MAE 104 - SUMMER 2015
Problem Session 3

08-20-2015

Problem 1:
A thin symmetric airfoil is flying with velocity U_∞ and angle of attack α, as shown in Figure 1. When α is small, the components of the velocity parallel and perpendicular to the airfoil are, for a region near it, $y << c$:

$$u(x, y) = U_\infty$$
$$v(x, y) = -w(x) + U_\infty \alpha$$

where $w(x)$ is the self induced velocity. We want to calculate the velocity field using *Thin Airfoil Theory*.

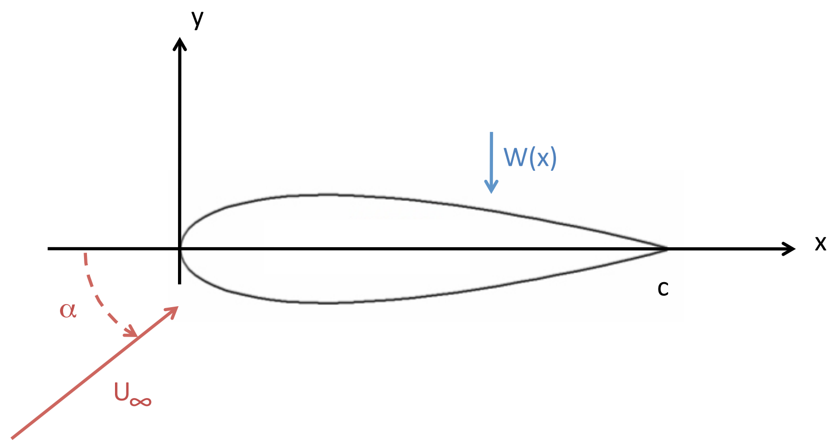

Figure 1: Thin airfoil.

1. First, substitute the thin airfoil by a vortex sheet such that the chord is a streamline. Calculate the intensity of the vortex sheet, $\gamma(\theta)$. Plot $\gamma(x)$.
2. Using the previously calculated $\gamma(\theta)$, calculate the vertical self-induced velocity w.
3. Calculate the lift.
4. Calculate the coefficient of moment around the leading edge of the wing.

Problem 2:
A thin airfoil of chord c is flying with velocity U_∞ and angle of attack α, as shown in Figure 2. The equation of the airfoil is:

$$\frac{y_a(x)}{c} = \begin{cases} 0 & ; \quad -\frac{c}{2} \leq x \leq 0 \\ -4\varepsilon \left(\frac{x}{c}\right)^2 & ; \quad 0 \leq x \leq \frac{c}{2} \end{cases}$$

where $\varepsilon << 1$.

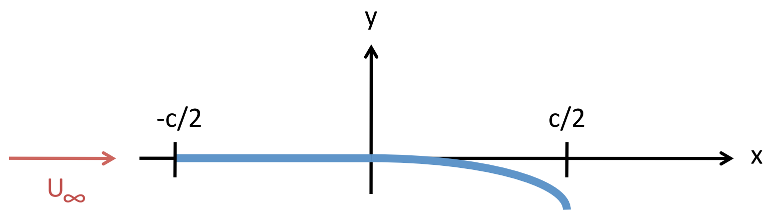

Figure 2: Thin airfoil.

We want to calculate:

1. The angle of attack of the airfoil.
2. The ideal angle of attack.
3. The lift coefficient.
4. The zero-lift angle of attack. Sketch the position of the zero-lift line.
5. The moment coefficient around the leading edge and trailing edge.
6. The position of the center of pressure.