Flow Over an Airfoil: An Introduction

Lecture 03

Reynolds Number and Mach Number

Reynolds number (Re) = $\rho U_\infty c/\mu$ = Vc/ν

- Ratio of inertial to viscous forces
- How fast is the fluid moving vs. how viscous it is.

Low Re = high viscous forces, more orderly flow. Viscosity prevents the growth of perturbations in flow.

High Re = low viscous forces, less orderly flow. Perturbations grow at a faster rate.

U_∞ = Freestream velocity
c = characteristic linear dimension
μ = kg/s.m (dynamic or absolute viscosity)
$\nu = \mu/\rho$ = m^2/s (kinematic viscosity)

Mach Number (M) = U_∞/c

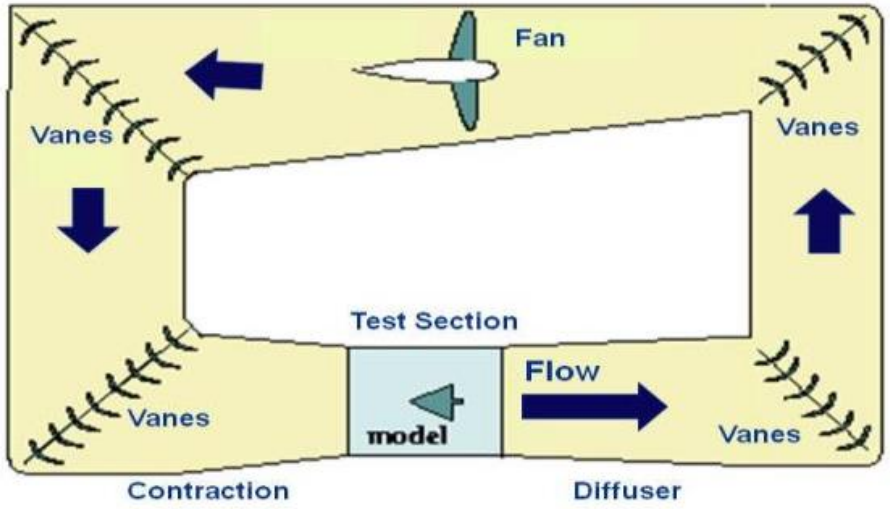
M<0.3 : Incompressible flow
M<0.8 : Subsonic flow
0.8<M<1.2 : Transonic flow
1.2<M : Supersonic flow
5.0<M : Hypersonic flow

c = speed of sound = $(k\ p/\ \rho)^{1/2}$ = $(k\ R\ T)^{1/2}$
(k = Ratio of specific heats; p = pressure; T = temperature; ρ = density)
c = 340.29 m/s at sea level.

Types of Low Speed Wind Tunnel Facilities

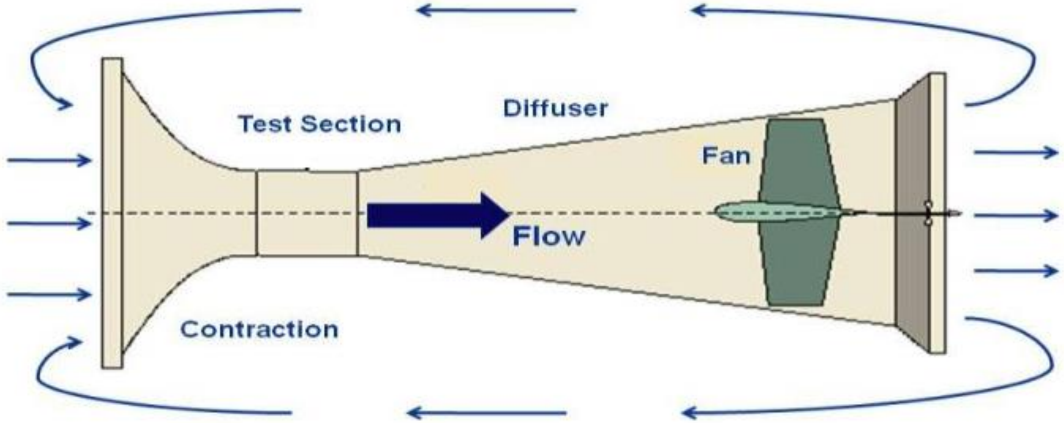
Highest attainable velocity ~ 135 m/s; M=0.4

Closed return



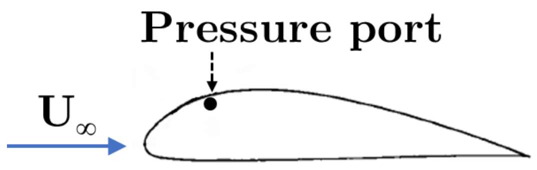
Advantage: Better flow quality,
lower operational cost.
Disadvantage: Higher set-up cost

Open return



Advantage: lower set-up cost.
Disadvantage: Lower flow-quality,
Higher operational cost.

Experimental Technique: Surface Pressure Measurement



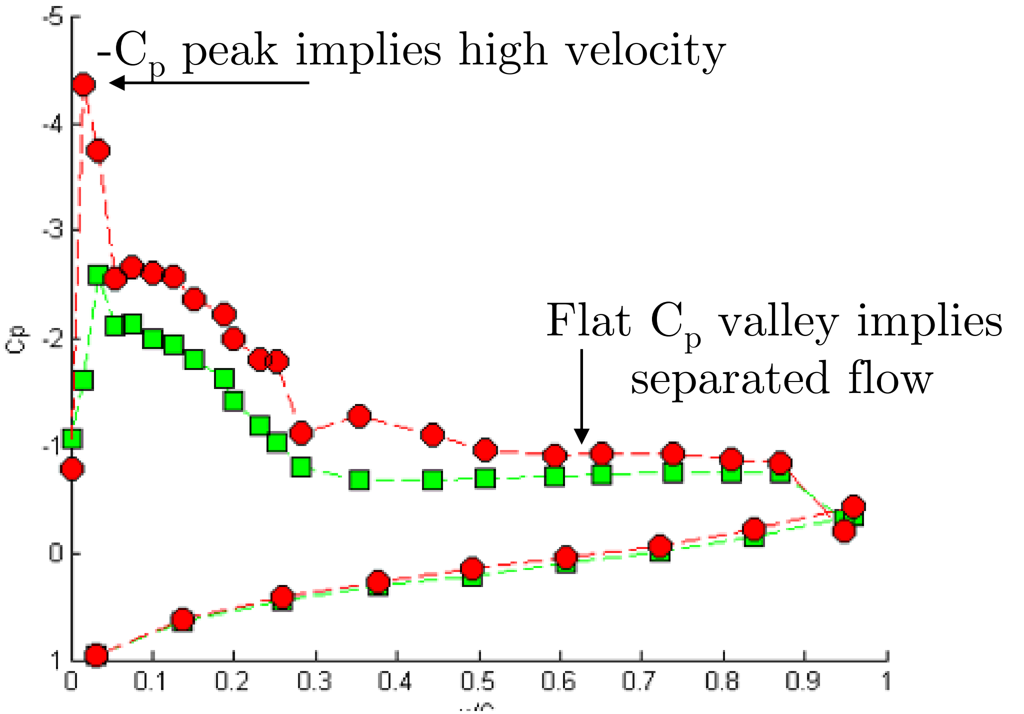
Coefficient of pressure:

$Cp = (p - p_{\infty}) / q_{\infty}$

p = Surface static pressure

p_{∞} = Freestream static pressure

q_{∞} = Dynamic pressure($\rho U_{\infty}^2/2$)

