Aerodynamic Performance of Airfoils

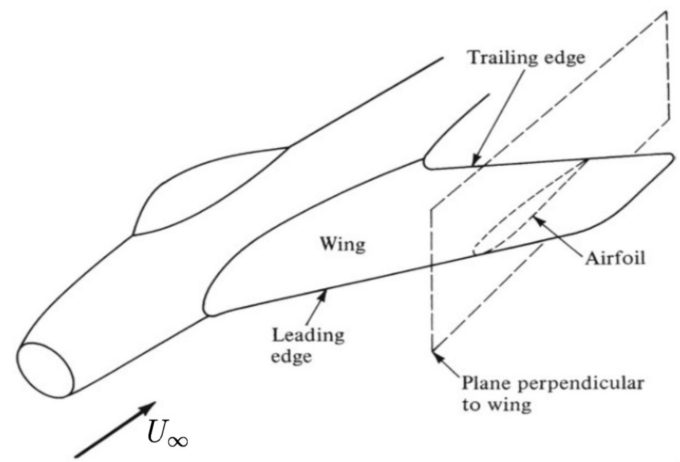
Thermal Fluids and Energy Systems Lab

(ME EN 4650)

Prof. Pardyjak

Department of Mechanical Engineering

University of Utah



(based on Prof. M's lecture slides)

THE UNIVERSITY OF UTAH

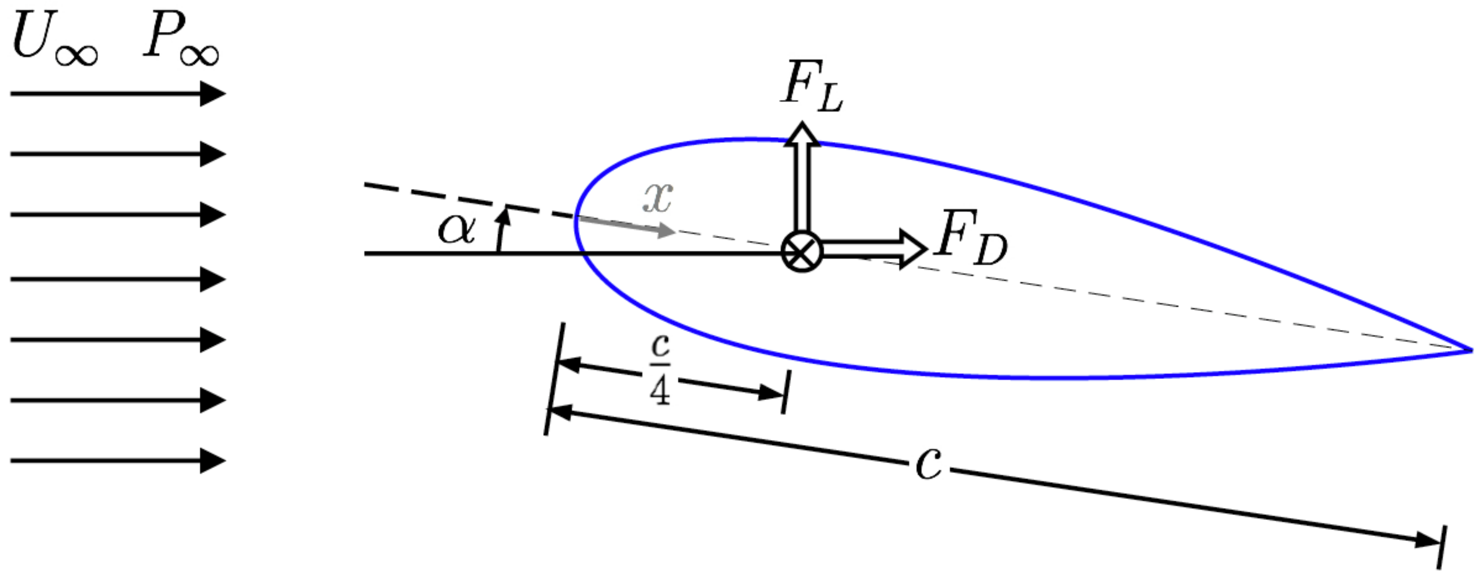
Objectives

- <u>Measure</u> the **lift and drag** forces on a model NACA 0012 airfoil as a function of angle of attack using a lift/drag balance
- <u>Measure</u> the **static pressure** along the top and bottom surfaces of the airfoil at two different angles of attack before and after stall
- Compare the measured data with published

Aerodynamic Performance of Airfoils

- What we want to know:
  - Lift and Drag coefficient vs angle of attack and Reynolds number
  - Static pressure distribution version distance along airfoil
- What we can measure:
  - Lift and drag force (load balance)
  - Static pressure (pressure taps and transducer)
  - Dynamic pressure (pitot-static +transducer)

Airfoil Geometry and Terminology



x : coordinate along chord line
c : chord length
t : max thickness
U_∞ : freestream velocity
P_∞ : freestream static pressure
α : angle of attack
F_D : drag force
F_L : lift force
$\frac{c}{4}$: quarter-chord

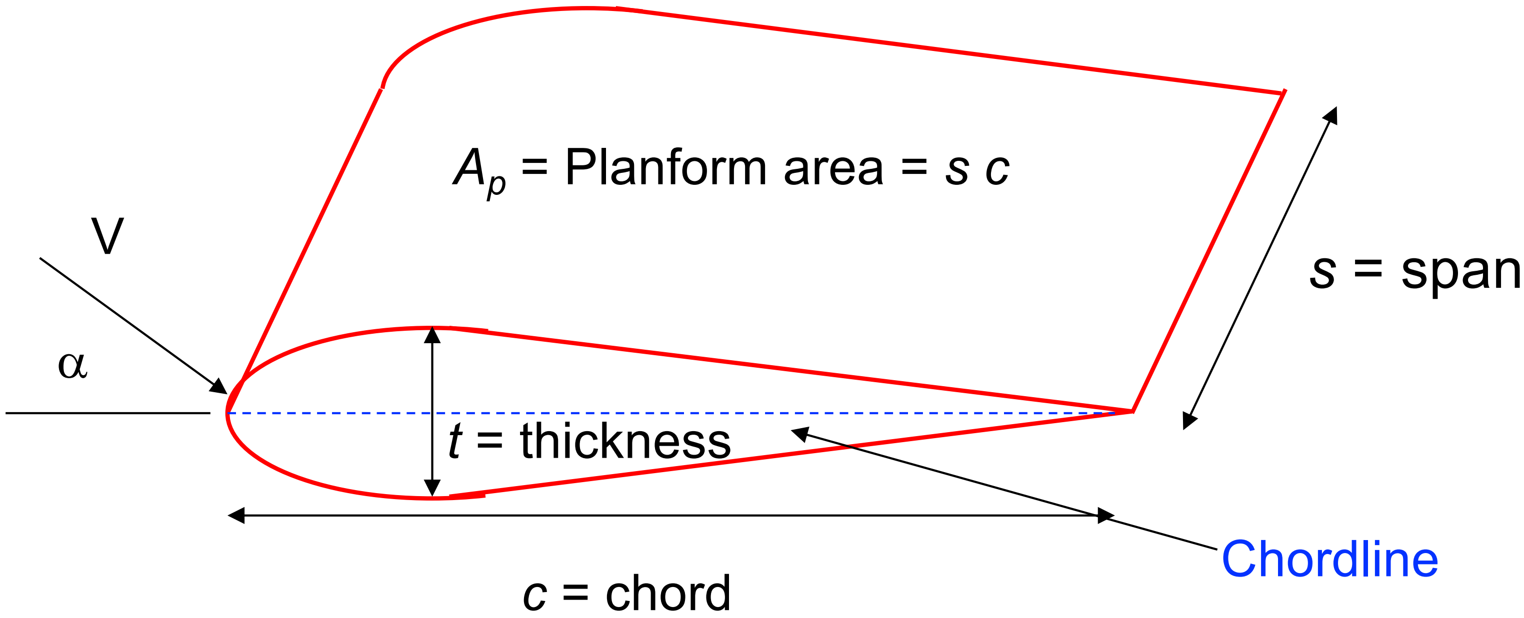
Chord Reynolds Number

$$Re_c = \frac{\rho U_\infty c}{\mu}$$

Lift and Drag Coefficients

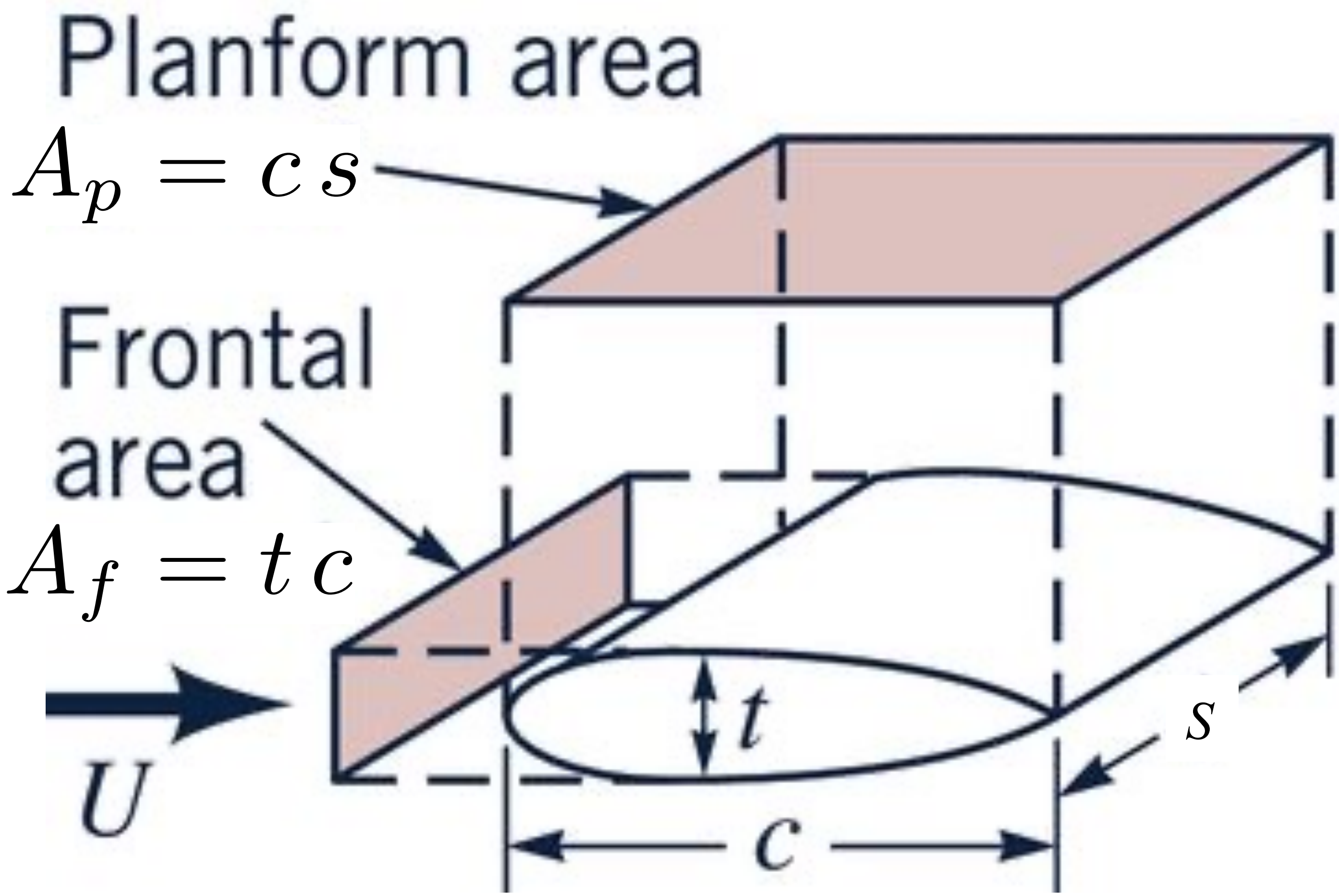
$$C_L = \frac{F_L}{\frac{1}{2}\rho U_\infty^2 (c\,s)} \qquad C_D = \frac{F_D}{\frac{1}{2}\rho U_\infty^2 (c\,s)}$$

