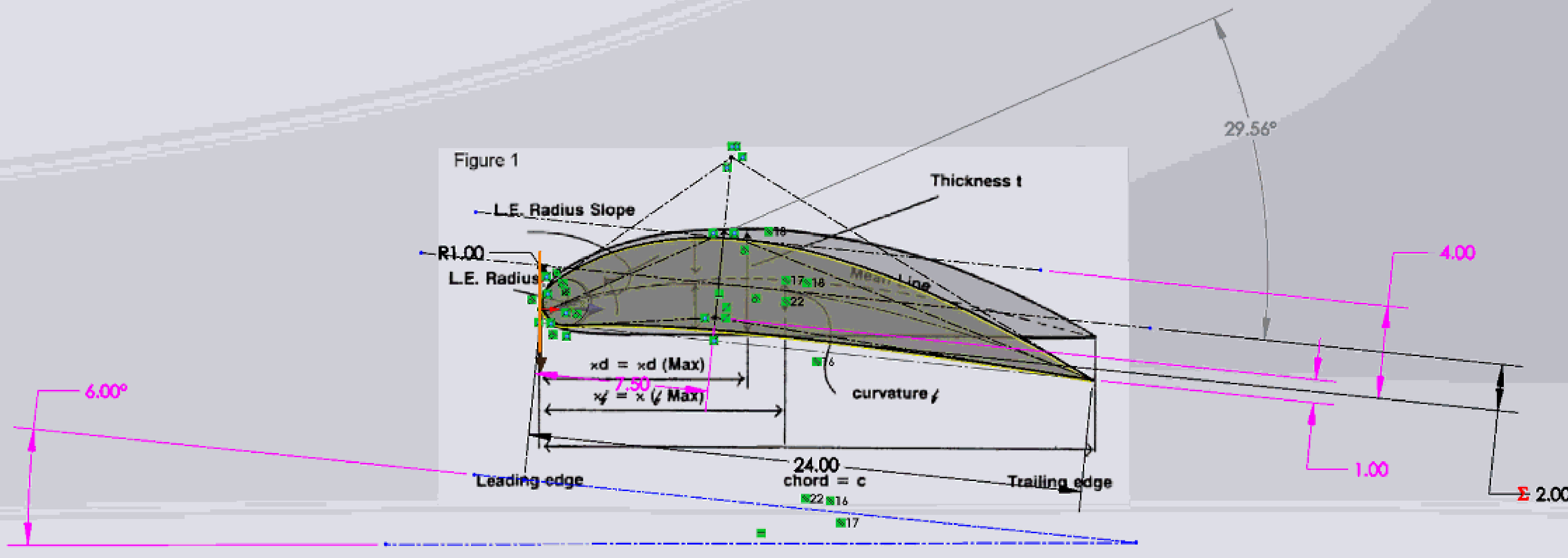


Top View
Wing Planform
Trailing Edge
Chord
c
Span
s
Centerline
Wing Area A
Leading Edge
Aspect Ratio = AR
AR = s^2 / A
AR = s / c for rectangle
Airfoil
Mean Camber Line
Symmetric Airfoil
Chord Line
Thickness
Camber
Side View
Dihedral Angle
Tip
Tip
Front View
Wings

The Foil Section



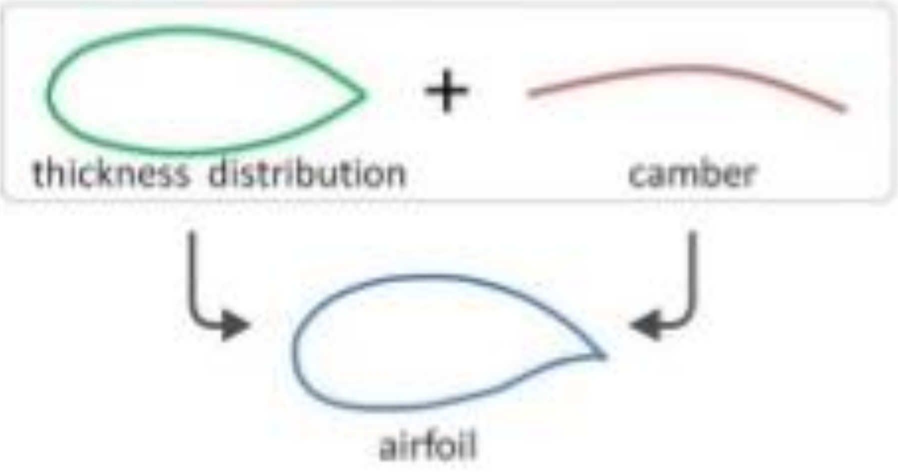
Parametric Modelling



Mathematical Representation: Foil Geometry

Parameterization Methods:

