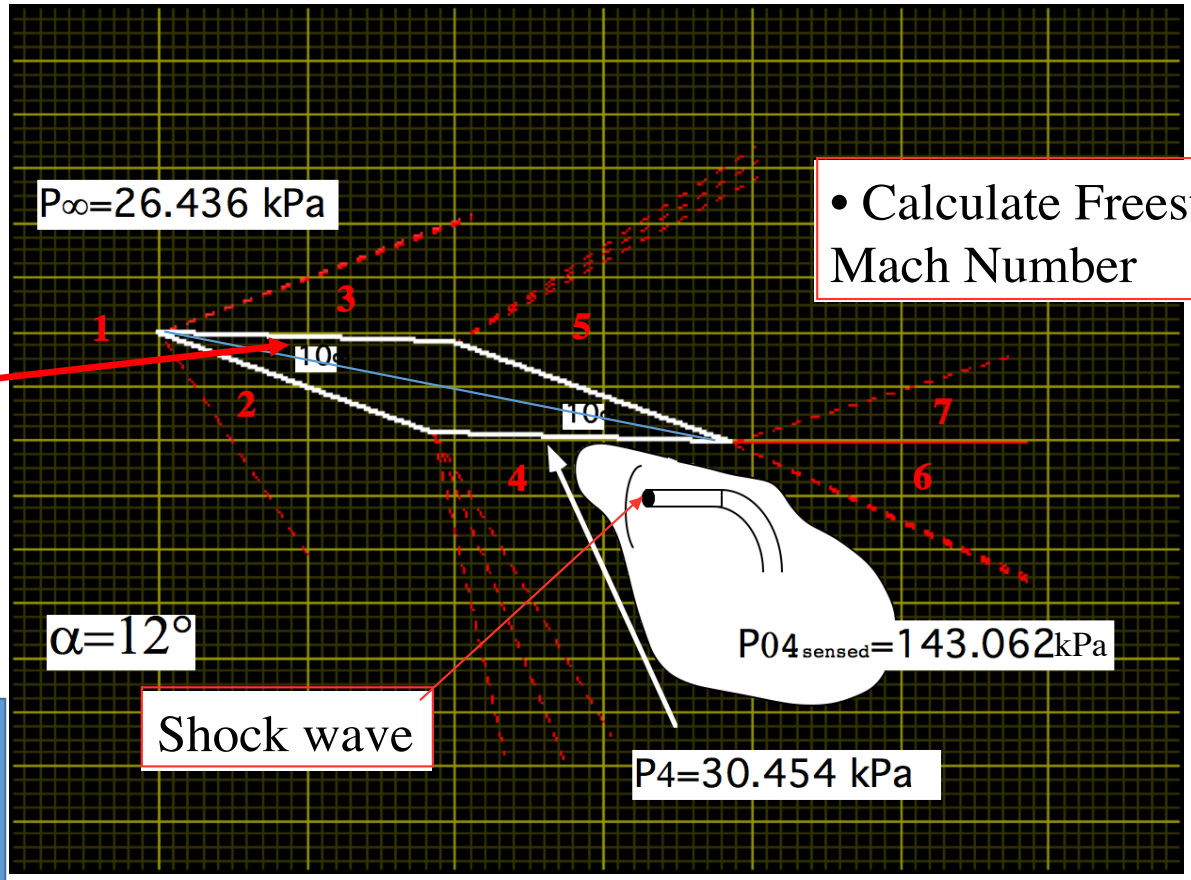
Section 7, Homework (8) … Due Monday November 29, 2021

