AE333

Aerodynamics

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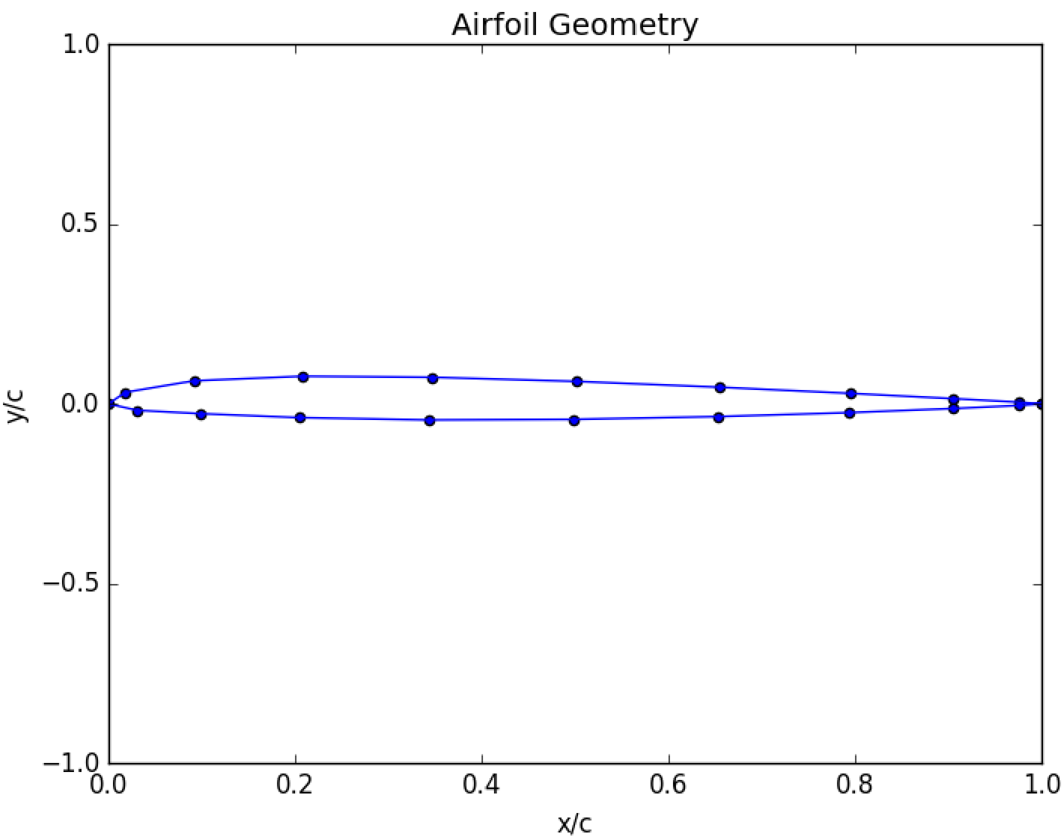
Question:

Input - Random number = 8,
NACA 23112,
roll number =13,
Number of panels = 20,
Angle of attack - 3,13,17.
Velocity = 20mps.
Linear variation.

Panels:

Airfoil points were taken from airfoiltools.com
121 points were obtained from net. As the points were given to define the airfoil in most appropriate way, the points taken for the assignment are every 6th point.
It gives 21points starting from [1,0] anticlockwise and completing the airfoil to [1,0] again.
Also the initial and final points were not exactly [1,0] but very close to it, and to get a complete plot of airfoil, it was rounded off to [1,0]. 20 points are required for 20 panels, in 21points that are considered the first and last points are same, it helps to execute the for loop.



Source Panel Method:

To express the flow or an object defined by set of points/equation, we consider sources of certain strength/unit length panel to be placed over the surface of an airfoil.
If equation is given, points of the object can be calculated from the equation. Certain number of panels are taken to consider the object, points that when joined can describe a panel in a most geometrically correct way.
In this assignment linear source panel method is discussed.
The midpoints of the panels are the control surfaces, i.e. the boundary conditions are applied on these points, where as the continuity of the source strengths is considered at boundaries of panel.

