ASEN 3111 Computational Lab #1: Computation of Lift and Drag

Lab Date: September 1, 2017
Due Date: September 15, 2017

Collaboration Policy:

Collaboration is permitted on the computational labs. You may discuss the means and methods for formulating and solving problems and even compare answers, but you are not free to copy someone else's work. *Copying material from any resource (including solutions manuals) and submitting it as one's own is considered plagiarism and is an Honor Code violation.*

Lab Reports Policy:

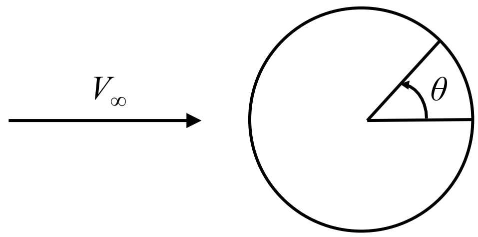
Computational lab reports must be written individually. If you have collaborated with others while writing your code, be sure to credit them in the Acknowledgements section. Computational lab reports should be submitted by 8:00 AM on the due date. Reports will not be accepted after the given due date.

Reflection Questions:

In this lab, there are two reflection questions. Please answer these questions in the Discussion portion of your lab report.

Problem #1:

Consider ideal (incompressible and inviscid) flow over a cylinder as depicted in the figure below:



For this flow problem, the coefficient of pressure, defined as:

$$C_p \equiv \frac{p - p_\infty}{q_\infty}$$

where p_∞ is the freestream pressure and q_∞ is the dynamic pressure, is known analytically to be equal to:

$$C_p = 1 - 4\sin^2(\theta).$$

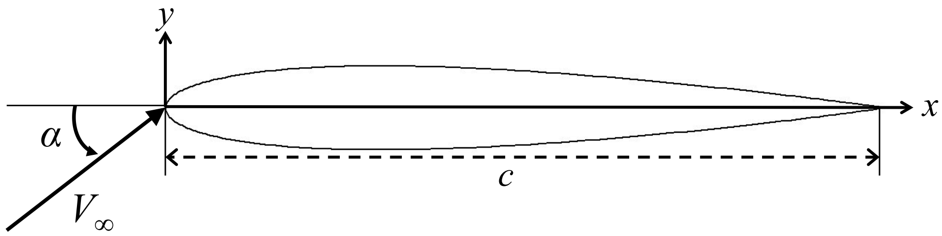
Using the composite Simpson's rule, determine the lift and drag (per unit span) on a stationary cylinder of diameter $d = 2$ m in an ideal airflow with freestream airspeed $V_\infty = 25$ m/s, air density $\rho_\infty = 0.9093$ kg/m^3, and pressure $p_\infty = 7.012 \times 10^4$ Pa.

It is required that you obtain lift and drag solutions that are within 0.001 Newtons of the exact solutions.