Part1

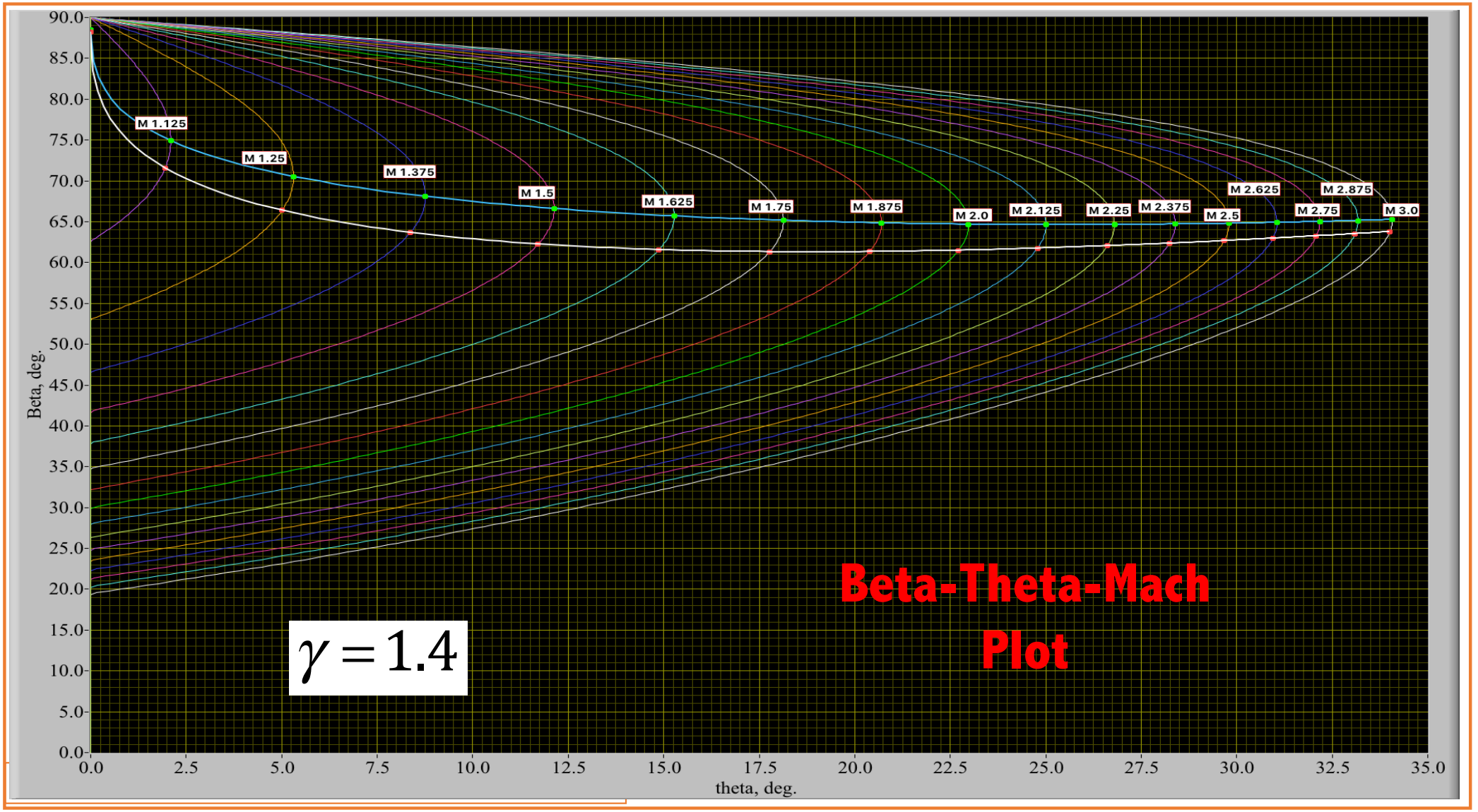
Solution Hints: Assume supersonic flow at probe and Wedge has a ***10 deg. half angle***, work backwards from probe...