Airfoil Geometry and Terminology

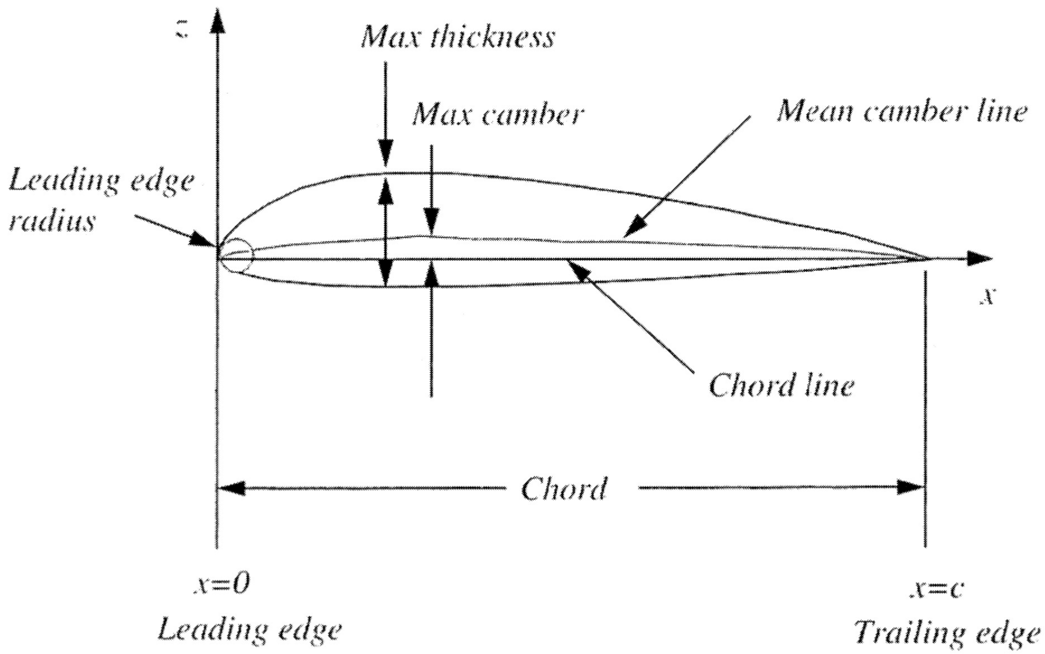


- Chordline – straight line going from the leading edge to the trailing edge
- α - Angle of Attack; angle between the freestream and chordline
- Symmetric Airfoils have no lift at α = 0

Airfoil Planform Area & Frontal Area



Flow around an Airfoils – Cambered (Chordline is not an axis of symmetry)



Cambered Airfoils <u>have</u> lift at $\alpha = 0$

Naming Convention (others exist):

