

16.100 Homework Assignment # 7

Due: Monday, November 7, 9am

Reading Assignment

Anderson, 3rd edition: Chapter 17, pages 787 – 801
Chapter 18, pages 803 – 810

Problem 1

In this problem, we will use Xfoil to simulate the flow of over an airfoil at relatively low Reynolds number, and compare the results to some simple flat plate results. To run Xfoil, you can use MIT server or download a copy of Xfoil for Windows. In either case, directions are available on the MIT server.

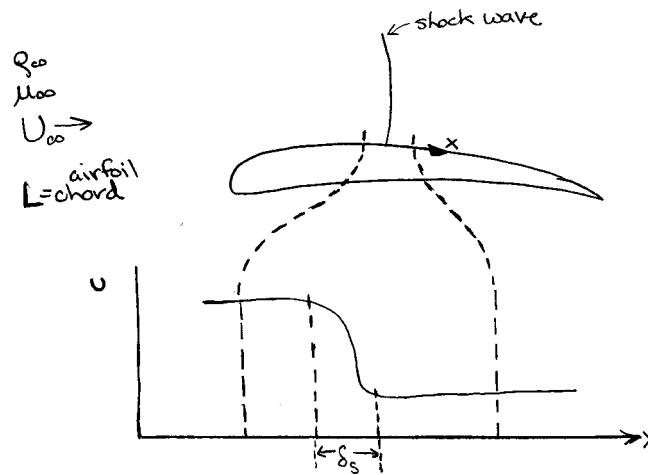
- a) Using Xfoil, calculate the drag coefficient versus Reynolds number for the NACA 0004 airfoil from $Re = 1 \times 10^3$ to $Re = 1 \times 10^6$ at zero degrees angle of attack. Plot the results on a log-log scale and include the drag coefficient from the laminar flat plate solution by Blasius. How does the Xfoil result compare to the flat plate theory?
- b) A form factor approach is often used to estimate the drag on an airfoil (or other body) by multiplying the skin friction drag estimated from flat plate results. Specifically, assume the drag can be estimated as follows:

$$c_d = K c_d^{flat}$$

where K is the form factor and c_d^{flat} is the drag coefficient from the Blasius flat plate theory for a plate of equal chord length. From the Xfoil results in a), estimate the form factor K . Does a constant value of K (i.e. independent of Reynolds number) exist which provides a good estimate of the Xfoil drag coefficient?

- c) The displacement thickness of the boundary layer predicted using Xfoil can also be compared to the flat plate results. Plot the displacement thickness at the trailing edge as predicted by Xfoil versus the displacement thickness at the trailing edge estimated from Blasius flat plate theory for the range of Reynolds numbers given above. How do the displacement thicknesses for the airfoil and the flat plate results compare?

Problem 2



An airfoil in transonic flow will have a shock wave as shown in the figure above. The flow properties vary rapidly through the shock which has a width of δ_s (as shown in the inset above).

- a) Perform an order of magnitude analysis through the shock (hint: use the steady, one-dimensional momentum equation). In particular, determine the power a with which δ_s / L scales with the Reynolds number $Re \equiv \rho_\infty U_\infty L / \mu_\infty$, in other words find the power a where

$$\frac{\delta_s}{L} = O(Re^a).$$

- b) For an airfoil with chord of 10 feet flying at Mach 0.85 at an altitude of 30,000 feet, estimate how thick (in inches) a shock would be? Assume that the constant in the order of magnitude analysis conducted in part a) is one, i.e. assume $\delta_s / L = Re^a$.
- c) At the same conditions as in part b), estimate how thick (in inches) a laminar boundary layer would be towards the end of the airfoil (assuming the flow is attached)?