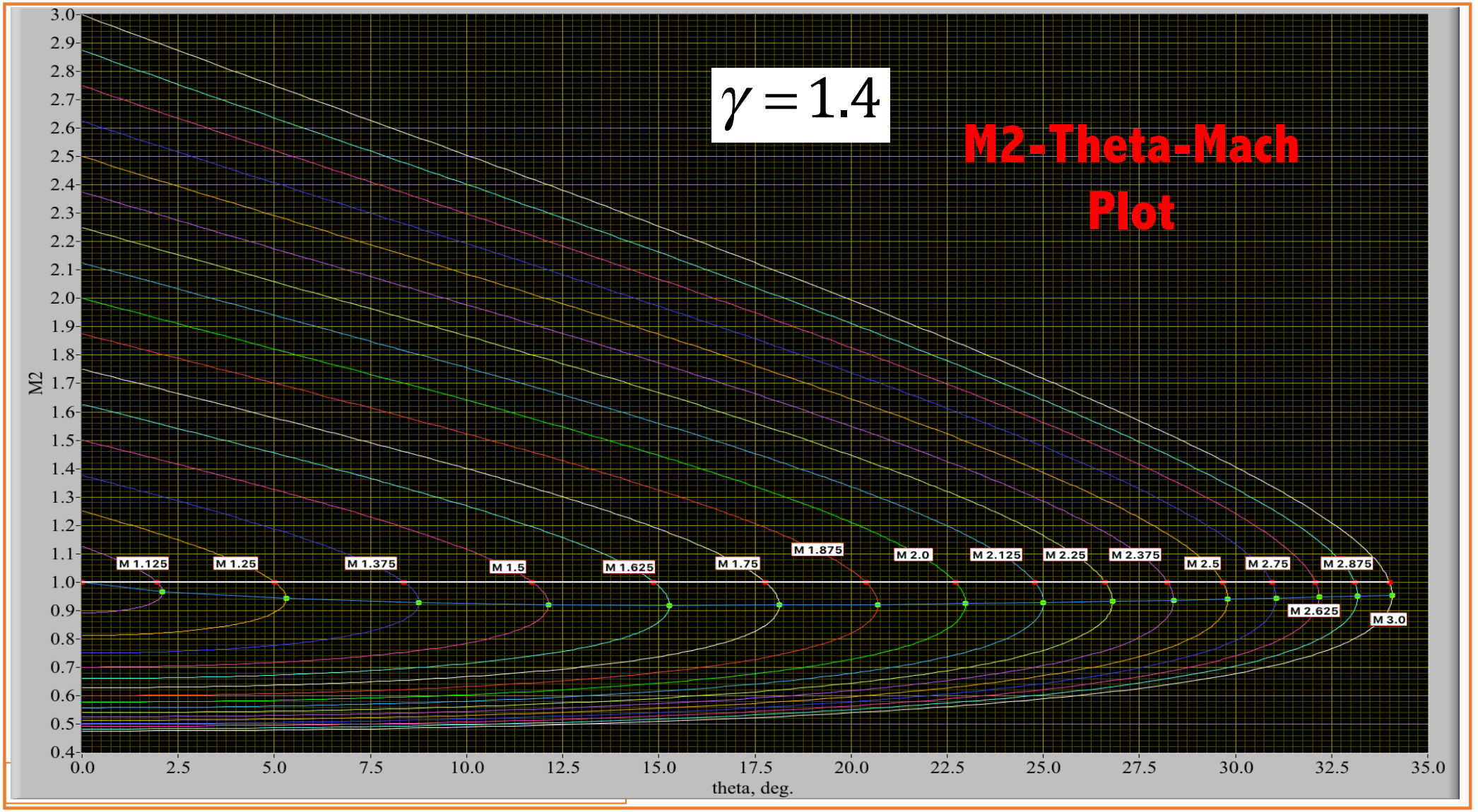
Static pressure Measurement is ahead of shock

"Rayleigh Pitot Equation"



