MAE 104 - SUMMER 2015
Problem Session 5

09-03-2015

Problem 1:
We have designed a family of airfoils with the camber line

$$\frac{y_c(x)}{c} = \begin{cases} \frac{a}{10}\left[\frac{x}{c} - 2\left(\frac{x}{c}\right)^2\right] & ; \quad \frac{x}{c} \leq \frac{1}{4} \\ \frac{a}{90}\left[1 + \frac{x}{c} - 2\left(\frac{x}{c}\right)^2\right] & ; \quad \frac{x}{c} \geq \frac{1}{4} \end{cases},$$

as shown in Figure 1.

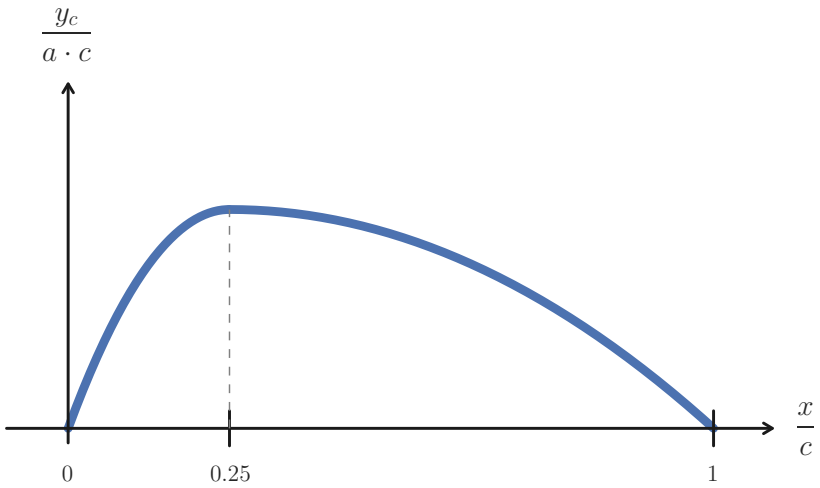

Figure 1: Camber line of the airfoil familly.

1. Find the parameter a such that $\alpha_{l=0} = -5°$.
2. Find the lift coefficient c_l of this airfoil.
3. Find the coefficient of moment about the leading edge $c_{m,L.E.}$ of this airfoil.
4. Find the coefficient of moment about the trailing edge $c_{m,T.E.}$ of this airfoil.
5. Find the coefficient of moment about the quarter-chord point $c_{m,c/4}$ of this airfoil.
6. Find the center of pressure $x_{c.p.}$ of this airfoil.

Problem 2:
An airfoil of zero thickness is described by the equation

$$\frac{y_a}{c} = -\left(\frac{x}{c}\right)^2 - \frac{1}{3}\left(1 - 2\frac{x}{c}\right)^3.$$

It is flying at zero angle of attack with speed U_∞ on air (density ρ_∞) at rest. Using incompressible thin airfoil theory,

1. Calculate the lift coefficient c_l.
2. Calculate the moment coefficient about the leading edge $c_{m,L.E.}$.
3. Calculate the center of pressure $x_{c.p.}$ of the airfoil.
4. Calculate the values of c_l, $c_{m,L.E.}$ and $x_{c.p.}$ when the airfoil is flying at Mach number $M_\infty = 0.5$.

Problem 3:
Consider a wing with elliptical planform shape, zero aerodynamic and geometric twist, span b and large aspect ratio $\Lambda \gg 1$, immersed on a uniform flow with freestream velocity U_∞ and density ρ. Calculate the lift and induced drag forces, induced velocity and induced angle of attack on the wing.