Aerodynamics Computational Assignment #1: Computation of Lift and Drag

Assigned Date: August 30, 2019
Due Date: September 19, 2019

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.*

Matlab Code Policy:

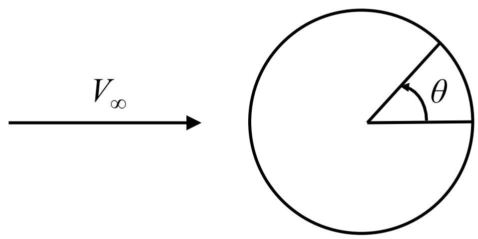
Computational codes must written individually and are expected to be written in MATLAB. If you have collaborated with others while writing your code be sure to acknowledge them in the header of your code, otherwise you may receive a zero for plagiarism. All code files required to successfully run the computational assignment driver script should be submitted via the course website by 11:59pm on the due date. Code files will not be accepted after the given due date.

Reflection Questions:

In this assignment, there are multiple reflection questions. These reflection questions are provided to help you review the functionality of your code, help you analyze and understand your results, and to test your understanding of the concepts being studied.

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 $d = 1$ m in an ideal airflow with freestream airspeed $V_\infty = 30$ m/s, air density $\rho_\infty = 1.225$ kg/m^3, and pressure $p_\infty = 101.3 \times 10^3$ Pa.

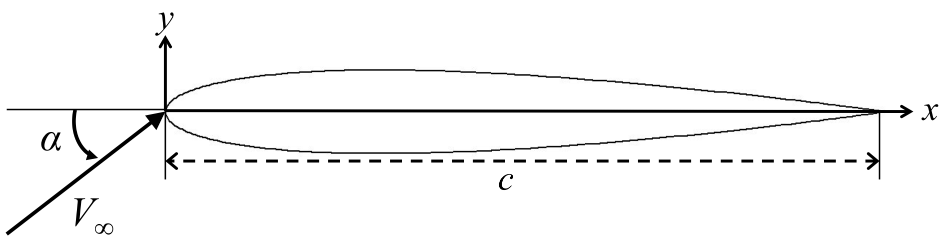
It is required that you obtain lift and drag solutions that are within 0.001 Newtons of the exact solutions.

Produce two plots of lift and drag, respectively, versus the number of panels used to discretize the surface of the cylinder. Print to the command window the number of panels, N, required to achieve the accuracy desired above.

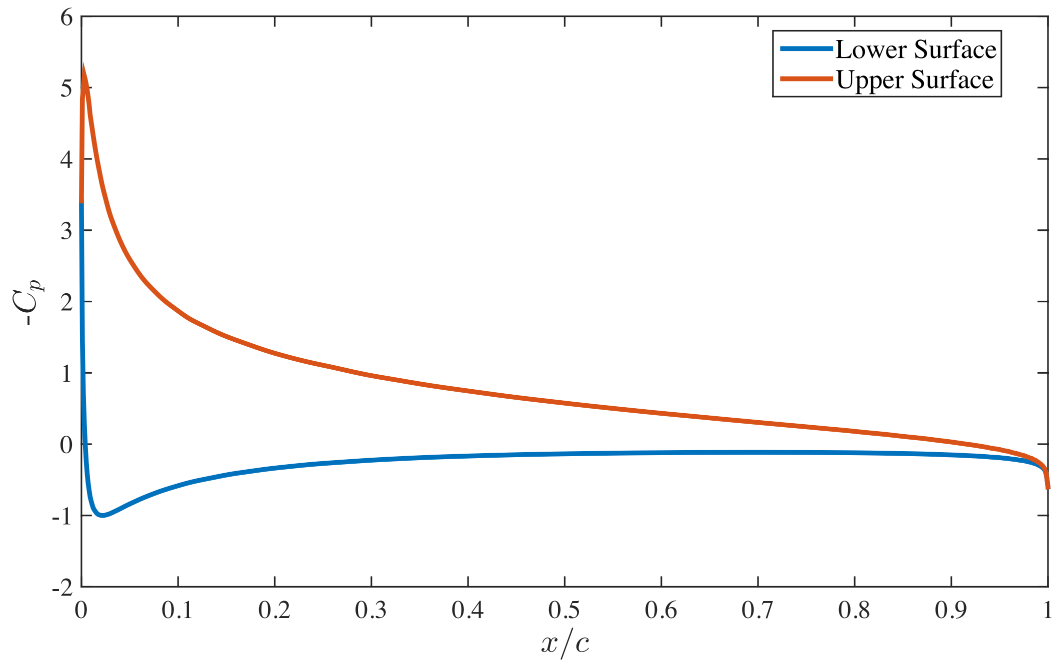
Reflection: Are your obtained lift and drag results physically reasonable?

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 are stored within a MATLAB .mat file `Cp.mat` located in the Lab directory on the course website. 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 $c = 2$ m at 9° angle of attack in an ideal airfoil with freestream airspeed $V_\infty = 30$ m/s, air density $\rho_\infty = 1.225$ kg/m^3, and pressure $p_\infty = 101.3 \times 10^3$ 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.

Produce a plot of the lift versus the number of integration points, n, where integration points n = N+1 panels along the surface of the airfoil. Additionally print to the command wind the number of integration points required to achieve the desired accuracies above.

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? If the number of experimental pressure ports is limited how can you best locate these ports along to surface to improve the accuracy in the lift estimate?

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., $t = \text{xx}/100$).