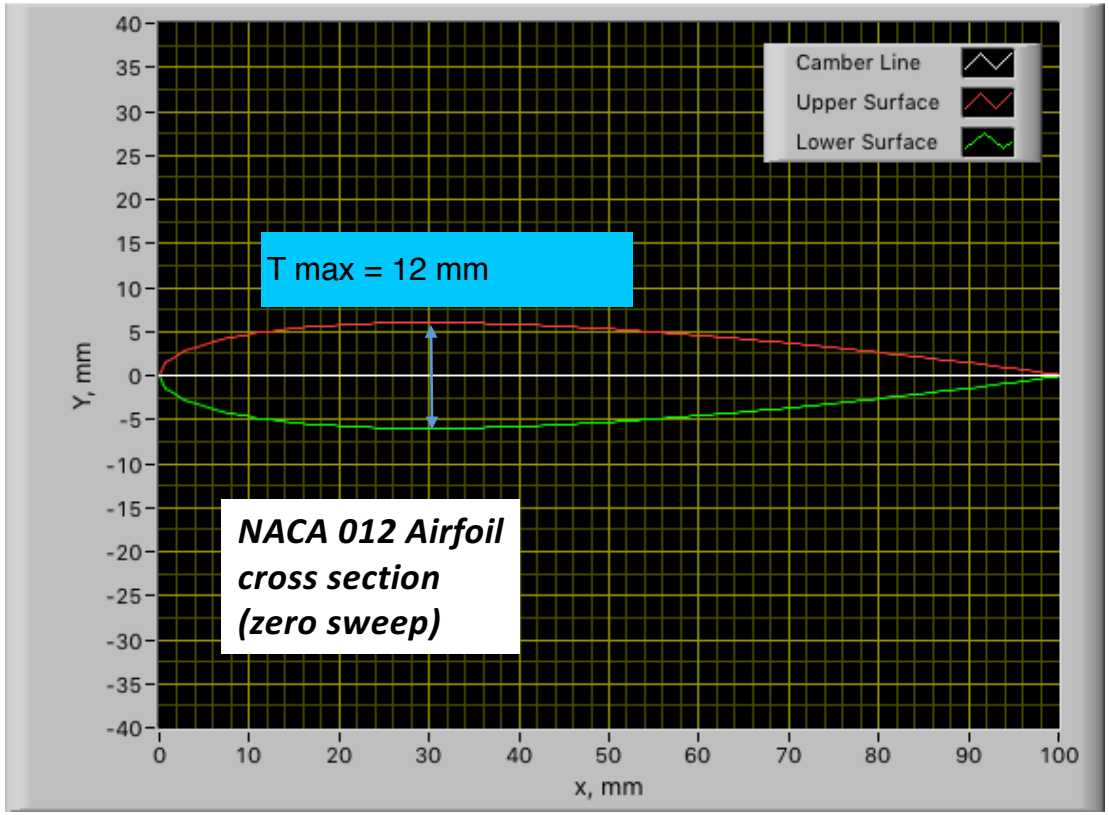
- Calculate Freestream Mach Number
- Assume $\gamma = 1.4$


Beta-Theta-Mach Plot
$\gamma = 1.4$
M 1.125
M 1.25
M 1.375
M 1.5
M 1.625
M 1.75
M 1.875
M 2.0
M 2.125
M 2.25
M 2.375
M 2.5
M 2.625
M 2.75
M 2.875
M 3.0
Beta, deg.
theta, deg.
0.0
2.5
5.0
7.5
10.0
12.5
15.0
17.5
20.0
22.5
25.0
27.5
30.0
32.5
35.0
5.0
10.0
15.0
20.0
25.0
30.0
35.0
40.0
45.0
50.0
55.0
60.0
65.0
70.0
75.0
80.0
85.0
90.0


$\gamma = 1.4$
M2-Theta-Mach Plot
M2
theta, deg.
M 1.125
M 1.25
M 1.375
M 1.5
M 1.625
M 1.75
M 1.875
M 2.0
M 2.125
M 2.25
M 2.375
M 2.5
M 2.75
M 2.875
M 2.625
M 3.0

Section 7, Homework, Part 2

NACA 0012 Airfoil Coordinates



During the late 1920s and into the 1930s, the National Advisory Committee for Aeronautics (NACA) developed a series of thoroughly tested **airfoils** and devised a numerical designation **for** each **airfoil** — a **four digit** number that represented the **airfoil** section's critical geometric properties.*

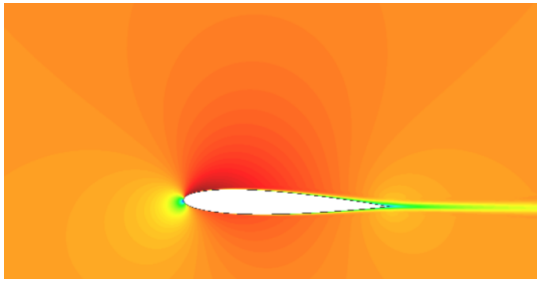
- First digit describing maximum camber as percentage of the chord.
- Second digit describing the distance of maximum camber from the airfoil leading edge in tenths of the chord.
- Last two digits describing maximum thickness of the airfoil as percent of the chord.

