ASEN 3111 Computational Lab #4: Flow Over Thick Airfoils

Lab Date: October 6, 2017
Due Date: October 27, 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.

Problem #1:

Write a MATLAB function to apply the vortex panel method for an arbitrary two-dimensional body defined by a set of (x, y) coordinates that define its surface. Your function should include the following steps:

1. Read in the list of points.
2. Read in necessary flow conditions (e.g., the free-stream flow speed and angle of attack).
3. Form the system of equations.
4. Solve the system of equations.
5. Plot the coefficient of pressure resulting from your calculations.
6. Return the sectional coefficient of lift resulting from your calculations.

Your function should take the form:

```
function c_l = Vortex_Panel(x,y,V_inf,alpha)
```

where `c_l` is the sectional coefficient of lift (to be computed and returned), `x` is a vector containing the x-location of your coordinates (i.e., the boundary points in the panel method), `y` is a vector containing the y-location of your coordinates, `V_inf` is the free-stream flow speed, and `alpha` is the angle of attack. It is recommended that the first and last entries of `x` and `y` correspond to the trailing edge to ease implementation.

You are free to plot the coefficient of pressure in any form you like. For instance, you may plot the coefficient of pressure versus x/c, where $c = \max x - \min x$ is the chord length. Alternatively, you may plot the coefficient of pressure versus the arc length s.

Problem #2:

Apply the MATLAB function you wrote for Problem #1 to lifting flow over a NACA 0012 airfoil at various angles of attack. For an angle of attack of 0°, compute the flow for a couple of different resolutions and compare the results. From this study, choose a nominal number of panels required for a desired *quantitative* level of accuracy. It is up to the student to define a measure of error in this regard. Then, using the nominal number of panels, compute lifting flow over a NACA 0012 airfoil and plot the results (i.e., coefficients of lift and pressure) for the following angles of attack: $\alpha = -5^\circ, 0^\circ, 5^\circ, 10^\circ$.

Problem #3:

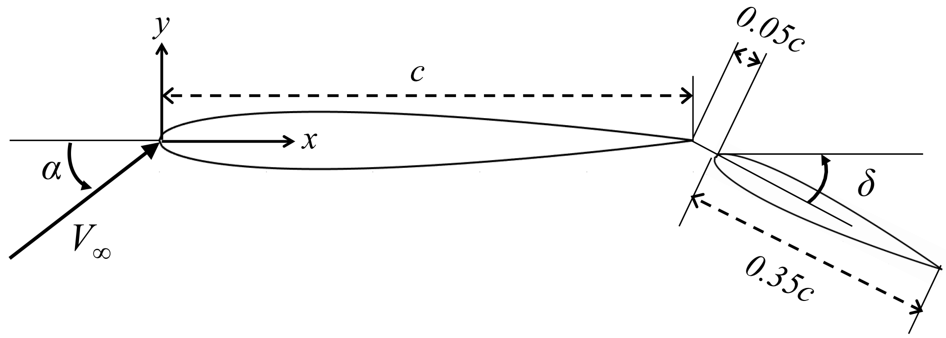
Using the MATLAB function you wrote for Problem #1, obtain plots of the sectional coefficient of lift versus angle of attack for the following airfoils:

- NACA 0012
- NACA 2212
- NACA 4412
- NACA 2430

Using these plots, estimate the lift slope and zero-lift angle of attack for each of the airfoils, and compare these results with thin airfoil theory.

Bonus (Up to +10 Points):

Use the vortex panel method to compute lifting flow over the following multi-element wing:



where both the airfoil and trailing edge flap are constructed from NACA 0012 airfoils. Obtain a plot of the sectional coefficient of lift versus angle of attack for a deflection angle of 10°, and compare this plot with the corresponding plot of sectional coefficient of lift versus angle of attack for a NACA 0012 airfoil without flap.

Suggested Approach:

We need to build 4-digit NACA airfoils repeatedly throughout this lab. As such, it is suggested that you build a MATLAB function to construct panels for a given NACA airfoil. For instance, your MATLAB function may take the form:

```
[x,y] = function NACA_Airfoil(m,p,t,c,N)
```

where `x` is a vector containing the x-location of the boundary points, `y` is a vector containing the y-location of the boundary points, `m` is the maximum camber, `p` is the location of maximum camber, `t` is the thickness, `c` is the chord length, and `N` is the number of employed panels.

Note: The formula for the shape of a NACA 4-digit series airfoil with camber is a bit involved. The first ingredient is the height of the airfoil from the mean camber line, which 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 (mean camber line to surface), and t is the maximum thickness as a fraction of the chord. As with the case of a symmetric NACA airfoil, the last two digits in the NACA xxxx description gives $100t$. The second ingredient is the formula for the mean camber line, which is:

$$y_c = \begin{cases} m\frac{x}{p^2}\left(2p - \frac{x}{c}\right), & 0 \leq x \leq pc \\ m\frac{c-x}{(1-p)^2}\left(1 + \frac{x}{c} - 2p\right), & pc \leq x \leq c \end{cases}$$

where m is the maximum camber and p is the location of maximum camber. The first digit in the NACA xxxx description gives $100m$ while the second digit gives $10p$. Then, the coordinates (x_U, y_U) and (x_L, y_L) of the upper and lower airfoil surface, respectively, become:

$$x_U = x - y_t \sin\xi \qquad y_U = y_c + y_t \cos\xi$$
$$x_L = x + y_t \sin\xi \qquad y_L = y_c - y_t \cos\xi$$

where

$$\xi = \arctan\left(\frac{dy_c}{dx}\right).$$

Note that for the NACA 4412 airfoil, $m = 4/100$, $p = 4/10$, and $t = 12/100$.