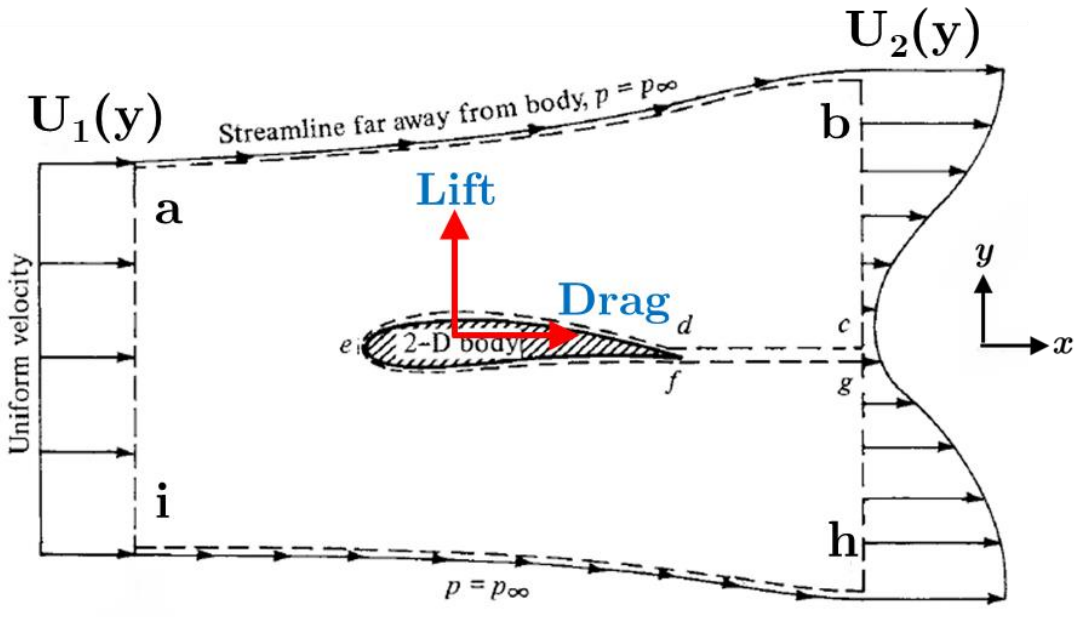


Static measurement: Surface ports are connected to the pressure transducer using tubes.

Dynamic measurement: The transducer is flush mounted to the surface.

Pressure is generally presented as Cp vs. x/C curve.

Experimental Technique: Drag Measurement



Conservation of mass

$$\int_i^a \rho\, u_1\, dy = \int_h^b \rho\, u_2\, dy$$

Conservation of linear momentum in the direction of flow (x)

$$\int_i^a \rho\, u_1^2\, dy = \int_h^b \rho\, u_2^2\, dy + \mathbf{Drag}$$

$$\mathbf{Drag} = \int_h^b \rho\, u_2(u_1 - u_2)\, dy$$

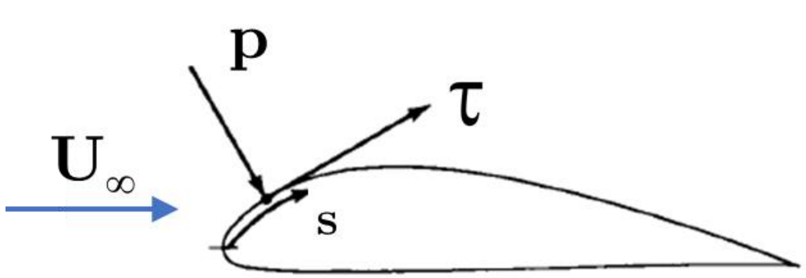
Pressures measured using pitot-static tube

$U_{(x,\, y)} = [2*(Po_{(x,\, y)} - P_{(x,\, y)})/\rho]^{1/2}$

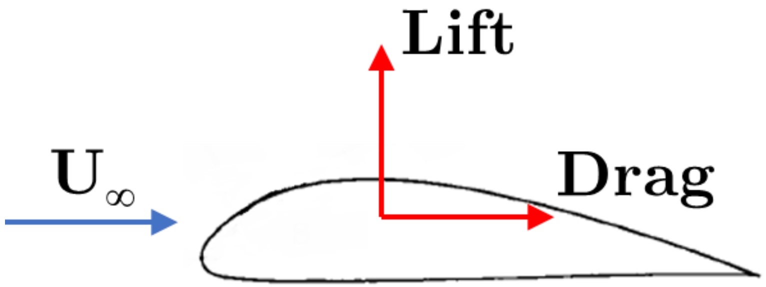
$Po_{(x,\, y)}$ = Local stagnation pressure

$P_{(x,\, y)}$ = Local static pressure

Lift and Drag Forces on a Body



P(s)= pressure distribution
τ (s)=shear stress distribution



Resultant force: $\vec{F} = \int_S d\vec{F}_\tau + \int_S d\vec{F}_p$

$$\vec{F} = F_D\hat{\imath} + F_L\hat{\jmath}$$

$$F_D = F_{D,p} + F_{D,\mu}$$

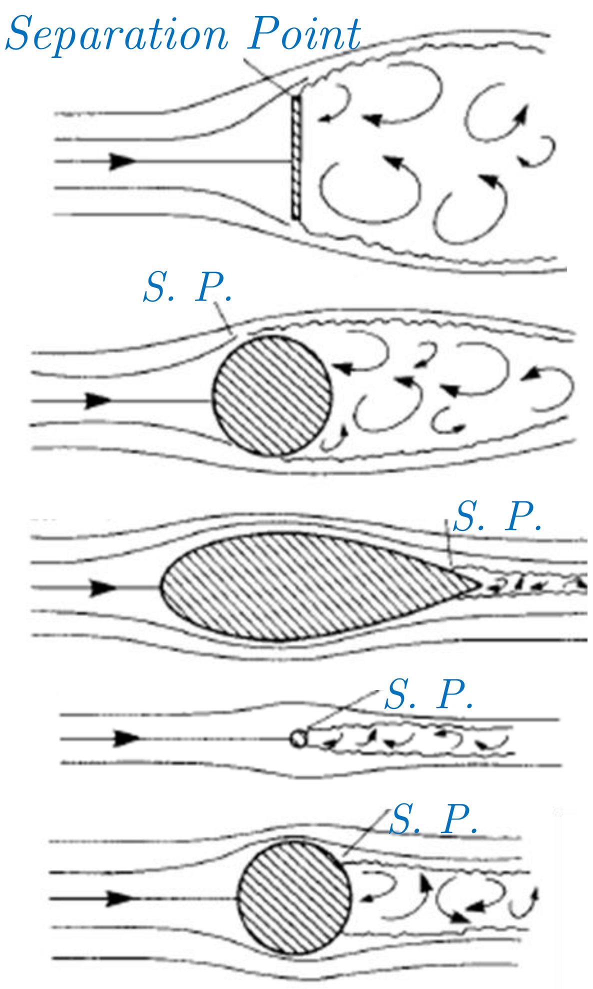
Total drag = Pressure drag + Skin friction drag
(Profile) (Form) (Viscous)

<u>Aerodynamic coefficients:</u>

C_L = Lift/($\rho U_\infty^2/2$) ; C_D = Drag/($\rho U_\infty^2/2$)