1. Class Shape Transformation
2. B- Splines
3. Hicks- Henne bump functions
4. Domain Element Method
5. Bezier Surfaces
6. Modal Extraction Methods
7. Parameterised Section Method
8. Mesh Points
9. Class Shape Transformation



NACA 4 Digit Series:

The first digit specifies the maximum camber (m) in percentage of the chord, the second indicates the position of the maximum camber (p) in tenths of chord, and the last two numbers provide the maximum thickness (t) of the airfoil in percentage of chord.

NACA 2415 airfoil has a maximum thickness of 15% with a camber of 2% located 40% back from the airfoil leading edge (or 0.4c).

Compute the mean camber line coordinates by plugging the values of m and p into the following equations for each of the x coordinates.

$$\pm y_t = \frac{t}{0.2}\left(0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4\right)$$

3.Calculate the thickness distribution above (+) and below (-) the mean line by plugging the value of t into the following equation for each of the x coordinates.

$$y_c = \frac{m}{p^2}\left(2px - x^2\right) \qquad \text{from } x = 0 \text{ to } x = p$$

$$y_c = \frac{m}{(1-p)^2}\left[(1-2p) + 2px - x^2\right] \qquad \text{from } x = p \text{ to } x = c$$

NACA 5 Digit Series

The first digit, when multiplied by 3/2, yields the design lift coefficient (c_l) in tenths. The next two digits, when divided by 2, give the position of the maximum camber (p) in tenths of chord.

The final two digits again indicate the maximum thickness (t) in percentage of chord.

The NACA 23012 has a maximum thickness of 12%, a design lift coefficient of 0.3, and a maximum camber located 15% back from the leading edge.

1. Compute the mean camber line coordinates for each x location using the following equations

$$y_c = \frac{k_1}{6}\left[x^3 - 3mx^2 + m^2(3-m)\,x\right] \qquad \text{from } x = 0 \text{ to } x = p$$

$$y_c = \frac{k_1\, m^3}{6}(1-x) \qquad \text{from } x = p \text{ to } x = c$$

2. Calculate the thickness distribution using the same equation as the Four-Digit Series.

3. Determine the final coordinates using the same equations as the Four-Digit Series.

NACA 1 Series

The 1-Series airfoils are identified by five digits, as exemplified by the NACA 16-212.

- The first digit, 1, indicates the series (this series was designed for airfoils with regions of barely supersonic flow).
- The 6 specifies the location of minimum pressure in tenths of chord, i.e. 60% back from the leading edge in this case.
- Following a dash, the first digit indicates the design lift coefficient in tenths (0.2)
- the final two digits specify the maximum thickness in tenths of chord (12%)
- Since the 16-XXX airfoils are the only ones that have ever seen much use, this family is often referred to as the 16-Series rather than as a subset of the 1-Series.

NACA 6 Series

NACA 64_1-212, a=0.6.

- 6 denotes the series and indicates that this family is designed for greater laminar flow than the Four- or Five-Digit Series.
- The second digit, 4, is the location of the minimum pressure in tenths of chord (0.4c).
- The subscript 1 indicates that low drag is maintained at lift coefficients 0.1 above and below the design lift coefficient (0.2) specified by the first digit after the dash in tenths.
- The final two digits specify the thickness in percentage of chord, 12%.
- The fraction specified by **'a'** indicates the percentage of the airfoil chord over which the pressure distribution on the airfoil is uniform, 60% chord in this case.
- If not specified, the quantity is assumed to be 1, or the distribution is constant over the entire airfoil.

NACA 7 Series

The 7-Series was a further attempt to maximize the regions of laminar flow over an airfoil differentiating the locations of the minimum pressure on the upper and lower surfaces.

