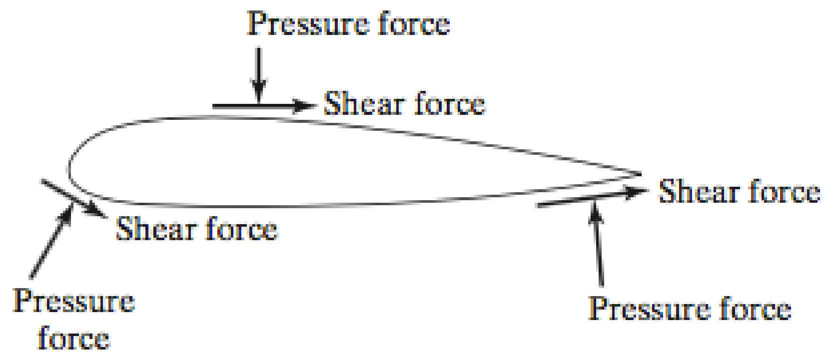
SPC 307
Introduction to Aerodynamics

Lecture 8
Airfoil Parameters

April 30, 2017

Figure 5.2 Normal (or pressure) and tangential (or shear) forces on an airfoil surface.



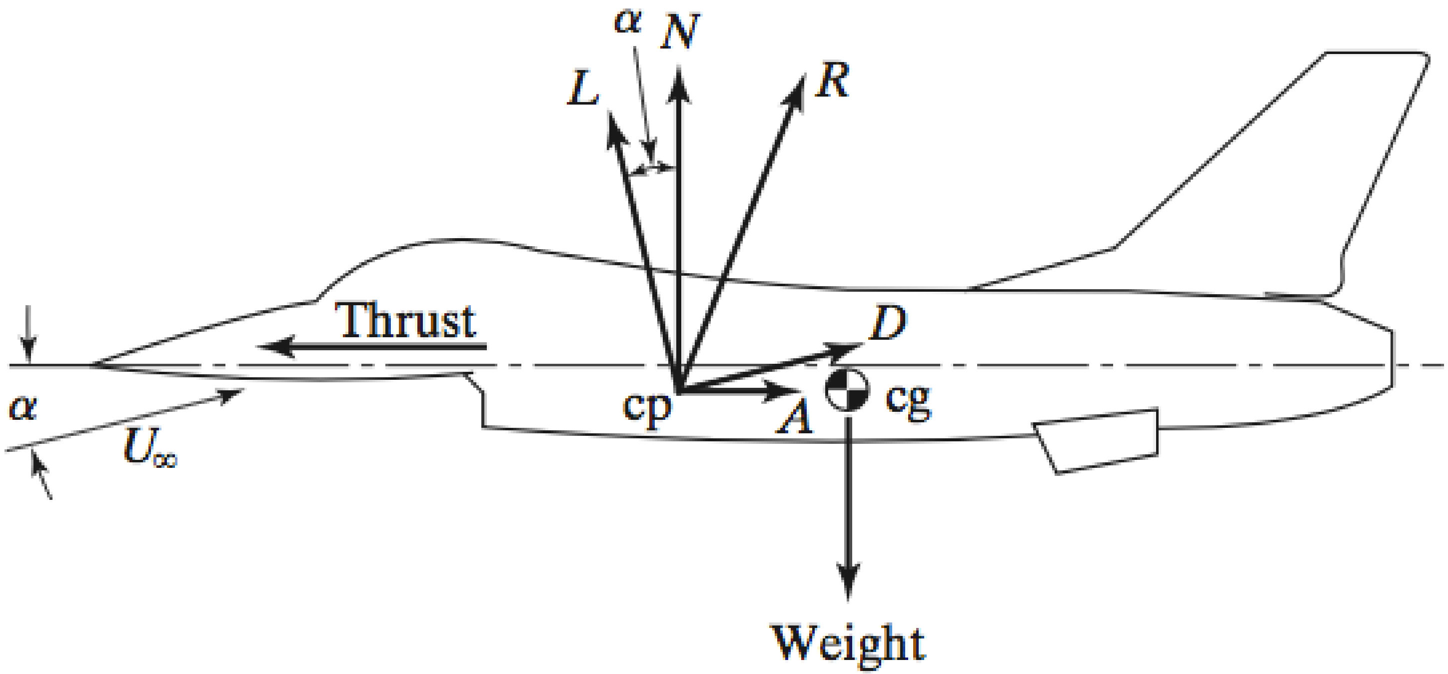


Figure 5.3 Nomenclature for aerodynamic forces in the pitch plane.

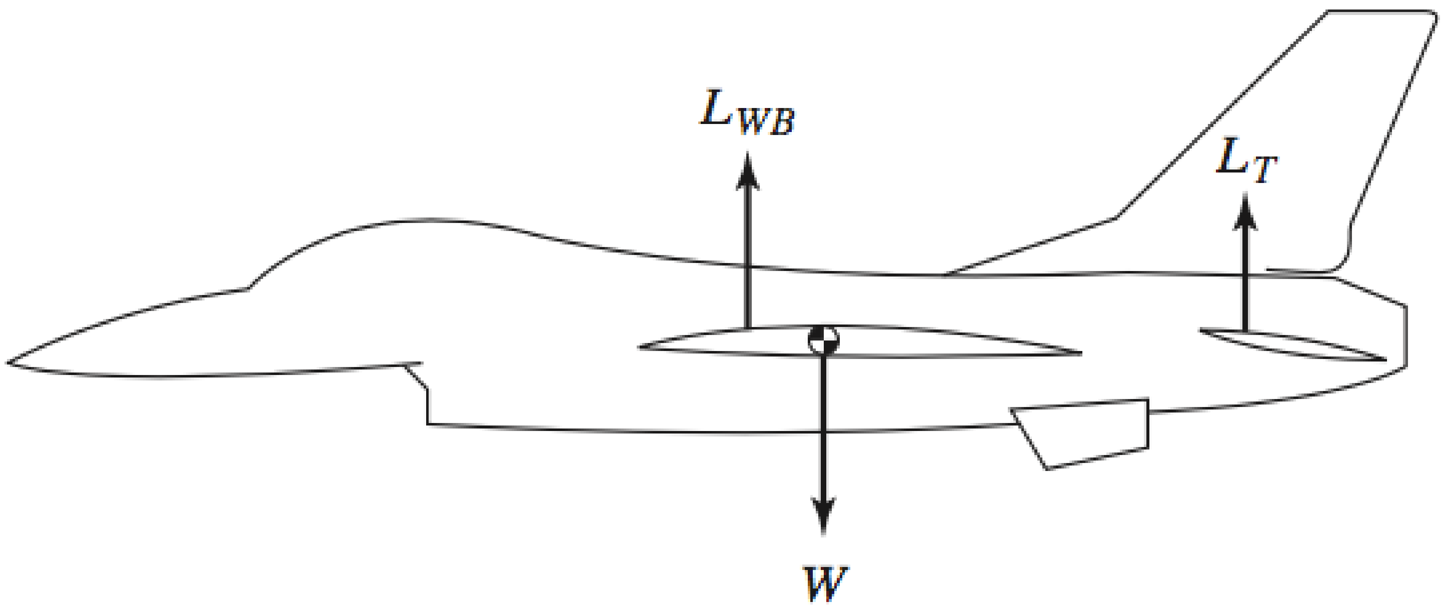


Figure 5.4 Moment balance to trim an aircraft.

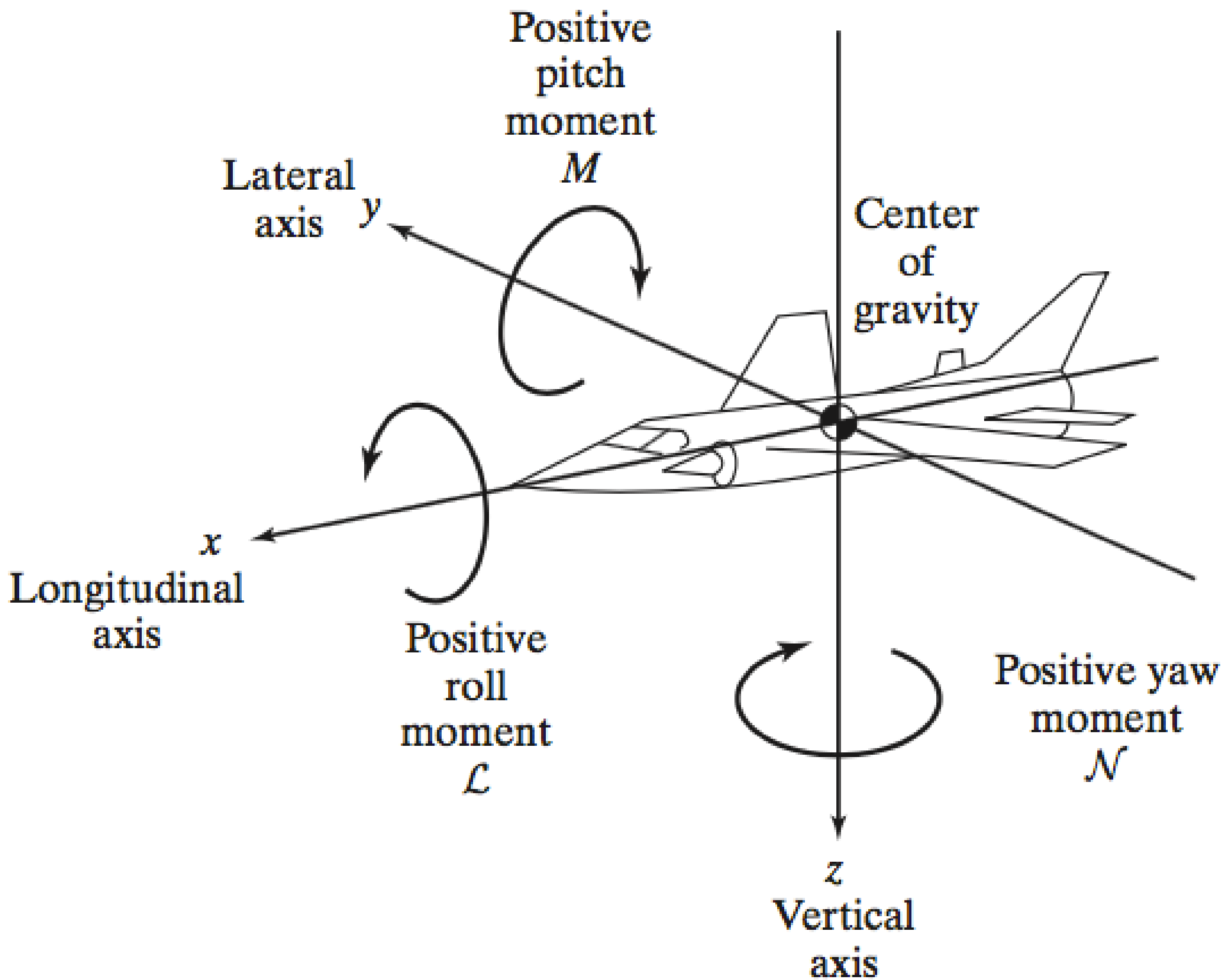
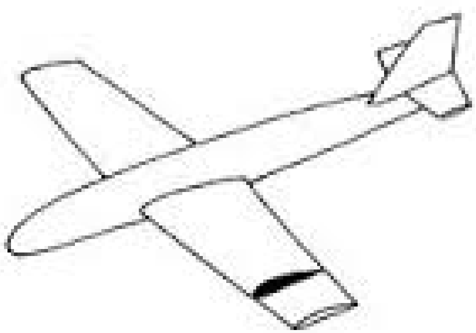


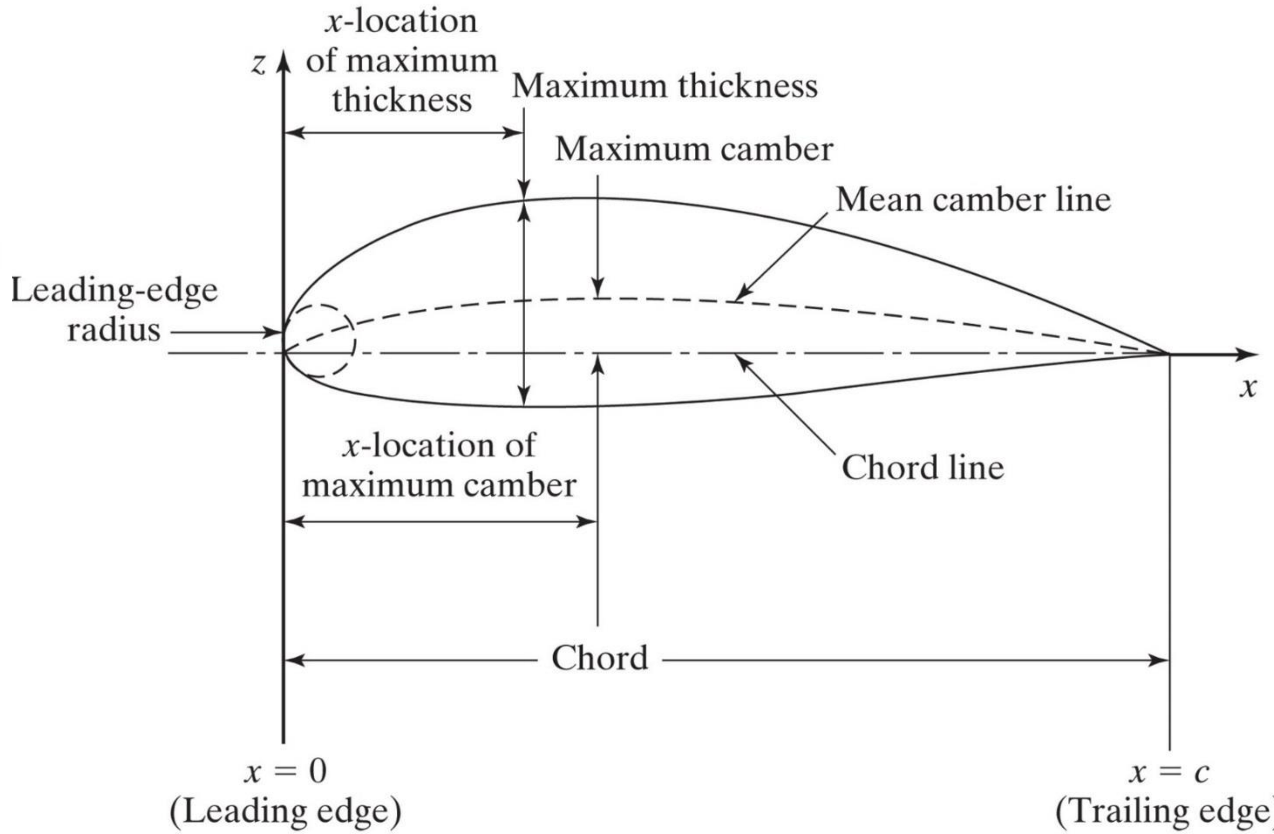
Figure 5.5 Reference axes of the airplane and the corresponding aerodynamic moments.

Parameters That Govern Aerodynamic Forces

- Configuration geometry
- Angle of attack (i.e., vehicle attitude in the pitch plane relative to the flight direction)
- Vehicle size or model scale
- Free-stream velocity
- Density of the undisturbed air
- Reynolds number (as it relates to viscous effects)
- Mach number (as it relates to compressibility effects)

AIRFOIL GEOMETRY PARAMETERS





AIRFOIL GEOMETRY PARAMETERS

- If a horizontal wing is cut by a vertical plane, the resultant section is called the *airfoil section* .
- The generated lift and stall characteristics of the wing depend strongly on the geometry of the airfoil sections that make up the wing.
- Geometric parameters that have an important effect on the aerodynamic characteristics of an airfoil section include
  - (1) the leading-edge radius,
  - (2) the mean camber line,
  - (3) the maximum thickness and the thickness distribution of the profile, and
  - (4) the trailing-edge angle.
- The effect of these geometric parameters, will be discussed after an introduction to airfoil-section nomenclature.

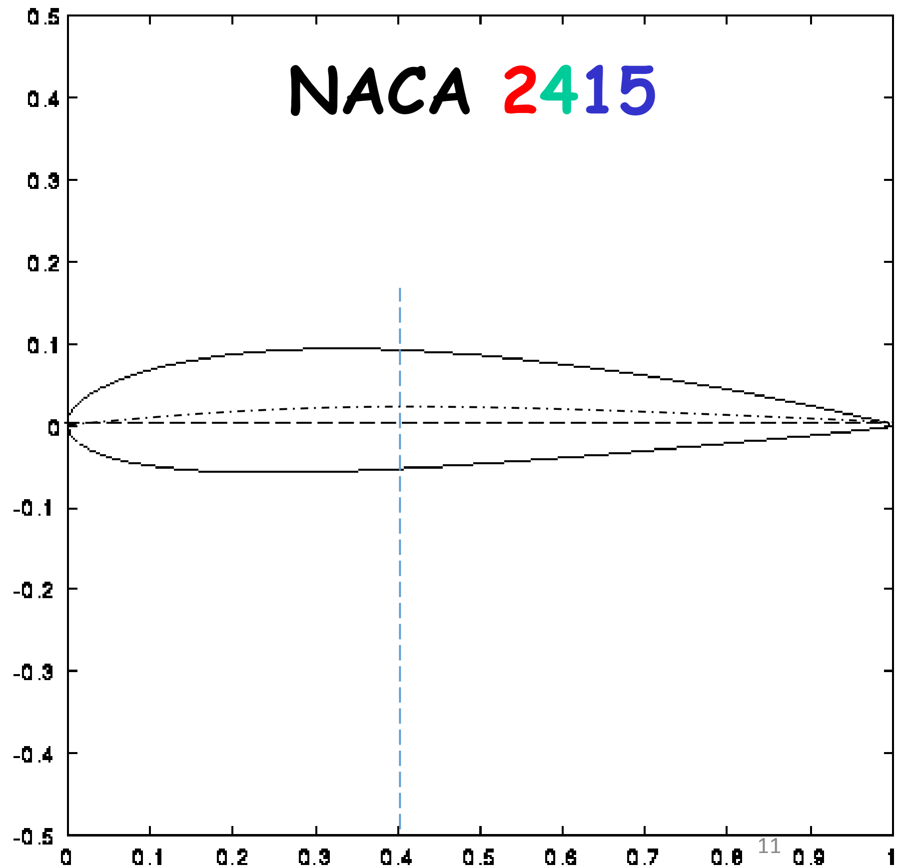
Parameters used to describe the airfoil

- Some of the basic parameters to describe the airfoil geometry are:
    1. Leading edge—the forward most point on the airfoil (typically placed at the origin for convenience)
    2. Trailing edge—the aft most point on the airfoil (typically placed on the *x* axis for convenience)
    3. Chord line—a straight line between the leading and trailing edges (the *x* axis for our convention)
    4. Mean camber line—a line midway between the upper and lower surfaces at each chord-wise position
    5. Maximum camber—the largest value of the distance between the mean camber line and the chord line, which quantifies the camber of an airfoil
    6. Maximum thickness—the largest value of the distance between the upper and lower surfaces, which quantifies the thickness of the airfoil
    7. Leading-edge radius—the radius of a circle that fits the leading-edge curvature
- These geometric parameters are used to determine certain aerodynamic characteristics of an airfoil.

- The geometry of many airfoil sections is uniquely defined by the NACA designation for the airfoil.
- There are a variety of NACA airfoil classifications, including NACA four-digit wing sections, NACA five-digit wing sections, and NACA six-series wing sections, among others.
- As an example, consider the NACA four-digit wing sections, which are designate as: NACA XYZZ
- The first integer, X, indicates the maximum value of the mean camber-line ordinate in percent of chord.
- The second integer, Y, indicates the distance from the leading edge to the maximum camber location in tenths of chord.
- The last two integers, ZZ, indicate the maximum section thickness in percent of chord.
- Therefore, the NACA 0010 is a symmetric airfoil section whose maximum thickness is 10% of the chord (the two leading “0” digits indicate that there is no camber, so the airfoil is symmetric about the *x* axis).
- The NACA 4412 airfoil section is a 12% thick airfoil which has a 4% maximum camber located at 4/10ths (40%) of the chord.