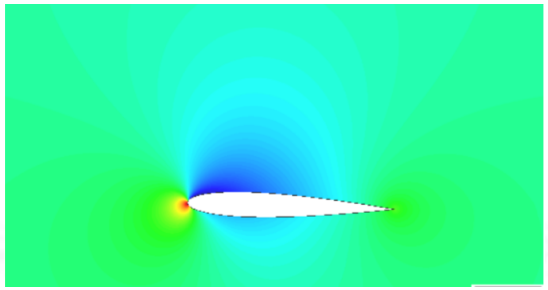
- **"Fundamentals of aerodynamics", John D. Anderson, Jr., 3rd ed., Chapt. 4.*

Section 7, Homework, Part 2

- **Consider NACA 0012 Airfoil at 1.5°α.**

- Calculate the critical drag rise (subsonic) Mach number for zero wing sweep
- Re-Calculate the M_{cr} assuming 30° wing sweep, Λ
- ***Compare the effective fineness ratios (t/c), for the unswept and swept wing sections***
- ***What do you conclude?***



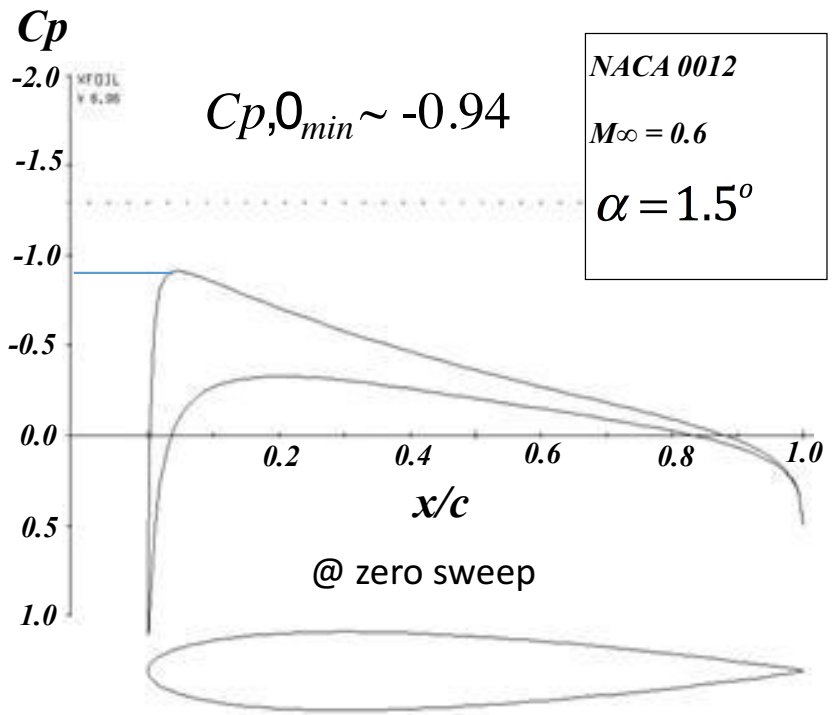


NACA 0012

M∞ = 0.6