NACA 747A315

- The 7 denotes the series, the 4 provides the location of the minimum pressure on the upper surface in tenths of chord (40%)
- 7 provides the location of the minimum pressure on the lower surface in tenths of chord (70%)
- The fourth character, a letter, indicates the thickness distribution and mean line forms used
- A series of standaradized forms derived from earlier families are designated by different letters.
- Again, the fifth digit incidates the design lift coefficient in tenths (0.3)
- the final two integers are the airfoil thickness in perecentage of chord (15%)

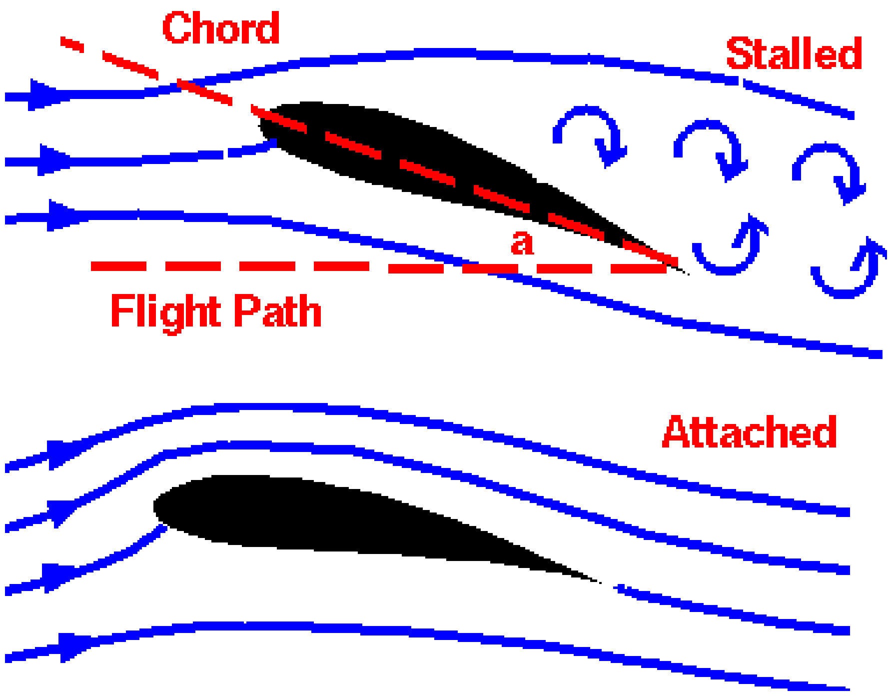
NACA 8 Series

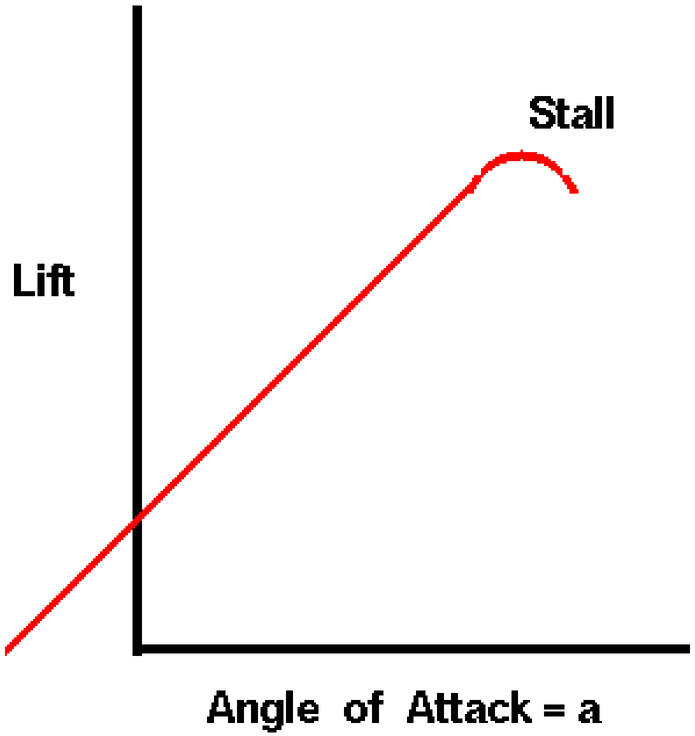
Like the earlier airfoils, the goal was to maximize the extent of laminar flow on the upper and lower surfaces independently.
The naming convention is very similar to the 7-Series

NACA 835A216

- The 8 designates the series, 3 is the location of minimum pressure on the upper surface in tenths of chord (0.3c)
- 5 is the location of minimum pressure on the lower surface in tenths of chord (50%)
- the letter A distinguishes airfoils having different camber or thickness forms
- 2 denotes the design lift coefficient in tenths (0.2)
- 16 provides the airfoil thickness in percentage of chord (16%).