NACA FOUR-DIGIT SERIES

- **First digit** specifies maximum camber in percentage of chord
- **Second digit** indicates position of maximum camber in tenths of chord
- **Last two digits** provide maximum thickness of airfoil in percentage of chord



Example: **NACA 2415**

- Airfoil has maximum thickness of 15% of chord (0.15c)
- Camber of 2% (0.02c) located 40% back from airfoil leading edge (0.4c)

NACA Four-Digit Series

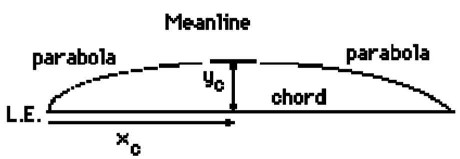
- Earliest NACA airfoils were designated as four-digit series.
- The thickness distribution is given as :

$$\pm y_t = \frac{t}{20}\left[0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4\right]$$

- where, t = maximum thickness as fraction of chord.
- The leading radius is : $r_t = 1.1019\ t^2$.
- The camber line for the four-digit series airfoils consists of two parabolic arcs tangent at the point of maximum ordinate.
- The expressions for camber(y_c) are :

$$y_c = \frac{m}{p^2}\left(2px - x^2\right);\ x \le x_{ycmax}$$

$$= \frac{m}{(1-p)^2}\left[(1-2p) + 2px - x^2\right];\ x > x_{ycmax}$$



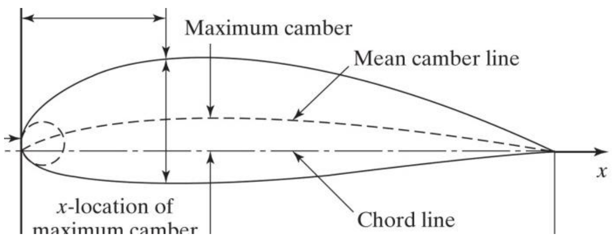
- m = maximum ordinate of camber line as fraction of chord
- p = chord-wise position of maximum camber as fraction of chord

NACA Five-Digit Series

- During certain tests it was observed that C_{lmax} of the airfoil could be increased by shifting forward the location of the maximum camber.
- This finding led to development of five-digit series airfoils.
- The new camber lines for the five-digit series airfoils are designated by three digits.
- The camber line shape is given as :

$$y_c = \frac{1}{6}k_1\left[x^3 - 3mx^2 + m^2(3-m)x\right], 0 < x \le m$$

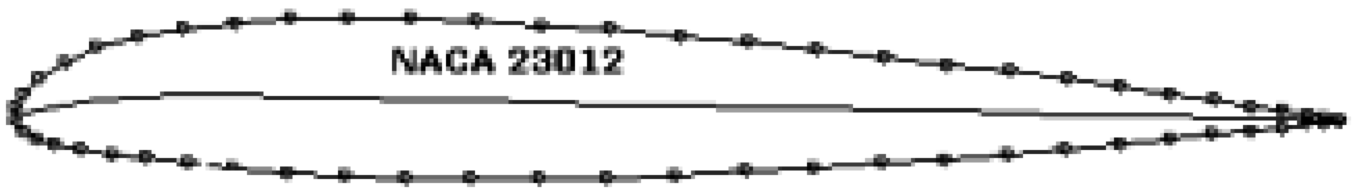
$$= \frac{1}{6}k_1 m^3 [1-x]; m < x < 1$$



- The value of 'm' decides the location of the maximum camber and that of k_1 the design lift coefficient(C_{li} or C_{lopt}).
- The same thickness distribution was retained as that for NACA four-digit series airfoils.

NACA Five-Digit Series

- The NACA Five-Digit Series uses the same thickness forms as the Four-Digit Series but the mean camber line is defined differently and the naming convention is a bit more complex.
  - ➢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.
- For example, 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.



Standard modifications to the airfoils

- A series of "standard" modifications to the airfoils were designated by the NACA using a suffix consisting of a dash followed by two digits.
- These modifications consist essentially of:

  (1) changes of the leading-edge radius from the normal value,

  (2) changes of the position of maximum thickness from the normal position (which is at 0.3 *c*).

  - NACA 0010-64

- The first integer (6) indicates the relative magnitude of the leading-edge radius (normal leading-edge radius is "6"; sharp leading edge is "0").
- The second integer (4) of the modification indicates the location of the maximum thickness in tenths of chord.

Modified NACA Four- and Five-Digit Series

- Another example, the NACA 0003.46-64.069.
- The basic shape is the 0003, a 3% thick airfoil with 0% camber.
- This shape is a symmetrical airfoil that is identical above and below the mean camber line.
- The first modification we will consider is the 0003-64.
- The first digit following the dash refers to the roundedness of the nose.
- A value of 6 indicates that the nose radius is the same as the original airfoil while a value of 0 indicates a sharp leading edge.
- Increasing this value specifies an increasingly more rounded nose.
- The second digit determines the location of maximum thickness in tenths of chord.
- The default location for all four- and five-digit airfoils is 30% back from the leading edge.

Modified NACA Four- and Five-Digit Series

- In this example, the location of maximum thickness has been moved back to 40% chord.
- Finally, notice that the 0003.46-64.069 features two sets of digits preceeded by decimals.
- These merely indicate slight adjustments to the maximum thickness and location thereof.
- Instead of being 3% thick, this airfoil is 3.46% thick.
- Instead of the maximum thickness being located at 40% chord, the position on this airfoil is at 40.69% chord.
- NACA six-series
- The NACA six-series airfoils were, in general, designed to maximize the amount of laminar flow on the section, and were not designed purely as a geometric family.

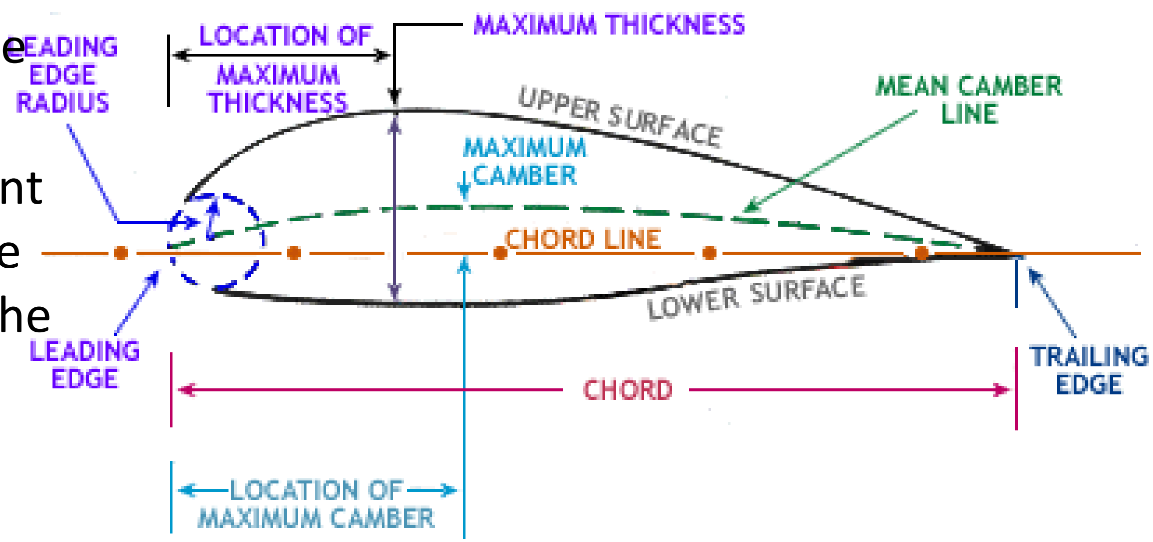
Leading-Edge Radius and Chord Line

- The *chord line* is defined as the straight line connecting the leading and trailing edges.
- The leading edge of airfoils used in subsonic applications is rounded, with a radius that is on the order of 1% of the chord length.
- The leading-edge radius is the radius of a circle centered on a line tangent to the leading-edge camber connecting tangency points of the upper and the lower surfaces with the leading edge.
- The center of the leading-edge radius is located so that the cambered section projects slightly forward of the leading-edge point.
- The magnitude of the leading-edge radius has a significant effect on the stall (or boundary-layer separation) characteristics of the airfoil section. Specifically, if the leading-edge radius is too small, the flow will have a tendency to separate near the leading edge, causing fairly abrupt stall characteristics for the airfoil.
- The geometric angle of attack is the angle between the chord line and the direction of the “free stream” flow. For many airplanes the chord lines of the airfoil sections are inclined relative to the vehicle axis, which is called *incidence* . Incidence is used to create lift while the airplane is on the runway, which decreases take-off distance, among other benefits.

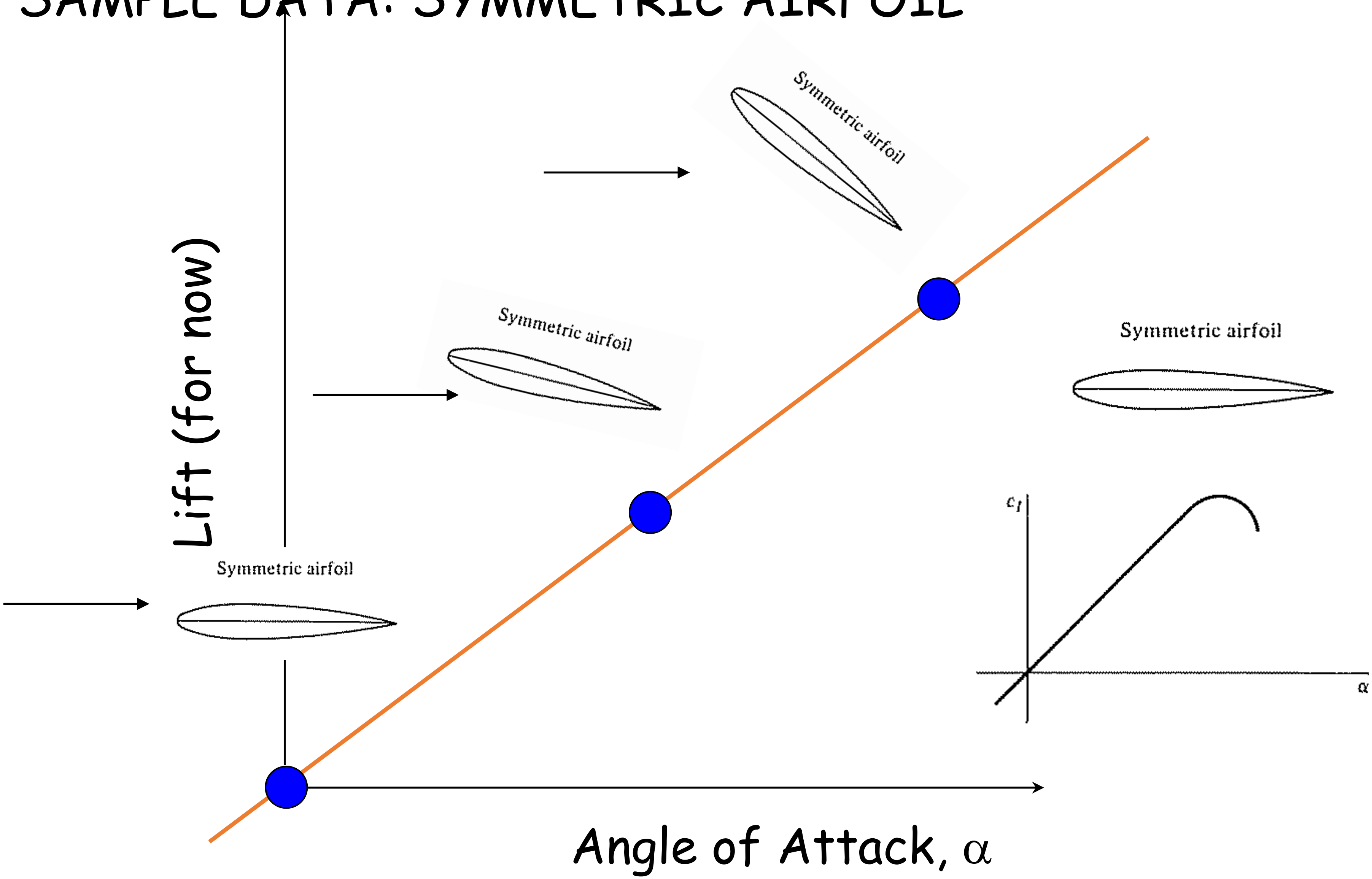
Mean Camber Line

- The locus of the points midway between the upper surface and the lower surface, as measured perpendicular to the chord line, defines the *mean camber line* .
- The shape of the mean camber line is very important in determining the aerodynamic characteristics of an airfoil section.
- Cambered airfoils in a subsonic flow generate lift even when the section angle of attack is zero.

- Camber has a beneficial effect on the maximum value of the section lift coefficient, $C_{l\max}$.
- If the maximum lift coefficient is high, the stall speed will be low, all other factors being the same.

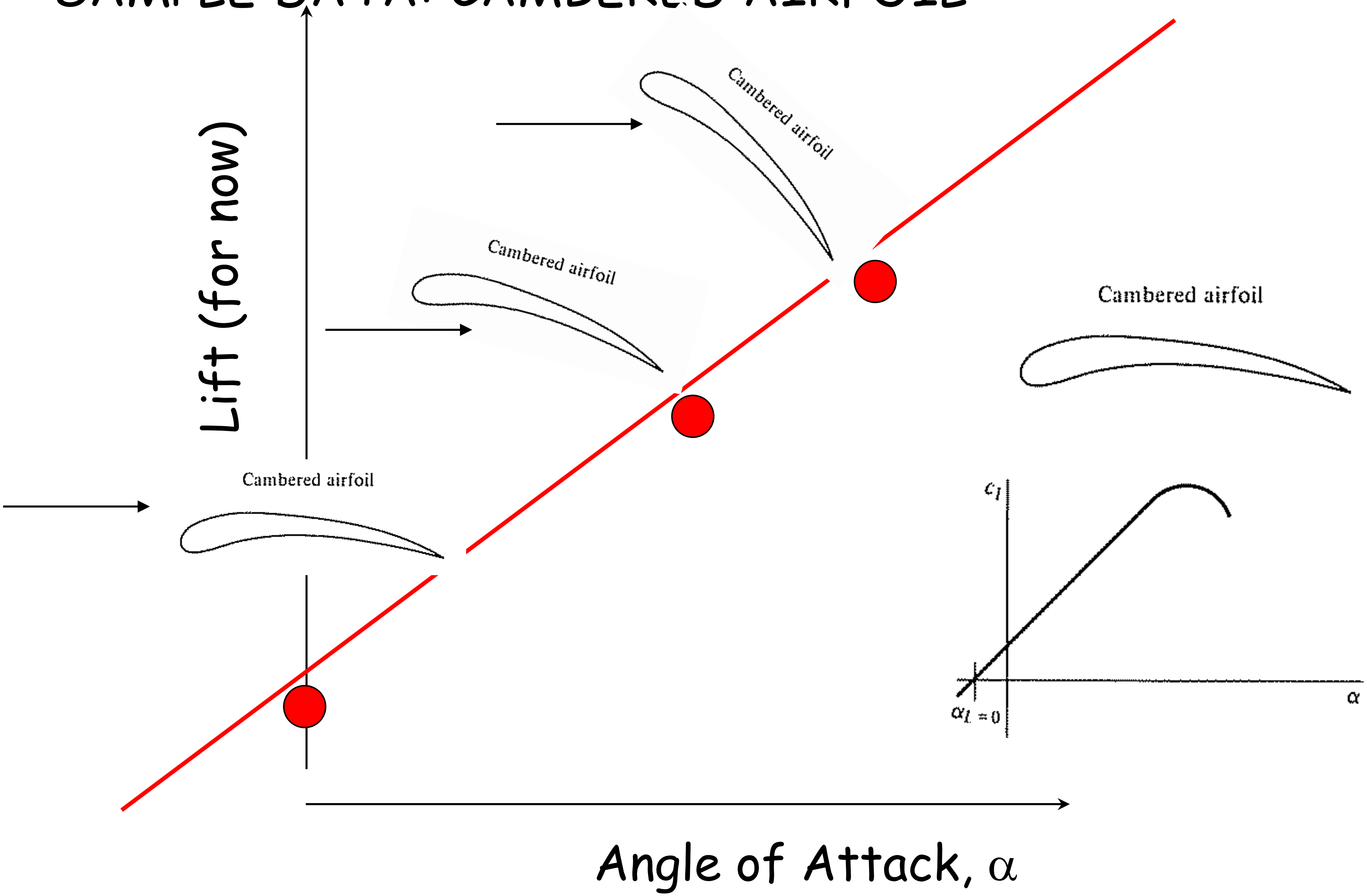


SAMPLE DATA: SYMMETRIC AIRFOIL



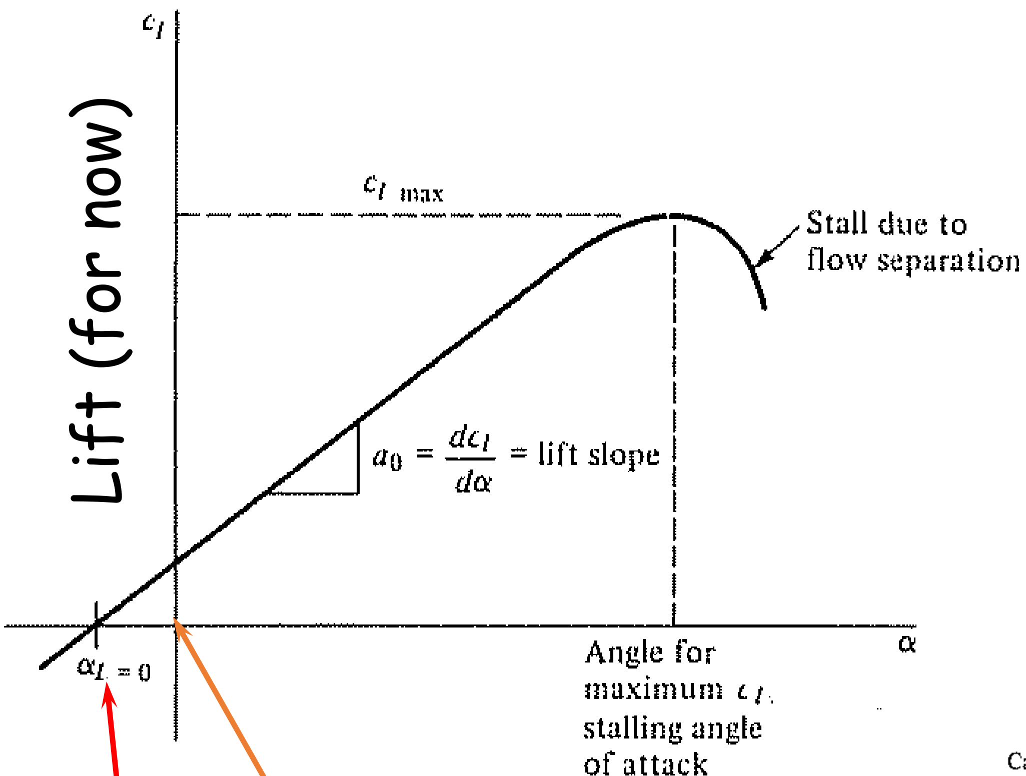
A symmetric airfoil generates zero lift at zero angle of attack

SAMPLE DATA: CAMBERED AIRFOIL



A cambered airfoil generates positive lift at zero α

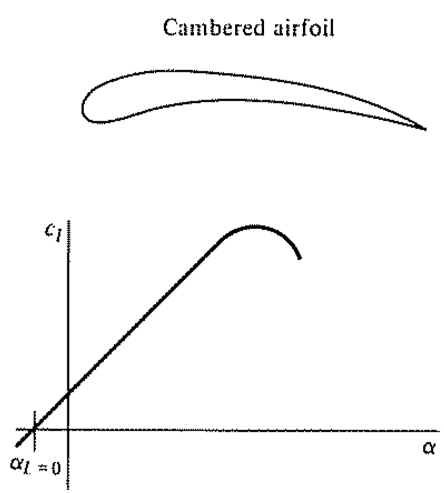
SAMPLE DATA

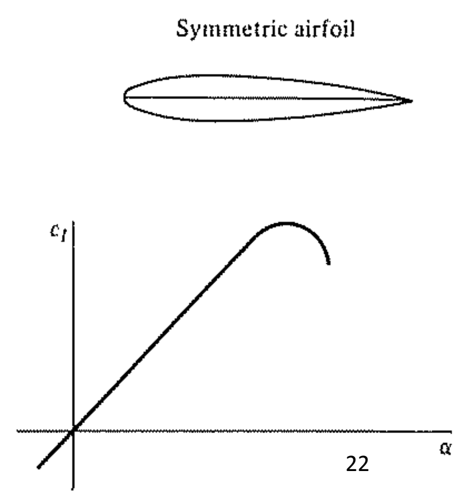


- Lift coefficient (or lift) linear variation with angle of attack, α
  - Cambered airfoils have positive lift when $\alpha = 0$
  - Symmetric airfoils have zero lift when $\alpha = 0$
- At high enough angle of attack, the performance of the airfoil rapidly degrades → stall

Cambered airfoil has lift at α=0

At negative α airfoil will have zero lift

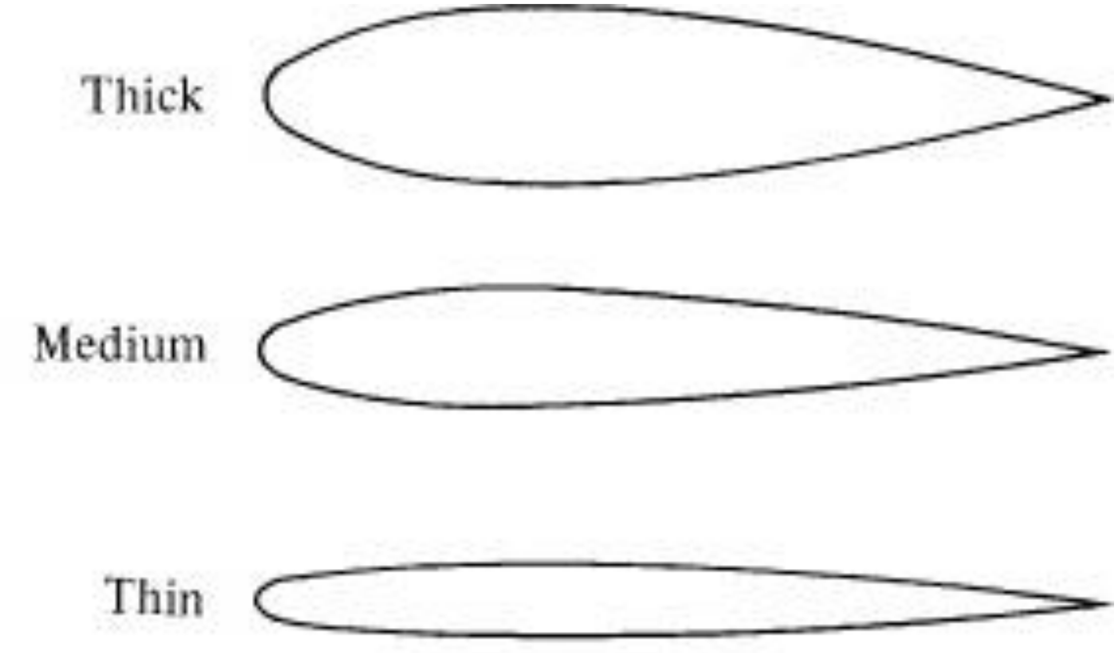




Maximum Thickness and Thickness Distribution

- The maximum thickness and the thickness distribution strongly influence the aerodynamic characteristics of the airfoil section.
- The maximum local velocity of the fluid increases as the maximum thickness increases. Therefore, the minimum pressure value is smallest for the thickest airfoil. As a result, the adverse pressure gradient associated with the deceleration of the flow from the location of this pressure minimum to the trailing edge is greatest.
- The boundary layer becomes thicker (and is more likely to separate producing large values for the form drag). Because of this, the beneficial effects of increasing the maximum thickness are limited.
- The high thickness and camber produce low critical Mach numbers.
- Consider the maximum section lift coefficients for several different thickness-ratio airfoils presented in the table.
- The data show that a low-speed airfoil has an optimum thickness for maximizing the lift of the airfoil.
- In the case of the data shown below, the optimum thickness to maximize $C_{l\max}$ is approximately 12%.