L/D ratio = Net Lift / Net Drag

C_D on Different Geometries at Different Reynolds Numbers



Flat plate, length=**d**, Re~**10^5**, C_D=**2.0**

Cylinder diameter=**d**, Re~**10^5**, C_D=**1.2**

Streamline body=**d**, Re~**10^5**, C_D=**0.12**
thickness

Cylinder diameter=**d/10**, Re~**10^4**, C_D=**1.2**

Cylinder diameter=**d**, Re~**10^7**, C_D=**0.6**

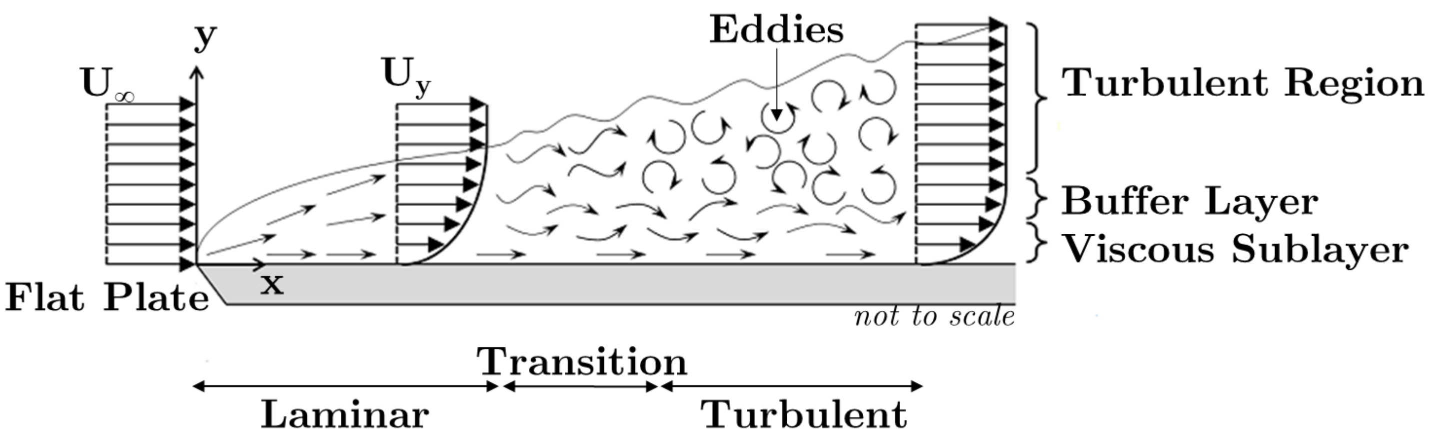
Effect of geometry

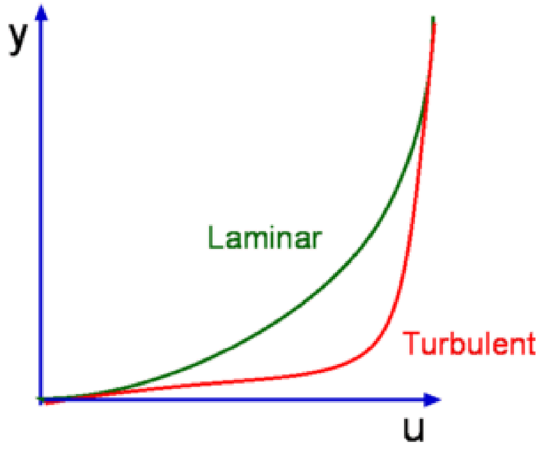
-Delayed separation due to less adverse pressure gradient.

Effect of Re,

-Delayed separation due to laminar to turbulent B.L. transition

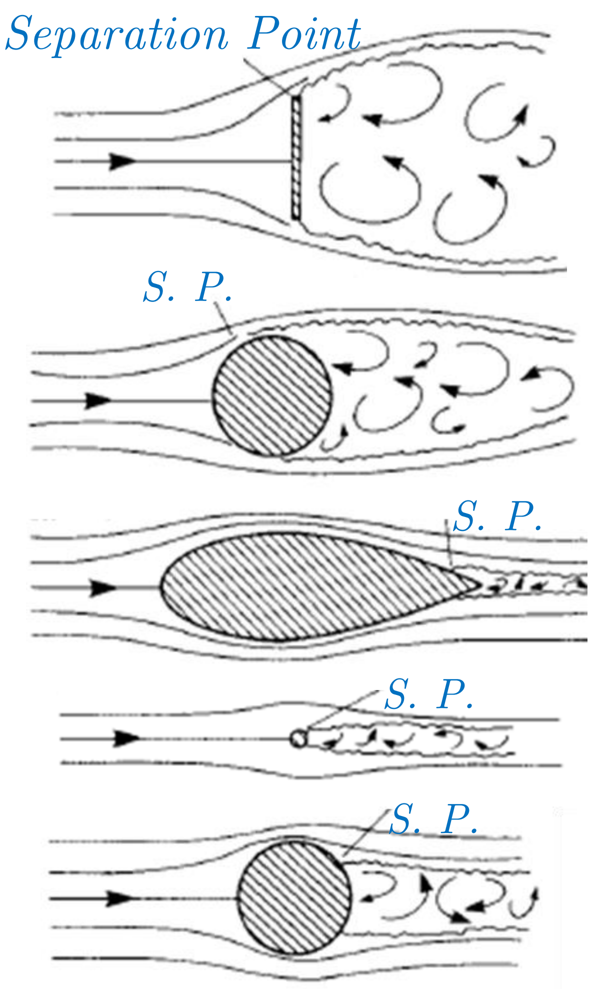
Laminar and Turbulent Boundary Layers



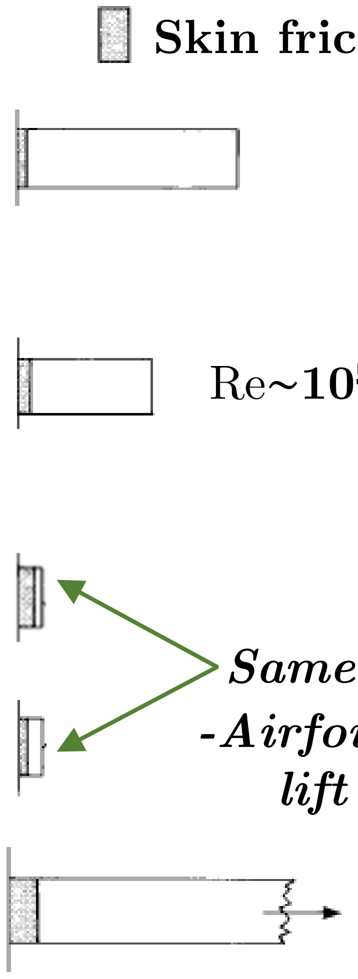


Laminar B.L.	**Turbulent B.L.**
At low Re	At high Re
Flow is laminar throughout	Flow is laminar in the viscous sublayer and turbulent in the rest.
Shear stress on the surface is lower for laminar B.L. **Lower skin friction drag.**	
Turbulent B.L. is more stable and resistant to separation due to higher mass and momentum exchange. **Delayed separation leading to lower pressure drag.**	

C_D on Different Geometries at Different Reynolds Numbers



Skin friction drag, Pressure drag



Re~10^5, C_D=**1.2**

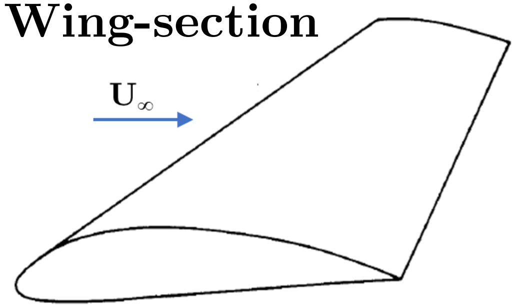
Same total drag

-Airfoil shape helps in achieving high lift at the expense of low drag.