Force Characteristics and Polars





For small angles, lift is related to angle.

Greater Angle = Greater Lift

For larger angles, the lift relation is complex.

Included in Lift Coefficient

Pressure Distribution and Forces: First Principles

Pressure forces act perpendicular to the surface.

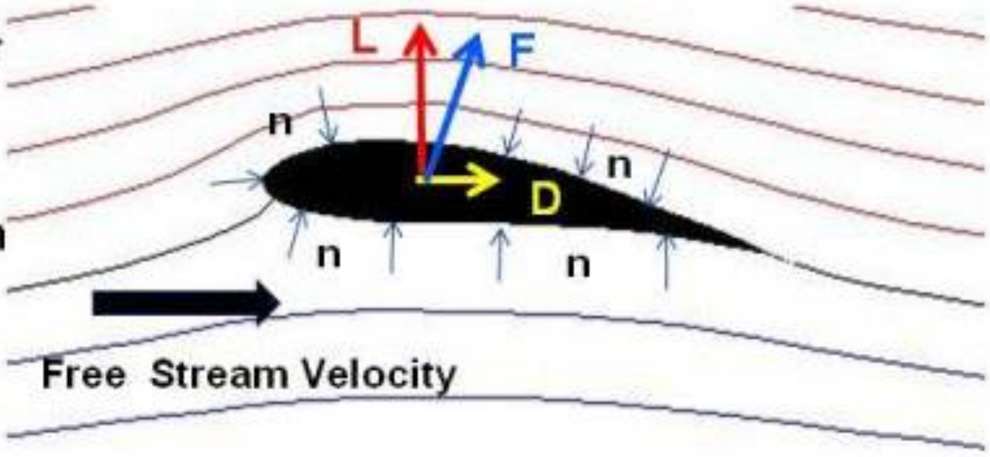
Force on the body is the vector sum of the pressure times the area around the body.

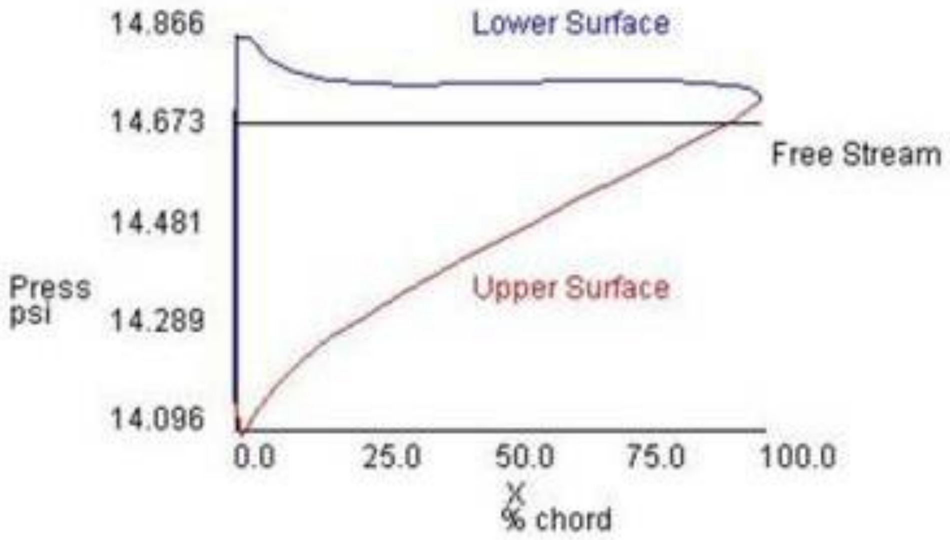
$$\vec{F} = \sum p \, \vec{n} \, \Delta A$$