AIRFOIL THICKNESS SUMMARY

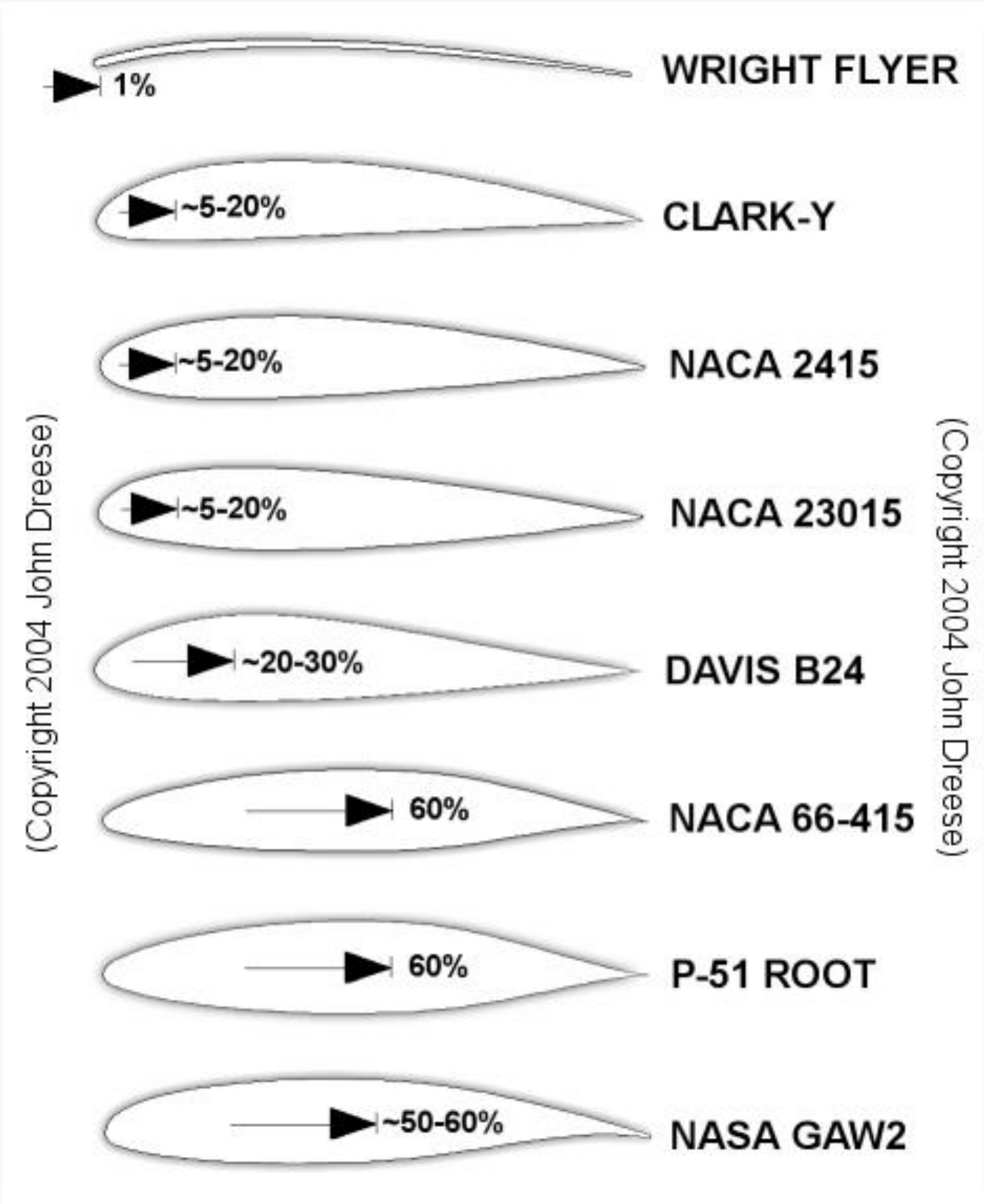


Airfoil Section	$C_{l_{max}}$
NACA 2408	1.5
NACA 2410	1.65
NACA 2412	1.7
NACA 2415	1.63
NACA 2418	1.48
NACA 2424	1.3

Note: thickness is relative to chord in all cases
Ex. NACA 0012 → 12 %

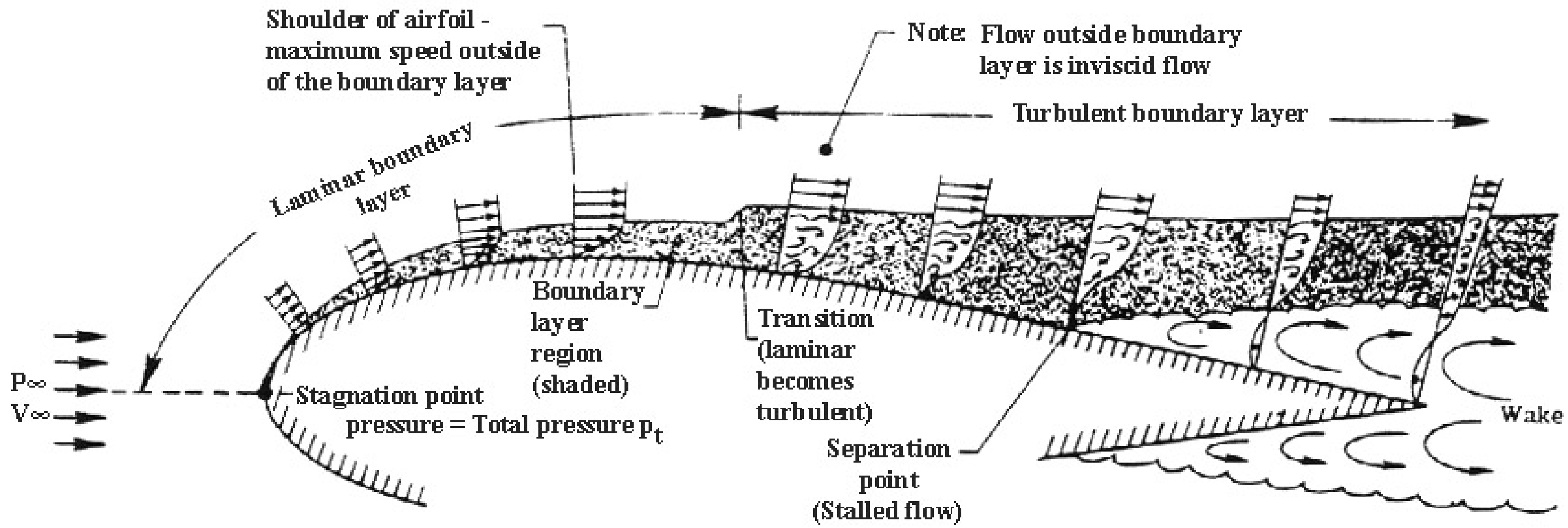
- Which creates most lift?
  - Thicker airfoil
- Which has higher critical Mach number?
  - Thinner airfoil
- Which is better?
  - Application dependent!

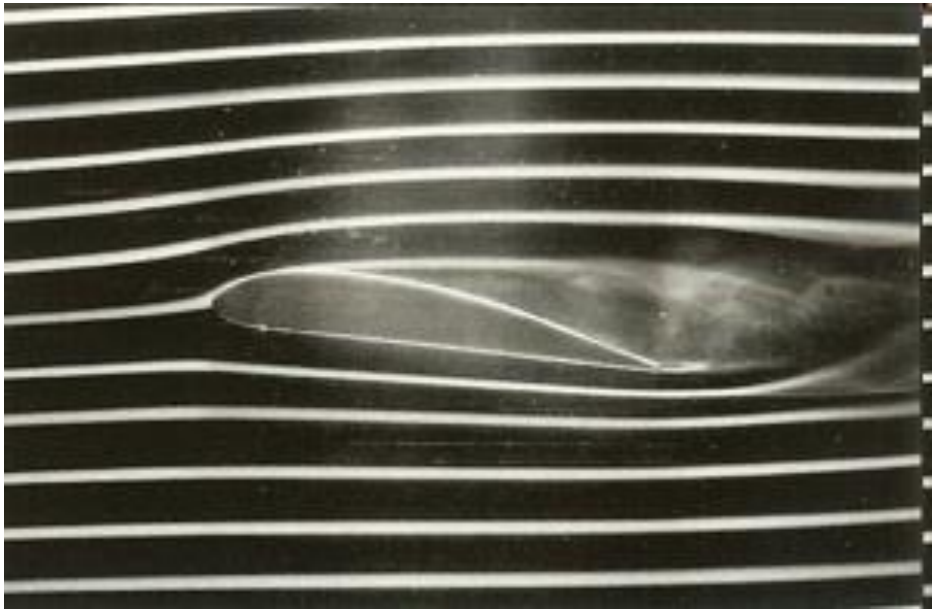
Evolution of Airfoil Design



Laminar boundary layer creates less skin friction drag

Boundary Layer Flow Separation

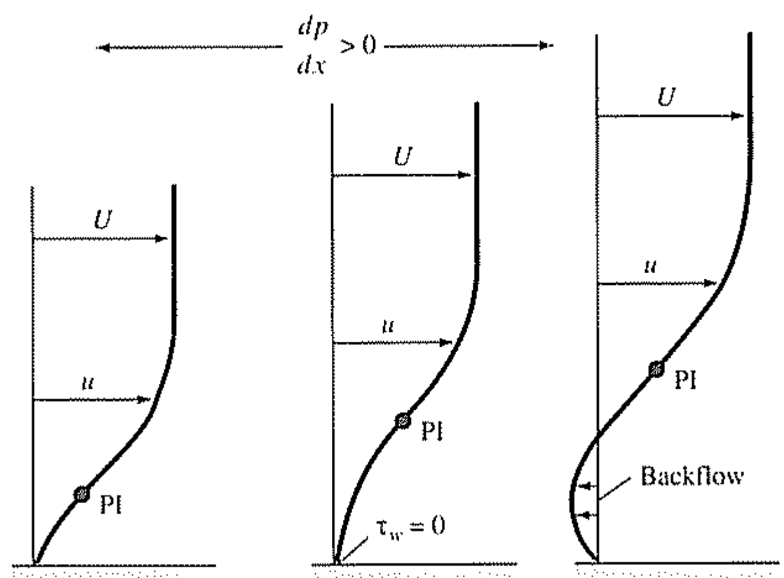




When flow separation occurs, there is also pressure drag.

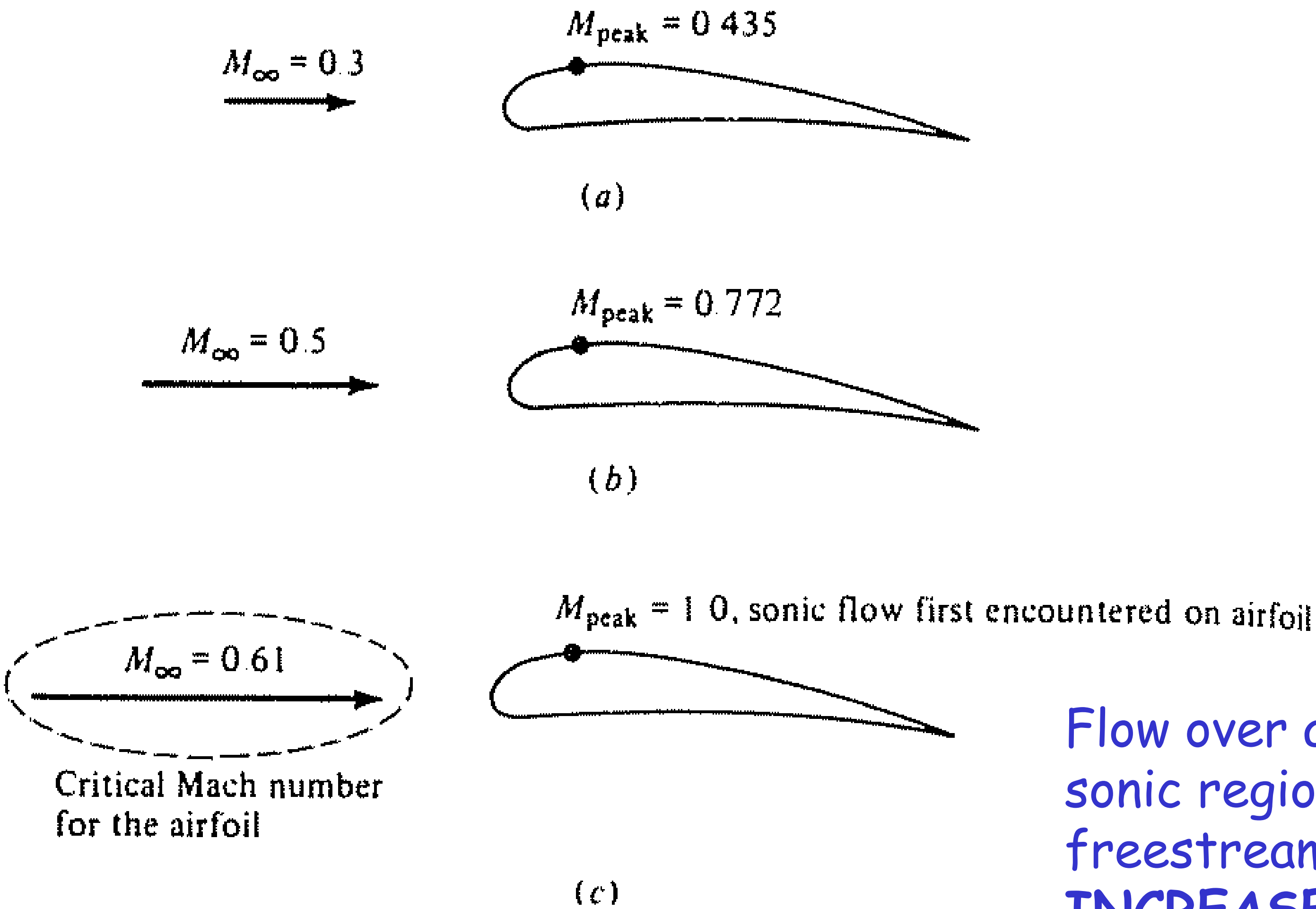
WHY DOES BOUNDARY LAYER SEPARATE?

- Adverse pressure gradient interacting with velocity profile through B.L.
- High speed flow near upper edge of B.L. has enough speed to keep moving through adverse pressure gradient
- Lower speed fluid (which has been retarded by friction) is exposed to same adverse pressure gradient is stopped and direction of flow can be reversed
- This reversal of flow direction causes flow to separate
  - Turbulent B.L. more resistance to flow separation than laminar B.L. because of fuller velocity profile
  - To help prevent flow separation we desire a turbulent B.L.



CRITICAL MACH NUMBER, M_{CR}

- As air expands around top surface near leading edge, velocity and M will increase
- Local M > M_∞

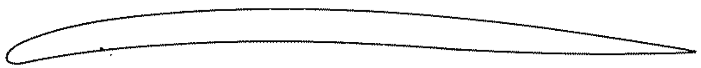


Flow over airfoil may have sonic regions even though freestream $M_\infty < 1$

INCREASED DRAG!