Distance from leading edge to the location of max camber in tenths of the chord

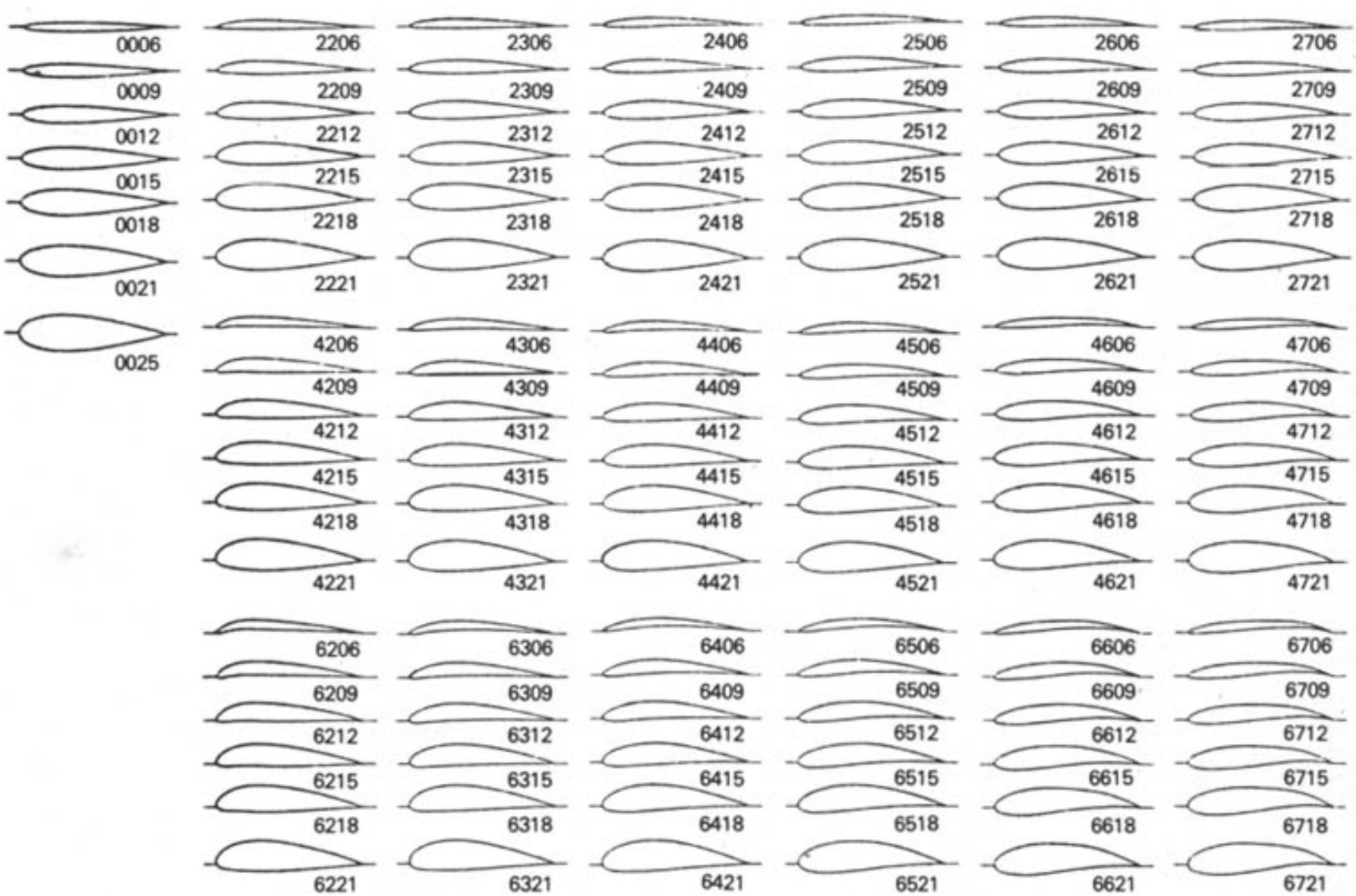
NACA 4-digit Series: Example NACA 0012

Max camber as a % chord

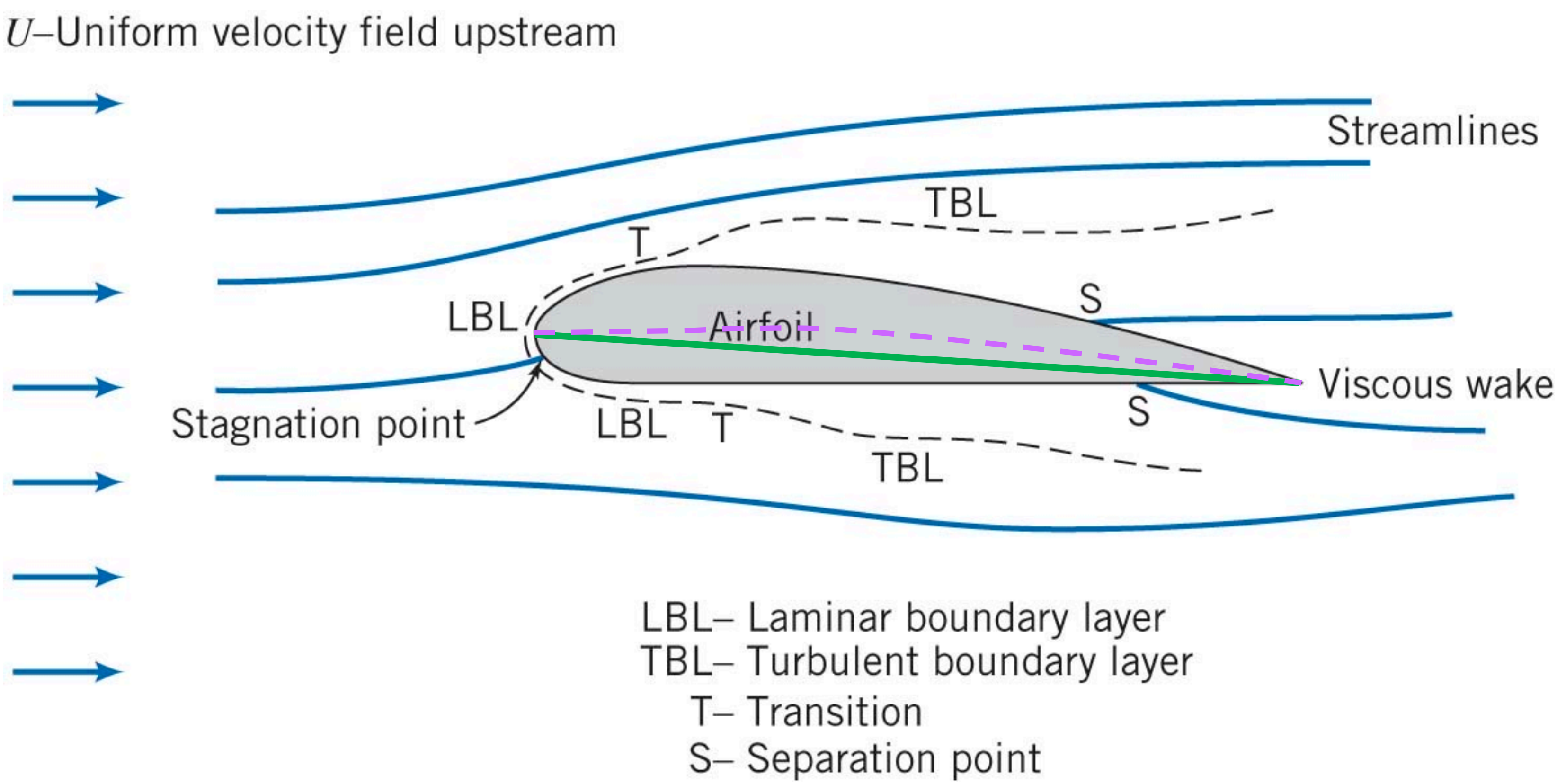
Max Section thickness in % chord

NACA-National Advisory Committee for Aeronautics

NACA Airfoils from1920-1958



Flow around an Airfoil



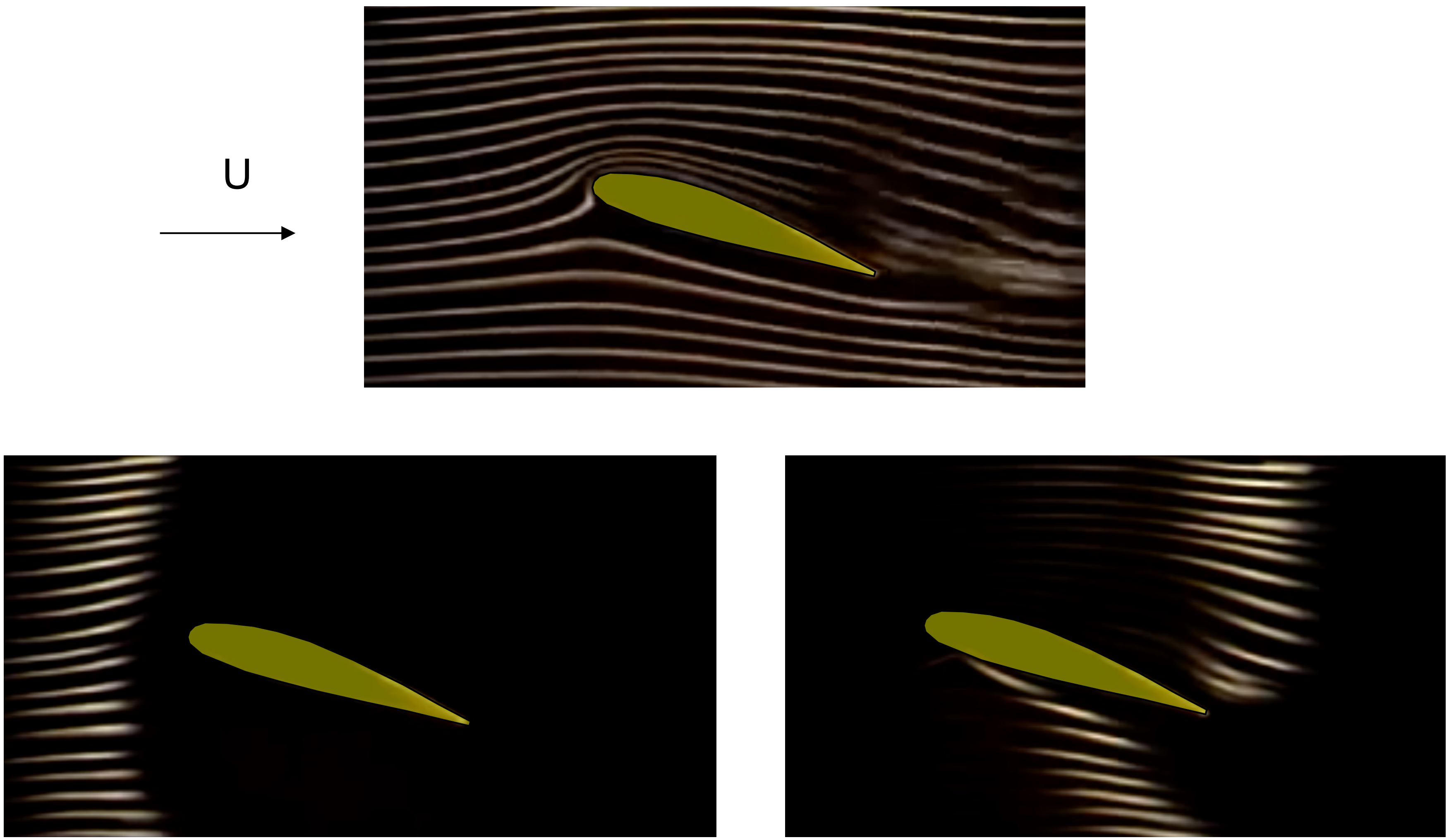
Flow Visualization (changing angle of attack)

Flow Visualization

$\alpha = 0°$

$\alpha = 11°$

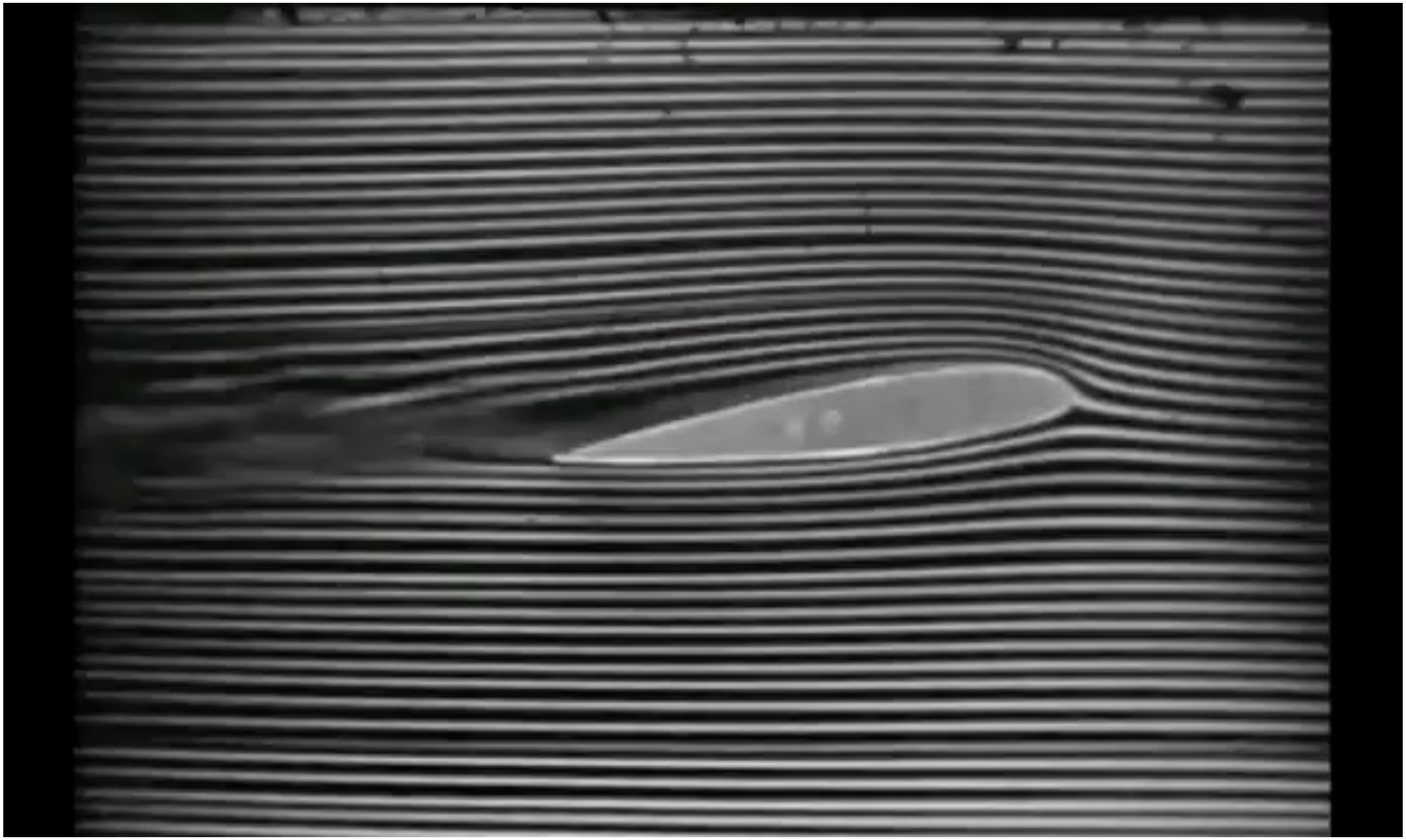
Generation of Lift



Lift Force

- https://techtv.mit.edu/collections/ifluids/videos/32593-boundary-layer-control
- Pressure distribution due to
  - Bernoulli's equation
  - Euler-n equation
- Momentum flux

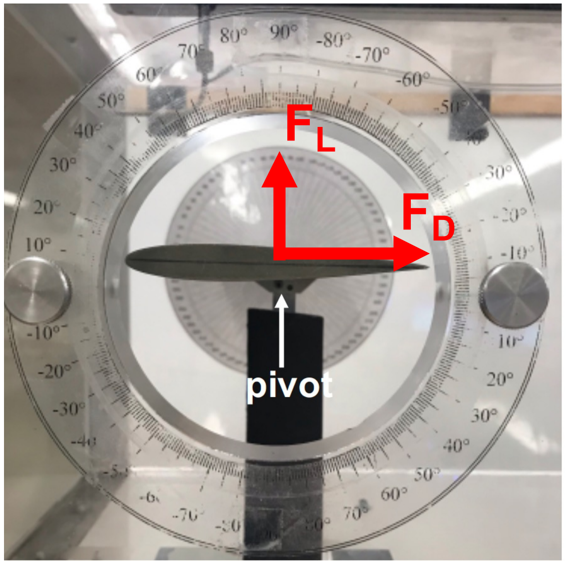
Flow Visualization: Stall



Measurements

12-inch x 12-inch
ELD wind tunnel

1. Airfoil attached to
a lift & drag balance



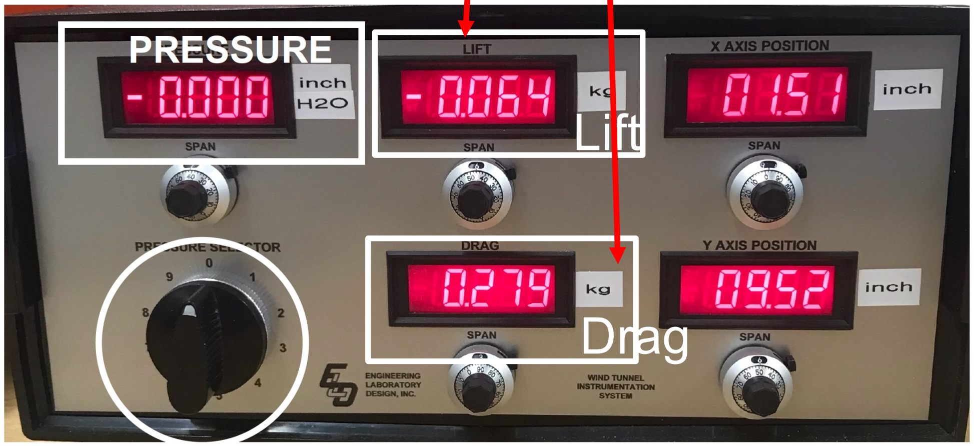
Measurements

12-inch x 12-inch ELD wind tunnel

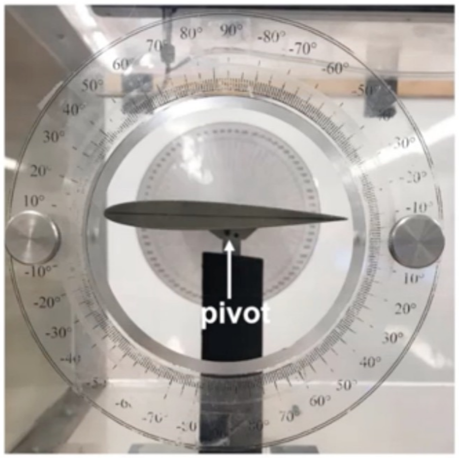
2. Airfoil with pressure taps

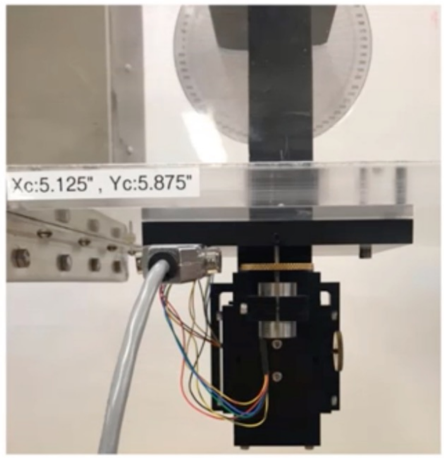
Measurements

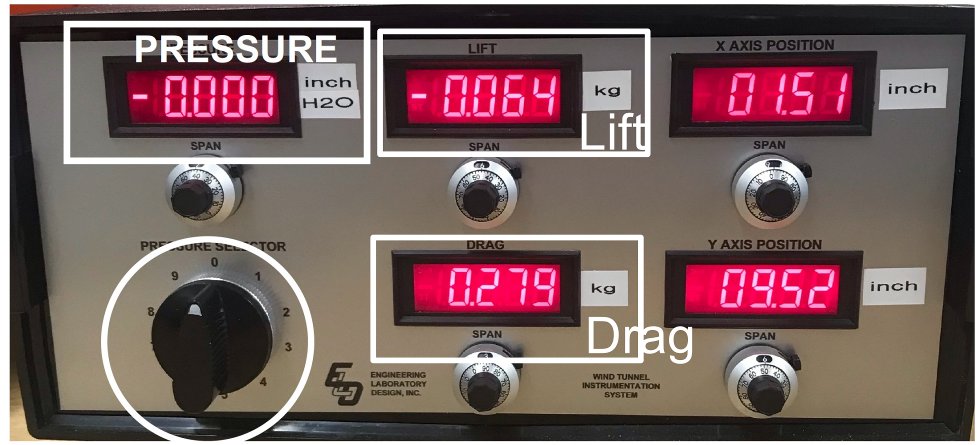
Quantity	Symbol	Units	Instrument
Static pressure on airfoil	$P_x - P_{atm}$	in H_2O	surface taps on airfoil
lift + drag forces	F_L, F_D	kg	Lift/drag balance
Freestream dynamic pressure	$\frac{1}{2}\rho U_\infty^2$	in H_2O	Pitot-static probe



