AE211A

INCOMPRESSIBLE AERODYNAMICS

MID-SEMESTER EXAMINATION SUBMISSION

2019-20 - II

Incompressible Flow Analysis of Eppler 325 Airfoil using Source Panel and Vortex Panel Methods

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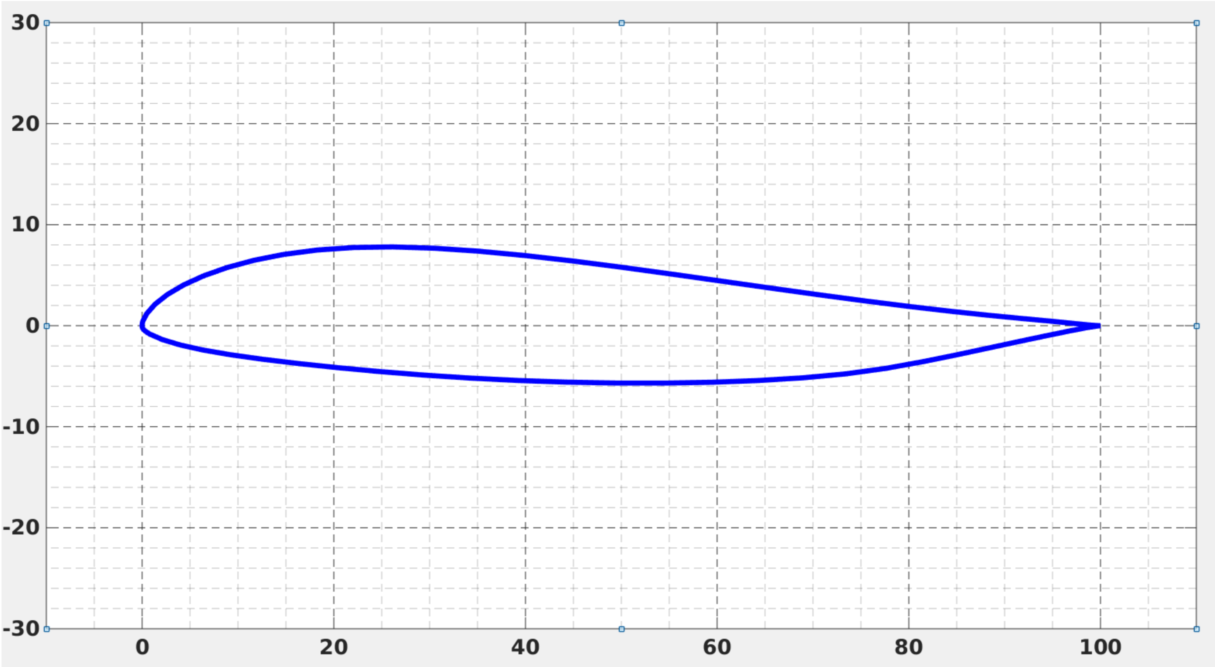
1 About the airfoil

The shape that we have chosen for our analysis is the Eppler 325 Airfoil.



THE EPPLER 325 AIRFOIL

For this airfoil :

- Max thickness is 12.6% at 34.3% of the chord length.
- Max camber is 1.8% at 16.3% of the chord length.

The data for the airfoil was obtained from airfoiltools.com. The dataset contains 71 unique points. We have chosen adjacent points to be the ends of a panel. Thus, we get 71 panels which form a closed polygon of 71 sides.

Note that in the given dataset, the density of the data points is relatively higher near the leading edge of the airfoil. This enables us to cover the leading edge region with a number of small panels, which helps us accurately represent the rapid surface curvature in the region.

2 Code

The MATLAB code used for analysis can be found at the following link :

`https://github.com/dryairship/ae211a-eppler-325`

The sources for the base code have been attributed in the repository as well as in the bibliography of this report. Further, for the entire analysis, V_∞ was chosen to be $10 m/s$.

3 Source Panel Method Analysis

3.1 Summation of source strengths

The summation of source strengths should come out to be 0 because it is not physically possible for the flow to pass through the boundary of the airfoil. Thus, the sum of source strengths can be used to verify the accuracy of the solution.

The value of $lambda$ calculated for each panel in the code is actually strength per unit length. Thus, to get the actual strength of each panel, we multiply $lambda$ with the length of the panel, S. We then sum the values of $lambda \cdot S$ for all panels. This sum should theoretically come out to be 0. The table below shows the obtained values for for the angle of attack $\alpha \in [-10°, 29°]$. We find that the calculated sum of the strengths of the source panels remains close to 0 for varying angles of attack.

