Consider the low-speed airflow over the NACA 0012 airfoil at low angles of attack. The Reynolds number based on the chord is $Re_c = 2.88 \times 10^6$. This flow can reasonably be modeled as incompressible and inviscid.

1. Explain why the incompressible, inviscid model for this flow should yield lift coefficient values that match well with experiment but will yield a drag coefficient that is always zero.

2. What is the boundary value problem (BVP) you need to solve to obtain the velocity and pressure distributions for this flow at an angle of attack of 10°? Indicate governing equations, domain and boundary conditions ($\vec{u} = 0$ at a certain boundary etc.). For each of the boundary conditions, indicate also the corresponding boundary type that you need to select in FLUENT.

3. Set up this BVP in FLUENT and solve it numerically to obtain the velocity and pressure distributions at an angle of attack of 10°. Use the two meshes provided in a zip file at
   http://www.mae.cornell.edu/swanson/mae423files.html
   The second mesh is a refined version of the first. You might find the following airfoil tutorial helpful to solve this problem in FLUENT:
   https://confluence.cornell.edu/x/JyqkC
   Note that you can skip the geometry and meshing steps since the meshes are provided to you (the weather was warm outside and we were in a generous mood).

   (a) Plot the pressure coefficient C_p obtained from FLUENT on both meshes in the same figure. Add the corresponding experimentally measured pressure values as symbols. The experimental data is from Gregory & O'Reilly, NASA R&M 3726, Jan 1970 and provided in the same zip file as the meshes. Follow the aeronautical convention of flipping the vertical axis so that negative C_p values are above and positive C_p values are below. This can be done in MATLAB using
   ```
   set(gca, 'YDir', 'reverse');
   ```
   Add a suitable legend to distinguish between the curves.

   (b) Obtain the lift and drag coefficients from the FLUENT results on the two meshes. Compare these with experimental or expected values (present this comparison as a table). The experimental values for 10° angle of attack are:
   $C_l = 1.2219;\quad C_d = 0.0138.$

   (c) Repeat **3a** and **3b** for 6° angle of attack but only on the unrefined mesh. The experimental values for 6° angle of attack are:
   $C_l = 0.663;\quad C_d = 0.0090.$

   (d) Comment on the comparison with experiment for the two angles of attack. Also, comment on the effect of mesh refinement. How does the pressure distribution over the airfoil change on increasing the angle of attack?

   Compare your results visually with that obtained by Prof. Caughey with his code. His results are provided in a pdf file within the zip file.