Measuring Airfoil Lift/Drag





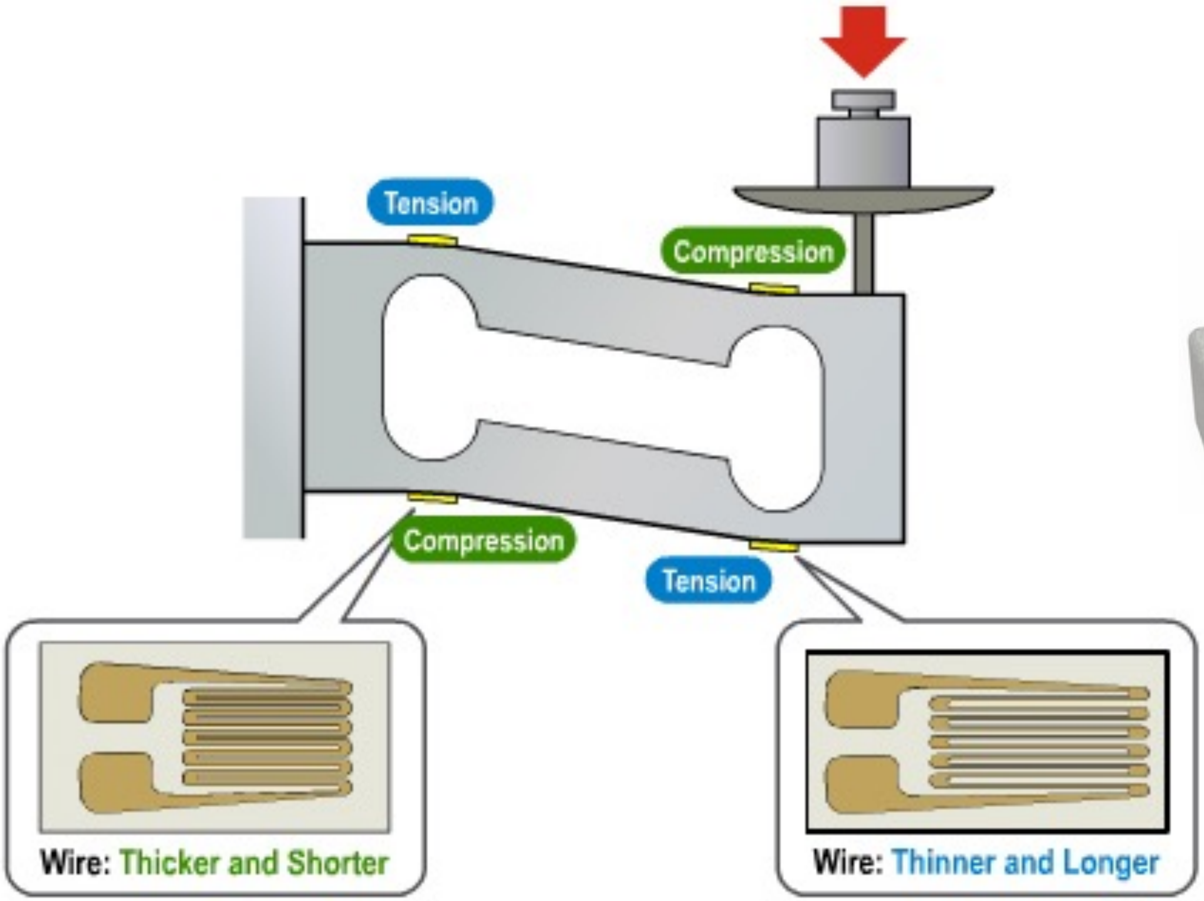


We will use the data acquisition system and PC to make the measurements

1. Zero the balance – take a baseline measurement
2. Zero the pressure transducer – measure the freestream velocity
3. Take measurements at the difference angles of attack (0-16 degrees in 1-degree increments)

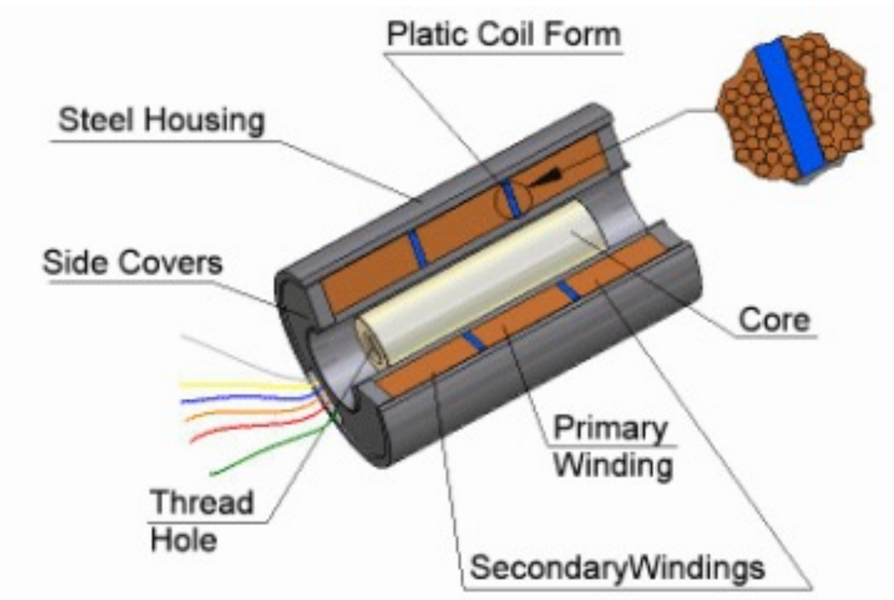
Measuring Airfoil Lift/Drag

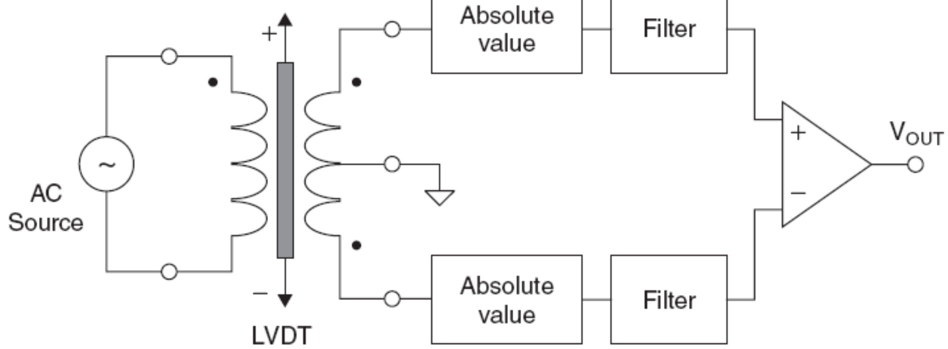
Strain Gauge Load Cell

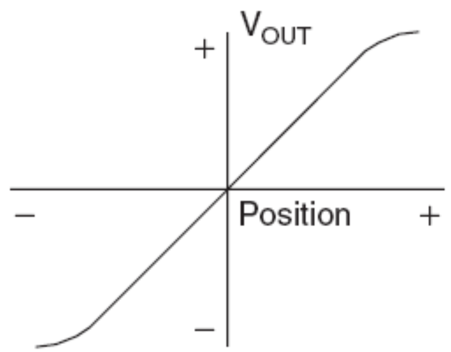


Measuring Airfoil Lift/Drag

LVDT (Linear Variable Displacement Transducer)