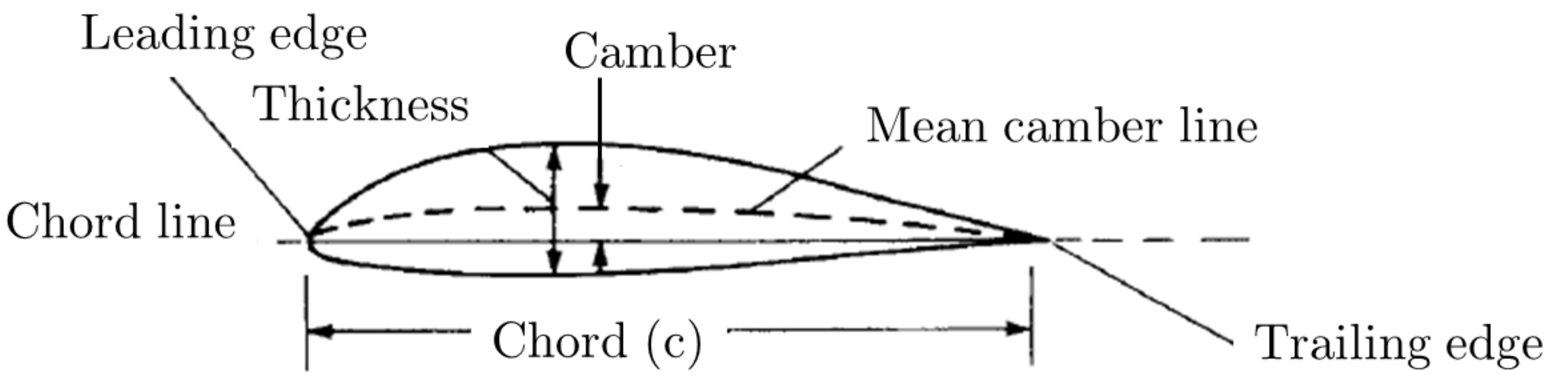
Re~10^7, C_D=**0.6**

Airfoil Geometry

Wing-section



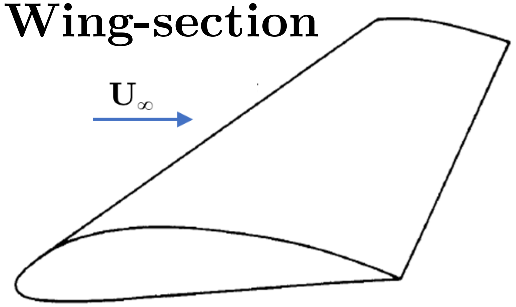
Airfoil Nomenclature



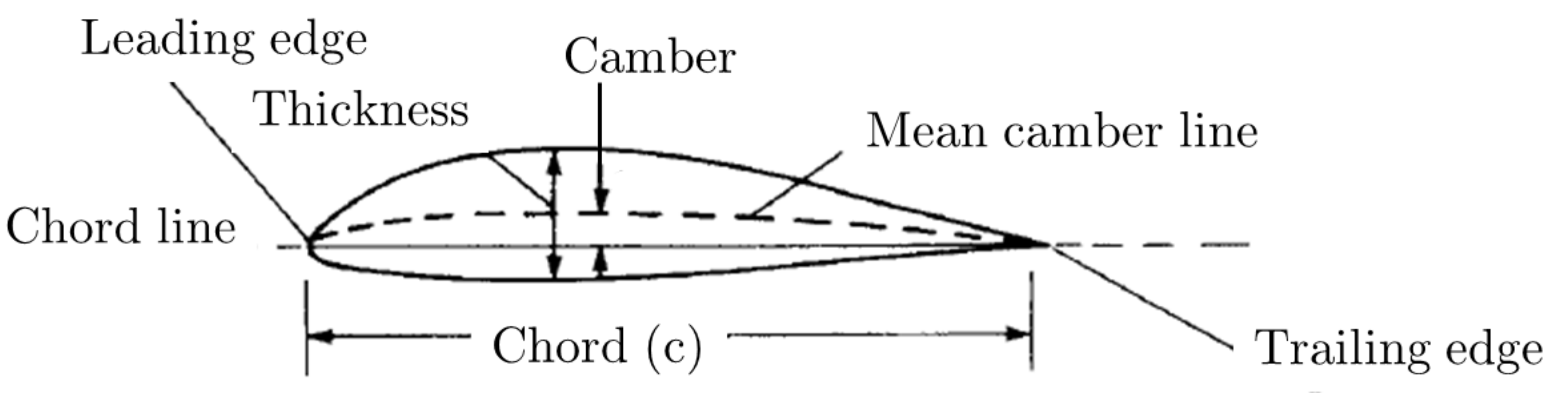
Mean camber line: Locus of points mid-way between upper and lower surface.
Leading edge: Front point of mean camber line.
Trailing edge: End point of mean camber line.
Chord line: Straight line joining the leading and trailing edge.
Chord length (c): Length of the chord line
Camber: Maximum perpendicular distance between chord and camber line.
Thickness: Maximum perpendicular distance between upper and lower surface.