3.2 Circulation and Lift

The Source Panel method is based on potential flow, which is derived from Euler's equations. Since Euler's equations neglect the effect of viscosity, all solutions of the Source Panel method are non-lifting flows. We calculated the theoretical circulation using $\int V.ds$. Also, from the Kutta–Joukowski theorem, we know that lift is directly proportional to the circulation. Therefore, the solutions obtained by using the source panel method must have 0 circulation. This is reflected in the calculated values:

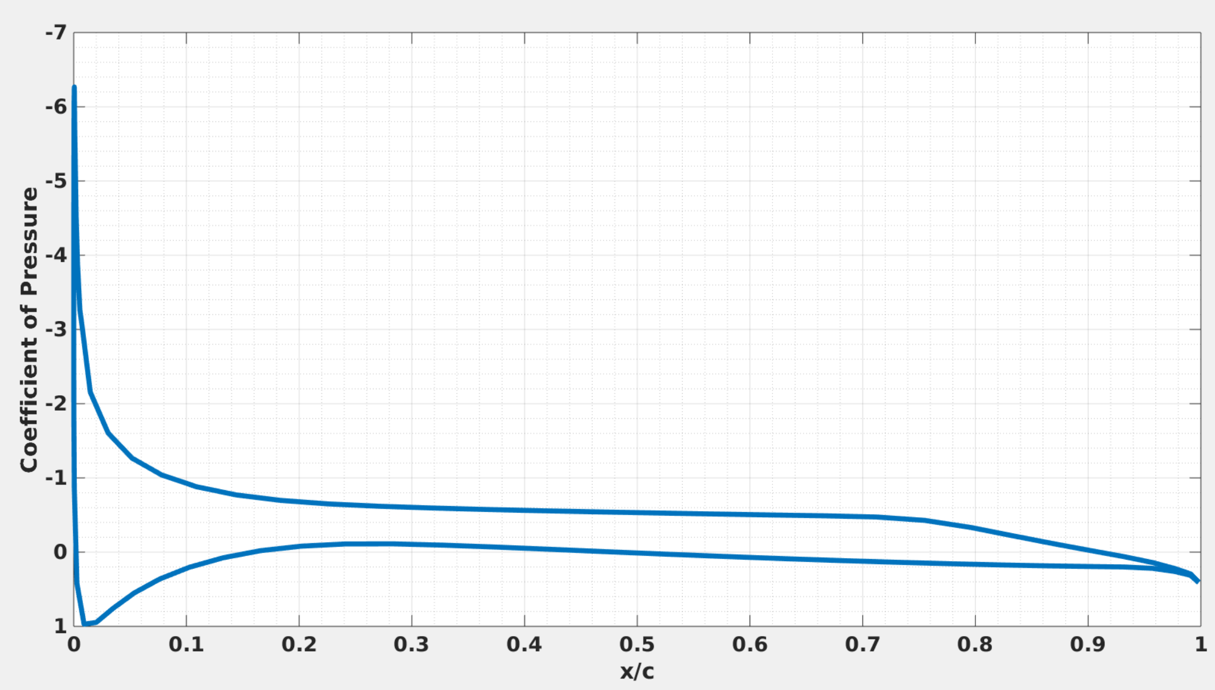
Angle of Attack	Sum of strength of sources	Circulation		Angle of Attack	Sum of strength of sources	Circulation
-10	-0.13798	-1.48423		10	0.10813	-0.23032
-9	-0.13531	-2.44962		11	0.09542	-0.19319
-8	-0.42456	-0.24419		12	0.08484	-0.16678
-7	-0.39909	-0.18674		13	0.07594	-0.13144
-6	-0.42056	-0.13134		14	0.06838	-0.10056
-5	-0.48114	-0.09577		15	0.06189	-0.06337
-4	-0.59907	-0.16004		16	0.05629	-0.02873
-3	-0.84116	-0.21832		17	0.05143	-0.00653
-2	-1.51558	-0.51761		18	0.04717	0.01263
-1	-10.70031	-4.27961		19	0.04343	0.03774
0	1.97284	0.20379		20	0.04012	0.05834
1	0.87497	-0.59979		21	0.03719	0.07048
2	0.54976	-0.66773		22	0.03457	0.08188
3	0.39361	-0.65112		23	0.03224	0.10112
4	0.30184	-0.58152		24	0.03014	0.10764
5	0.24147	-0.48177		25	0.02824	0.11421
6	0.19883	-0.42507		26	0.02654	0.1242
7	0.16718	-0.3568		27	0.02499	0.13122
8	0.14285	-0.30719		28	0.02358	0.13014
9	0.12363	-0.27052		29	0.0223	0.13881

VARIATION OF CIRCULATION AND SUM OF SOURCE STRENGTHS WITH THE ANGLE OF ATTACK

4 Vortex Panel Method Analysis

4.1 C_P v/s $\frac{x}{c}$

We get two Pressure Coefficients for each X, one for the upper surface and one for the lower surface. In the graph, the lower curve corresponds to the lower surface of the airfoil and the upper curve corresponds to the upper surface. A negative C_P means that velocity is greater than the free stream speed and opposite is true for positive C_P's.