$$\vec{F} = \oint p \, \vec{n} \, dA$$

$$\text{Lift} = L = F_{normal}$$

$$\text{Drag} = D = F_{streamwise}$$



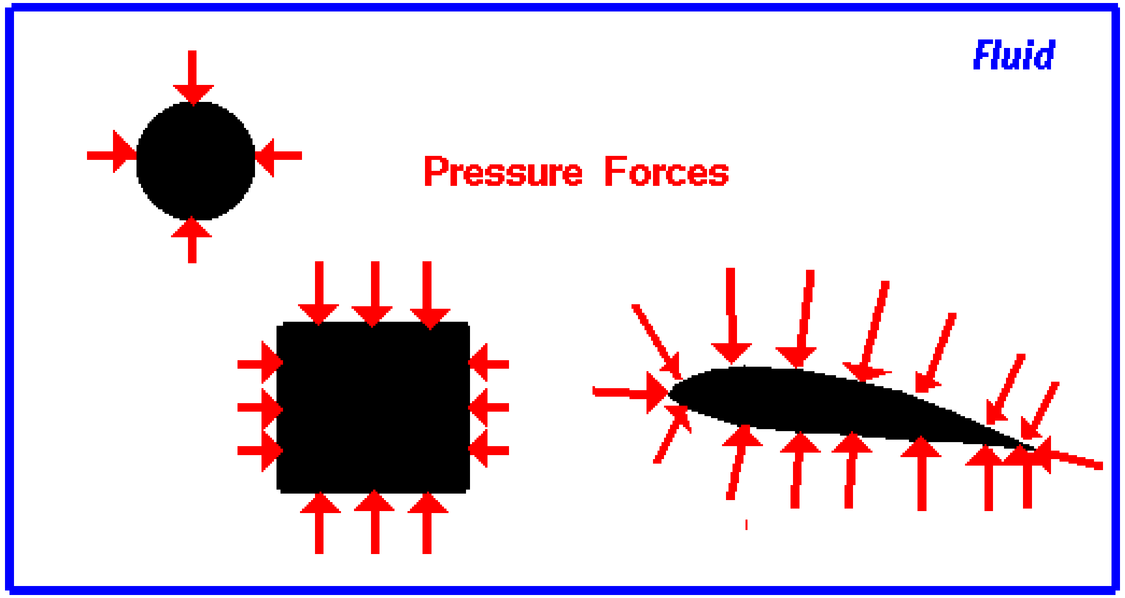


Choosing an aerofoil

Requirements: Parameters

1. Cl Lift Coefficient
2. Cd Drag Coefficient
3. Cm Pitching Moment Coeffic

- Cl = L /(q*A)
- Cd = D/(q*A)
- Cm = M/(q*A*c)



Pressure forces act normal (perpendicular) to surface.

Force on the body is the vector sum of the pressure x area around the entire solid body.

$$F = \sum_{\text{surface}} \vec{P}\,\Delta A = \oint \vec{P}\, dA$$

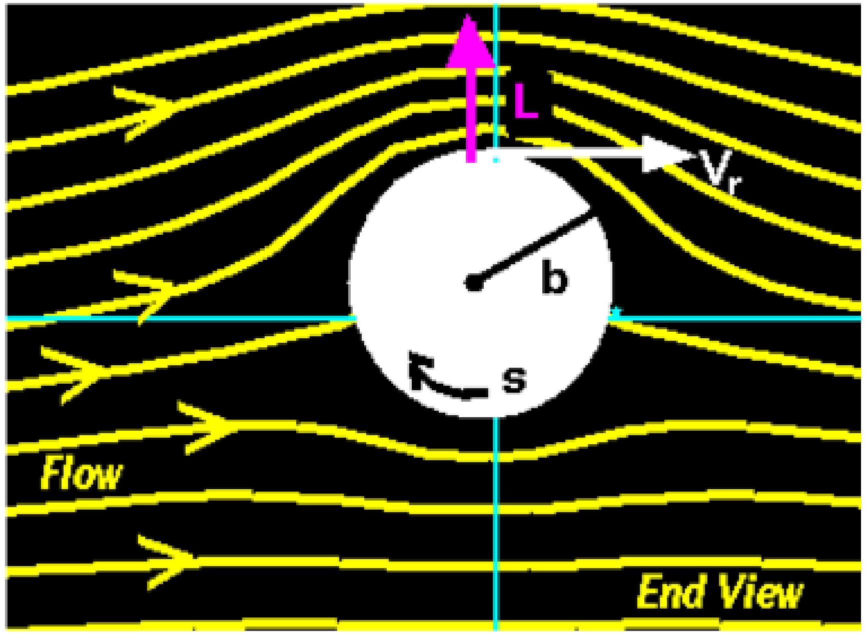
Choosing an aerofoil

Design Objectives/ Requirements:

