SPC 307 - Aerodynamics
Sheet 7
INCOMPRESSIBLE FLOWS AROUND AIRFOILS OF INFINITE SPAN

1. Water flows past a flat plate that is oriented parallel to the flow with an upstream velocity of 0.5 m/s. Determine the approximate location downstream from the leading edge where the boundary layer becomes turbulent. What is the boundary layer thickness at this location?
2. Calculate C_l and $C_{m,\,c/4}$ for a NACA 0009 airfoil that has a plain flap whose length is 0.2c and which is deflected 25°. When the geometric angle of attack is 4°, what is the section lift coefficient? Where is the center of pressure?
3. The NACA 23012 wing section thus has a design lift coefficient of 0.3, has its maximum camber at 15% of the chord, and has a maximum thickness of 0.12 c. The equation for the mean camber line is

$$\frac{z}{c} = 2.6595\left[\left(\frac{x}{c}\right)^3 - 0.6075\left(\frac{x}{c}\right)^2 + 0.11471\left(\frac{x}{c}\right)\right]$$

   for the region $0.0c \le x \le 0.2025c$ and

$$\frac{z}{c} = 0.022083\left(1 - \frac{x}{c}\right)$$

   for the region $0.2025c \le x \le 1.000c$.
   Calculate the A_0, A_1, and A_2 for this airfoil section. What is the section lift coefficient, C_l ? What is the angle of attack for zero lift, α_{0l}? What angle of attack is required to develop the design lift coefficient of 0.3? Calculate the section moment coefficient about the theoretical aerodynamic center. Compare your theoretical values with the experimental values in Fig. 1 that are reproduced from the work of Abbott and von Doenhoff (1949).
   When the geometric angle of attack is 3°, what is the section lift coefficient? What is the x/c location of the center of pressure?

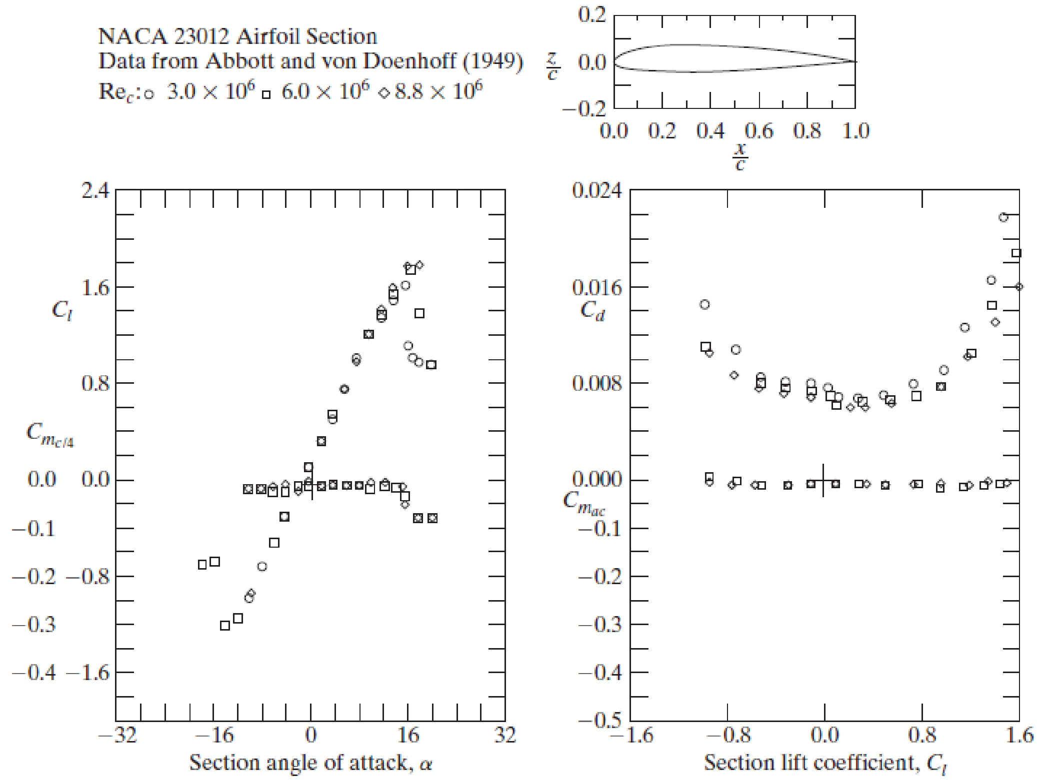

Fig. 1

4. The NACA 4412 airfoil has a mean camber line given by

$$\frac{z}{c} = \begin{cases} 0.25\left[0.8\dfrac{x}{c} - \left(\dfrac{x}{c}\right)^2\right] & \text{for } 0 \le \dfrac{x}{c} \le 0.4 \\ 0.111\left[0.2 + 0.8\dfrac{x}{c} - \left(\dfrac{x}{c}\right)^2\right] & \text{for } 0.4 \le \dfrac{x}{c} \le 1 \end{cases}$$

Using thin airfoil theory, calculate

a) $\alpha_{l=0}$

b) C_l and $C_{m,\,c/4}$ at α=3°

c) compare the result with the experimental data from the following reference: Abbott, Theory of Wing Sections, Mcgraw-Hill Book 1949 (Link).