

16.100 Homework Assignment # 4

Due: Monday, October 3 at 9am

Reading Assignment

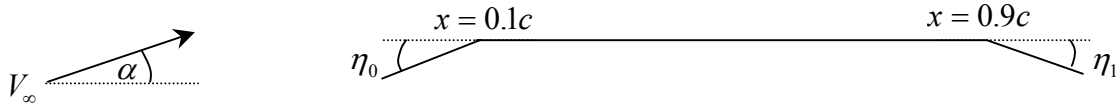
Handout “Thin Airfoil Theory Summary”

Anderson, 3rd edition: Chapter 4 Section 4.1 – 4.9

Note: I did not give a targeted reading assignment for the problems, because essentially all of the reading is useful for all of the problems.

Problem #1 (30%)

In this problem, we will consider the aerodynamic impact of leading and trailing edge flaps on an airfoil using thin airfoil theory. To be specific, we will consider the following camberline:



where the leading and trailing edge flap deflection angles (defined positive downwards) are η_0 and η_1 , respectively. In the following problems, use thin airfoil theory and assume small angles throughout.

- Calculate the derivative of the lift coefficient with respect to the leading edge flap deflection, $\partial c_l / \partial \eta_0$.
- Calculate the derivative of the lift coefficient with respect to the trailing edge flap deflection, $\partial c_l / \partial \eta_1$.
- Show that trailing edge flap has a significantly greater impact on the lift coefficient than the leading edge flap.

Problem #2 (40%)

Given the relatively small impact of leading edge flaps on the lift coefficient predicted from thin airfoil theory, the question arises why would one use leading edge flaps? We will take a look at this question in this problem. From the handout, we saw that the difference in pressure coefficient from the lower to the upper surface, $\Delta C_p \equiv C_p|_{lower} - C_p|_{upper}$ can be related to the vortex sheet strength distribution, $\gamma(x)$, calculated in thin airfoil theory by,

$$\Delta C_p = 2 \frac{\gamma(x)}{V_\infty}.$$

Also, in the specific case of an uncambered (symmetric) airfoil, we saw that the vortex sheet strength was,

$$\gamma(x) = 2\alpha V_\infty \sqrt{\frac{1-x/c}{x/c}}$$

Thus, at the leading edge of the airfoil (i.e. $x=0$), the vortex sheet strength and the pressure difference become infinite. While in the real case, a thin, uncambered airfoil does not have an infinite pressure difference, it does have, none-the-less, a very large drop in pressure and is commonly referred to as the “leading-edge suction peak.” This suction peak will rapidly accelerate the flow around the leading edge such that the boundary layer will have a difficult time remaining attached. If separation does occur due to this leading-edge effect, we call this “leading-edge separation.”

So, let’s consider what a leading-edge flap can do to alleviate this problem. We will again look at the airfoil above, but this time only with a leading-edge flap (still at 0.1c).

- a) For a general freestream angle of attack α , what angle does the leading edge flap need to be set at such that the suction peak would be completely eliminated according to thin airfoil theory?
- b) Suppose we need the airfoil to generate a lift coefficient $c_l = 0.5$. What are the freestream angle of attack and the leading-edge flap deflection that are required to produce this lift coefficient without a suction peak. Plot (on the same graph) the distribution of $\Delta C_p(x)$ for the symmetric airfoil with and without the leading-edge flap when $c_l = 0.5$. Describe the difference in the pressure difference distribution.

Problem #3 (30%)

As we saw in the previous problem, leading-edge flaps can be used to decrease (or eliminate) the leading-edge suction peak at a desired lift coefficient. When airfoils are designed for cruise performance, however, a better strategy is to design an airfoil that produces the correct lift with no suction peak through a cambered airfoil (i.e. without including leading-edge flaps). To see that this is possible, we will consider the NACA 44XX airfoils. Note: the general camber line for the NACA four-digit airfoils is:

$$\frac{z}{c} = \begin{cases} \left[2p \frac{x}{c} - \left(\frac{x}{c} \right)^2 \right] \frac{m}{p^2} & \text{for } 0 < \frac{x}{c} < p \\ \left[1 - 2p + 2p \frac{x}{c} - \left(\frac{x}{c} \right)^2 \right] \frac{m}{(1-p)^2} & \text{for } p < \frac{x}{c} < 1 \end{cases}$$

m is the maximum camber and is the first-digit/100. Also, p is the location of the maximum camber and is second digit/10. Apply thin-airfoil theory to answer the following questions.

- Determine the angle of zero-lift for the 44XX airfoils.
- Determine the angle at which suction-peak is eliminated. We will call this the design angle of attack for the 44XX airfoils.
- What is the design lift coefficient for the 44XX airfoils (i.e. the lift coefficient at the design angle of attack).