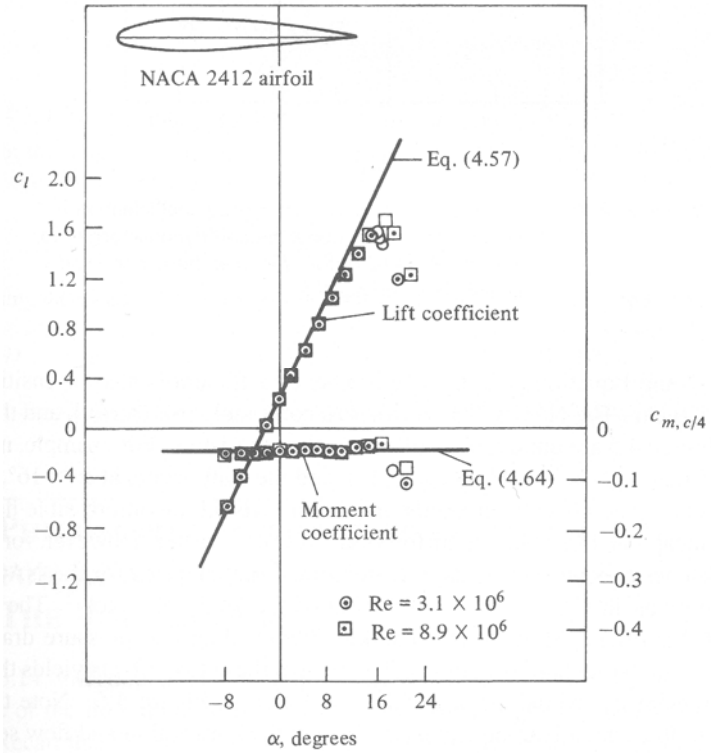
3. Incompressible flow over airfoils

3.1
Consider the data for the NACA 2412 airfoil given in the figure below. From the figure estimate the lift and moment about the quarter chord (per unit span) for this airfoil when the angle of attack is 4° and the free-stream is at standard sea level conditions with a velocity of 50 ft/s (15.2 m/s). The chord of the airfoil is 2 ft (0.61m).



3.2
Consider an NACA 2412 airfoil with a 2-m chord in an air-stream with a velocity of 50 m/s at standard sea level conditions. If the lift per unit span is 1353 N, what is the angle of attack?

3.3
Consider a thin, symmetric airfoil at 1.5° angle of attack. From the results of thin airfoil theory, calculate the lift coefficient and the moment coefficient about the leading edge.

3.4
The NACA 4412 airfoil has a mean camber line given by

$$\frac{z}{c} = \begin{cases} 0.25 \cdot (0.8 \cdot \frac{x}{c} - (\frac{x}{c})^2) & for \quad 0 \le \frac{x}{c} \le 0.4 \\ 0.111 \cdot (0.2 + 0.8 \cdot \frac{x}{c} - (\frac{x}{c})^2) & for \quad 0.4 \le \frac{x}{c} \le 1 \end{cases}$$

Using thin airfoil theory, calculate

(a) $\alpha_{L=0}$

(b) c_l when $\alpha = 3°$

3.5*(hard work)

For the airfoil given in 3.4, calculate $c_{m,c/4}$ and x_{cp}/c when $\alpha = 3°$.

3.6

For the NACA 2412 airfoil, the lift coefficient and moment coefficient about the quarter-chord at -6° angle of attack are - 0.39 and - 0.045, respectively. At 4° angle of attack, these coefficients are 0.65 and -0.037, respectively. Calculate the location of the aerodynamic centre.

3.7

An airfoil has a mean camber line that has the shape of a circular arc. The maximum mean camber is kc, where k is a constant and c is the chord. The free-stream velocity is V_∞ and the angle of attack is α. Under the assumption that $k<<1$ show that the γ distribution is approximately

$$\gamma = 2V_\infty(\alpha\cot(\frac{\theta}{2}) + 4k\sin\theta)$$

3.3

For the circular arc described in 3.7 calculate the lift at $\alpha=0$ and c_{mac}

3.9

The slope of the mean camber line of an airfoil is given by

$$\frac{d\eta}{dx} = 0.1(\frac{1}{3} + \cos\theta + \cos 2\theta)$$

Find and plot the mean camber as a function of x. Compute the angle of attack at zero lift and c_{mac}. Find expressions for c_l and x_{cp} when the airfoil is at an angle of attack α.

3.10

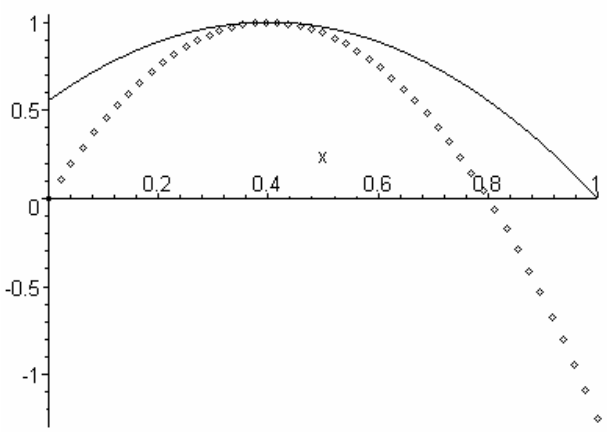
NACA 2510 is a 10% thick airfoil whose maximum camber 0.02c occurs at the mid chord position. Assume that the mean camber line can be represented approximately by a parabola. Suggest a modification of this airfoil configuration for doubling the lift at $\alpha=0$. What is the NACA designation of the modified airfoil?

3.11

The mean camber of an airfoil consists of two parabolas joined at their vertices (see figure below). The maximum mean camber is kc is located at 0.4c behind the leading edge. For this airfoil find the value of the angle of attack at zero lift and c_{mac}.

3.12

Use the method in 3.11 to calculate $c_l(\alpha)$ and c_{mac} for the airfoils NACA1412, NACA2412 and NACA4412 and compare with the experimental curves.



Answers:

3.1 60 N -0.98 Nm

3.2 2.0°

3.3 c_l=0.164 $c_{m,LE}$= - 0.041

3.4 a) -.073 rad b) c_l=0.79

3.5 $c_{m,c/4}$= -0.105 x_{cp}=0.38 c

3.6 $x_{ac} = 0.242c$

3.7

3.8 $c_l = 2\pi\alpha + 4k\pi \qquad c_{m,ac} = -k\pi$

3.9 $c_{m,ac} = 0 \quad x_{cp} = c/4 \quad c_l = 2\pi(\alpha + 0.017)$

3.10 $k = 0.04 \quad NACA4510$

3.11 $c_l = 6.28\alpha + 11.4k \quad c_{m,ac} = -2.65k$

3.12

$c_l = 2\pi + 0.1138$ *NACA*1412

$c_l = 2\pi + 0.2278$ *NACA*2412

$c_l = 2\pi + 0.4559$ *NACA*4412

$c_{m,ac} = -.027 \quad NACA1412$

$c_{m,ac} = -.053 \quad NACA2412$

$c_{m,ac} = -.106 \quad NACA4412$