

Aerodynamics Computational Assignment #3: Flow Over Thick Airfoils

Assigned Date: February 24, 2021

Due Date: March 18, 2021

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 be 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.

Learning Outcomes:

1. Practice using the vortex panel method to calculate aerodynamic forces on a thick airfoil.
2. Understand the difference between the application and results of thin airfoil theory and the vortex panel method.
3. Understand how the changes in wing section camber and thickness alter the lift slope.

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. Return the coefficient of pressure resulting from your calculations and consider plotting it within the function.
6. Return the sectional coefficient of lift resulting from your calculations.

Your function should take the form:

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

where c_l is the section coefficient of lift and C_p is the coefficient of pressure (both which should be computed and returned), \mathbf{x} is a vector containing the x-location of your coordinates (i.e. the boundary points in the panel method), \mathbf{y} is a vector containing the y-location of your coordinates, V_{inf} is the free-stream flow speed, and α is the angle of attack. It is recommended that the first and last entries of \mathbf{x} and \mathbf{y} correspond to the trailing edge to ease implementation.

While debugging your function you are encouraged to plot the coefficient pressure within the function, but after debugging is complete it is recommended that you pass the coefficient of pressure back out of the function for plotting and analysis (as detailed in Problem # 2 below). 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. Produce a plot of the change in error versus the number of total panels (e.g. upper and lower surfaces) and print the nominal number of panels needed to the command window. 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: $\alpha = -5^\circ, 0^\circ, 5^\circ, 10^\circ$.

Reflection: How does the pressure coefficient field change with increasing angle of attack? How similar is it to thin airfoil theory (i.e. Comp. Assignment 02) and what differences exist?

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:

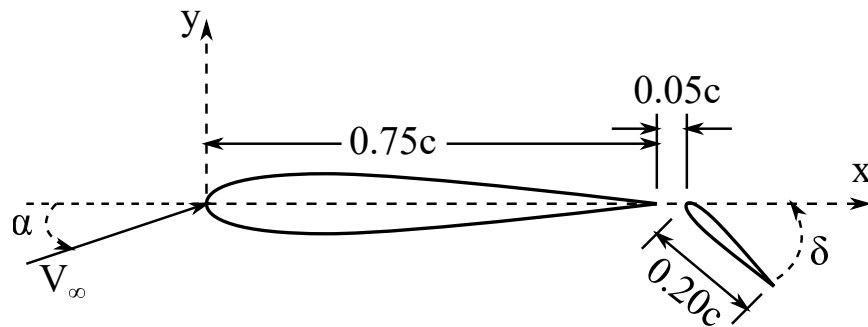
- NACA 0012
- NACA 2412
- NACA 4412
- NACA 2424

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. It is recommended that you plot all of these together to provide a clearer comparison.

Reflection: How do changes in the wing section camber and thickness alter the lift slope and the zero lift angle of attack?

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 section coefficient of lift versus angle of attack for a deflection angle of $\delta = 30^\circ$, 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:

You will 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_Airfoils(m,p,t,c,N)
```

where \mathbf{x} is a vector containing the x-location of the boundary points, \mathbf{y} is a vector containing the y-location of the boundary points, \mathbf{m} is the maximum camber, \mathbf{p} is the location of maximum camber, \mathbf{t} is the thickness, \mathbf{c} is the chord length, and \mathbf{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 thickness distribution of the airfoil from the mean camber line, which is given by:

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

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:

$$\begin{aligned} x_U &= x - y_t \sin \xi & y_U &= y_c + y_t \cos \xi \\ x_L &= x + y_t \sin \xi & y_L &= y_c - y_t \cos \xi \end{aligned}$$

where

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

Note that for the NACA 4415 airfoil, $m = 4/100$, $p = 4/10$, and $t = 15/100$.