$\alpha = 1.5^{o}$

NACA 0012 at *1.5°α*



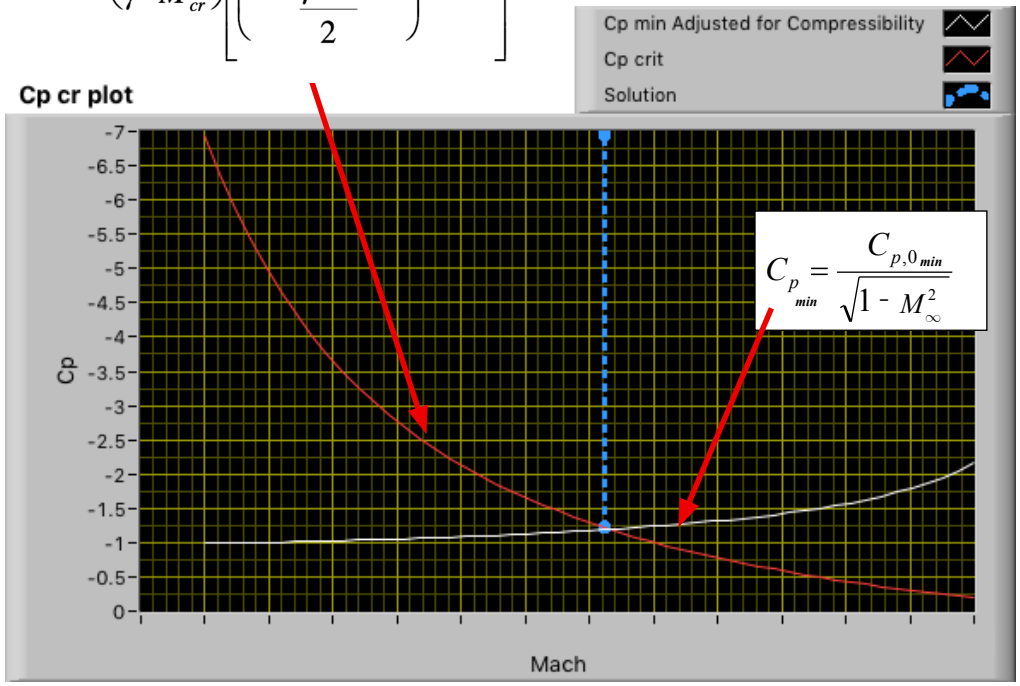
$Cp,0_{min} \sim -0.94$

NACA 0012

$M\infty = 0.6$

$\alpha = 1.5^o$

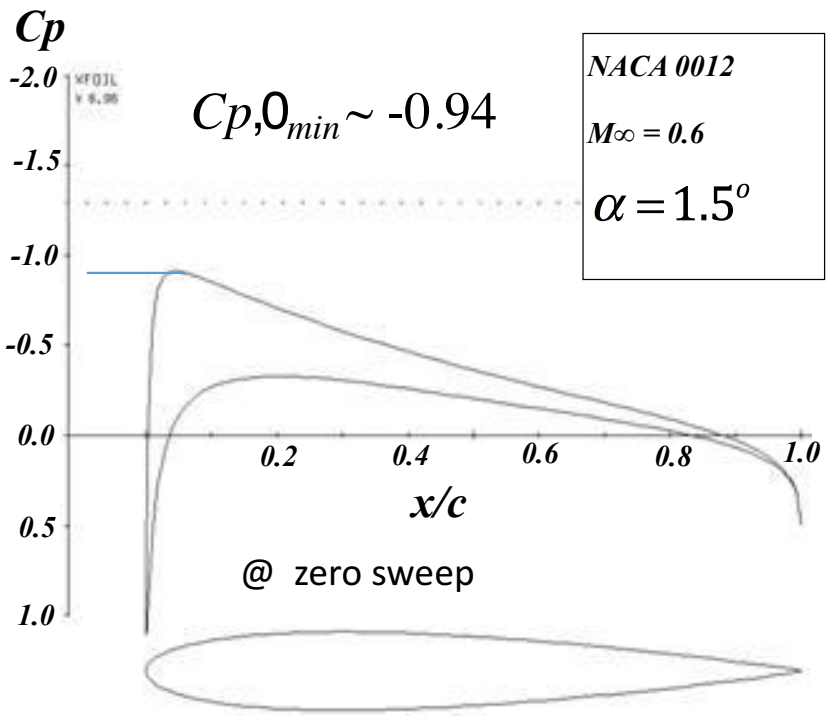
$$C_{p_{cr}} = \left(\frac{2}{\gamma \cdot M_{cr}^2}\right)\left[\left(\frac{1+\frac{\gamma-1}{2}M_{cr}^2}{\frac{\gamma+1}{2}}\right)^{\frac{\gamma}{\gamma-1}} - 1\right]$$



@ zero wing weep

.. Calculate $M_{\infty\ cr}$

NACA 0012 at *1.5°α with 30 deg. Wing sweep (Λ)*



What is M_{crit} @ 30° wing sweep (Λ)