AIRFOIL THICKNESS: WWI AIRPLANES

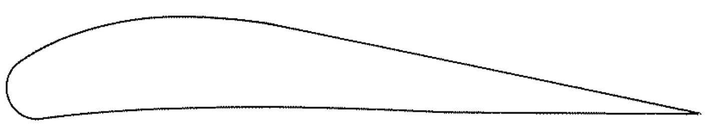
English Sopwith Camel



RAF 14. British

Thin wing, lower maximum C_L
Bracing wires required
– high drag

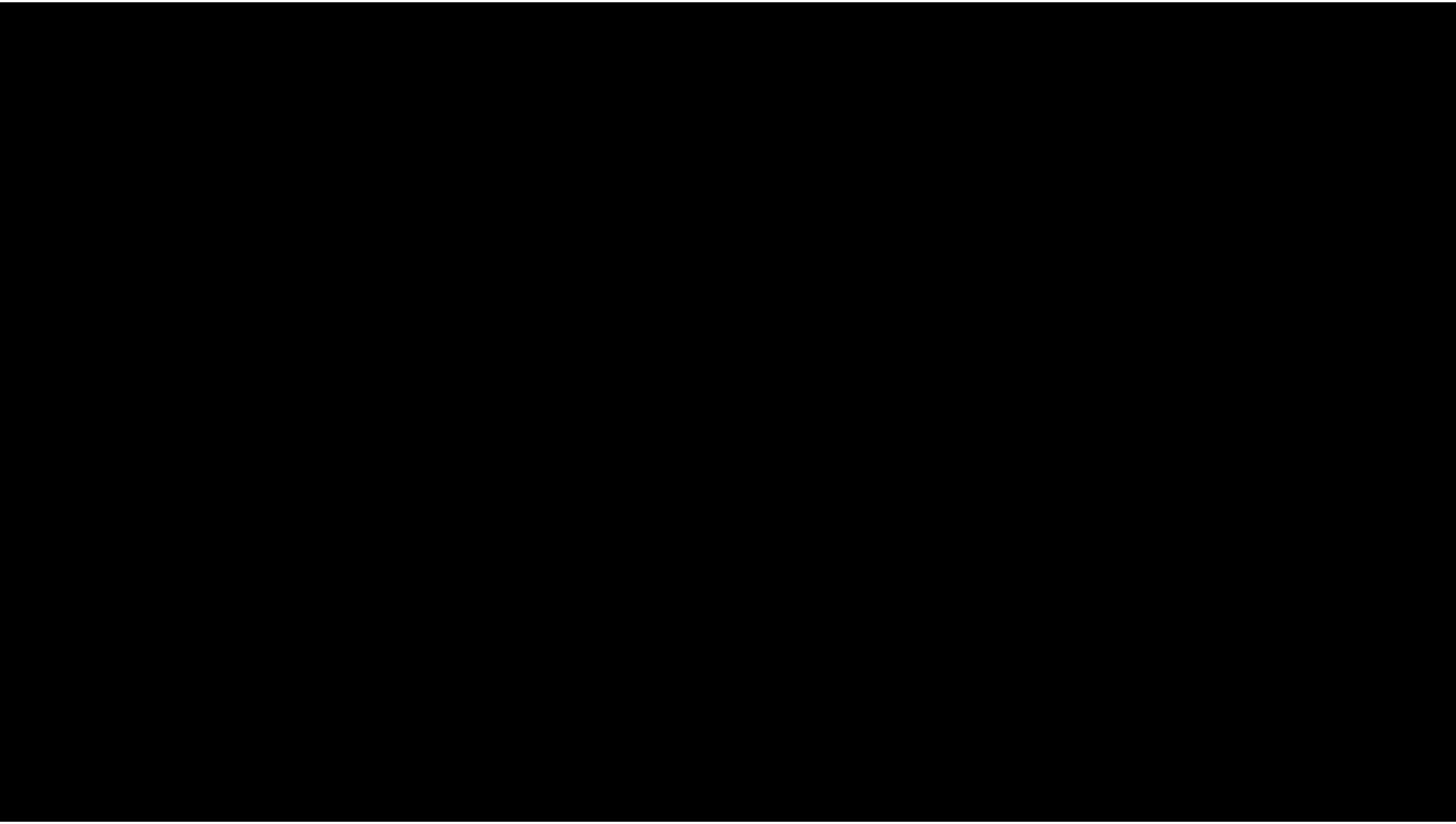
German Fokker Dr-1



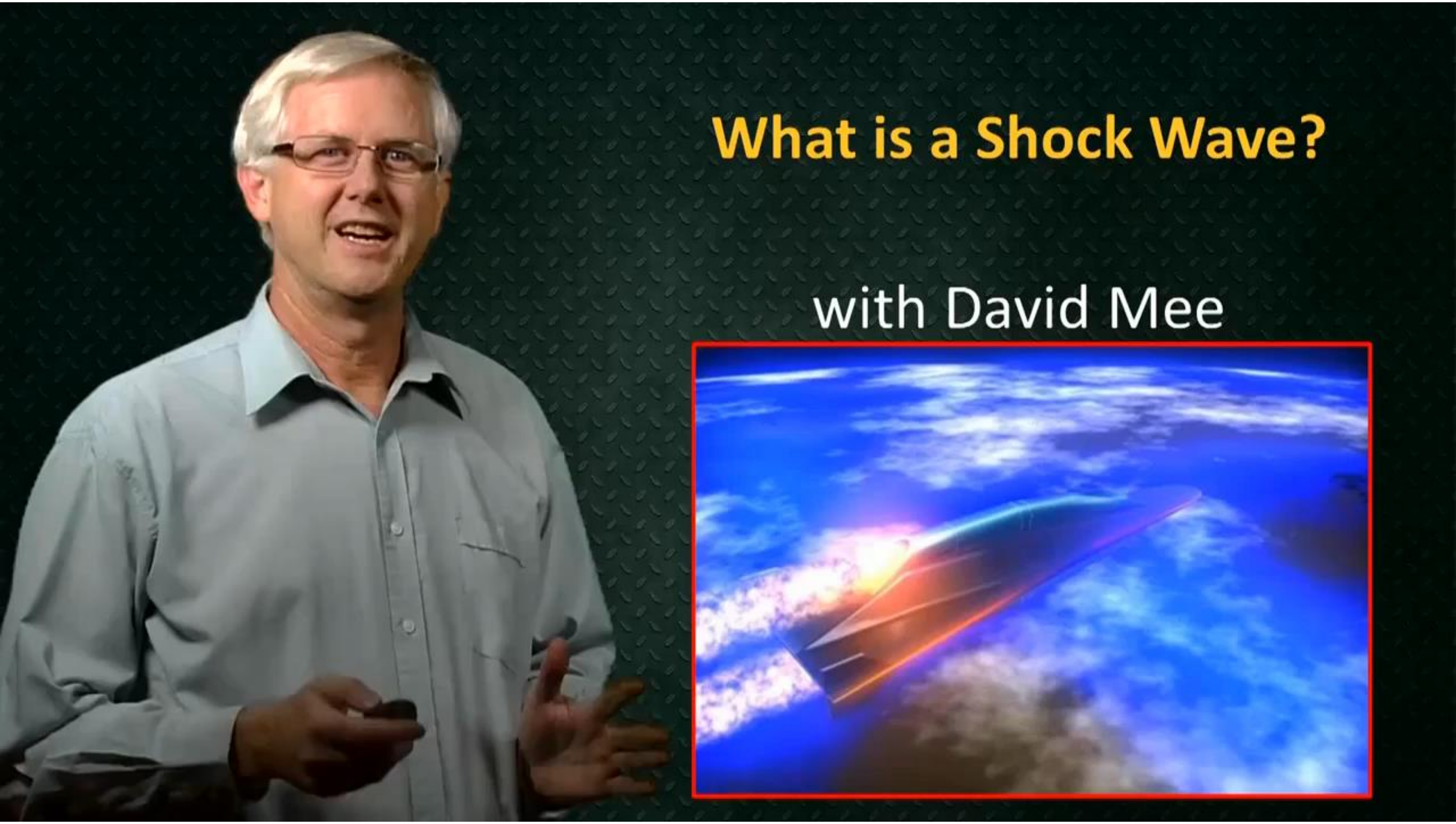
Göttingen 298. German

Higher maximum C_L
Internal wing structure
Higher rates of climb
Improved maneuverability

https://www.youtube.com/watch?v=j493HvCkMbM

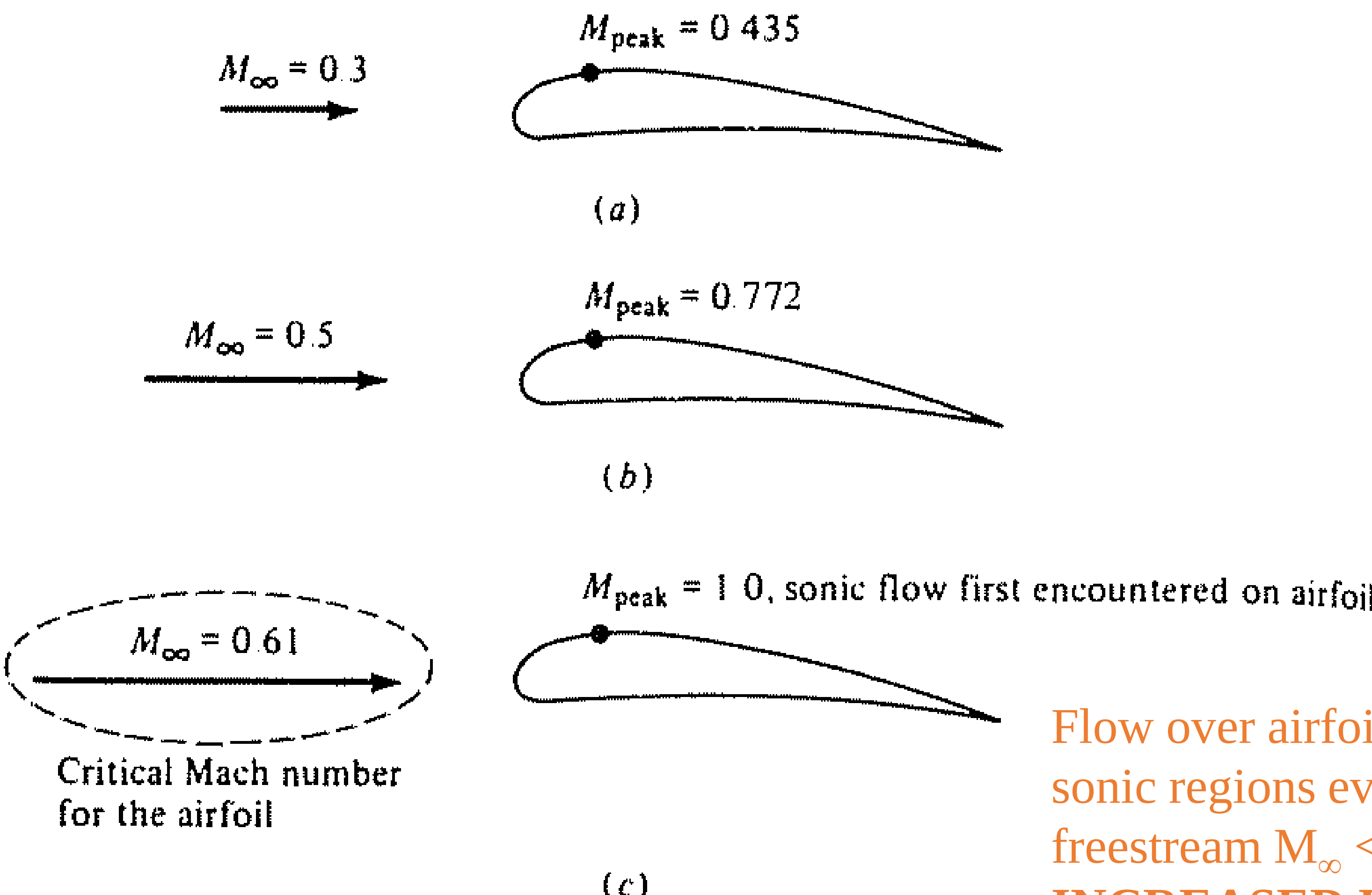


https://www.youtube.com/watch?v=gEXPVuAPyb0



CRITICAL MACH NUMBER, M_{CR} (5.9)

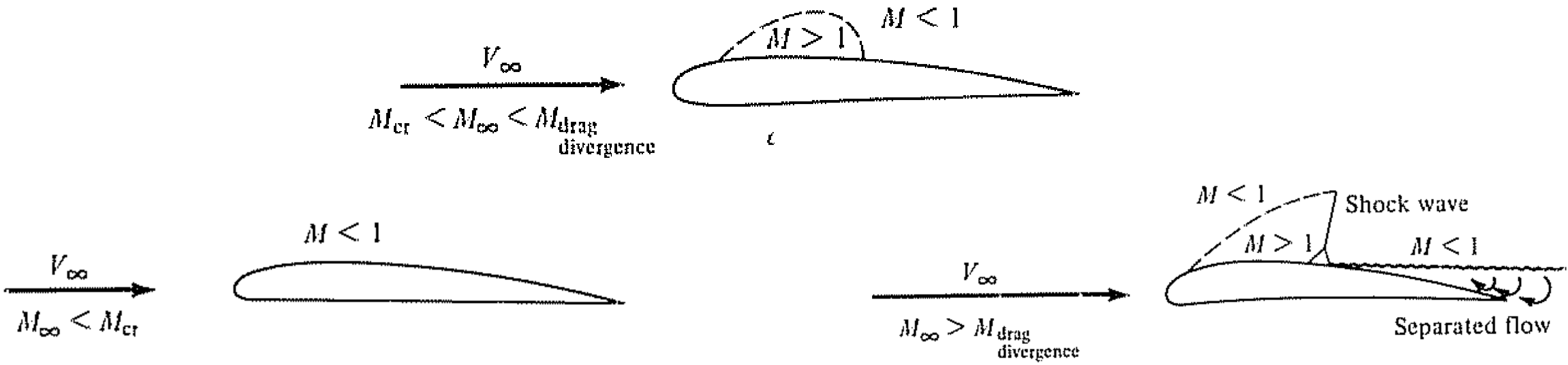
- As air expands around top surface near leading edge, velocity and M will increase
- Local $M > M_\infty$



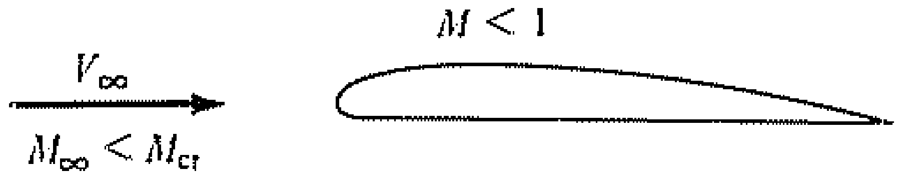
Flow over airfoil may have sonic regions even though freestream $M_\infty < 1$

INCREASED DRAG!

CRITICAL FLOW AND SHOCK WAVES

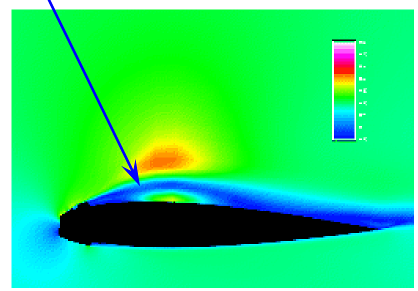


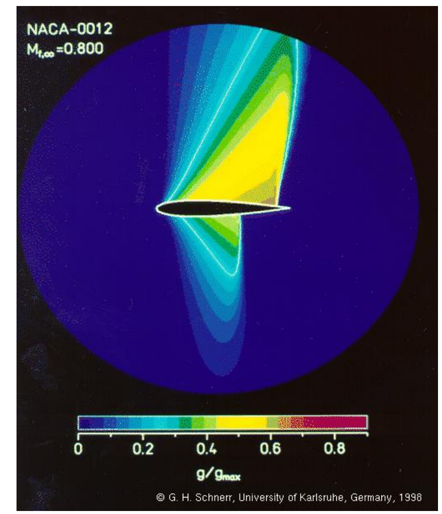
CRITICAL FLOW AND SHOCK WAVES



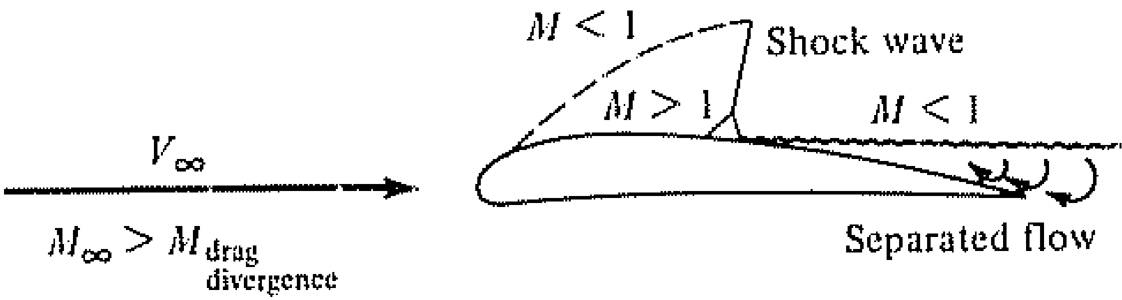
(a)

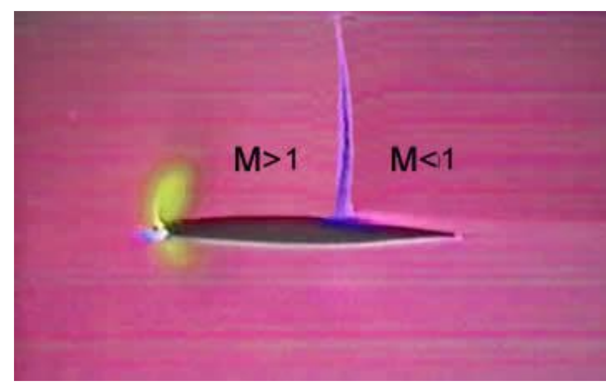
'bubble' of supersonic flow





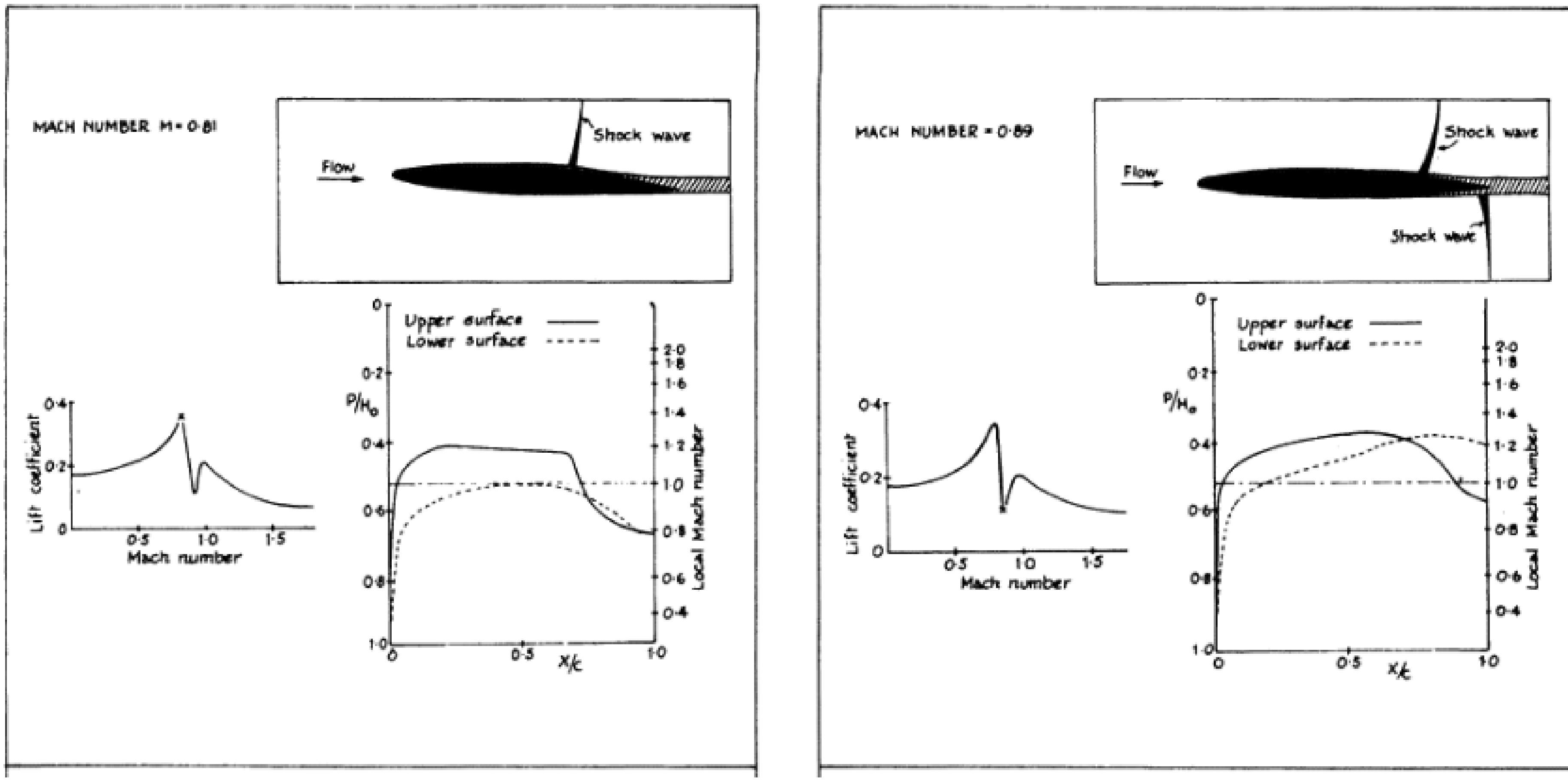
(b)

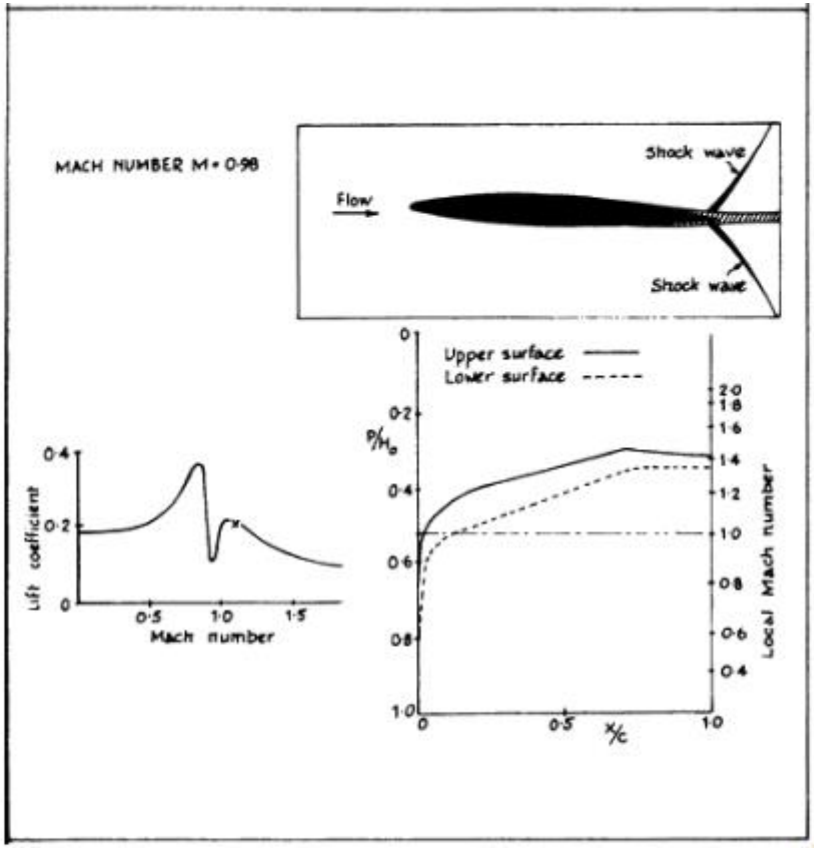


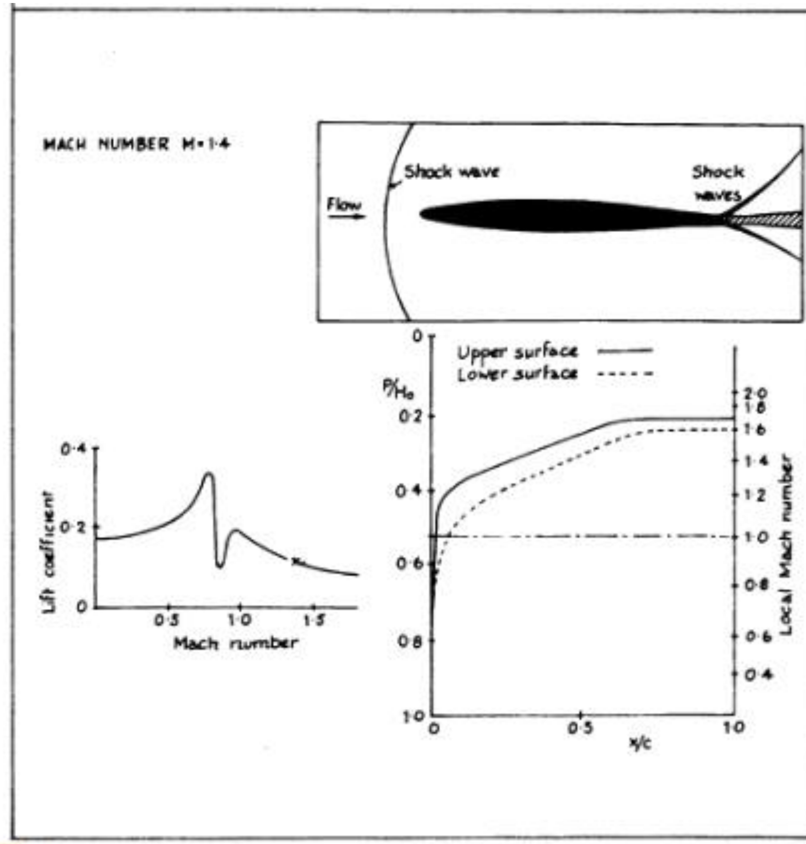


(c)