Basic Airfoil Nomenclature

Wing-section



Airfoil Nomenclature

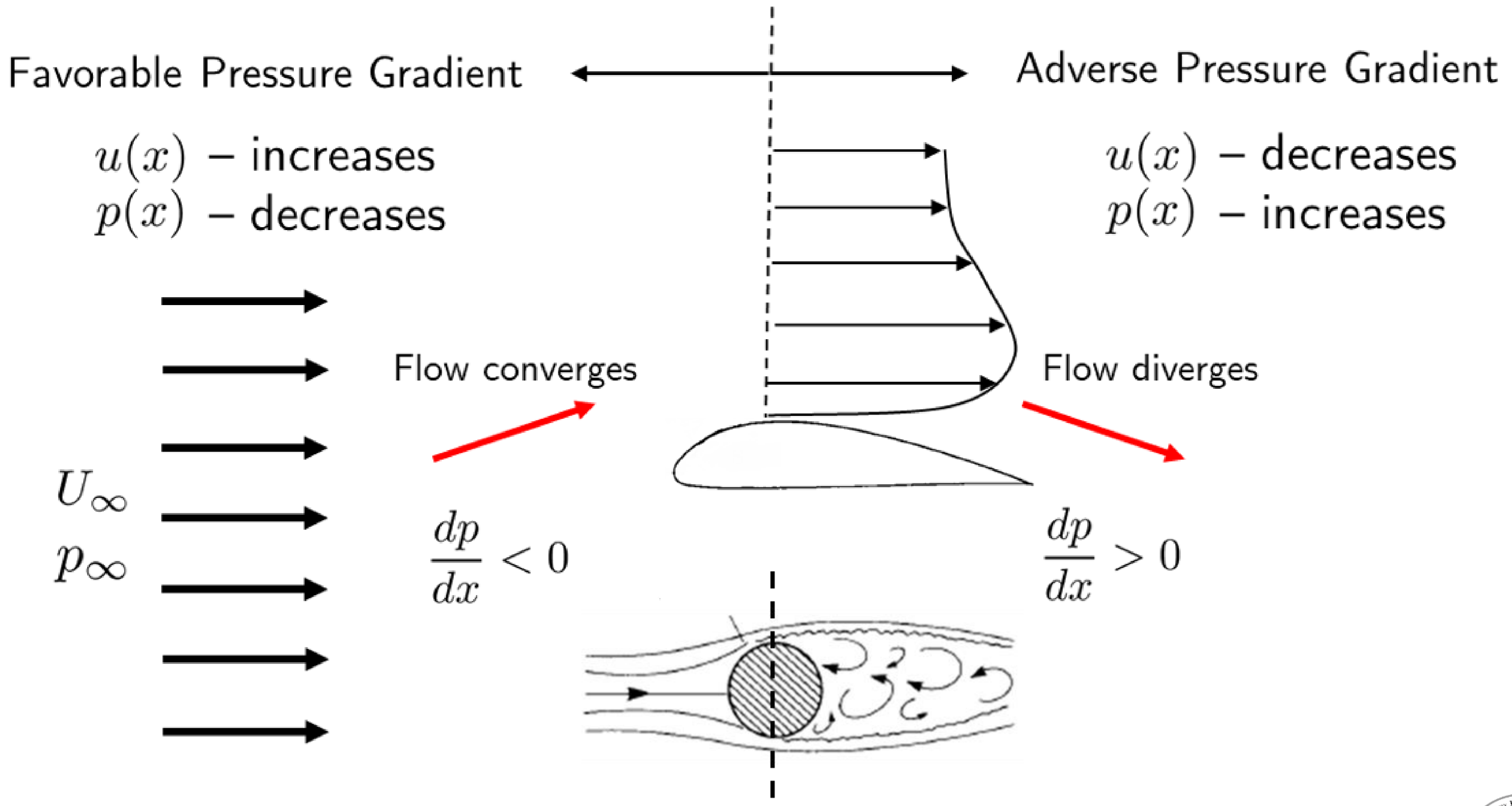


- **NACA four digit airfoil nomenclature (E.g.: NACA 2412)**

Chord (c): Variable
Maximum camber = 2% of c
Location of max camber = 40% of c
Maximum thickness = 12% of c
A Symmetric airfoil (E.g. NACA 0012) has 0 camber.

- ***Other Nomenclatures include NACA "five-digit" series and "6-series".***

Characteristics of Flow over an Airfoil

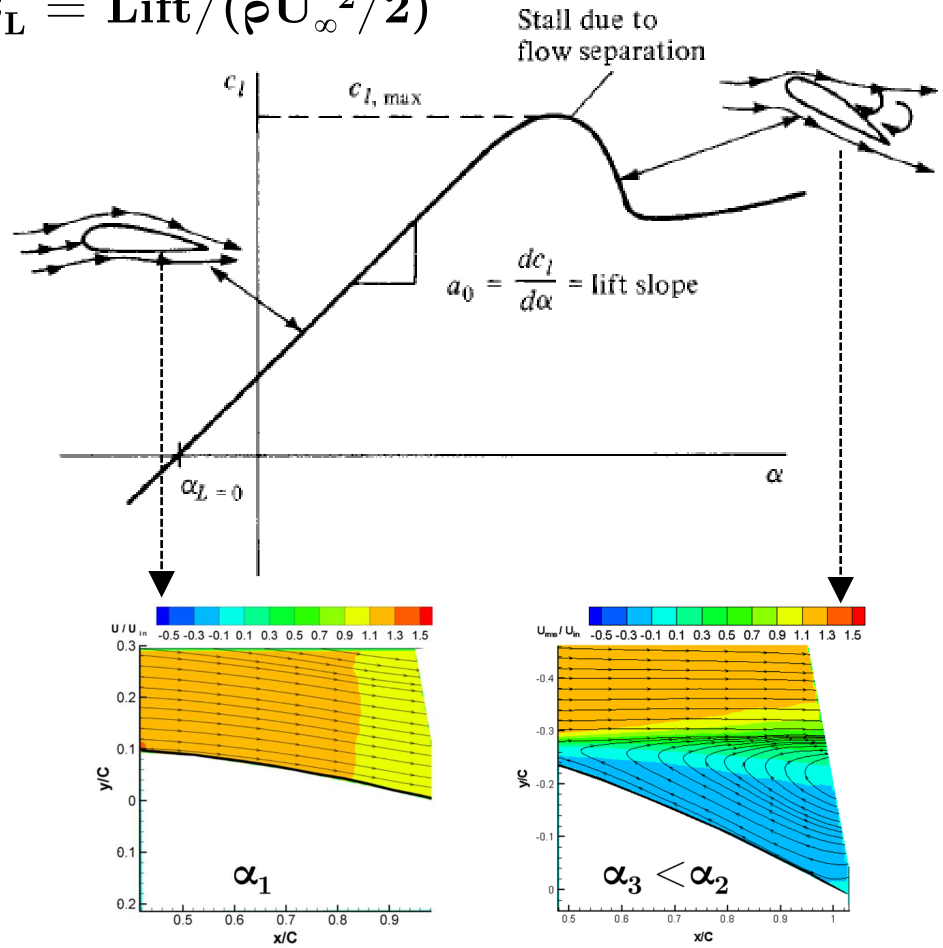


Characteristics of Flow over an Airfoil

- When flow velocity increases its static pressure decreases and vice-versa.
- The flow first accelerate and then decelerate on the top surface of the wing. This results from the curved airfoil geometry.
- Static pressure decreases in the region where flow accelerates leading to a negative or favorable pressure gradient.
- Static pressure increases in the region where flow accelerates leading to a negative or favorable pressure gradient.
- For a cambered airfoil pressure on upper surface is lower than pressure on the lower surface leading to the generation of lift. The same is true for a symmetric airfoil at an angle of attack.