- To reduce the drag at high speeds while trying to keep the maximum C_l greater than a certain value.
-
- This could involve slowly increasing the amount of laminar flow at low C_l's and checking to see the effect on the maximum lift.
- Minimize C_d with a constraint on C_{lmax}.
- Maximize L/D or $C_l^{1.5}/C_d$ or C_{lmax} / C_d @ $C_{ldesign}$.
- Avoid shocks
- High stall angle

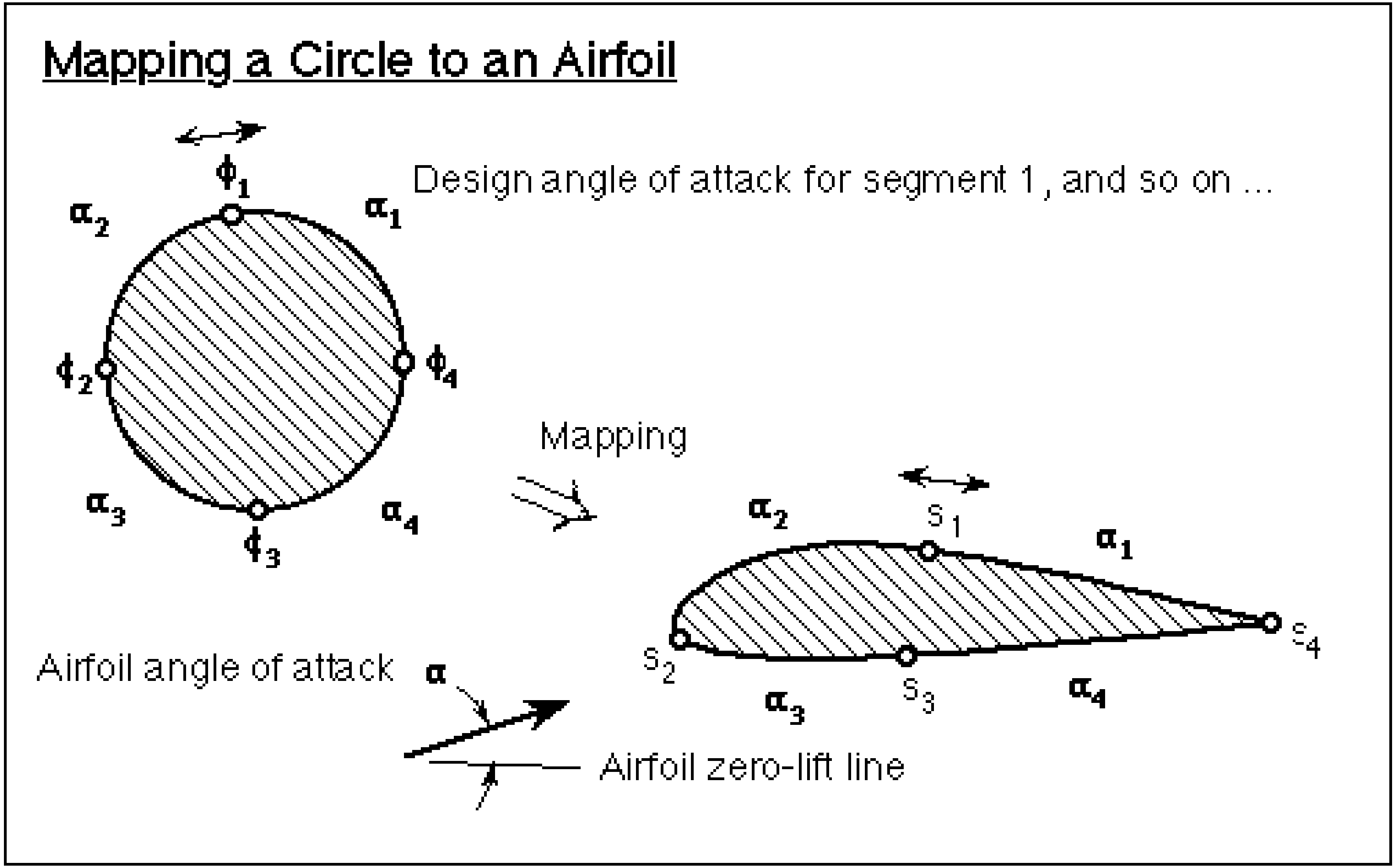
Math: Lift Theory

- Flow field can be divided into inviscid freestream and viscous boundary layer
- Potential flow around a cylinder is the basis of lift theory
- Potential flow around a stationary cylinder produces no lift, owing to its symmetry
- Kutta – Joukowski theorem– Lift is produced by a rotating cylinder in a potential flow field