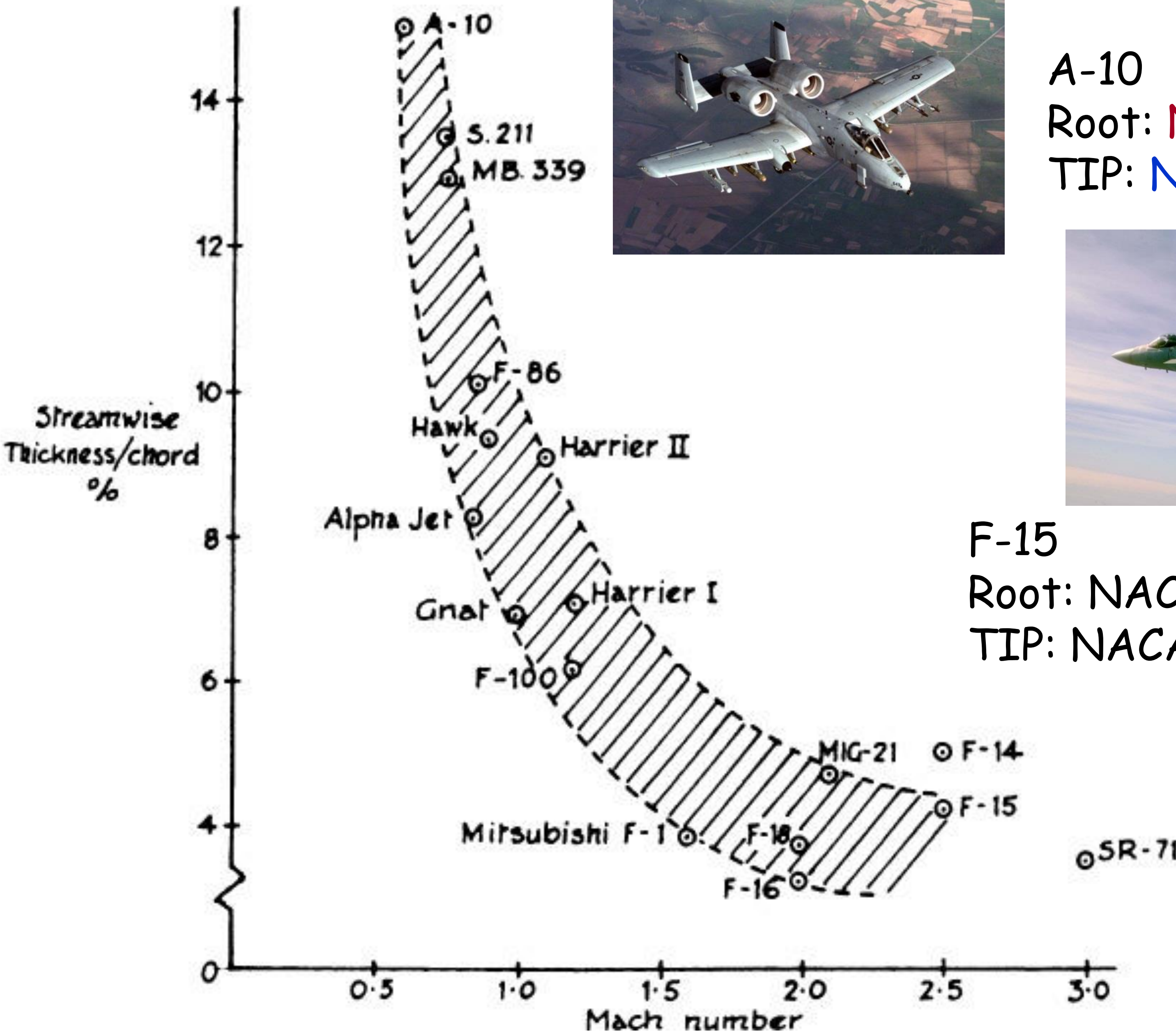
Fig 19 Flow changes over an aerofoil through the transonic range (angle of attack constant).[5]




MACH NUMBER M = 0·98
Shock wave
Flow
Shock wave
Upper surface
Lower surface
p/H₀
Local Mach number
x/c
Lift coefficient
Mach number


MACH NUMBER M = 1·4
Shock wave
Flow
Shock waves
Upper surface
Lower surface
p/H₀
Local Mach number
x/c
Lift coefficient
Mach number

THICKNESS-TO-CHORD RATIO TRENDS

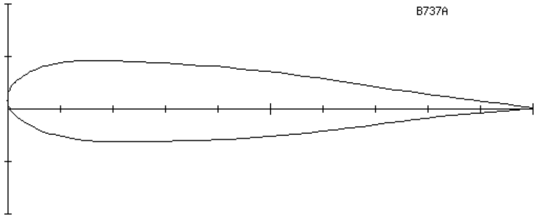


A-10
Root: NACA 6716
TIP: NACA 6713

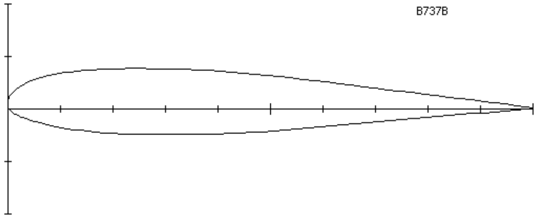
F-15
Root: NACA 64A(.055)5.9
TIP: NACA 64A203

Boeing 737

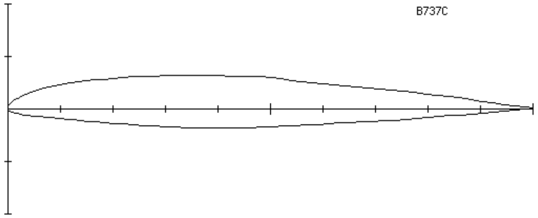
Root



Mid-Span



Tip



http://www.nasg.com/afdb/list-airfoil-e.phtml

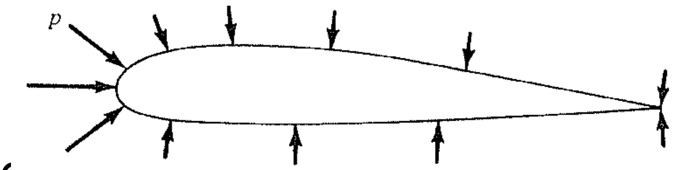
The thickness distribution

- The thickness distribution for an airfoil affects the pressure distribution and the character of the boundary layer.
- As the location of the maximum thickness moves aft, the velocity gradient (and hence the pressure gradient) in the mid-chord region decreases.
- The resultant favorable pressure gradient in the mid-chord region promotes boundary-layer stability and increases the possibility that the boundary layer remains laminar.
- Laminar boundary layers produce less skin-friction drag than turbulent boundary layers but are also more likely to separate under the influence of an adverse pressure gradient.
- In addition, the thicker airfoils benefit more from the use of high lift devices but have a lower critical Mach number.

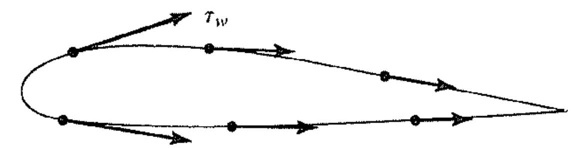
Trailing-Edge Angle

- The trailing-edge angle affects the location of the aerodynamic center.
- The *aerodynamic center* is that point about which the section moment coefficient is independent of the angle of attack.
- Therefore, the aerodynamic center is that point along the chord where all changes in lift effectively take place.
- The aerodynamic center of thin airfoil sections in a subsonic stream is theoretically located at the quarter-chord ($c/4$), but can vary depending on the geometry of the airfoil.

WHAT CREATES AERODYNAMIC FORCES?



- Aerodynamic forces exerted by airflow come sources:



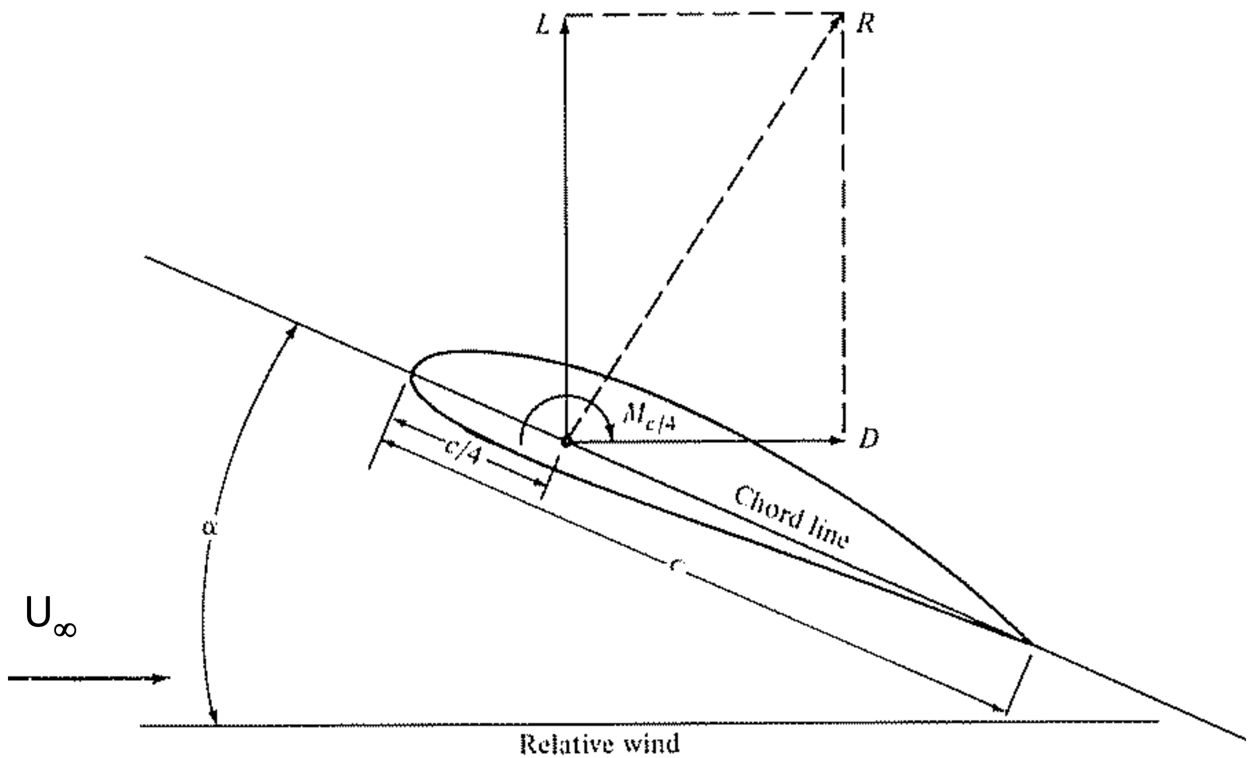
1. **Pressure, p**, distribution on surface
   - Acts **normal** to surface
2. **Shear stress, τ_w**, (friction) on surface
   - Acts **tangentially** to surface

- Pressure and shear are in units of force per unit area (N/m^2)
- Net unbalance creates an aerodynamic force

"No matter how complex the flow field, and no matter how complex the shape of the body, the only way nature has of communicating an aerodynamic force to a solid object or surface is through the pressure and shear stress distributions that exist on the surface."

RESOLVING THE AERODYNAMIC FORCE

- **Relative Wind:** Direction of U_∞
  - We use subscript ∞ to indicate far upstream conditions
- **Angle of Attack, α:** Angle between relative wind (U_∞) and chord line
- Total aerodynamic force, **R**, can be resolved into two force components
  - **Lift, L:** Component of aerodynamic force **perpendicular** to relative wind
  - **Drag, D:** Component of aerodynamic force **parallel** to relative wind



The Lift Coefficient, C_L

- For a relatively thin airfoil section at a relatively low angle of attack, the lift (and similarly the normal force) results primarily from the action of the pressure forces.
- The shear forces act primarily in the chord-wise direction (i.e., contribute primarily to the drag).
- Therefore, to calculate the force in the z direction, we only need to consider the pressure contribution.
- The pressure force acting on a differential area of the vehicle surface is $dF = p\,ds\,dy$.
- The elemental surface area is the product of ds , the wetted length of the element in the plane of the cross section, times dy , the element's length in the direction perpendicular to the plane of the cross section (or spanwise direction).
- Since the pressure force acts normal to the surface, the force component in the z direction is the product of the pressure times the projected planform area: $dF_z = p\,dx\,dy$

$$F_z = \oiint p \, dx \, dy$$

- But $\boxed{\oiint p_\infty dx \, dy = 0}$

$$F_z = \oiint (p - p_\infty) dx \, dy$$

$$\frac{F_z}{q_\infty cb} = \oiint \frac{p - p_\infty}{q_\infty} d\left(\frac{x}{c}\right) d\left(\frac{y}{b}\right)$$

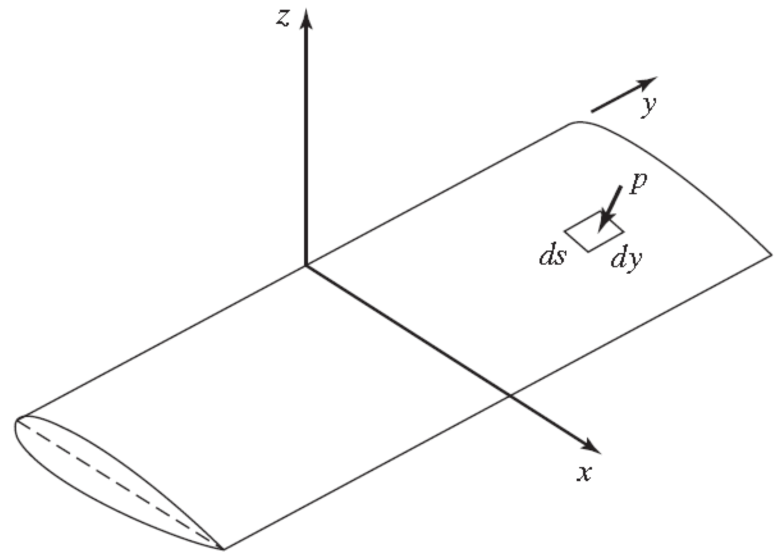
- But:

$$\boxed{C_p = (p - p_\infty)/q_\infty,}$$

$$\frac{F_z}{q_\infty S} = \oiint C_p d\left(\frac{x}{c}\right) d\left(\frac{y}{b}\right)$$

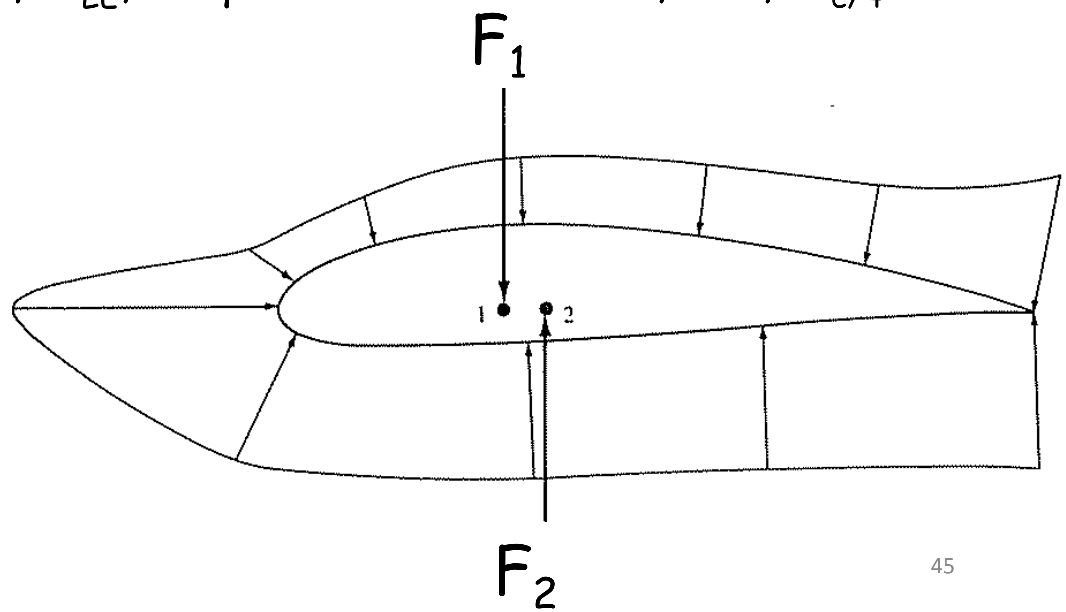
- Experimentally,

$$\boxed{C_L = \frac{L}{q_\infty S}}$$



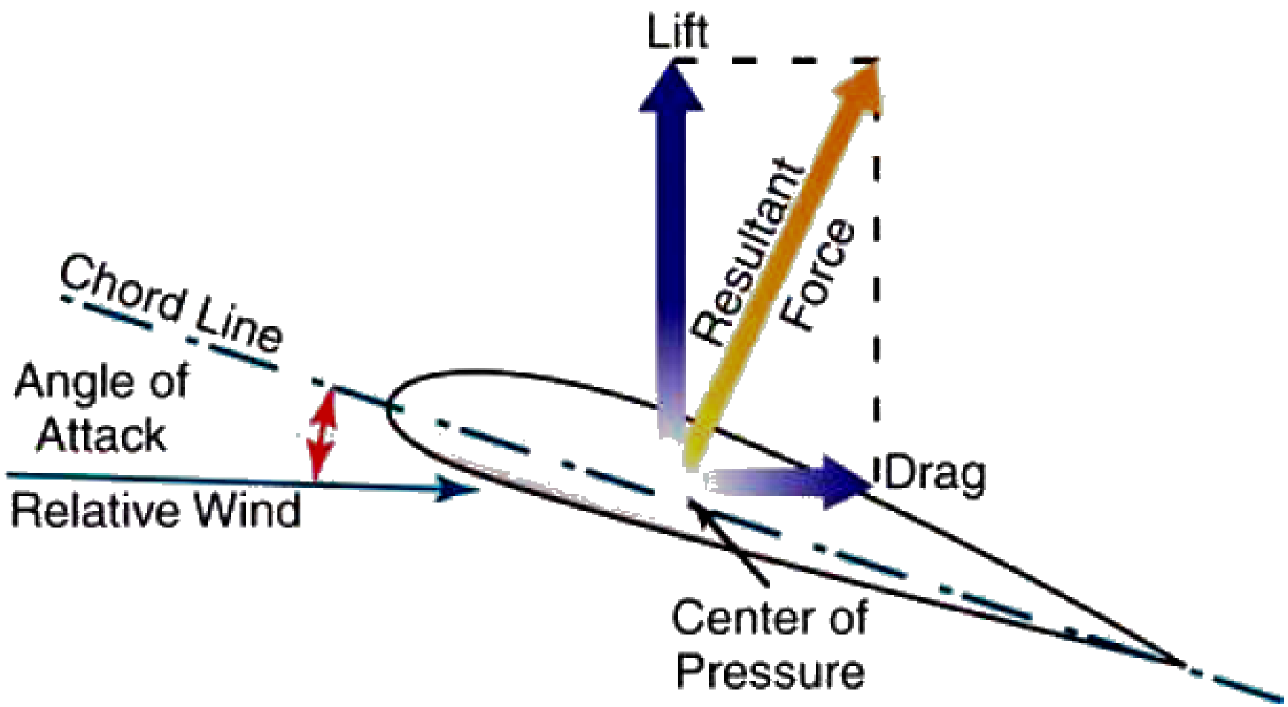
MORE DEFINITIONS

- Total aerodynamic force on airfoil is summation of F_1 and F_2
- Lift is obtained when $F_2 > F_1$
- Misalignment of F_1 and F_2 creates Moments, M, which tend to rotate airfoil/wing (A moment (torque) is a force times a distance)
- Value of induced moment depends on point about which moments are taken
  - Moments about leading edge, M_{LE}, or quarter-chord point, c/4, $M_{c/4}$
  - In general $M_{LE} \neq M_{c/4}$



Angle of Attack, α

- The angle of attack is the angle between the chord line and the average relative wind.
- Greater angle of attack creates more lift (up to a point).