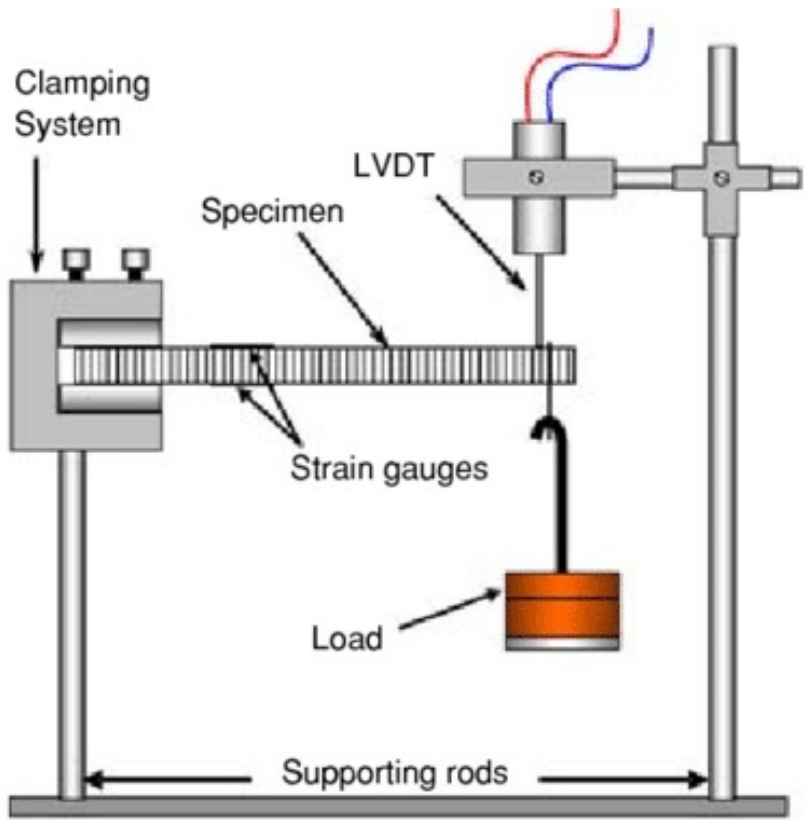




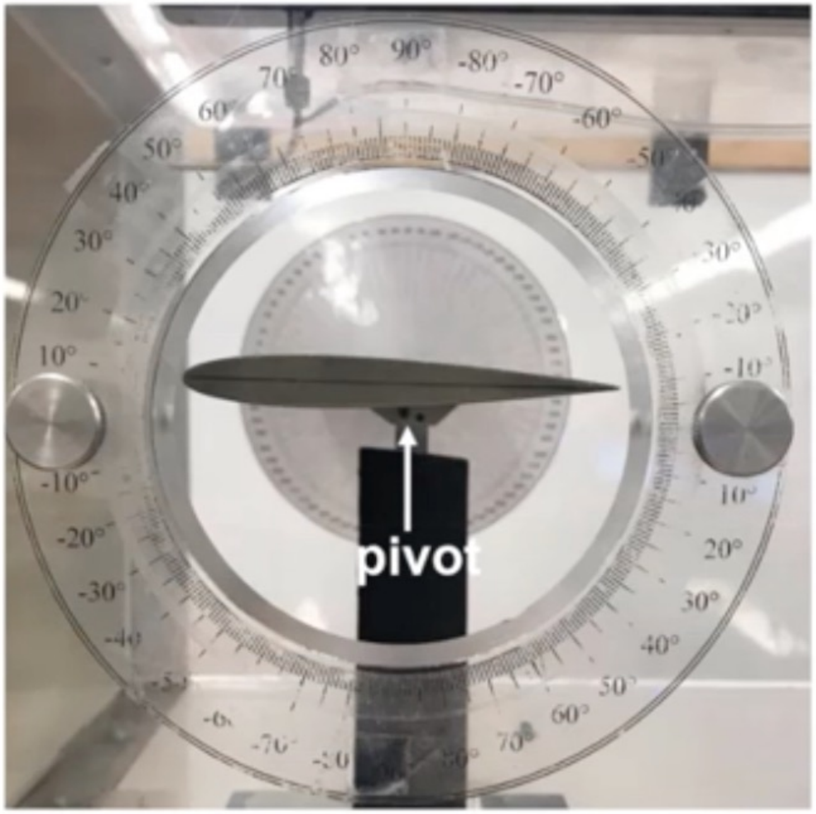


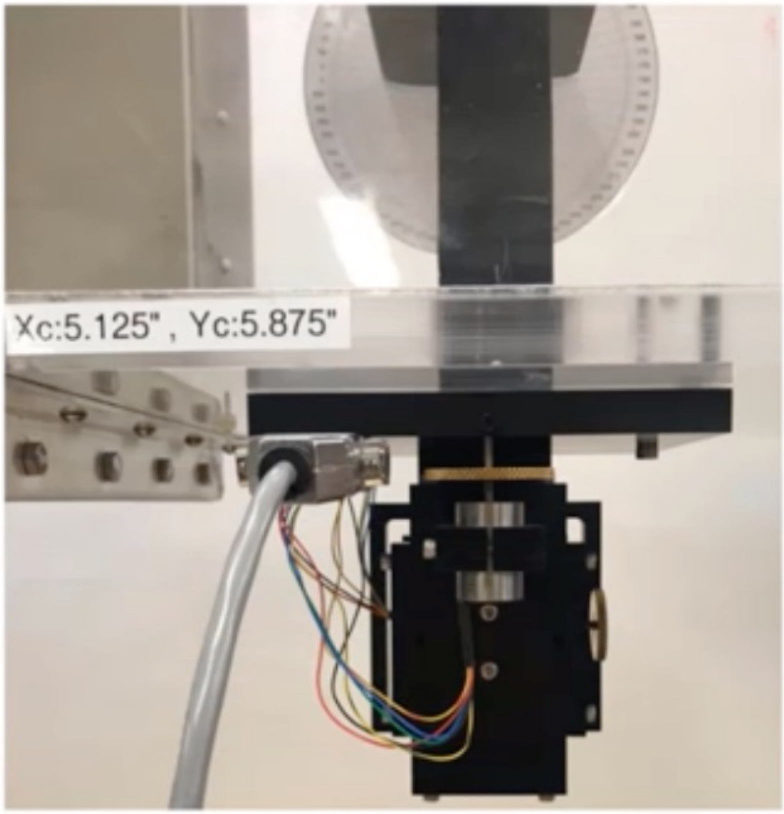
Measuring Airfoil Lift/Drag

LVDT (Linear Variable Displacement Transducer)



Measuring Airfoil Lift/Drag

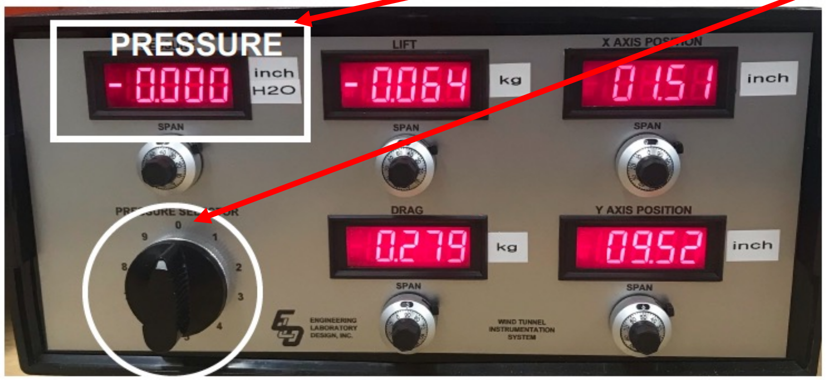


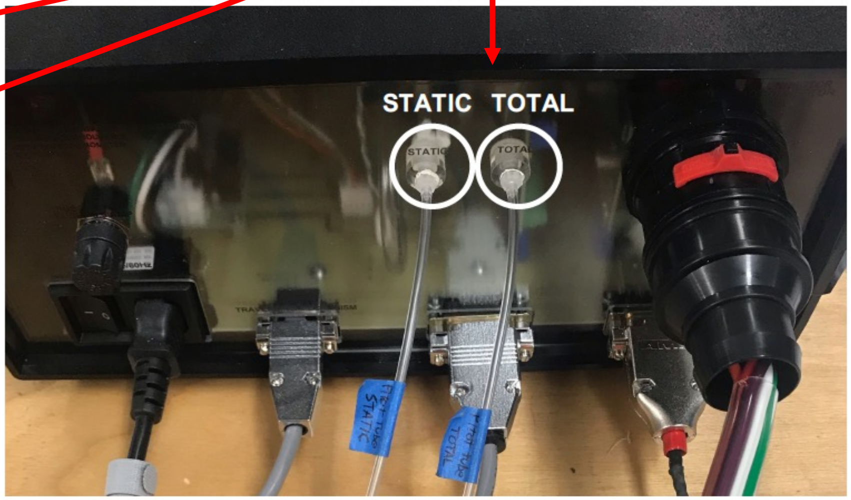


Measurements

Quantity	Symbol	Units	Instrument
Static pressure on airfoil	$P_x - P_{\text{atm}}$	in H_2O	surface taps on airfoil
lift + drag forces	$F_L,\ F_D$	kg	Lift/drag balance
Freestream dynamic pressure	$\frac{1}{2}\,\rho U_\infty^2$	in H_2O	Pitot-static probe

$$U_\infty = \sqrt{\frac{2(P_T - P_S)}{\rho}}$$

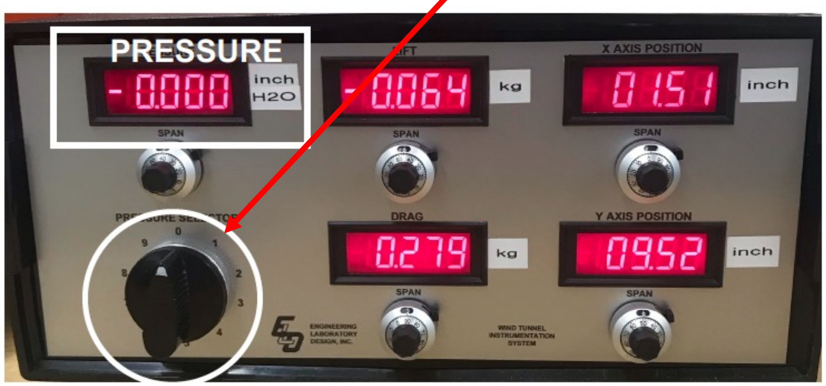


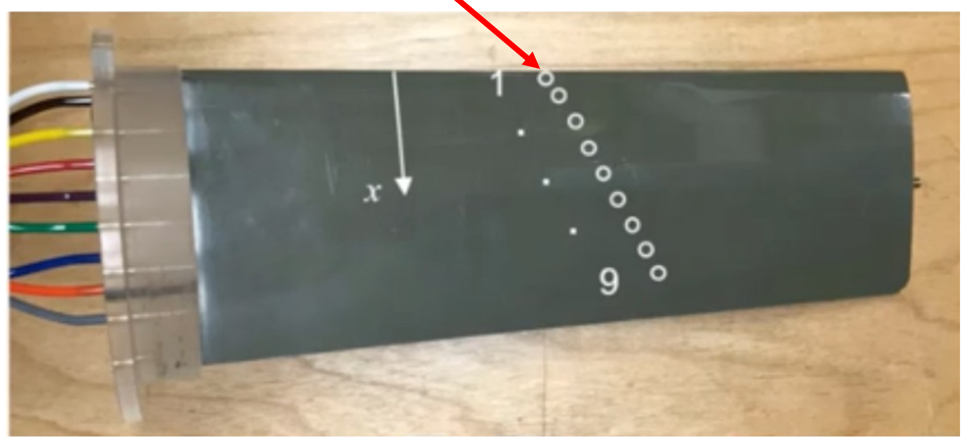


Measurements

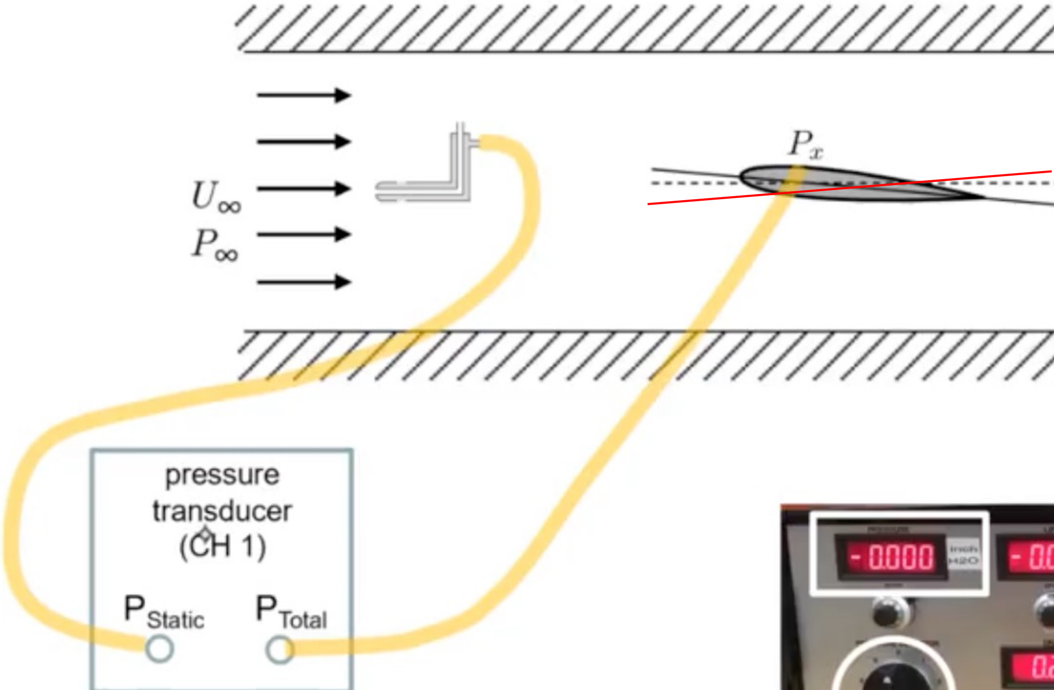
Quantity	Symbol	Units	Instrument
Static pressure on airfoil	$P_x - P_{\text{atm}}$	in H_2O	surface taps on airfoil
lift + drag forces	F_L, F_D	kg	Lift/drag balance
Freestream dynamic pressure	$\frac{1}{2}\,\rho U_\infty^2$	in H_2O	Pitot-static probe