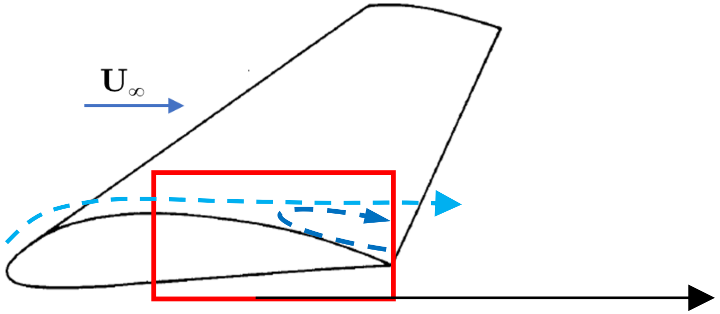
Variation in Lift with Angle of Attack

$C_L = \mathbf{Lift}/(\rho U_\infty^2/2)$

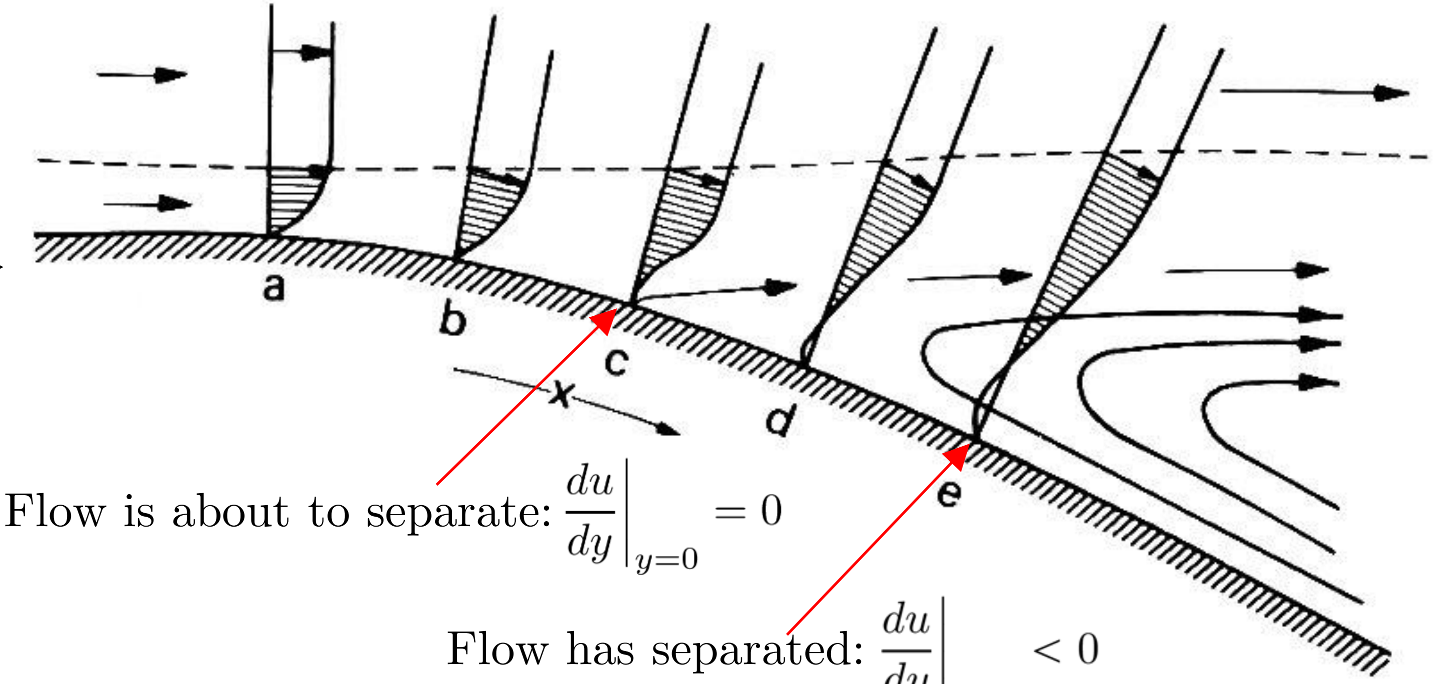


- The plot shows variation in CL with angle of attack (α) for a generic airfoil.
- $\alpha_{L=0}$: α at which C_L is 0.
- $\alpha_{L=0}$=0 for Symmetric airfoil and <0 for cambered airfoil.
- C_L increases linearly with α until stall.
- Stall: Complete separation of flow over airfoil leading to a significant drop in C_L.
- Stall angle: α at which stall occurs, depends on flow regime and airfoil geometry.
- Leading edge stall: Flow separation starts from the leading edge.
- Trailing edge stall: Flow separation starts from the trailing edge.

Flow Separation at High α



Flow features

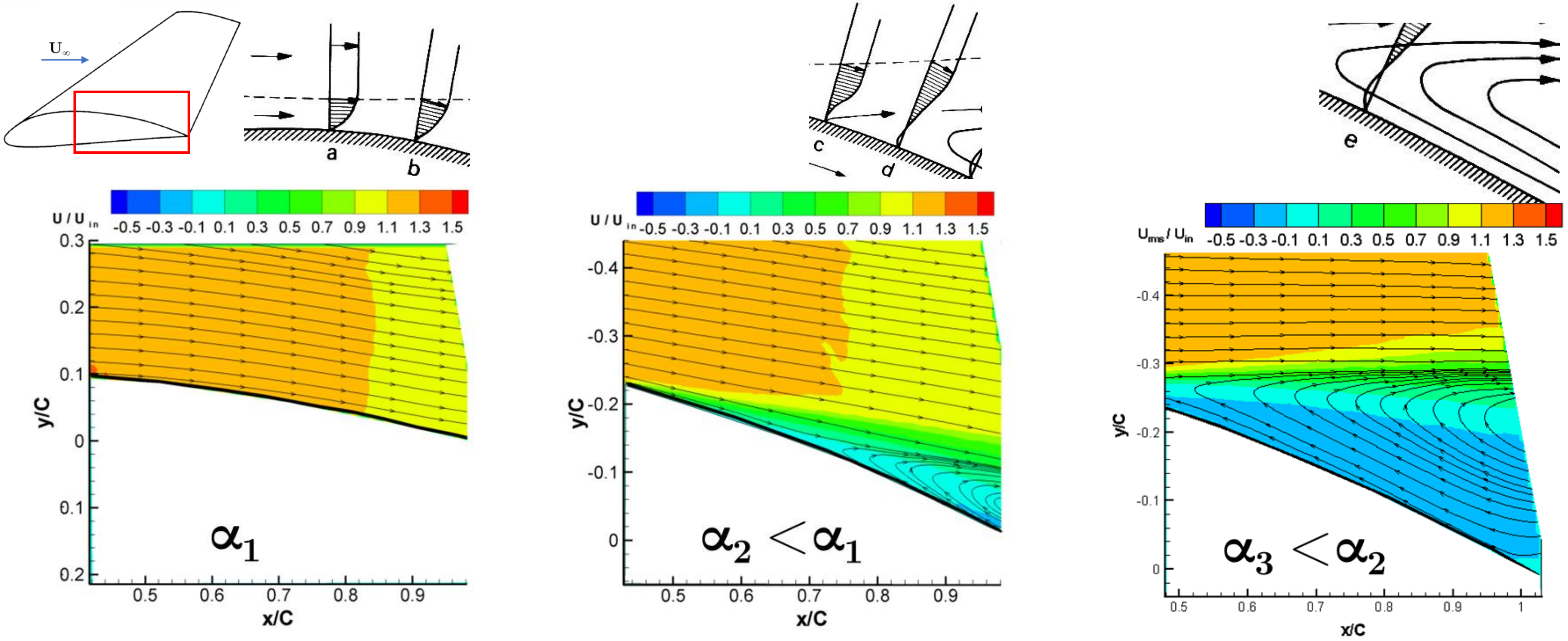


Flow is about to separate: $\left.\dfrac{du}{dy}\right|_{y=0} = 0$

Flow has separated: $\left.\dfrac{du}{dy}\right|_{y=0} < 0$

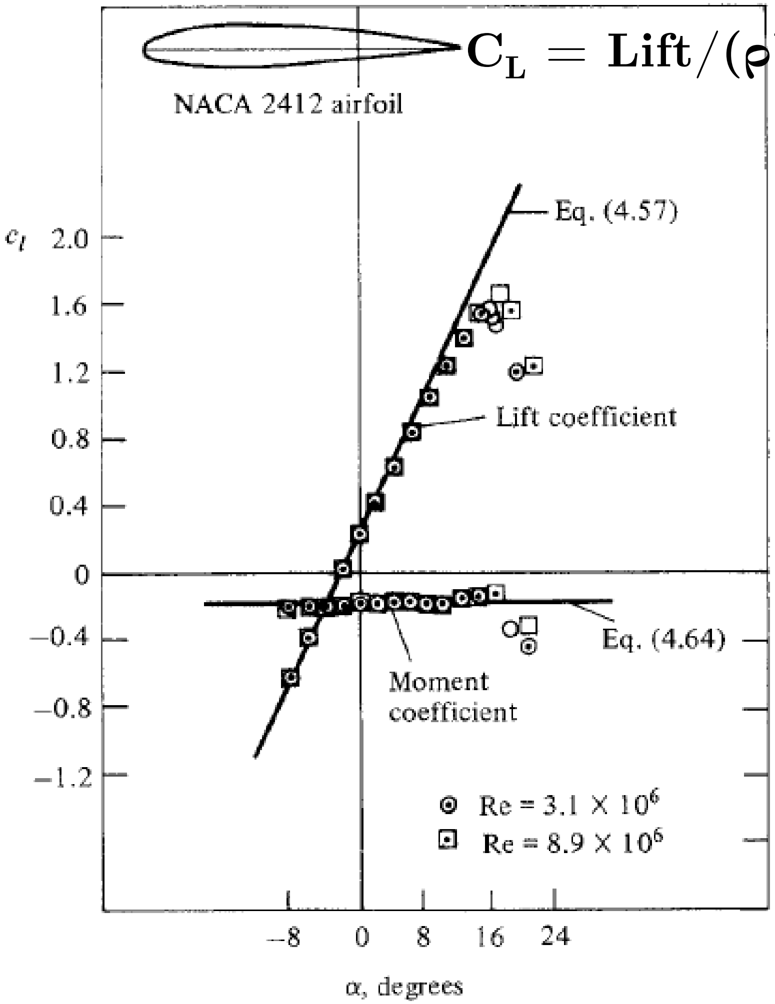
- Adverse pressure gradient can cause the flow to separate from the surface.
- Flow separation leads to a drastic reduction in lift and a large increase in pressure drag.