- This is practically observed as an opposite (vectorially) vortex in the wake of a wing



- **Stream function** $\Psi = U_0 r sin\theta \left(1 - \frac{a^2}{r^2}\right)$
- **Velocity Potential** $\Phi = U_0 r cos\theta \left(1 + \frac{a^2}{r^2}\right)$
- **Complex field** $Z = re^{i\theta} = X + iY$ $F = \Phi + i\Psi$

Conformal Mapping: Circle to Airfoil



Performance Parameters

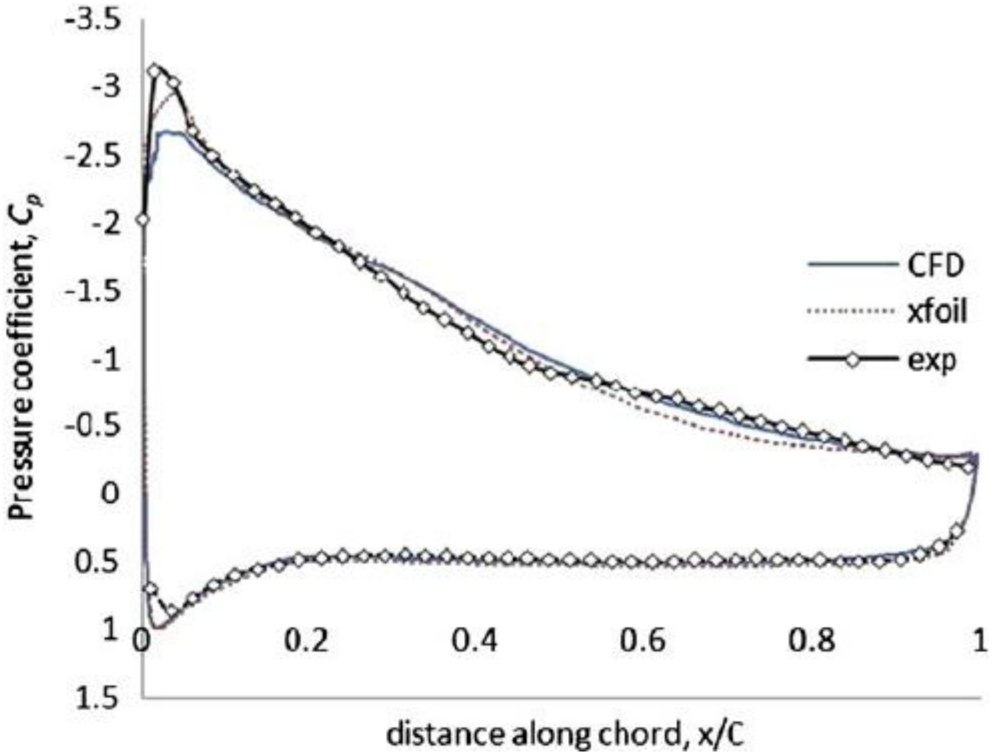
Evaluating Performance Parameters:

From the pressure distribution

$$C_N = \int C_{pl}\, d(X/C) - \int C_{pu}\, d(X/C)$$

$$C_L = C_N \cos \theta - C_C \sin \theta$$

Similarly, Cd



Methods to produce Cp – (x/c) curve:

1. CFD
2. Panel Methods
3. Wind Tunnel Testing

Thank you!

See you in the afternoon!