Measure pressure at a **positive** <u>and</u> **negative** angle of attack





Post-Processing Raw Data: Airfoil Relative Static Pressure



Tilt airfoil positive and negative angles

$$\text{output:}\ \Delta P = P_{\text{Total}} - P_{\text{Static}}$$

$$\Delta P = P_x - P_\infty$$

Coefficient of pressure: $C_p = \dfrac{P_x - P_\infty}{\frac{1}{2}\rho U_\infty^2}$

Experiments

Parameter	Symbol	Value
airfoil	—	NACA 0012
freestream velocity	U_∞	$\sim$25 m/s
chord length	c	4 in
max thickness	t	0.48 in
span length	s	12 in
chord Reynolds number	Re_c	$\sim 1.5 \times 10^5$
angle of attack	α	$0°$–$16°$

Experiment 1:

- Measure F_D & F_L versus α (1-degree increments)

Experiment 2:

- Measure $P_x - P_\infty$ versus x (top and bottom surfaces)

Quantities of Interest (Required Figures)

- C_L vs α (data w/ errorbars + published results*)

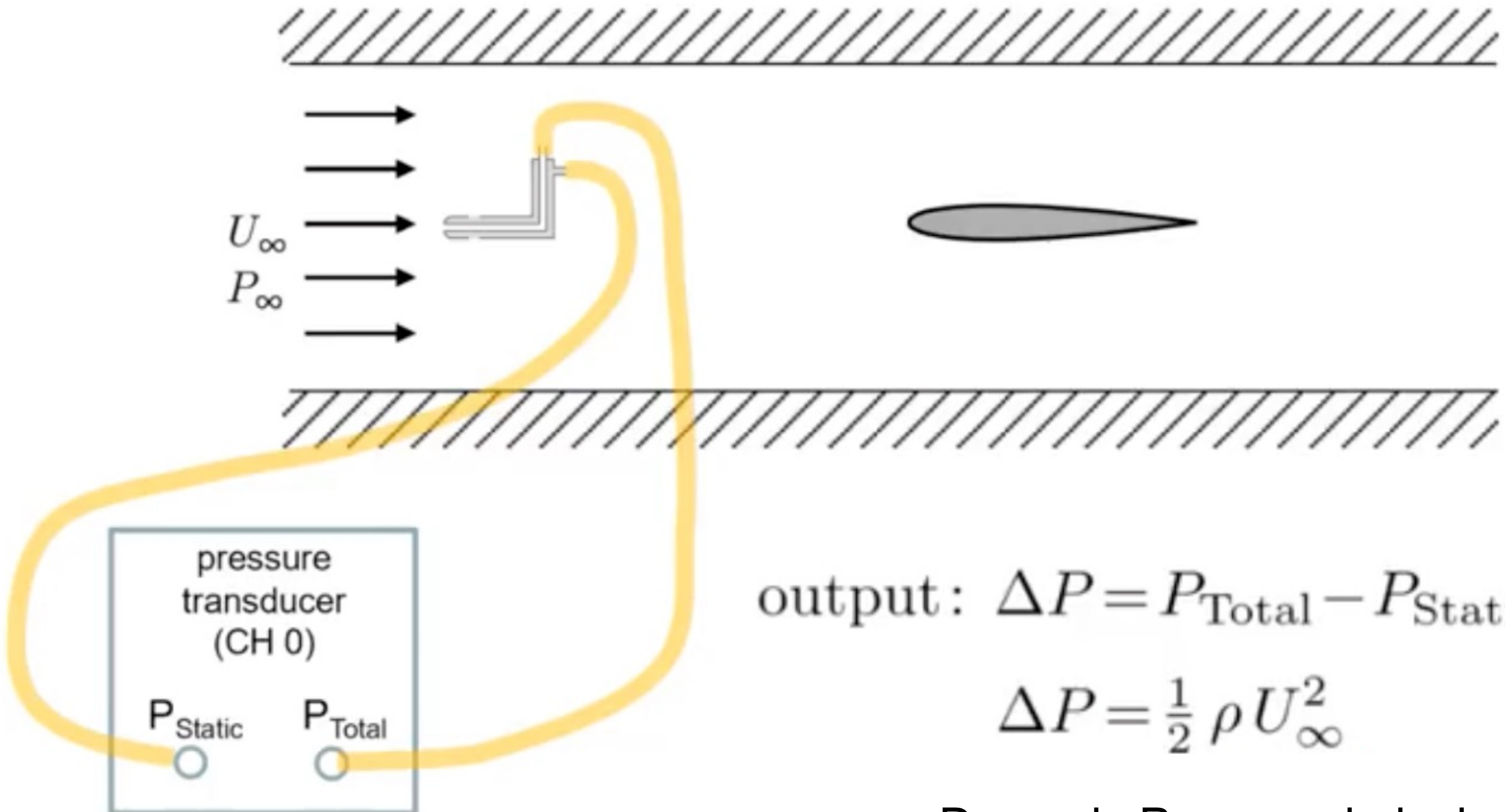
Use Matlab .fig file to plot

- C_D vs α (data w/ errorbars + published results*)

- *i.* C_p vs x/c at $\alpha \cong 5^\circ$ (top and bottom surface)
  ii. C_p vs x/c at $\alpha \cong 12^\circ$ (top and bottom surface)

*R. Sheldahl and P. Klimas, SAND80-2114 (1981)

Dynamic Pressure



$$\text{output}: \ \Delta P = P_{\text{Total}} - P_{\text{Static}}$$

$$\Delta P = \tfrac{1}{2}\rho U_\infty^2$$

Dynamic Pressure in inches H20
- Convert to Pa + use atm measurements of T and Patm for density

Post-Processing Raw Data: Lift & Drag Forces

1. Read in baseline data

```
data = readtable('baseline.txt');
L_base = mean((data(:,1)); %take ave of baseline data
D_base = mean(data(:,2));
```

2. Create a FOR-LOOP: read in lift/drag files for all angles of attack

Post-Processing Raw Data:
Lift & Drag Forces

2. Create a FOR-LOOP: read in lift/drag files for all angles of attack

```
for i = ... %loop through time series data files for diff
angles of attack
filename =[. . .];
data = readtable(filename);
L = (data(:,1)-L_base)*g; %multiply by gravity to get Newtons
D = (data(:,2)-D_base)*g;
% Average value for each angle of attack
Lavg(i) = mean(L);
Davg(i) = mean(D);

% Compute the standard error for the Lift and Drag
Lerr(i) = std(L)/sqrt(n);
Derr(i) = std(D)/sqrt(n);
end
```

$$\sigma_{\overline{F}_D} = \frac{\sigma_{F_D}}{\sqrt{N}}$$

ndependent samples in
ies
s/s * 15 = 120

Post-Processing Raw Data: Lift & Drag Forces

$$C_L = \frac{F_L}{1/2\,\rho\,U_\infty^2\,A_p} \qquad C_D = \frac{F_D}{1/2\,\rho\,U_\infty^2\,A_p}$$

3. Compute lift and drag coefficients and standard error

```
%Compute lift and draf coefficients
CL = Lavg/(Pdyn*c*s);
CD = Davg/(Pdyn*c*s);
```

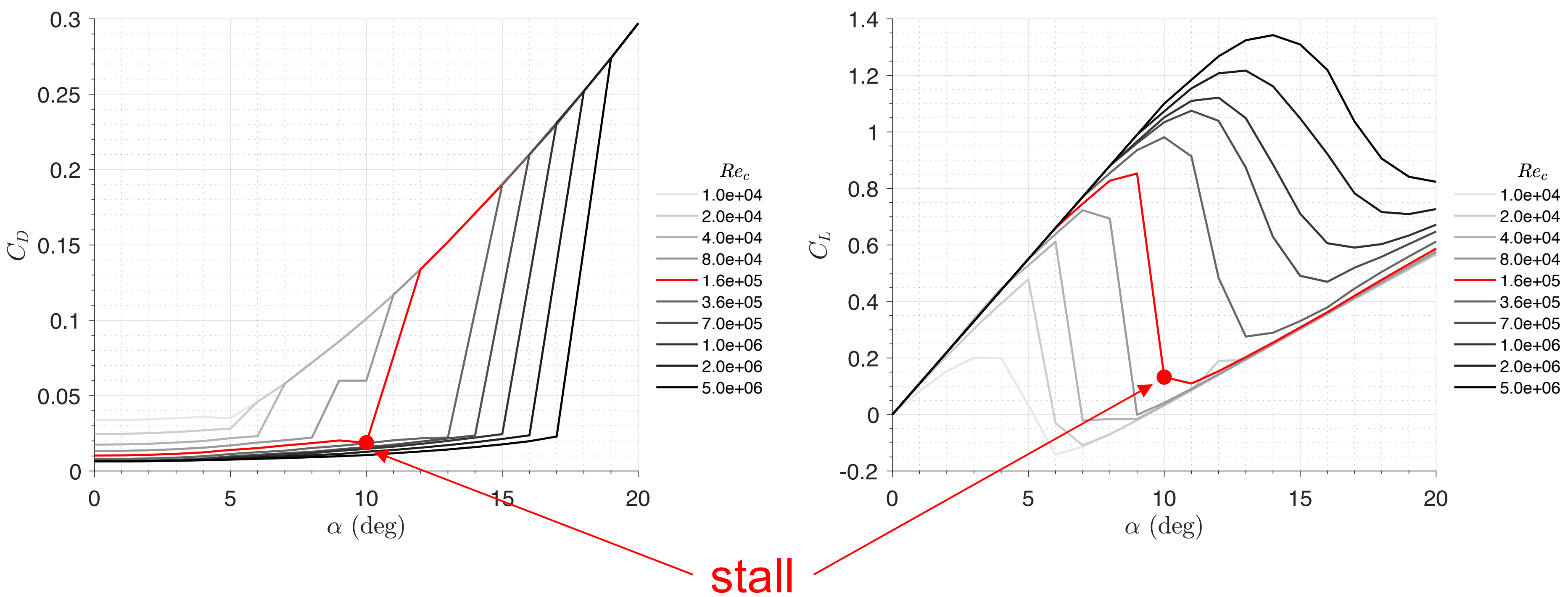
Careful with units!

```
%Compute standard error
CL_err = Lerr/(Pdyn*c*s);
CD_err = Derr/(Pdyn*c*s);
```

$$C_D = \frac{\overline{F}_D}{\frac{1}{2}\,\rho\,U_\infty^2\,A_p} \pm 2\left(\frac{\sigma_{\overline{F}_D}}{\frac{1}{2}\,\rho\,U_\infty^2\,A_p}\right)$$