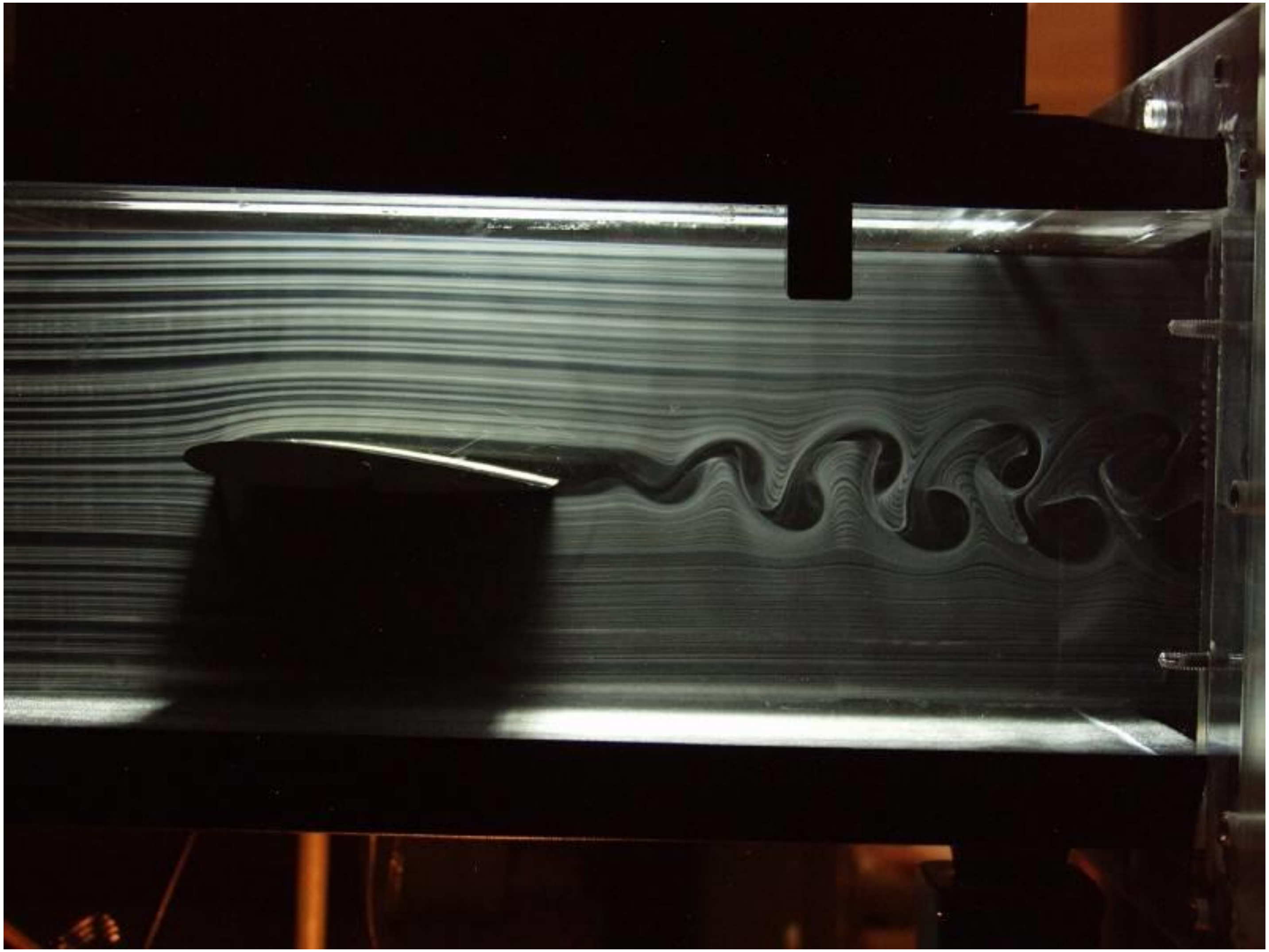


VARIATION OF L, D, AND M WITH α

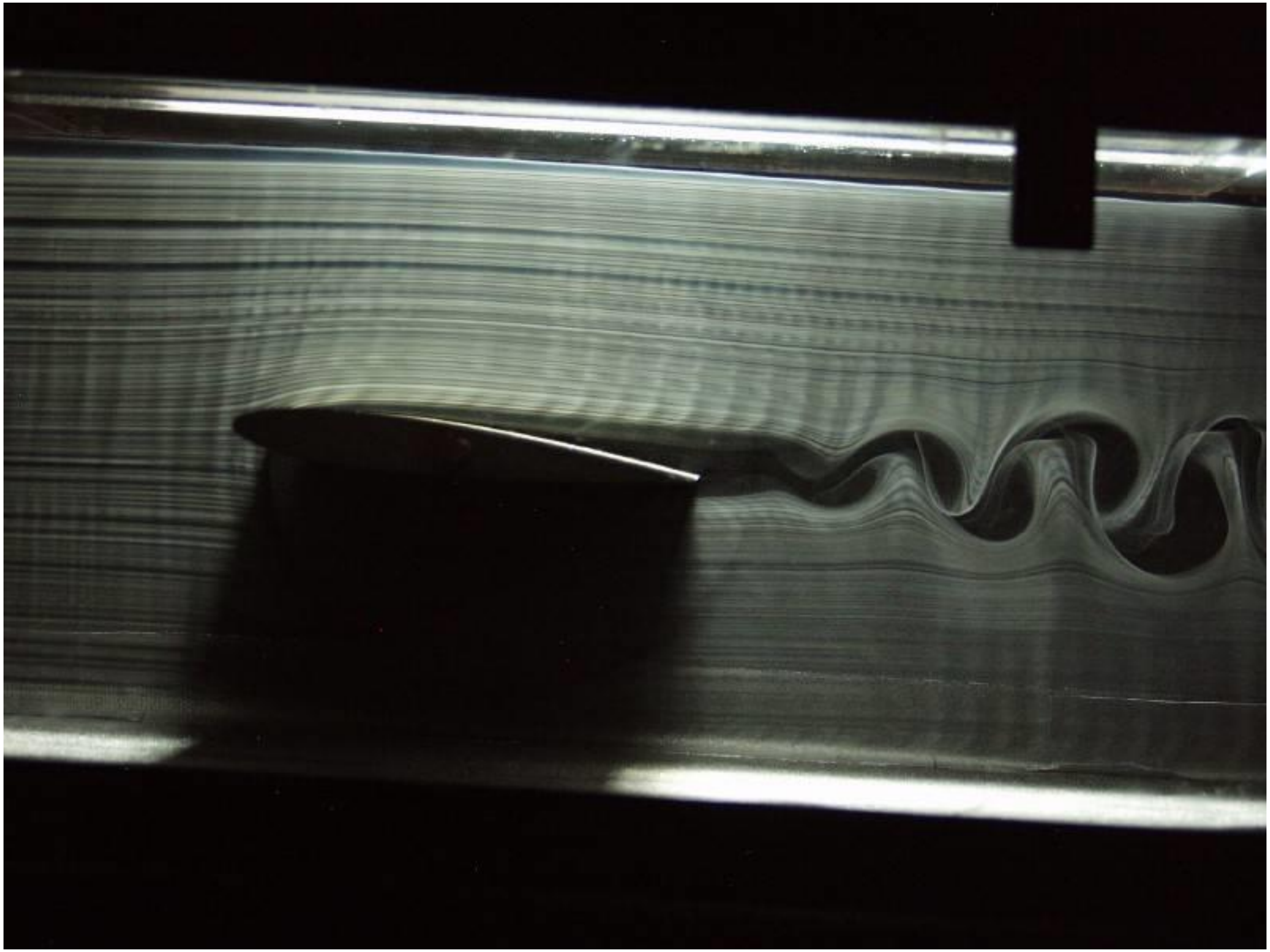
- Lift, Drag, and Moments on a airfoil or wing will change as α changes
- Variations of these quantities are some of most important information that an airplane designer needs to know

- **Aerodynamic Center**
  - Point about which moments essentially do not vary with α
  - M_{ac}=constant (independent of α)
  - For low speed airfoils aerodynamic center is near quarter-chord point, c/4