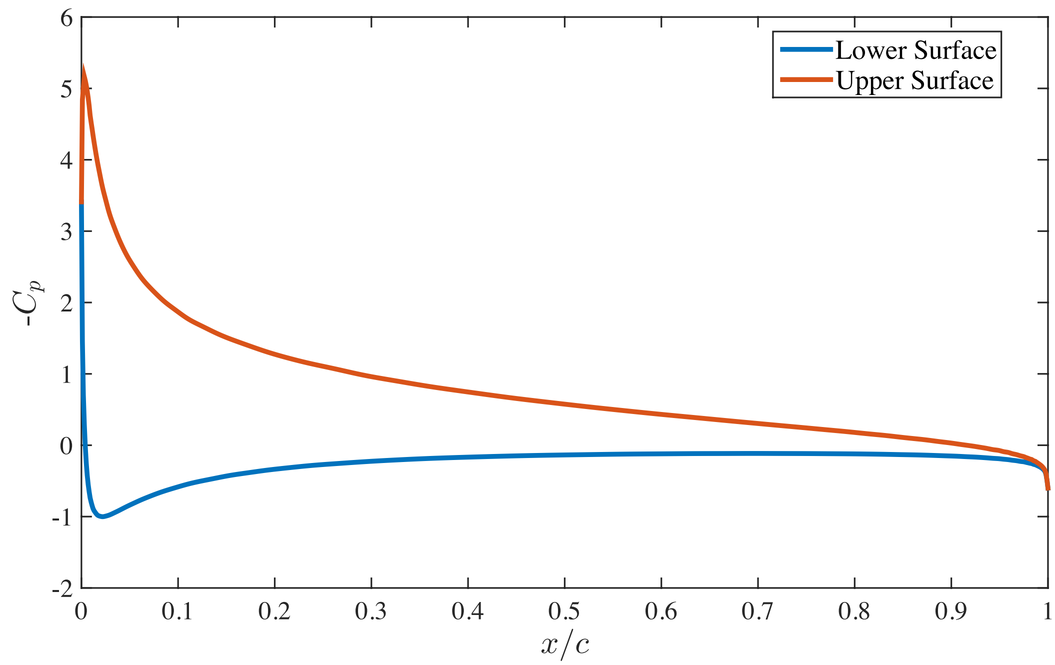
Reflection: Are your obtained lift and drag results physically reasonable?

Problem #2:

Consider ideal flow over a NACA 0012 airfoil at a 9° angle of attack as depicted in the figure below:



For this flow problem, the coefficient of pressure cannot be determined analytically. Nevertheless, the vortex panel method has been employed to approximate the coefficient of pressure along both the upper and lower surfaces of the NACA 0012 airfoil, yielding the results displayed below:



The vortex panel results have been further interpolated using splines and the results stored within a MATLAB .mat file `Cp.mat` located in the Lab directory on D2L. To open the MATLAB file, type `load Cp` into the Command Window. This will load two spline variables, `Cp_upper` and `Cp_lower`, into the Workspace. Then, to evaluate the coefficient of pressure along some location x/c along the upper surface, simply type `fnval(Cp_upper, x/c)`. Similarly, to evaluate the coefficient of pressure then along some location x/c along the lower surface, type `fnval(Cp_lower, x/c)`.

Using the MATLAB spline variables `Cp_upper` and `Cp_lower` and the composite trapezoidal rule, determine the lift and drag (per unit span) on a stationary NACA 0012 airfoil with chord length $c = 0.5$ m at 9° angle of attack in an ideal airfoil with freestream airspeed $V_\infty = 20$ m/s, air density $\rho_\infty = 1.225$ kg/m^3, and pressure $p_\infty = 10.13 \times 10^4$ Pa.

In addition to the above, complete the following tasks:

- Determine the number of equispaced (with respect to chord line distance, x) integration points required to obtain a lift solution with five percent relative error.
- Determine the number of equispaced (with respect to chord line distance, x) integration points required to obtain a lift solution with one percent relative error.
- Determine the number of equispaced (with respect to chord line distance, x) integration points required to obtain a lift solution with 1/10 percent relative error.

Reflection: Given the number of required equispaced integration points required to obtain an accurate lift solution, how should one go about measuring pressure in the wind tunnel to experimentally determine coefficient of lift?

Note: The formula for the shape of a NACA 00xx airfoil, with "xx" being replaced by the percentage of thickness to chord, is:

$$y_t = \frac{t}{0.2} c \left[0.2969 \sqrt{\frac{x}{c}} - 0.1260 \left(\frac{x}{c} \right) - 0.3516 \left(\frac{x}{c} \right)^2 + 0.2843 \left(\frac{x}{c} \right)^3 - 0.1036 \left(\frac{x}{c} \right)^4 \right]$$

where c is the chord length, x is the position along the chord from 0 to c, y_t is the half thickness at a given value of x (centerline to surface), and t is the maximum thickness as a fraction of the chord (i.e., $t = \text{xx}/100$).