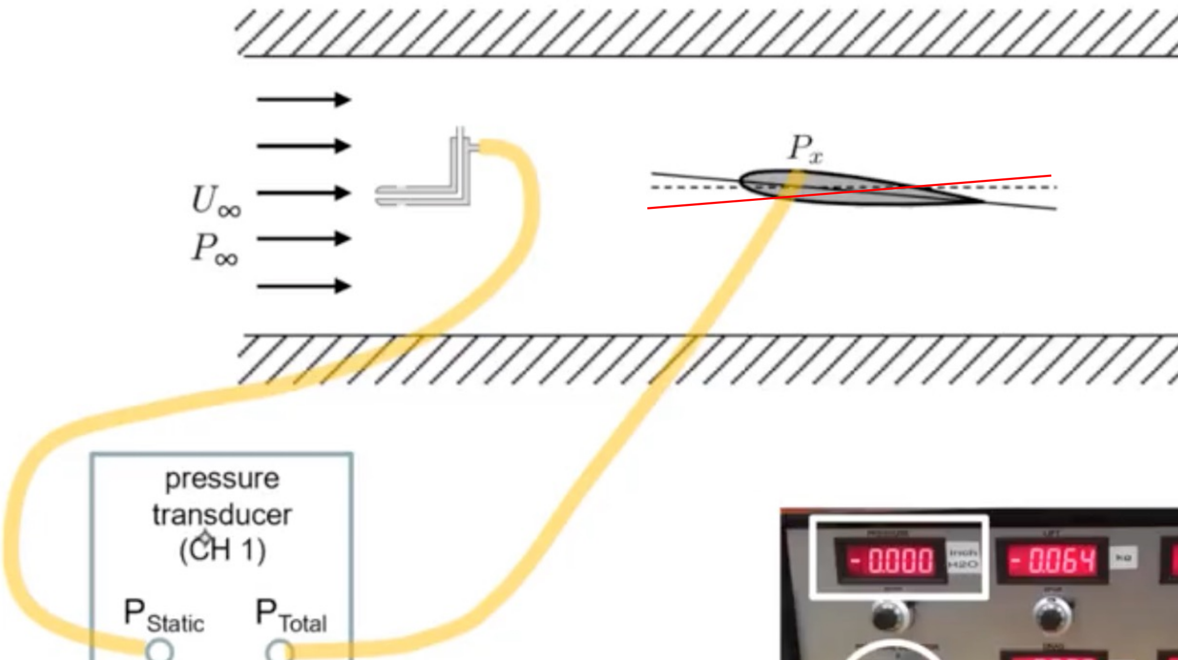
Lift and Drag Coefficients Figures 1a & 1b

You are given these plots, and you will pot your data on these plots



aerodynamic performance: C_L/C_D

Post-Processing Raw Data: Airfoil Relative Static Pressure



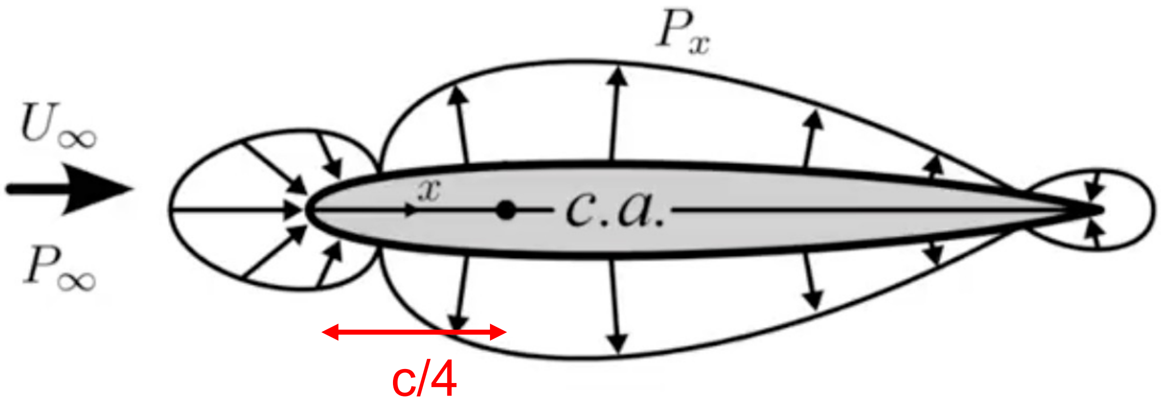
Tilt airfoil positive and negative

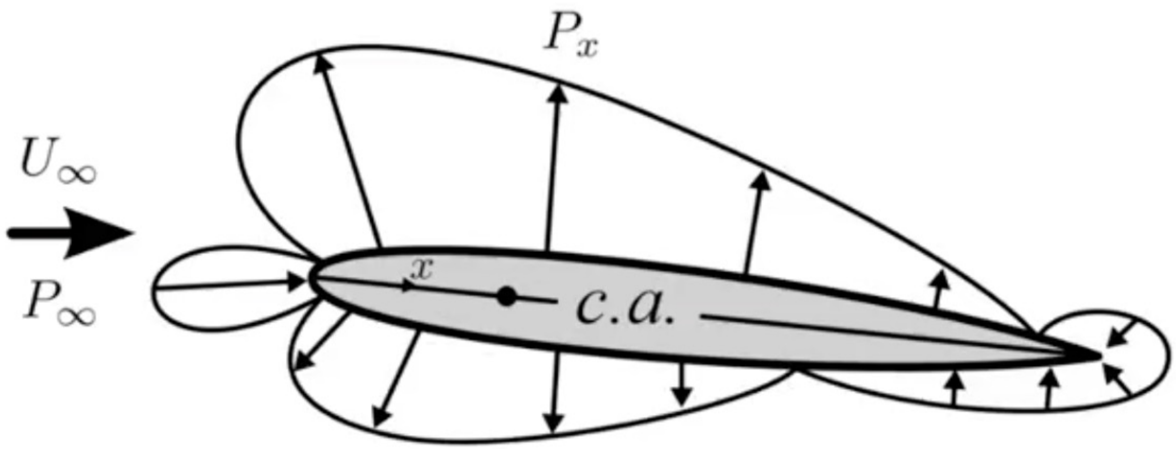
output: $\Delta P = P_{\text{Total}} - P_{\text{Static}}$

$$\Delta P = P_x - P_\infty$$

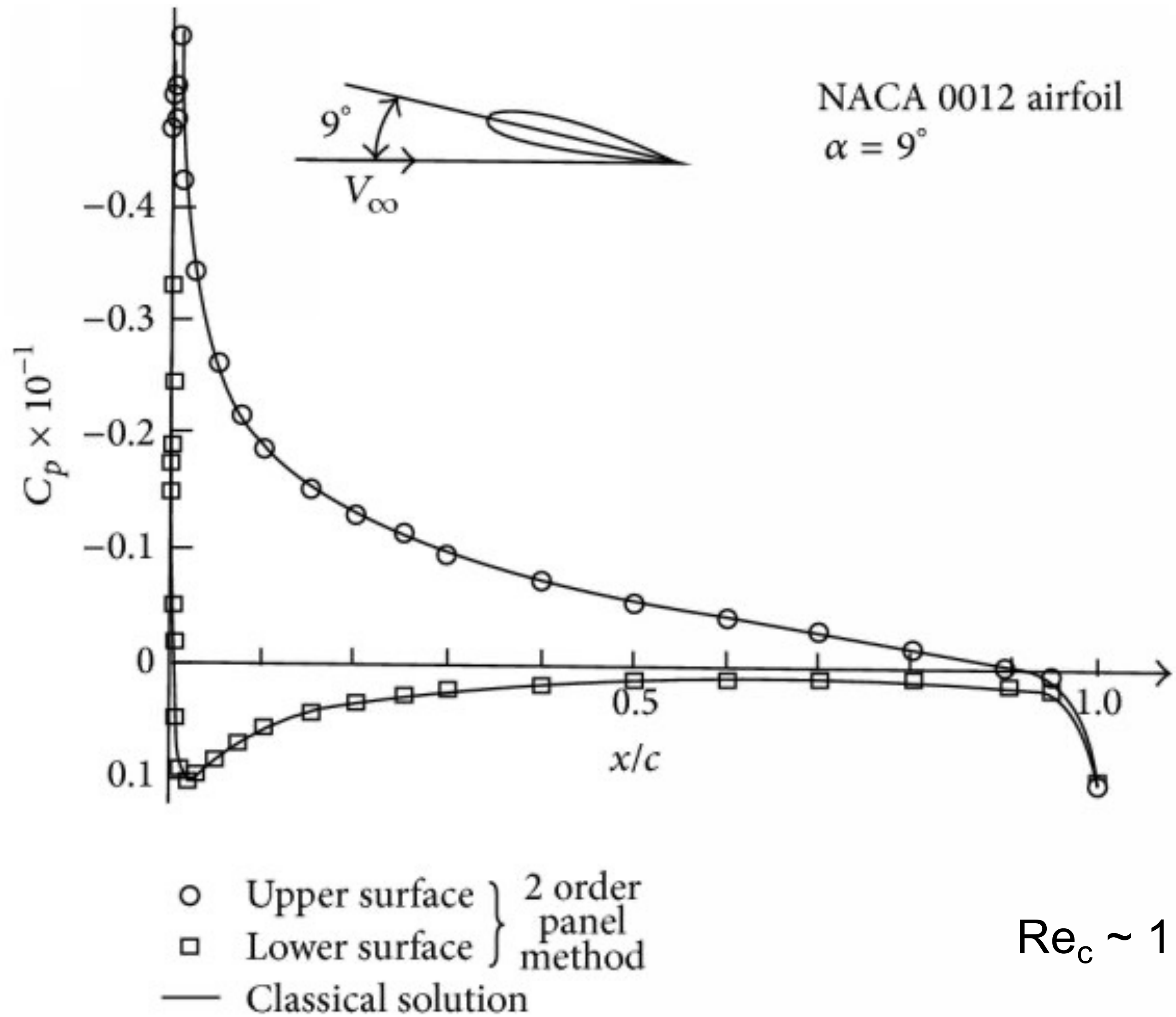
Coefficient of pressure: $C_p = \dfrac{P_x - P_\infty}{\frac{1}{2}\rho U_\infty^2}$