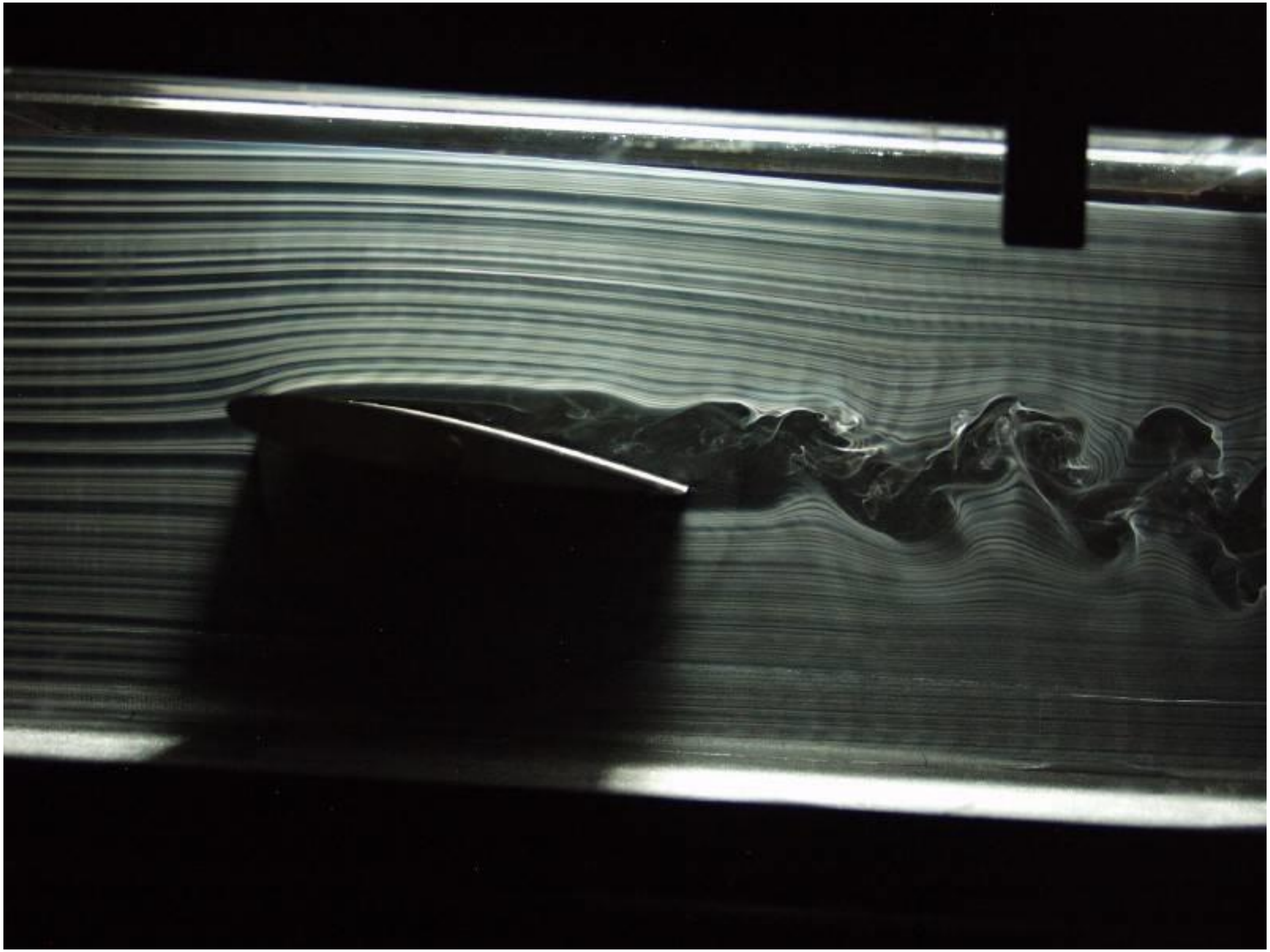
AOA = 2°

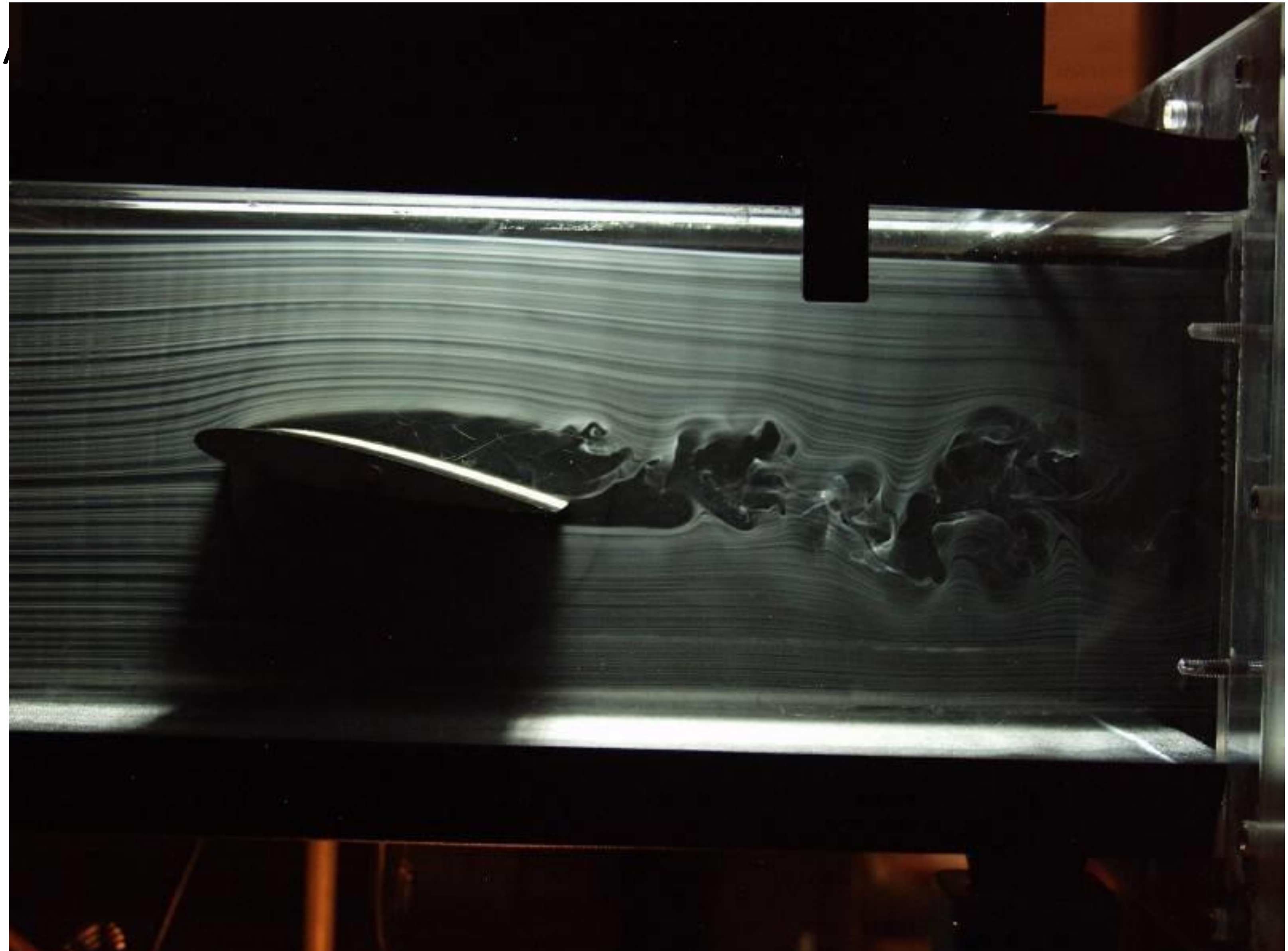


AOA = 3°

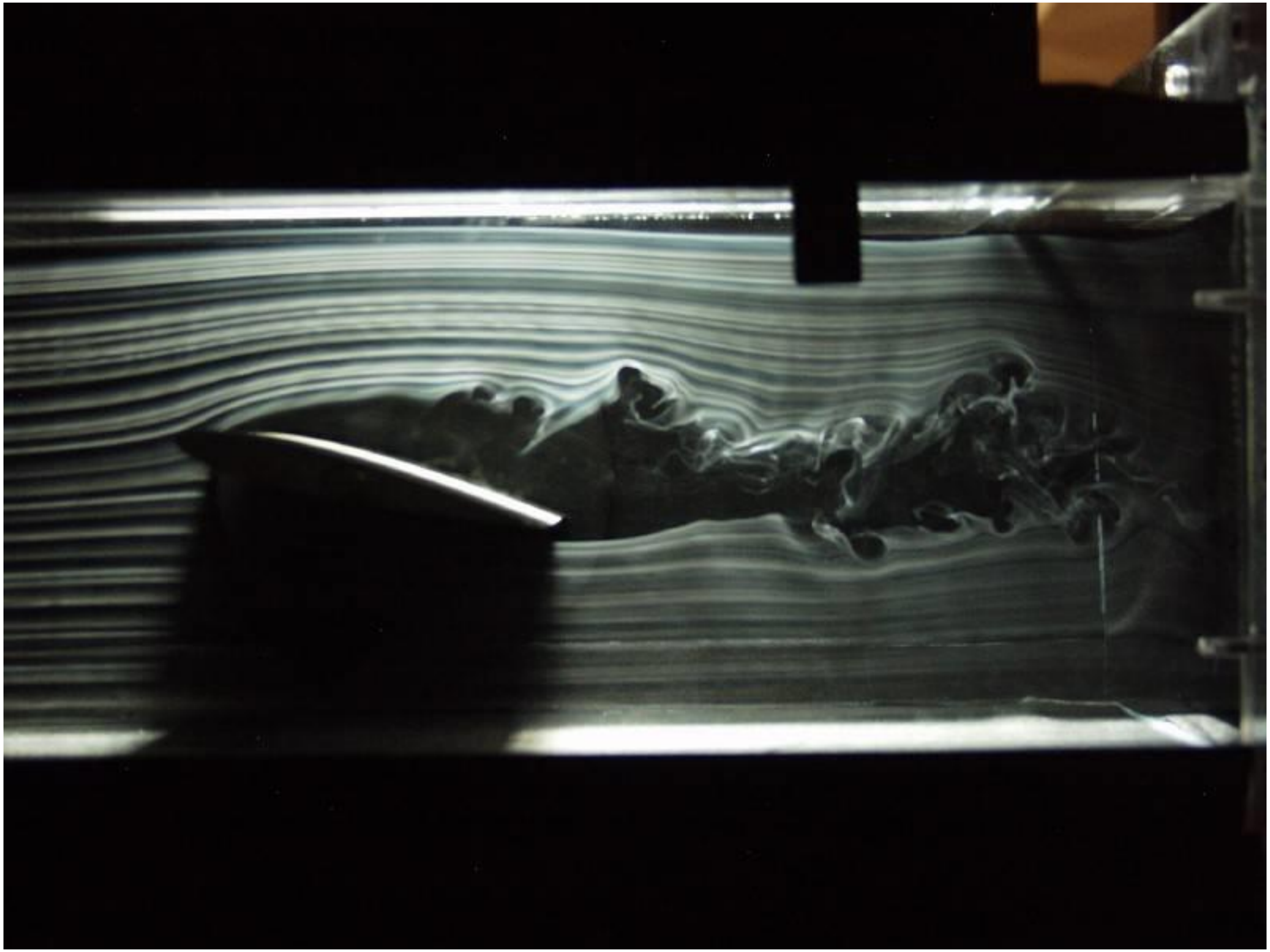


AOA = 6°

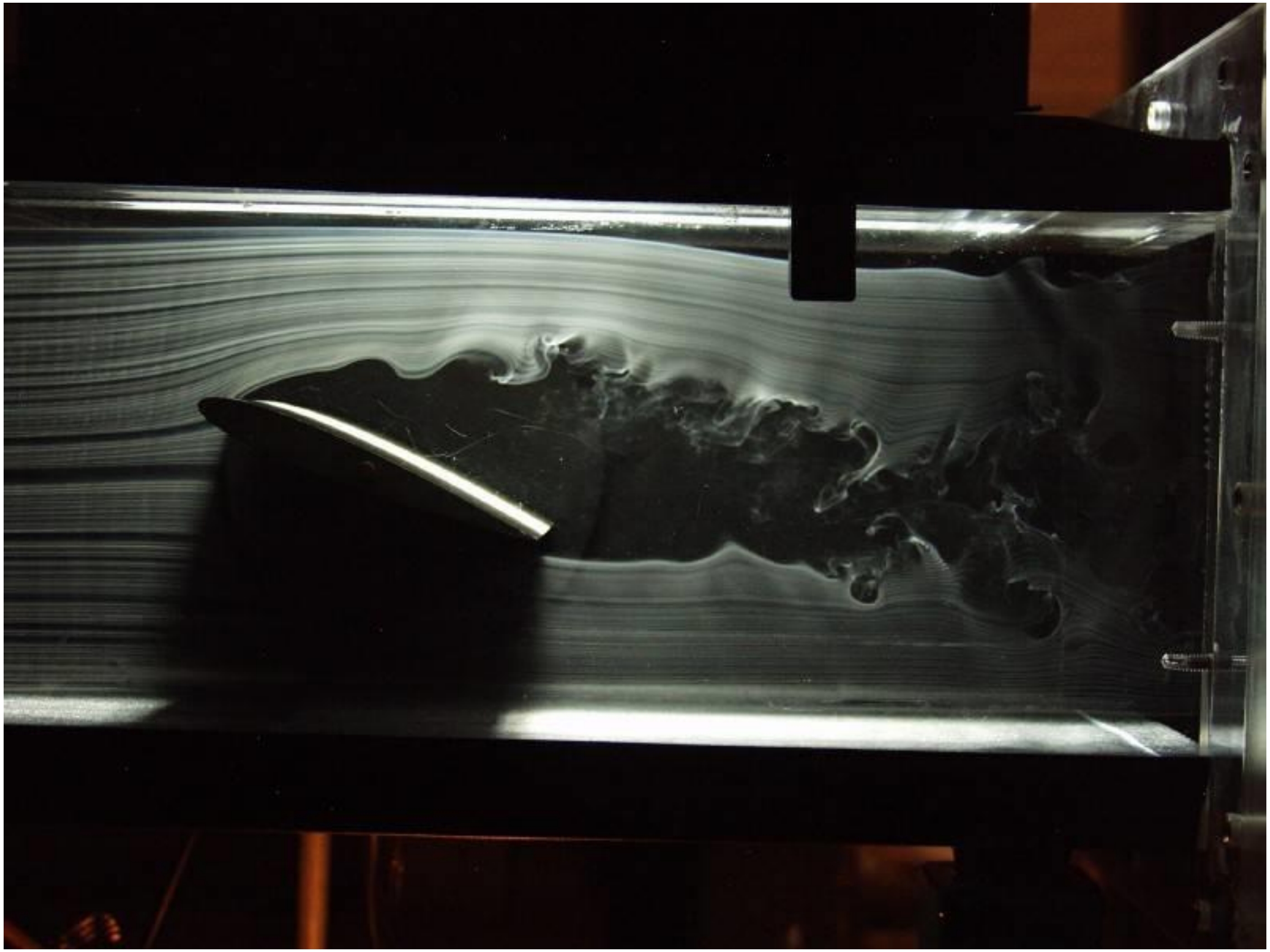




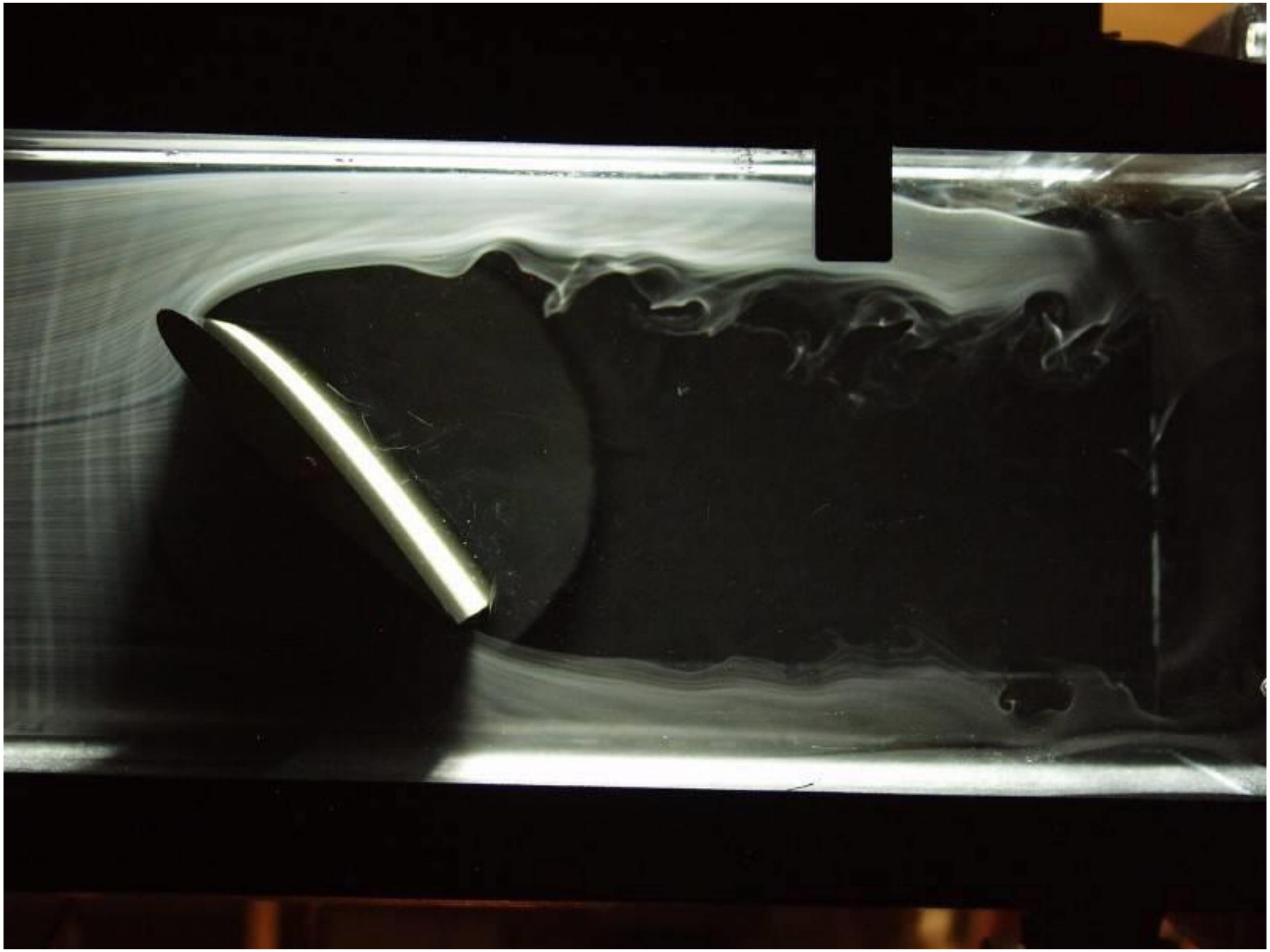
AOA = 12°



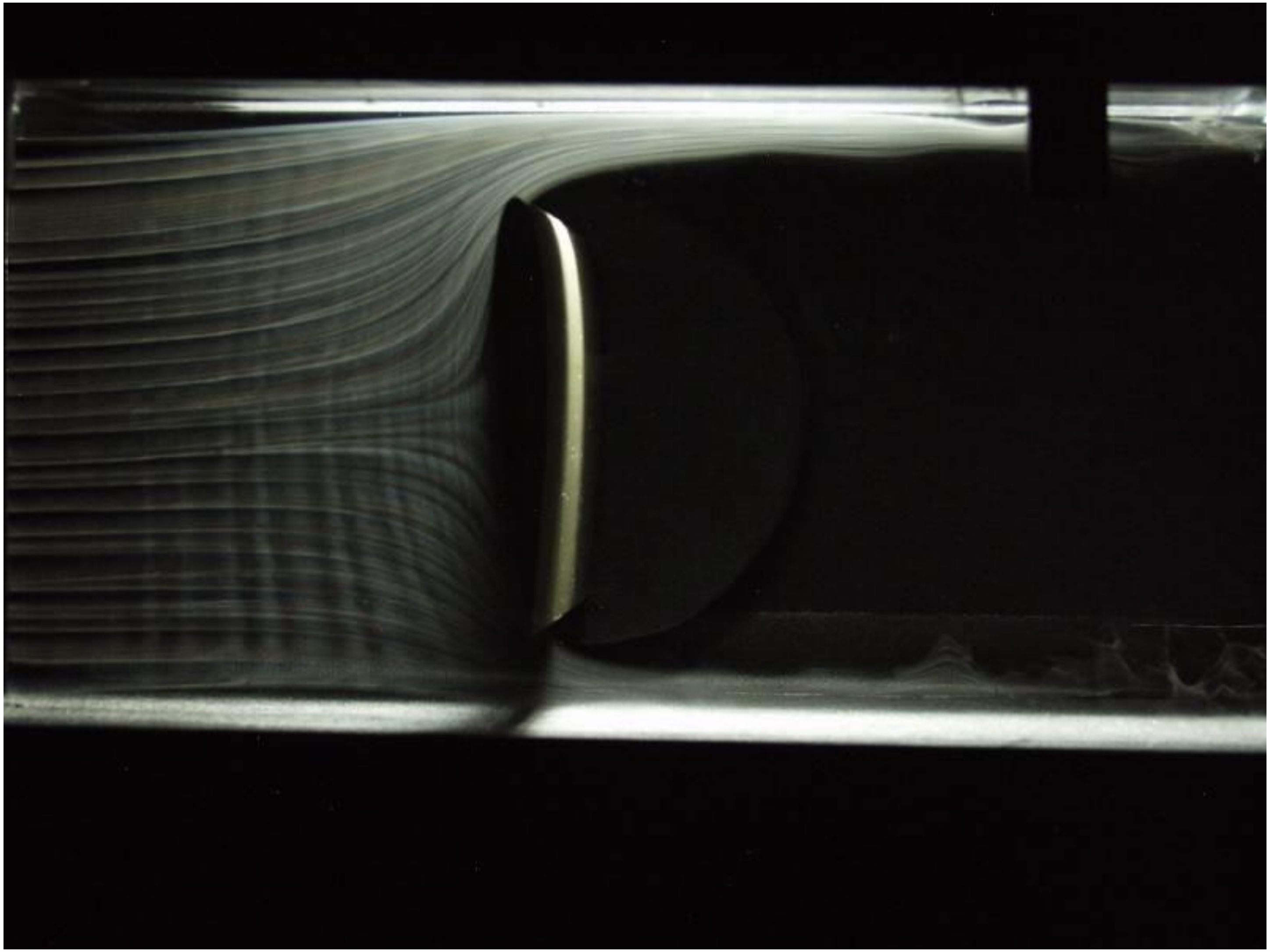
AOA = 20°



AOA = 60°



AOA = 90°



Lift Coefficient, C_L

- The lift coefficient is defined as:

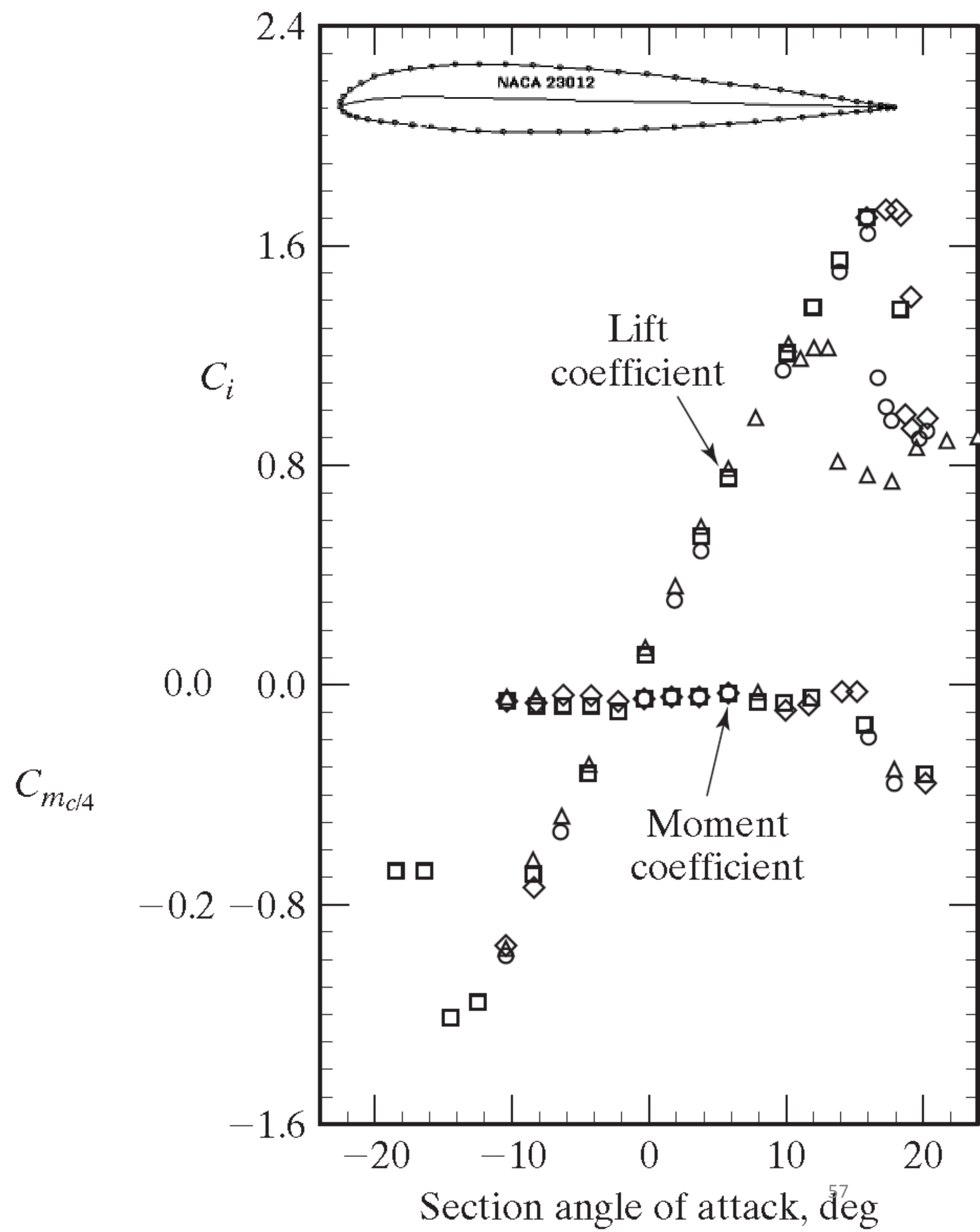
$$C_L = \frac{L}{\frac{1}{2}\rho U^2{}_{\infty}\, A} = \frac{L}{q_{\infty} S}$$

- Data are presented for a NACA 23012 airfoil which were obtained from a wind-tunnel model.
- The lift acting on a wing of infinite span does not vary in the *y* direction. For this two-dimensional flow, we are interested in determining the lift acting on a unit width of the wing [i.e., the lift per unit span (C_l)].
- The section lift coefficient is the lift per unit span (l) divided by the product of the dynamic pressure times the plan form area per unit span, which is the chord length (*c*):

$$C_l = \frac{l}{q_{\infty} c}$$

- The experimental section lift coefficient is independent of Reynolds number and is a linear function of the angle of attack from approximately -10 to +10. The slope of this linear portion of the curve is called the *two-dimensional lift-curve slope* .

- Section lift coefficient and section moment coefficient (with respect to *c* /4) for an NACA 23012 airfoil



Re_c

○ 3.0×10^6

□ 6.0×10^6

◇ 8.8×10^6 (smooth model)

△ 6.0×10^6 (standard rough

- Since the NACA 23012 airfoil section is cambered (the maximum camber is approximately 2% of the chord length), lift is generated at zero degrees angle of attack.
- In fact, zero lift is obtained at -1.2, which is designated α_{0l} or the section angle of attack for zero lift.
- The variation of the lift coefficient with angle of attack in the linear region is given by: $$C_l = C_{l_\alpha}(\alpha - \alpha_{0l})$$
- As the angle of attack is increased above 10, the section lift coefficient continues to increase (but is no longer linear with angle of attack) until a maximum value, C_{lmax}, is reached.
- $C_{l,max}$ is 1.79 and occurs at an angle of attack of 18, which is called the stall angle of attack.
- At angles of attack in excess of 10, the section lift coefficients exhibit a Reynolds number dependence.
- The adverse pressure gradient increases as the angle of attack increases.
- At these higher angles of attack, the air particles which have been slowed by the viscous forces cannot overcome the relatively large adverse pressure gradient, and the boundary layer separates.

EX. 5.3: Calculate the lift per unit span on a NACA 23012 airfoil section

- Consider tests of an unswept wing that spans the wind tunnel and whose airfoil section is NACA 23012.
- Since the wing model spans the test section, we will assume that the flow is two dimensional.
- The chord of the model is 1.3 m.
- The test section conditions simulate a density altitude of 3 km.
- The velocity in the test section is 360 km/h.
- What is the lift per unit span (in N/m) that you would expect to measure when the angle of attack is 4°? What would be the corresponding section lift coefficient?

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.

Solution:

- First, we need to calculate the section lift coefficient.
- We will assume that the lift is a linear function of the angle of attack and that it is independent of the Reynolds number (i.e., the viscous effects are negligible) at these test
conditions. These are reasonable assumptions as can be seen by referring to Fig. 5.13 .
- Therefore, the section lift coefficient from equation (5.9) is:
- Using the values presented in the discussion associated with Fig. 5.13 ,

$$C_l = C_{l_\alpha}(\alpha - \alpha_{0l})$$

- At an angle of a ... coefficient for an NACA 23 ... n Fig. 5.13 . To calculate the ... equation (5.8) to obtain

$$C_l = (0.1041/\text{deg})(4.0^\circ - (-1.2^\circ)) = 0.541$$

$$l = C_l q_\infty c$$

To calculate the dynamic pressure (q_∞), we need the velocity in m/s and the density in $\mathrm{kg/m^3}$ in order for the units to be consistent:

$$U_\infty = 360\,\frac{\mathrm{km}}{\mathrm{h}}\,\frac{1000\ \mathrm{m}}{\mathrm{km}}\,\frac{\mathrm{h}}{3600\ \mathrm{s}} = 100\,\frac{\mathrm{m}}{\mathrm{s}}$$

Given the density altitude is 3 km, we can refer to Appendix B to find that:

$$\rho = 0.9093\ \mathrm{kg/m^3}$$

(*Note:* The fact that we are given the density altitude as 3 km does not provide specific information either about the temperature or pressure.)

So now the lift per unit span is:

$$l = (0.541)\left[\frac{1}{2}\left(0.9093\,\frac{\mathrm{kg}}{\mathrm{m^3}}\right)\left(100\,\frac{\mathrm{m}}{\mathrm{s}}\right)^2\right](1.3\ \mathrm{m})$$

$$l = (0.541)\left[4546.5\,\frac{\mathrm{N}}{\mathrm{m^2}}\right](1.3\ \mathrm{m}) = 3197.6\,\frac{\mathrm{N}}{\mathrm{m}}$$

Moment Coefficient

- The moment created by the aerodynamic forces acting on a wing (or airfoil) is determined about a particular reference axis (also called the moment reference center).
- The reference axis could be the leading edge, the quarter-chord location, the aerodynamic center, and so on.
- The pitch moment about the leading edge due to the pressures acting on the surface will first be calculated. The contribution of the chord-wise component of the pressure force and of the skin-friction force to the pitch moment is small and is neglected.
- Therefore, the pitch moment about the leading edge due to the pressure force acting on the surface element whose area is *ds* times *dy* and which is located at a distance *x* from the leading edge is:

$$dM_0 = \underbrace{p\,dx\,dy}_{\text{Force}} \underbrace{x}_{\substack{\text{Lever}\\ \text{Arm}}}$$

$$M_0 = \oiint px\,dx\,dy$$ but $$\oiint p_\infty x\,dx\,dy = 0$$

$$M_0 = \oiint (p - p_\infty)x\,dx\,dy$$

$$\frac{M_0}{q_\infty c^2 b} = \oiint \frac{p - p_\infty}{q_\infty}\frac{x}{c}d\left(\frac{x}{c}\right)d\left(\frac{y}{b}\right)$$

$$\frac{M_0}{q_\infty Sc} = \oiint C_p\frac{x}{c}d\left(\frac{x}{c}\right)d\left(\frac{y}{b}\right)$$

- Now, the dimensionless moment coefficient is defined as:

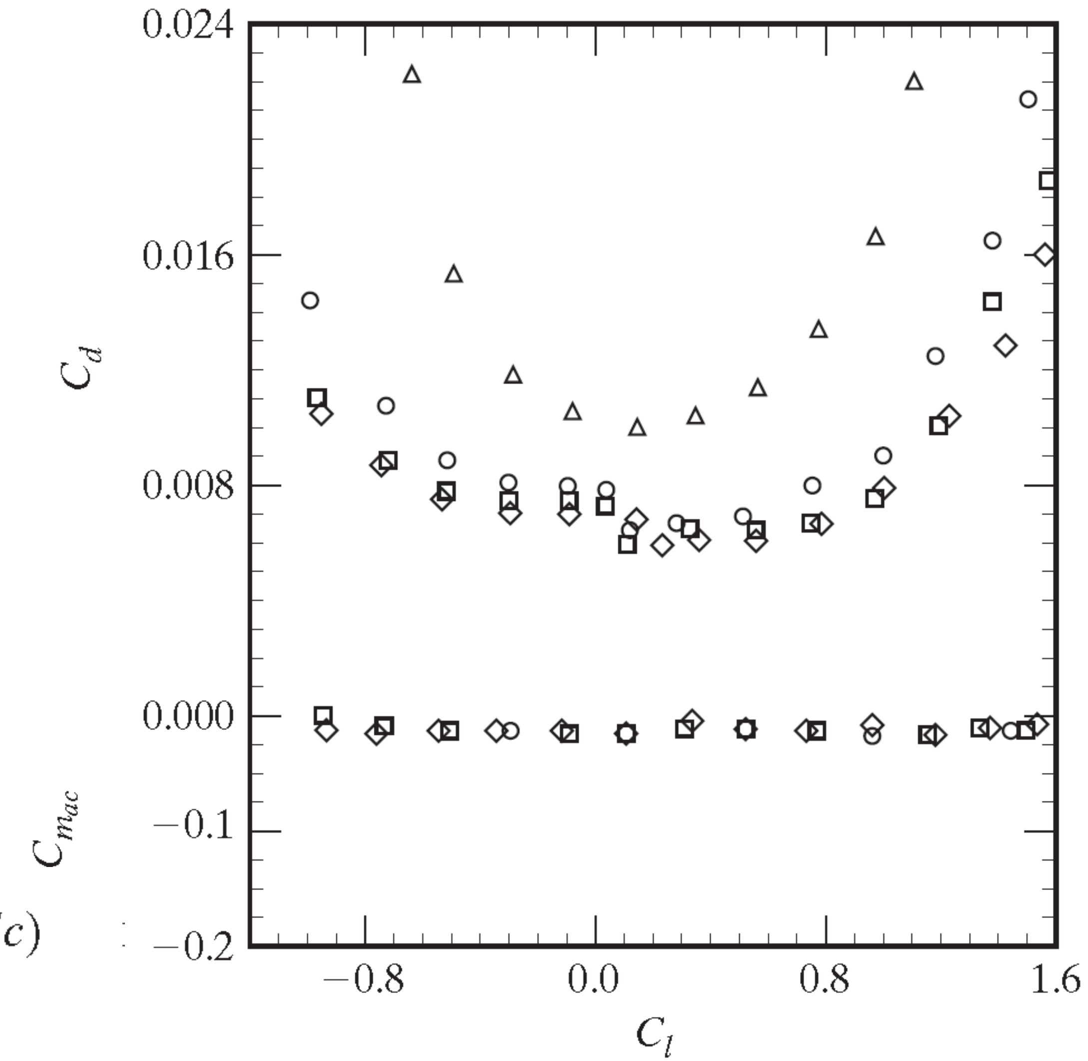
$$C_{M_0} = \frac{M_0}{q_\infty Sc}$$

- The section moment coefficient is used to represent the dimensionless moment per unit span (m_0)

