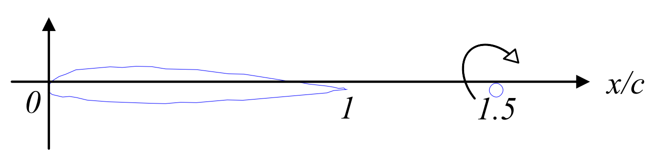
Homework 7

2D Panel Methods

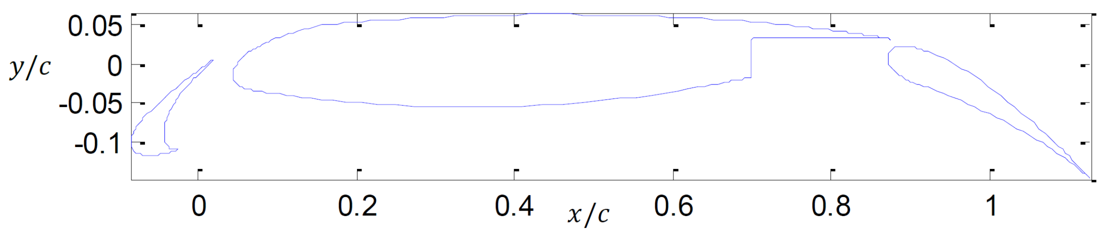
Your answers to these questions, and what you learn from them, will be greatly enhanced through collaboration and discussion amongst your discussion group and in the recitation. This is actively encouraged. However, once you have decided how to answer these, the final solutions must be prepared individually. Note that you will be asked to submit a PDF and your Matlab codes as separate files for this assignment.

1. The file r1080.dat in the zip file provided with this homework contains coordinates for the Roncz 1080 airfoil, as used on Burt Rutan's Voyager aircraft.
(a) Use the coordinates provided to perform a panel method solution for the Roncz 1080 with Kutta condition. You may use the coordinates to define the panel vertices – there is no need to interpolate. Feel free to adapt one of the 2D panel method codes on the course website (but highlight the changes you make in your submission). Plot the pressure coefficient distribution on the airfoil at 7 degrees angle of attack. Your plots should show $-C_p$ vs x/c. Use a range of -1 to 3 for $-C_p$.
(b) Derive an expression that relates the pressure coefficient distribution on the airfoil to the lift coefficient. Create a second version of the code that computes C_l from -15 to 15 degrees in steps of 1 degree. Show your derivation. Plot the results.

2. Use the data for the surface coordinates of the Eppler 377 modified airfoil (designed for ultralight aircraft), provided in the zip file with this homework.
(a) Plot the shape of the airfoil undistorted. (Note that the figure below is not a realistic view of the shape.) Create a MATLAB panel method code to compute and plot the pressure coefficient distribution on it for an angles of attack of 0° and 10° (plot $-C_p$ vs. x/c). Hint: You may use the airfoil coordinates supplied to define the panels.
(b) We now want to look at how a vortex in the flow downstream of the airfoil might affect its performance. Modify the linear vortex panel method code to include the effect of a vortex located half a chordlength downstream the airfoil trailing edge on the extension of the chord line, as shown in the figure. Plot the pressure distribution on the airfoil for an angle of attack of 0° when the vortex has a strength equal to $-0.5V_\infty c$ and $-1.5V_\infty c$, where V_∞ and c are the free stream velocity and chordlength. (Note: if you use a panel method code with streamline plotter from the course website, the streamline plotter may not work correctly once you add the vortex.)



3. We wish to investigate the flow around a 30P30N multi-element high lift airfoil system. The system can be seen in the figure below. The coordinates for the three elements are provided in the Matlab data file *threeElementAirfoil.mat*, provided in zipped form with this homework. This file contains three complex coordinate arrays *zslat*, *zmain*, and *zflap*. The coordinates are given so as to position each element at its correct relative location and angle for an overall airfoil angle of attack of zero degrees. Note that the chordlength c and the leading edge origins of the coordinates are referenced to the cruise configuration where the flap and slat are withdrawn. The trailing edge of each element is at its right-most point as shown in the figure. In extended high lift configuration shown, the main element, and thus the whole airfoil, is considered to be at zero angle of attack, regardless of the position or angle of the slat or flap. Develop a panel method code to compute the flow around this three element airfoil, by extending one of the methods provided in the course. Use the code to compute and plot the three pressure coefficient



distributions on the three airfoil elements for 0° angle of attack. Restrict your range of $-C_p$ from -1 to 4 to prevent these from creating scaling problems in your plots. Note that the sharp corners on the slat and main element may produce singularities in the pressure distributions. Plot the streamlines for this case.