The equation in integral form

$$I_{i,j} = \int_0^{S_j} \frac{(m_j * s_j + c_j)(Cs_j + D)}{s_j^2 + 2As_j + B} ds_j$$

where

$$A = -(x_i - X_j)\cos\Phi_j - (y_i - Y_j)\sin\Phi_j$$

$$B = (x_i - X_j)^2 + (y_i - Y_j)^2$$

$$C = \sin(\Phi_i - \Phi_j)$$

$$D = (y_i - Y_j)\cos\Phi_i - (x_i - X_j)\sin\Phi_i$$

$$S_j = \sqrt{(X_{j+1} - X_j)^2 + (Y_{j+1} - Y_j)^2}$$

$$E = \sqrt{B - A^2} = (x_i - X_j)\sin\Phi_j - (y_i - Y_j)\cos\Phi_j$$

$$\lambda_j = m_j * s_j + c_j$$

Phi in book is thetha in the code explained below. Phi is slope of plane with x axis.

(x_i, y_i) are the coordinates of the control point of the ith panel and (x_j, y_j) are the running coordinates
over the entire j th panel. The coordinates of the boundary points for the ith panel are (X_i, Y_i) and (X_{i+1}, Y_{i+1})

The integrated equation

$$I_{i,j} = m_j C S_j + \left[\frac{m_j(D - 2AC)}{2} \ln\left(\frac{S_j^2 + 2AS_j + B}{B}\right)\right]$$
$$- \frac{m_j(BC + A(D - 2AC))}{E}\left[\tan^{-1}\left(\frac{S_j + A}{E}\right) - \tan^{-1}\left(\frac{A}{E}\right)\right]$$
$$+ \frac{c_j C}{2} \ln\left(\frac{S_j^2 + 2AS_j + B}{B}\right) + \frac{c_j(D - AC)}{E}\left(tan^{-1}\frac{S_j + A}{E} - tan^{-1}\frac{A}{E}\right)$$

Boundary conditions are applied then, which has normal to plate velocity is zero and summation of all strength is zero. as we have added the sources for our convenience, and they were not originally present.
Hence we have 2n+1 equations and 2n variables (mi,ci)
The boundary conditions are explained better in the code with equations

for Cp

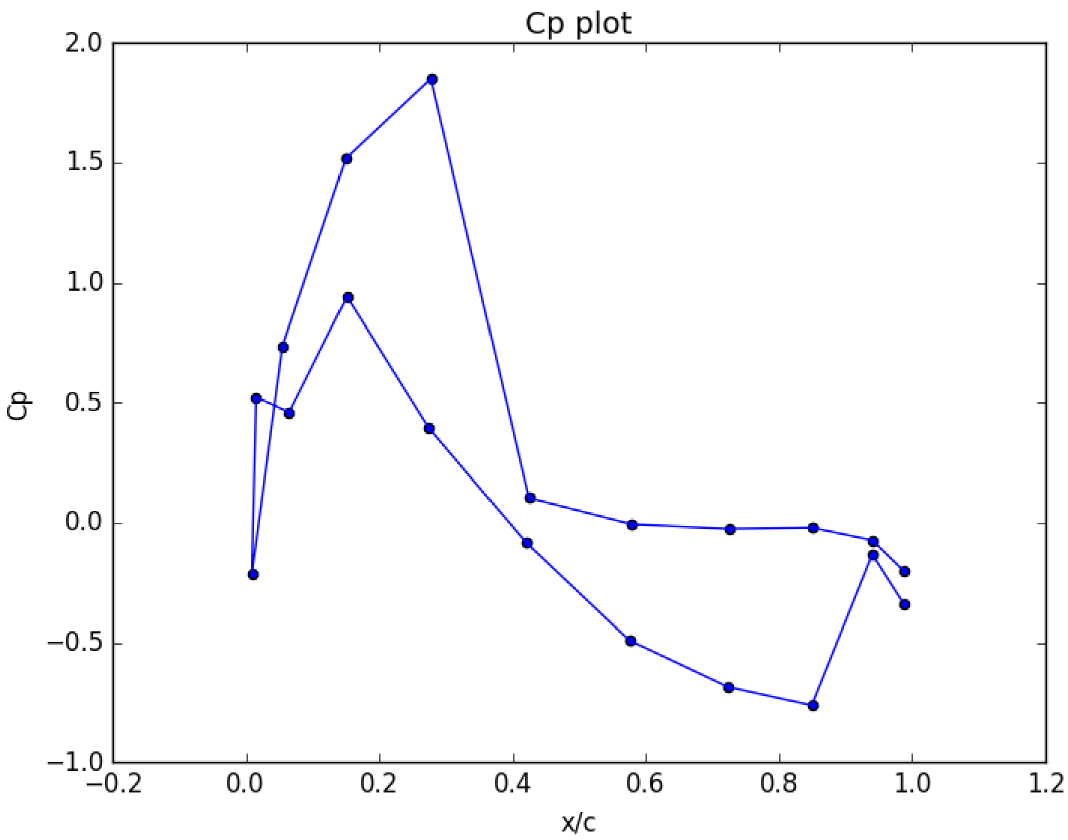
$$C_{p,i} = 1 - \left(\frac{V_i}{V_\infty}\right)^2$$

For Cl, Cd refer to equations in the code as they are summation of Cp of plates..