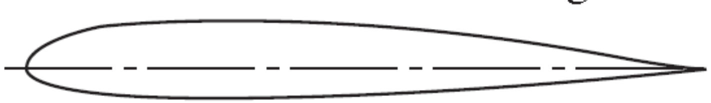
$$C_{m_0} = \frac{m_0}{q_\infty cc}$$

Section moment coefficient

- Section drag coefficient and section moment coefficient (with respect to the *ac*) for a NACA 23012 airfoil



	Re_c	ac position (x/c)
○	3.0×10^6	0.241
□	6.0×10^6	0.241
◇	8.8×10^6	0.247
△	6.0×10^6	Standard roughness



- The *aerodynamic center* is that point about which the section moment coefficient is independent of the angle of attack. Therefore, the aerodynamic center is that point along the chord where all changes in lift effectively take place.
- Since the moment about the aerodynamic center is the product of a force (the lift that acts at the center of pressure) and a lever arm (the distance from the aerodynamic center to the center of pressure), the center of pressure must move toward the aerodynamic center as the lift increases.
- The quarter-chord location is significant, since it is the theoretical aerodynamic center for incompressible flow about a two-dimensional airfoil.
- Notice that the pitch moment coefficient is independent of the Reynolds number (for those angles of attack where the lift coefficient is independent of the Reynolds number), since the pressure coefficient depends only on the dimensionless space coordinates (x/c, y/b).
- One of the nice features of the NACA 23012 airfoil section is a relatively high C_{lmax} with only a small $C_{m\text{ac}}$.

Drag Coefficient

- The drag force on a wing is due in part to skin friction and in part to the integrated effect of pressure. If *t* denotes the tangential shear stress at a point on the body surface, *p* the static pressure, and *n* the outward-facing normal to the element of surface *dS*, the drag can be formally expressed as

$$D = \oiint \vec{t} \cdot \hat{e}_\infty dS - \oiint p\hat{n} \cdot \hat{e}_\infty dS$$

- where e_∞ is a unit vector parallel to the free stream and the integration takes place over the entire wetted surface.
- The first integral represents the friction component and the second integral represents the pressure drag.
- The most straightforward approach to calculating the pressure drag is to perform the numerical integration indicated by the second term.

SUMMARY OF VISCOUS EFFECTS ON DRAG

- Friction has two effects:
  - Skin friction due to shear stress at wall
  - Pressure drag due to flow separation

$$D = D_{friction} + D_{pressure}$$

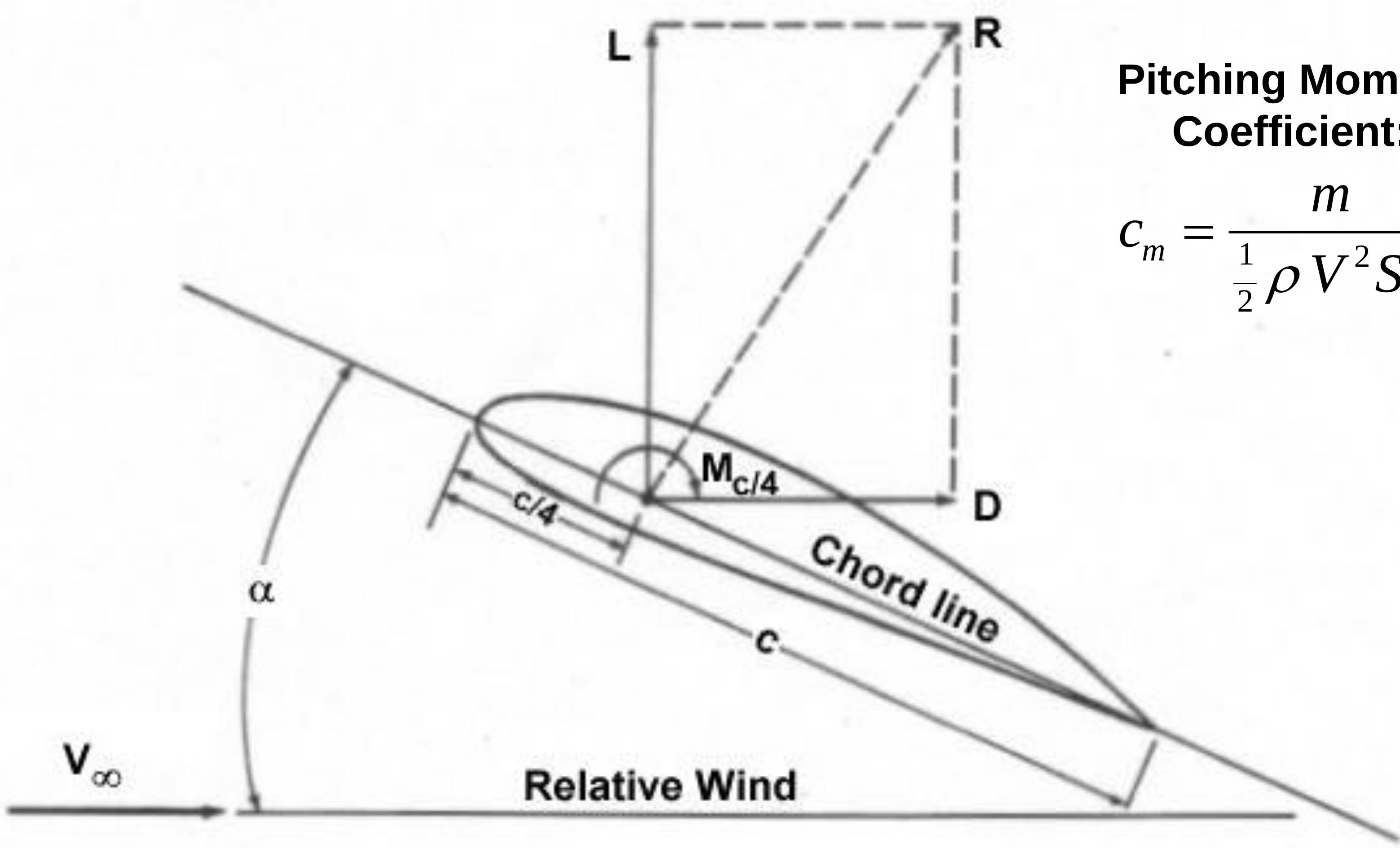
Total drag due to viscous effects Called **Profile Drag**	=	Drag due to skin friction	+	Drag due to separation
		Less for laminar More for turbulent		More for laminar Less for turbulent

So how do you design?

Depends on case by case basis, no definitive answer!

Aerodynamic Center

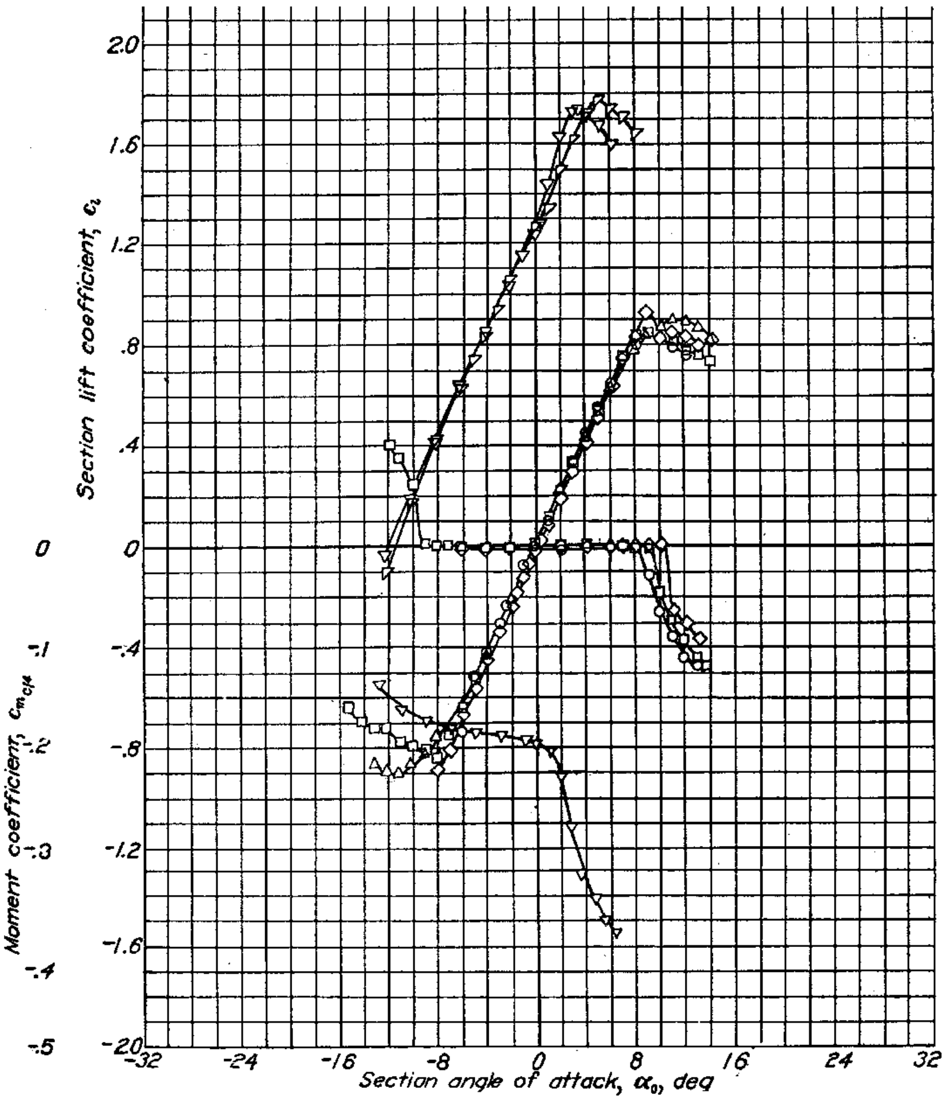
Since the c.p. varies with α, it is more desirable to use a fixed Aerodynamic Center (a.c.) as the point of action of the lift and drag. The pitching moment about this point can be calculated, and is found insensitive to α. For most airfoils, the a.c. locates at around quarter chord (x=c/4).

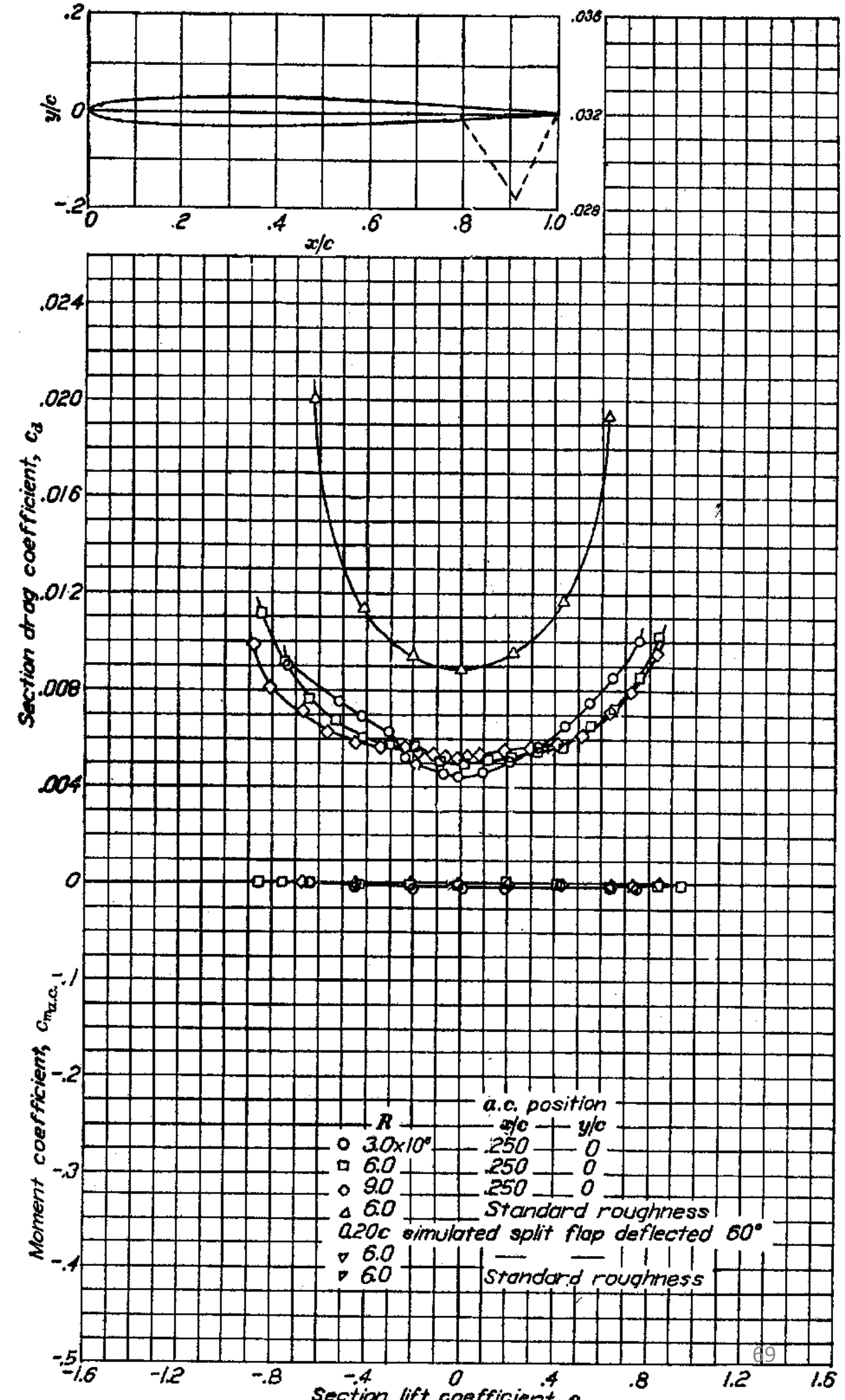


Pitching Moment Coefficient:

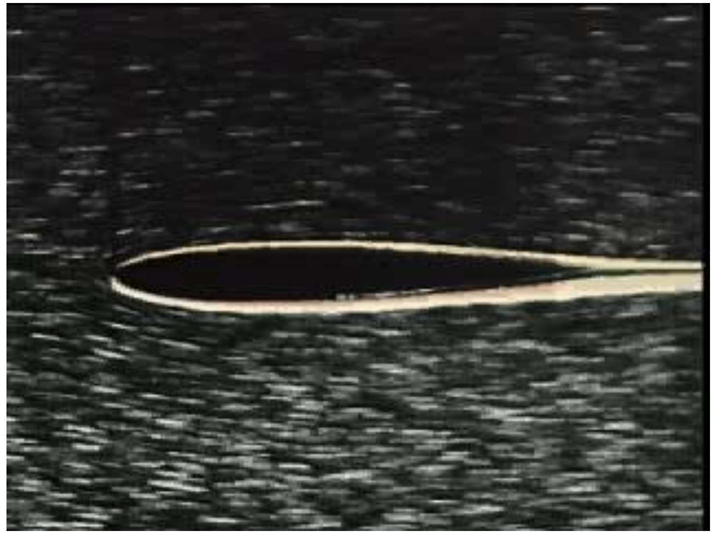
$$c_m = \frac{m}{\frac{1}{2}\rho V^2 Sc}$$

Typical Non-Cambered Airfoil Lift Curve & Drag Polar NACA 0006



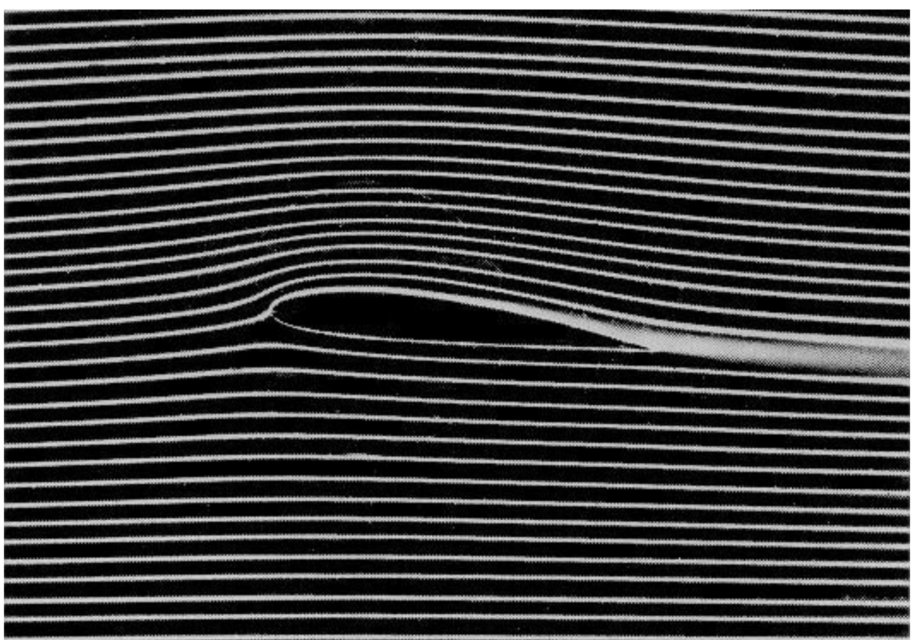


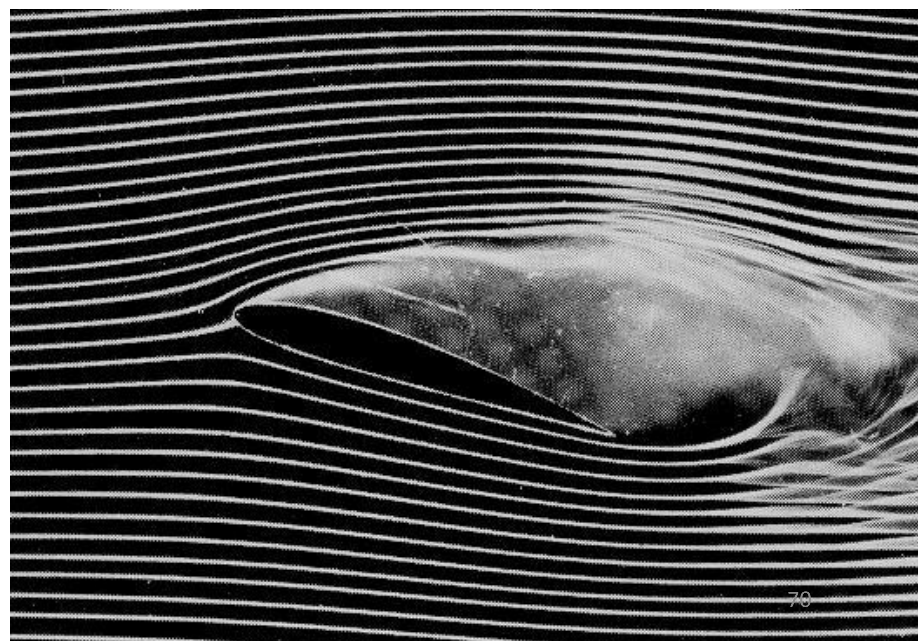
Flow structure on an airfoil



Attached flow

AIRFOIL STALL





Historical Airfoils

Wind Turbine Airfoils

- Design Perspective
  - The environment in which wind turbines operate and their mode of operation not the same as for aircraft
    - Roughness effects resulting from airborne particles are important for wind turbines
    - Larger airfoil thicknesses needed for wind turbines
  - Different environments and modes of operation imply different design requirements
  - The airfoils designed for aircraft not optimum for wind turbines