ME 460/560 Assign. #3 Fall 2016
DUE: Nov 14 (Monday on Canvas by 5 pm)

The goal of this assignment is to evaluate forces on surfaces assuming potential flow (no frictional forces). We are going to numerically evaluate the local pressure which can be integrated over the surface to find the net force. The method to be used is called the "panel method" where the surface is imagined to be made up of a series of panels, or flat plates. We are going to deal with a 2D steady flow – although this method could be used for 3D unsteady flow situations.

The geometry is defined by an equation for the curved surface (very thin) based on a coordinate system with its origin at the leading edge. The x axis extends in a straight line from the leading edge to the trailing edge (along the chord line) and the z coordinate is normal to x within the 2D flow plane. The surface chord length is c = .5 m. The surface is defined by η (the distance normal to the x axis that defines the surface). The value of η is given by the following equation that provides an asymmetric airfoil:

$$\eta(x) = 0.2\ \frac{x}{c}\left(1 - \frac{x}{c}\right)^3$$

<u>Flow conditions</u>: water at U=1 m/s for three different angles of attack, $\alpha = 0^o$, 2^o and 5^o (as measured between the chord line and U, the approach velocity). Hint: to account for variations of α keep the coordinate system the same and rotate the direction of U. Assume a pressure of 101 kPa far upstream.

The goal is to determine the pressure distribution and force on the surface. A number of panels will be used to cover the 2D surface with the vortex location and collocation points to be placed on each panel. Use the following steps. In your report show results for each step, with all information for each step self-contained (be sure to include equations and explanation of how results are obtained). Start each number on a new page.

1. Plot the surface profile, η, versus distance *x*. Describe the shape relative to a typical airfoil your might find.
2. Select flat panels, using at least 10 along the length of the surface. Draw a sketch showing your panel distribution. Create a table listing your (x,z) coordinates at the beginning and end of each panel as well as the locations for vortex placement and collocation points along the surface.
3. Evaluate the matrix a_{ij} (as defined in class which is the matrix for the condition of vortex circulation values = 1.0) and put results in the form of a table for each angle of attack.
4. Obtain the distribution of Γ, that is determine Γ for each panel, and plot this versus x for each α. Show all work.
5. Find the local pressure difference across the airfoil, Δp_k, for each element, k, based on the local lift force, L_k, and plot this versus *x* for each α in terms of the local pressure coefficient, ΔC_p. Show all work.
6. Determine the total lift force on the surface for each α. Compare this to the expected lift coefficient for a two dimensional flat plate at the same angle of attack.
7. (560 only) Now to get a feel for what the influence of viscous forces may be compare these results to flow over a <u>flat plate</u> of length *c*, for zero angle of attack. Look up the solution for the viscous force on the surface (assume boundary layer flow along the length c) and compare this in magnitude to the lift force at zero angle of attack. Indicate the conditions and equations that you use.