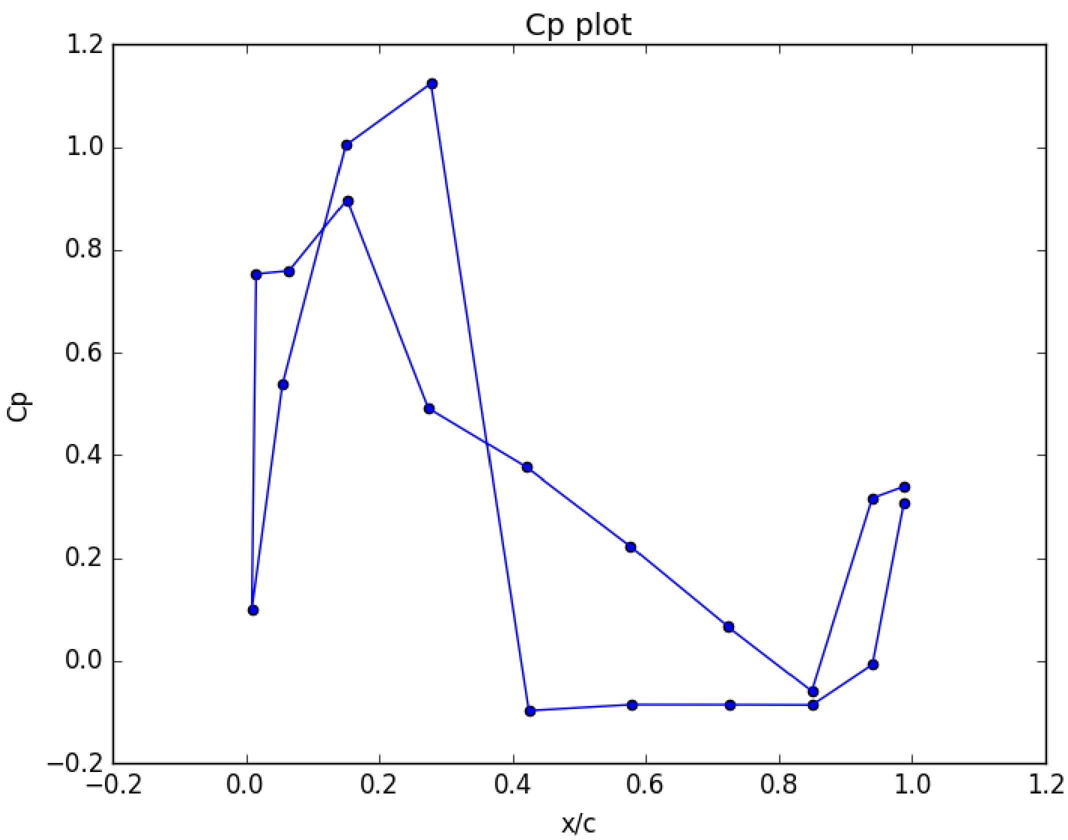
Algorithm
1) Import Values of Coordinates of airfoil
2)Plot airfoil
3)Show 20 points
4)Calculate length and slope angle of panels
5)Calculate beta angle of normal of panel and V(infinity)
6)Midpoints of panels
7)Calculating the value of A,B,C... and substituting in integrated equation
8)Writing continuity equation
9)Substituting one equation in those with the summation of strength's = 0 equation and solving
10)Getting the values of lambda's
11)Writing equation for Velocity along panel for Cp distribution
12) Substituting in Cp equation to get Cp values, and plot
13)Cl, Cd Equations to find Cl, Cd values and plot Cl,Cd vs alpha, for different angle of attack.
The program is created to take input value of alpha but is commented.
Code written in the attached file is as per the same sequence with proper comments.