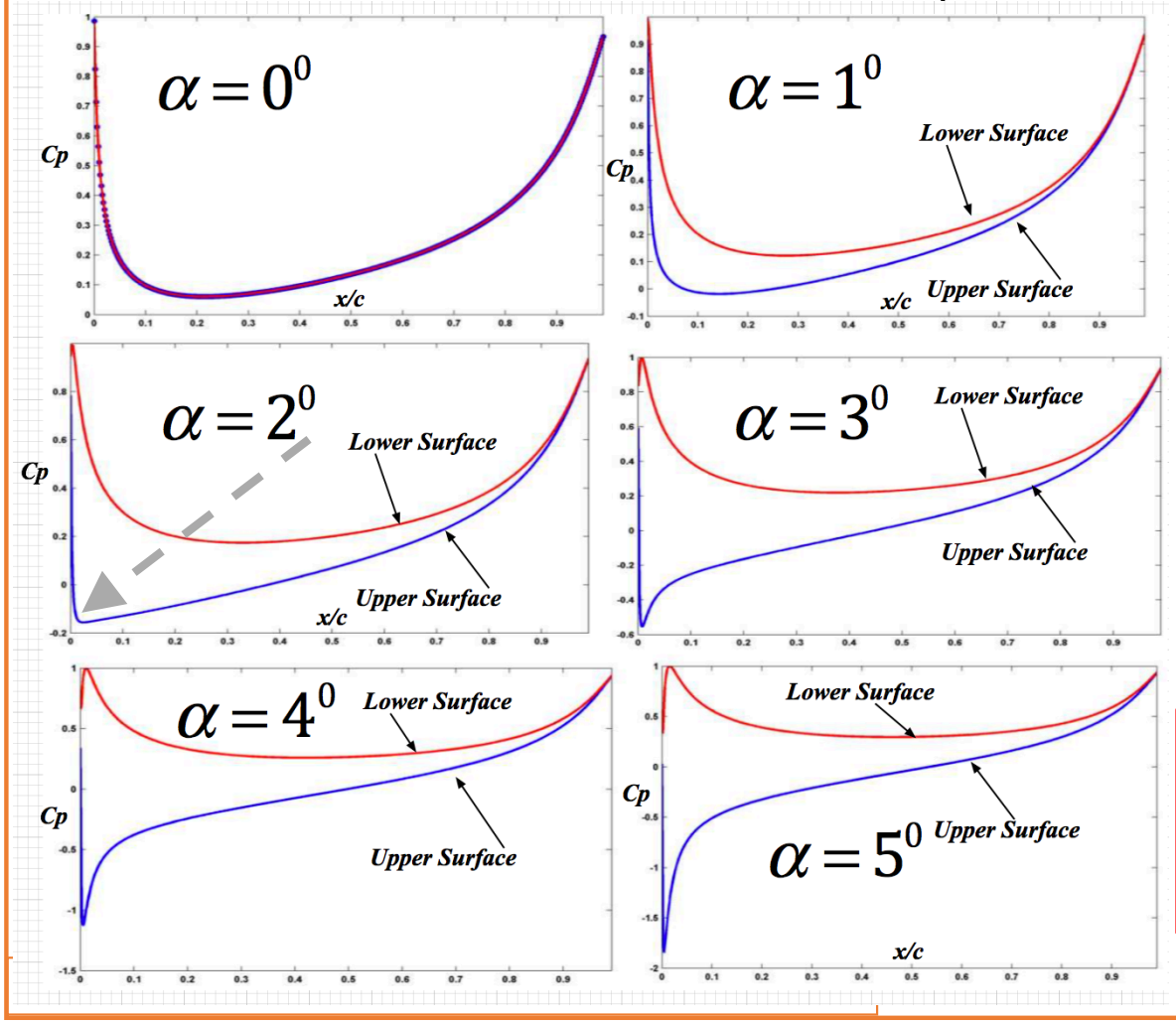
$$\left(M_{crit}\right)_{\Lambda} = \frac{\left(M_{crit}\right)_{0^o}}{\sqrt{1-\sin^2\Lambda\cdot\cos^2\alpha}}$$

What is *Fineness Ratio* @ 30° wing sweep (Λ)

Swept →

$$\frac{t}{c_{equiv}} = \frac{t}{c/\cos\Lambda}$$

Homework 8, Part 3



- Consider NACA 006 airfoil at 6 different Angles of Attack, **Incompressible flow**
- Plot $\mathbf{M_{crit}}$ as function of Angle of Attack
- Perform Calculations using the following compressibility corrections
  - Prandtl-Glauret
  - Karman-Tsien
  - Laitone's Rule
- Compare the resulting curves