Flow Separation at High α

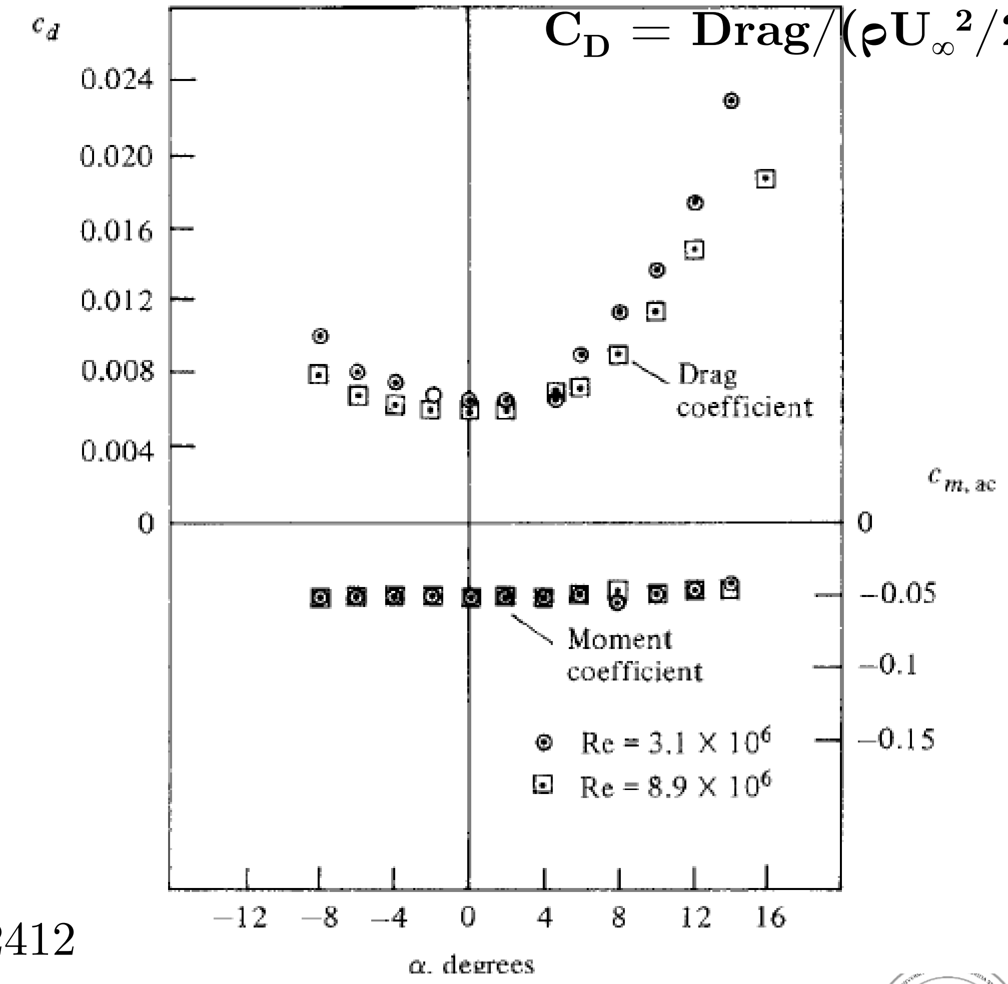


Generic C_L and C_D vs. α Characteristics

$C_L = \text{Lift}/(\rho U_\infty^2/2)$



$C_D = \text{Drag}/(\rho U_\infty^2/2)$



NACA 2412

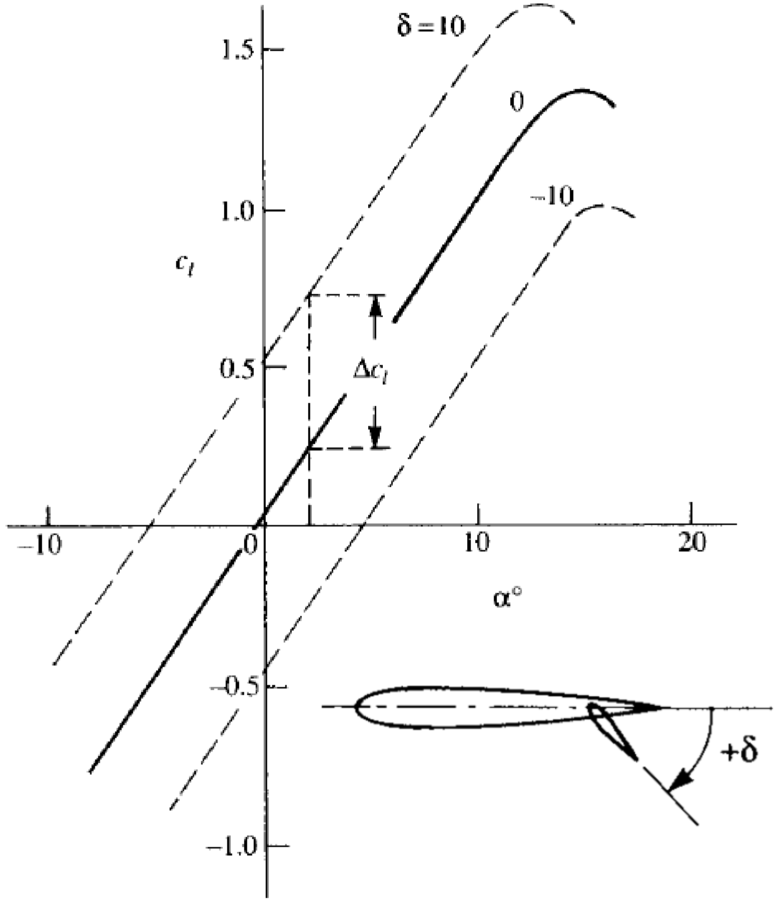
Most Efficient α for an Airfoil

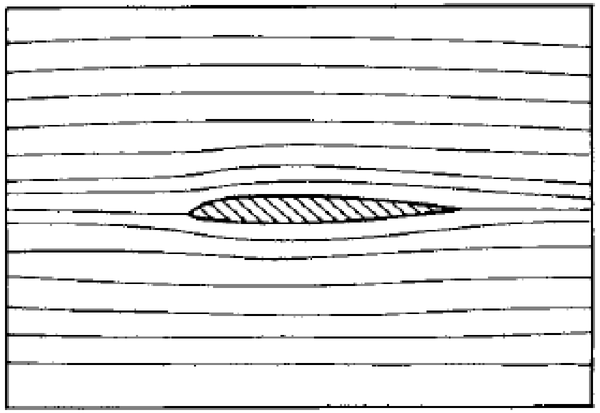
NACA 2412

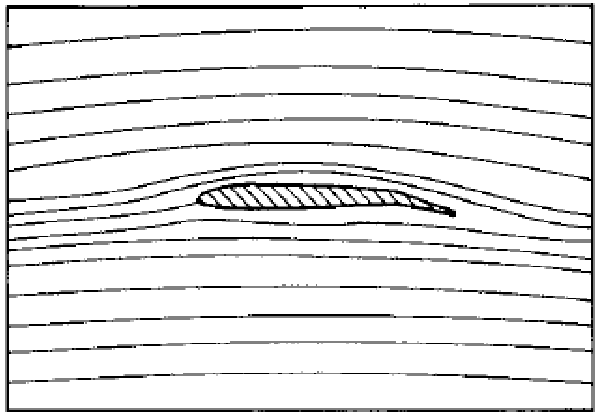
α	C_ℓ	C_d	C_ℓ/C_d
0	0.25	0.0065	38.5
4	0.65	0.0070	93
8	1.08	0.0112	96
12	1.44	0.017	85

- C_L/C_D: Lift to drag ratio (or L/D ratio) is a measure of aerodynamic efficiency.
- Higher the maximum achievable L/D, more efficient the airfoil design.
- Generally the L/D ratio increases with α, reaches a maxima and then decreases.
- For NACA 2414 the maxima occurs at $\alpha = 8°$. Hence maximum aerodynamic efficiency will be achieved at this angle.
- Α for max(L/D) is affected by other parameters when the flow in three dimensional.