Cp Distribution (Cp values on positive axis are actually negative. Plot was inverted to see proper distribution)

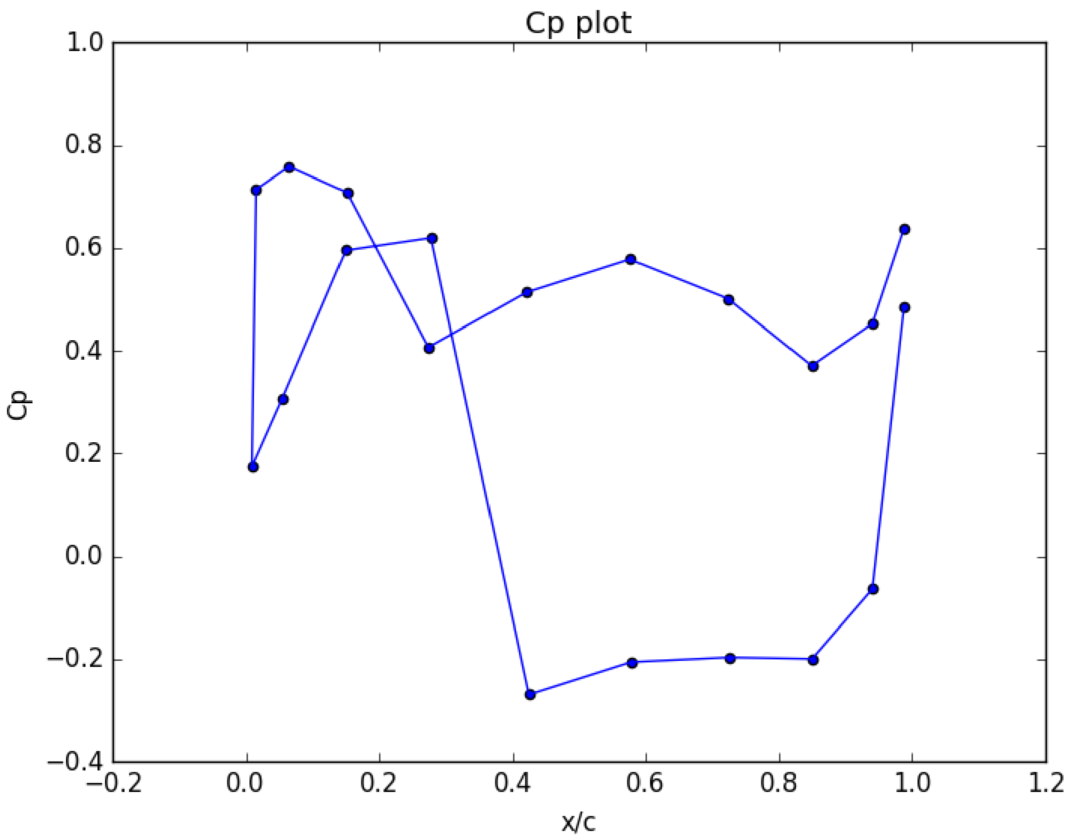
1) Alpha = -17degrees



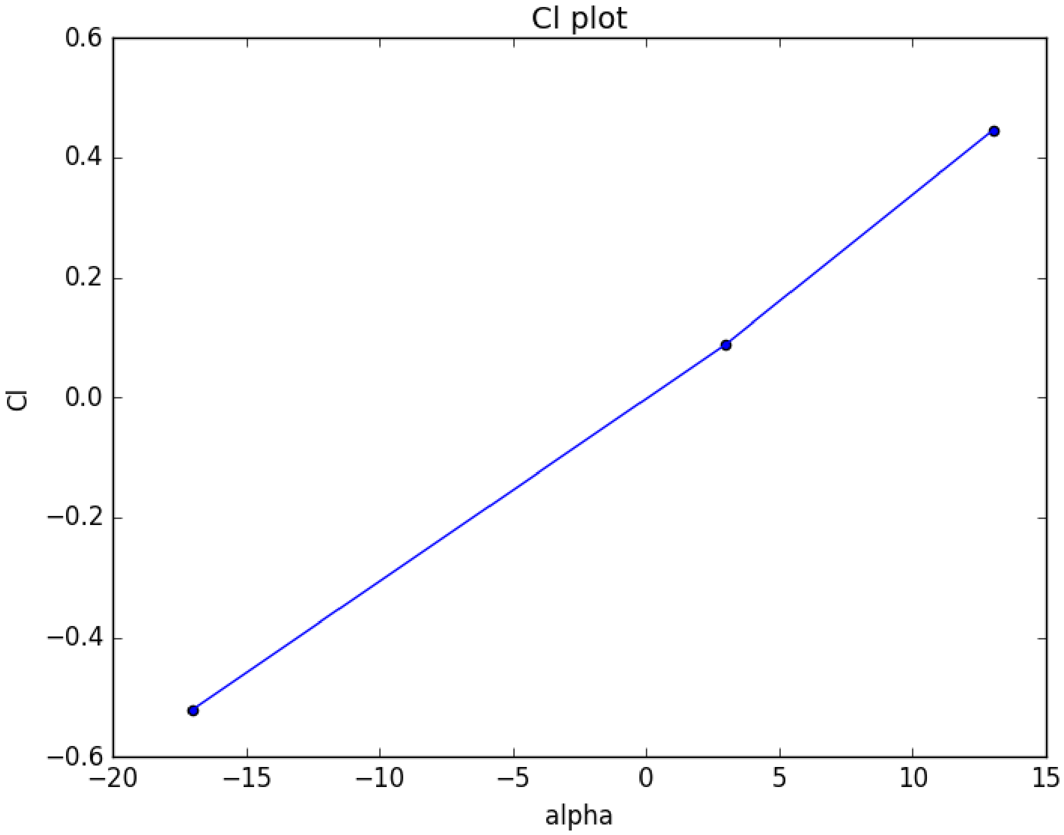
2)Alpha =3degrees



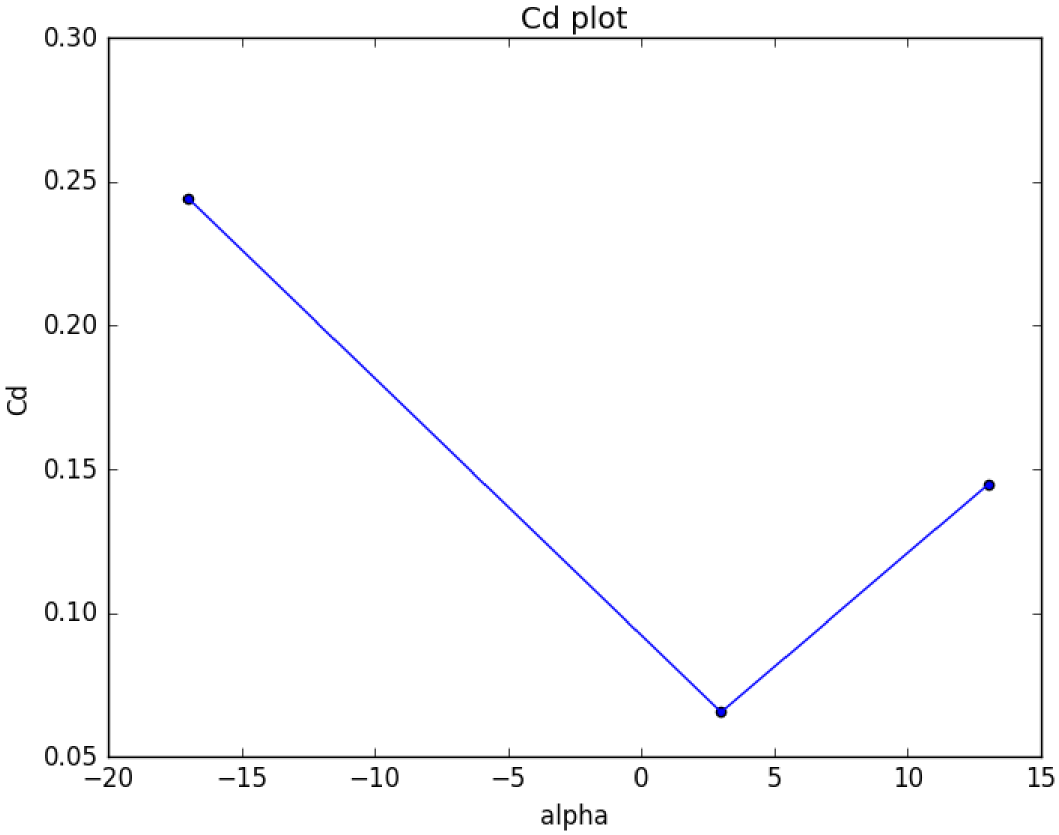
3)Alpha = 13degrees.



Cl Distribution vs Alpha



Cd Distribution vs Alpha



Conclusion

From the above assignment we understood that source panel method gives a pretty good approximation to the standard charts that we see in text books, though more number of panels could lead to better results. This coding algorithm can be used to simulate the lift, drag, or basic flow over any object and can be used extensive to study airfoil geometries and to get the perfect airfoil shape to find optimum solution for lift and drag coefficients, hence improving the plane in any way possible.