Pressure Distribution Relative to Upstream Static Pressure





Coefficient of Pressure

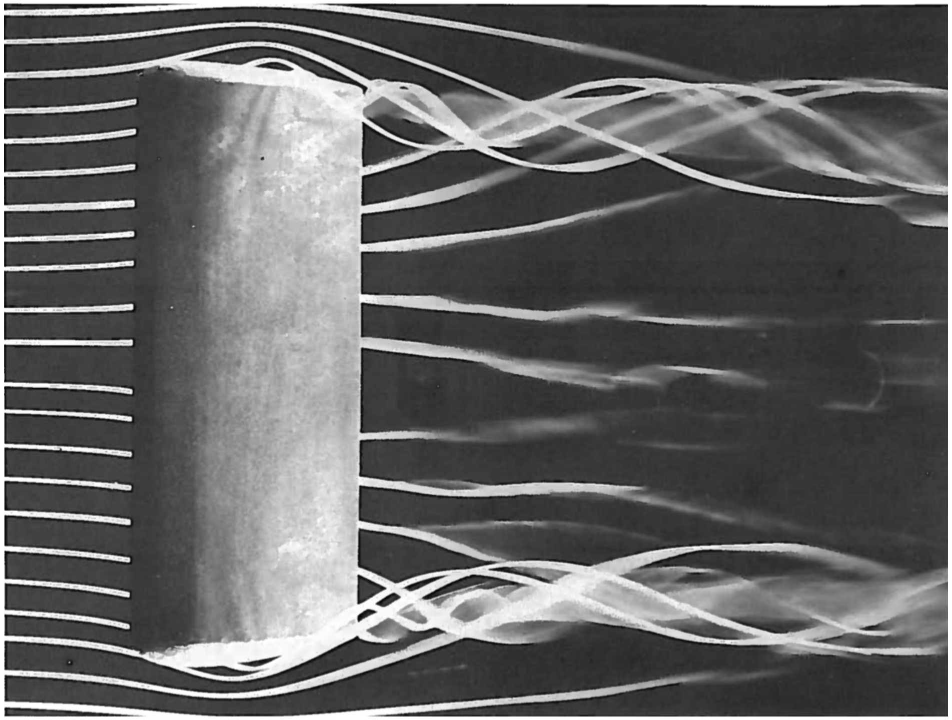


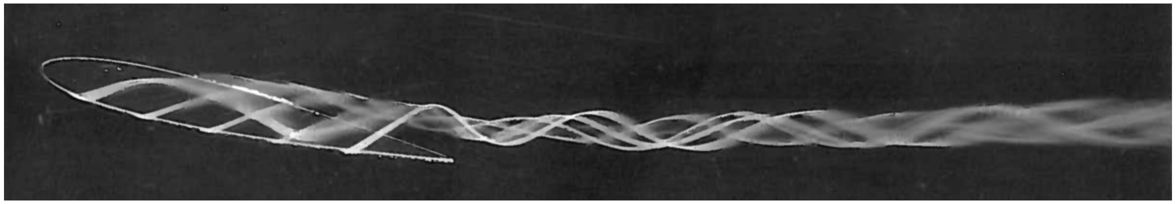
Re_c ~ 1,000,000

Figure 1c

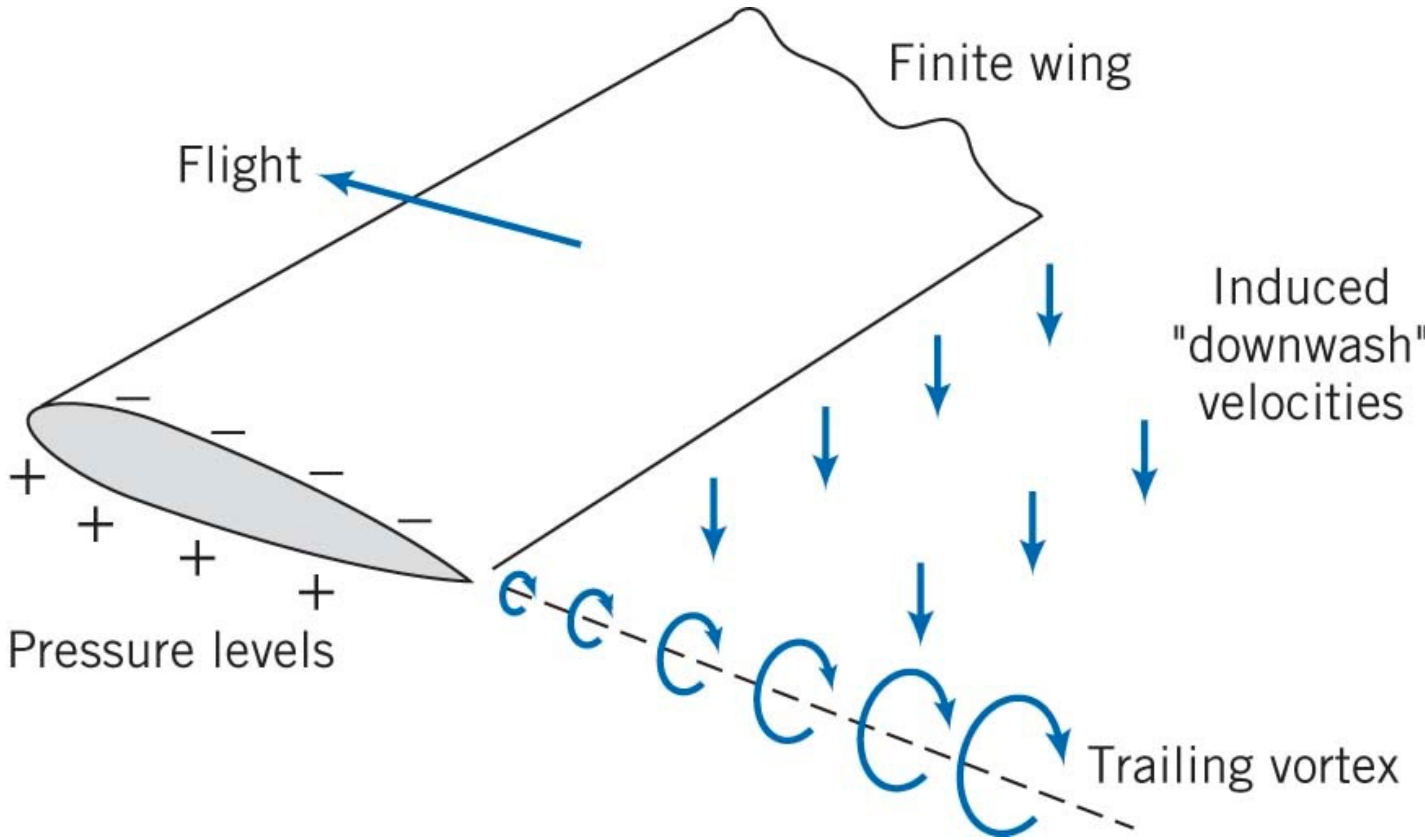
- Cp vs x/c for angles of attack: alpha = 5 and 12 degrees

End Effects (3D Airfoils)

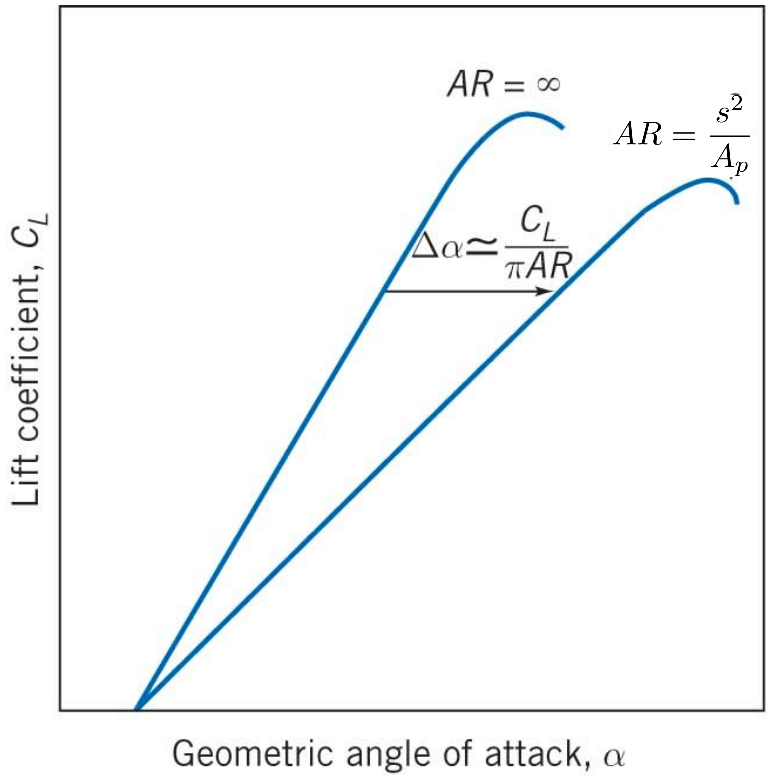




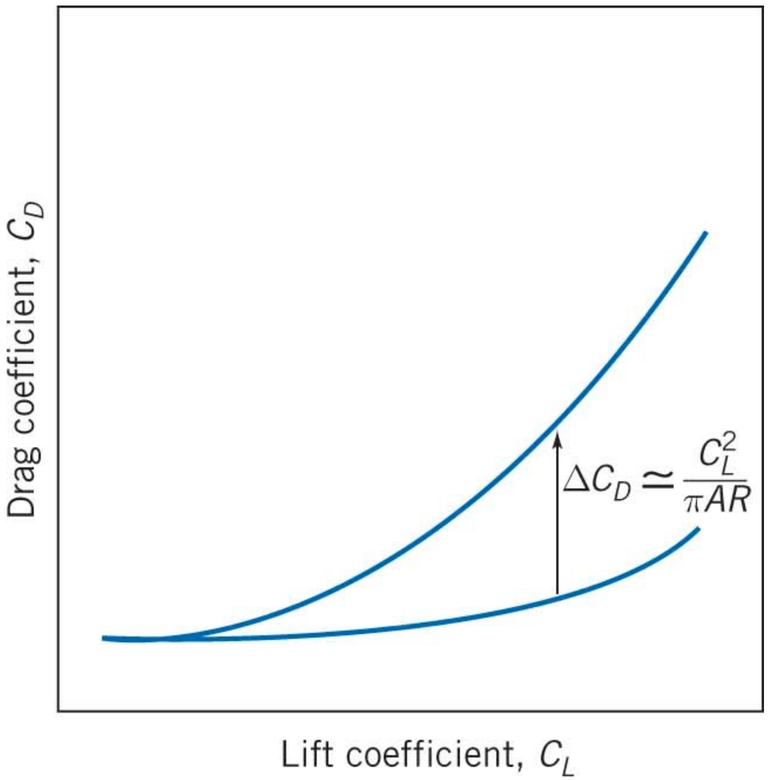
End Effects (3D Airfoils)



End Effects (3D Airfoils)

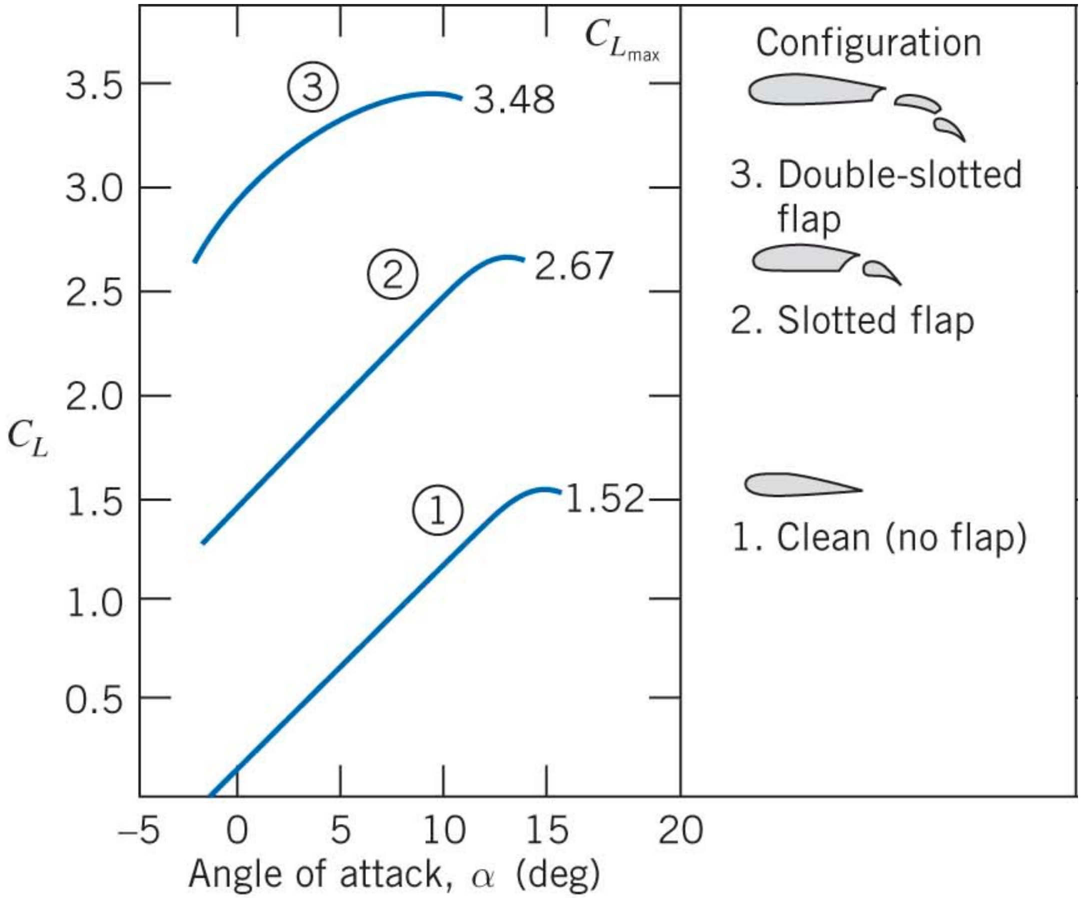


reduction of effective angle of attack



increase of induced drag

Wing Flaps



$C_{L_{max}}$
3.5
3.0
2.5
2.0
1.5
1.0
0.5
C_L
3
3.48
2
2.67
1
1.52
–5
0
5
10
15
20
Angle of attack, α (deg)
Configuration
3. Double-slotted flap
2. Slotted flap
1. Clean (no flap)