Wing with Trailing Edge Flap







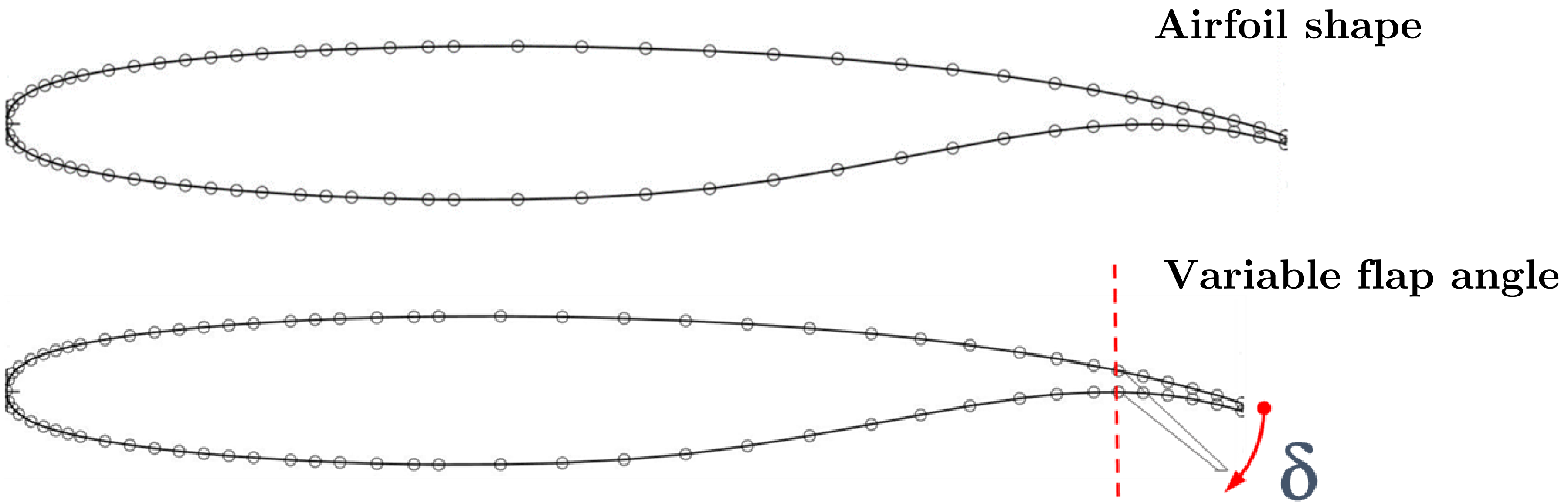
Trailing edge flap changes the effective camber of the airfoil.

$\delta > 0$: Effective camber increases, C_L increases,
Stall α decreases, $\alpha_{L=0}$ decreases.

$\delta < 0$: Effective camber decreases, C_L decreases,
Stall α increases, $\alpha_{L=0}$ increases.

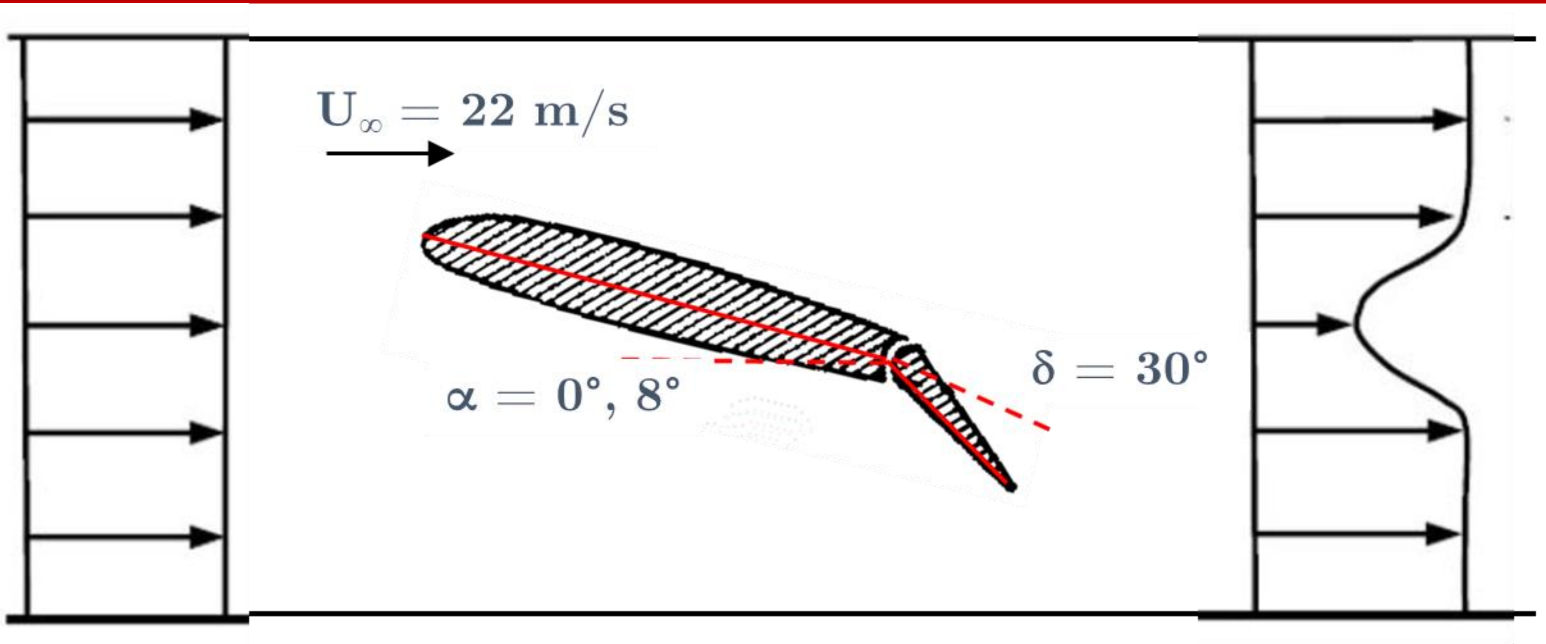
Flaps help in achieving higher lift at low angles of attack.

NASA EET Airfoil



- The airfoil used in this experiment is NASA Energy Efficient Transport (EET) airfoil.
- The higher camber at the trailing edge leads to higher effective camber leading to a higher lift.
- The flap angle can be varied to further increase the lift.
- **Approximate aircraft take-off configuration $\alpha = 8°$ and $\delta = 30$ °.**

Experiment 3



- Measure the stagnation and static pressure in the transverse direction at an upstream and downstream location of a 2D wing-section having NASA EET airfoil geometry.
- Extract velocity profiles from the pressure data.
- Measure total drag from the momentum deficit in the wake.