C_P v/s $\frac{x}{c}$ for $\alpha = 5\circ$

4.2 $C_{L,circ} = C_{L,CP}$

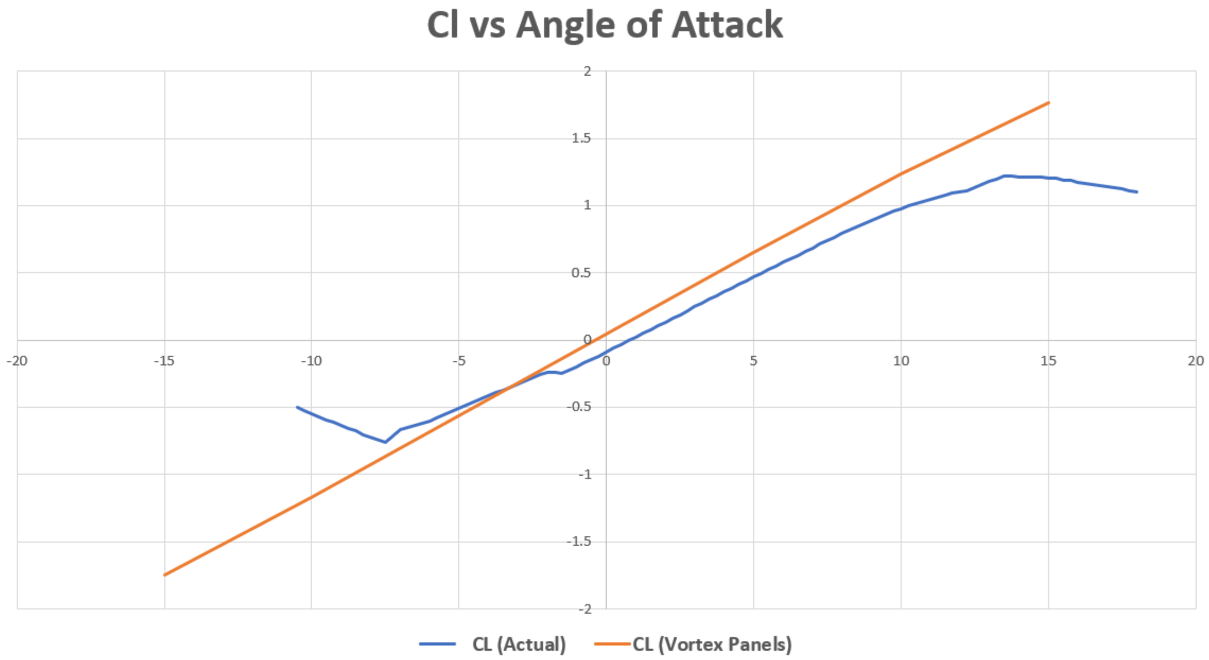
In the code, the lift coefficient is calculated in two ways, one by integrating over C_P and other by calculating circulation. These two values should come out to be the same. Here are the values for some angles of attack:

Angle of Attack	-15	-10	-5	0	5	10	15
CL (from CP)	-174.53	-117.02	-56.64	4.78	65.37	123.3	176.79
CL (from Circ.)	-170.02	-111.17	-51.47	8.61	68.64	128.14	186.67

COMPARISON OF C_L OBTAINED USING TWO DIFFERENT METHODS

4.3 Comparison of actual C_L v/s α and the obtained values

When we plot the values of the lift coefficients v/s angle of attacks as obtained by using the vortex panel method, we get the following graph:



C_L v/s α (ACTUAL V/S OBTAINED)

The obtained C_L v/s α graph is a straight line with a positive slope. This is because the vortex panel method does not account for flow separation, and hence it cannot be used to determine the stall angle.

Thus, the analysis of the lift coefficient done using the vortex panel method fails for angles of attack greater than the stall angle. However, for angles less than that stall angle, the obtained lift coefficient provides a good approximation for the actual lift coefficient.

4.4 Kutta condition

The solution was obtained by enforcing the Kutta condition at the trailing end. Since our dataset has points in the anticlockwise direction starting from the trailing edge, the two panels forming the cusp are the first and the last panels. Therefore, `gamma(first) + gamma(last) = 0`. This is shown in the code.

5 Bibliography

- The base code for the **Source Panel method** was obtained from `https://in.mathworks.com/matlabcentral/fileexchange/56156-source-panel-method-applied-to-flow-around-cylinder`

- The base code for the **Vortex Panel method** was obtained from `https://github.com/dpkprm/Vortex-Panel-Method`

- The **points dataset** for the Eppler 325 Airfoil was obtained from `http://airfoiltools.com/airfoil/details?airfoil=e325-il`.
  The same source also helped us get the actual C_L v/s α values for the airfoil.

- Fundamentals of Aerodynamics by John D. Anderson was consulted for solving the problem using the source panel method.

- Foundations of Aerodynamics by Arnold M. Kuethe and Chuen-Yen Chow was consulted for solving the problem using the vortex panel method.