Exploring the Effect of the Bow Region Shape and Bluntness on the Two-Dimensional Potential Flow for the DARPA 2 Submarine Sail

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Abstract

The effect of the bow region shape and bluntness on the two-dimensional potential flow for the DARPA 2 submarine sail, which is a symmetric airfoil whose model shape is given by the equations listed with the drawing in Fig. 1 of Appendix A, is explored and analyzed. The vortex panel method is used for a variable number of panels on the body surface to calculate the flow for different nose shapes. Various data are obtained including the pressure coefficient for the upper and lower surfaces of the hydrofoil, the circulation around the hydrofoil, the lift coefficient versus angle of attack, the moment coefficient about the quarter chord versus angle of attack, the location of the aerodynamic center, and a comparison of the lift coefficient with thin airfoil theory.

No aid given nor received on this assignment. –Michael D. Naper, Jr.

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1. Introduction

The DARPA 2 submarine sail is a generic unclassified, symmetric submarine used in hydrodynamics research, shown in Fig. 2 of Appendix A. For a Low Separation Submarine Sail to suppress flow separation, the adverse or positive pressure gradient over the bow and downstream must be sufficiently low to avoid stalled boundary layer flow. For a Low Cavitation Submarine Sail to avoid cavitation inception, the pressure over the surface at top speed at any angle of sideslip must not fall below the vapor pressure corresponding to the temperature of the water. When the pressure does fall below this vapor pressure, vapor bubbles form in the water and later collapse when the pressure increases downstream. The resulting pressure fluctuations impact on the surface and can fatigue the surface and cause it to become damaged. Furthermore, acoustic noise is produced, which can be detected by an enemy. Such a Low Cavitation Sail must have pressure above a minimal pressure $C_p = -K$ on its surface. Even with these requirements, the sail must produce enough lift for submarine performance.

2. Description of the Solution Method

The Vortex Panel Method is a computational method used in the study of the aerodynamics of airfoil sections. The Vortex Panel Method allows for the computation of ideal flows, i.e., flows in which the effects of compressibility and viscosity are negligible. Ideal flow is the simplest to model, with large parts of the flows past ships, submarines, cars, and light aircraft being closely ideal.

Ideal fluid flows are solutions to Laplace's equation, which is a linear differential equation, meaning that solutions can be superposed to produce new ideal flows. One approach to finding the solution to complex flow problems is therefore to begin with very simple ideal flows that are easily understood and described and then to superpose them together to produce the complex flow patterns desired.

The Vortex Panel Method models the flow past an airfoil as the summation of a uniform flow and a series of vortex panels (or sheets) arranged to form a closed polygon with a shape that approximates the actual curved shape of the airfoil. Superposition of the potential functions for a uniform stream at angle of attack and distributed potential vortices located along the surface of the airfoil are used to describe the fluid flow over the body. A linear variation of circulation per unit length along each panel is subsequently used. Acknowledging that the fluid cannot pass through the surface of the body, a boundary condition of zero velocity normal to the panels is imposed. Using the Kutta Condition, which states that the rear stagnation point must be at the trailing edge of the airfoil, a system of equations can be evaluated to obtain the various circulations per unit length. Lastly, the Kutta-Joukowski Theorem may be used to obtain the lift per unit span of the airfoil by integrating the calculated circulations per unit length along the surface of the airfoil and substituting into

$$l = \rho_\infty U_\infty \Gamma.$$

3. Results

Part 1:

To get a sense of how the bow region shape and bluntness of the DARPA 2 submarine sail affects the two-dimensional potential flow around the sail, various nose shape parameters were used (N = 2, 3/2, 3) in the computation of the desired data. Additionally, it should be apparent that the configuration of the panels used to approximate the contour of the hydrofoil needs to be taken into consideration when utilizing the vortex panel method to calculate the flows around the hydrofoil. As such, the effects of various paneling configurations, i.e., changes in the number of panels (24 versus 48) and the method of determining panel boundary points, were also explored. The sail contour for the various nose shape parameters and paneling configurations can be examined in Appendix B.

Intuitively, the resulting data from the chosen panel configuration used in the analysis of the hydrofoil will be most sensitive where changes in surface curvature are large, i.e., the nose and tail of the sail. As such, care was taken in selecting an appropriate paneling method such that these areas of the contour were emphasized in the determination of panel boundary points. Three separate paneling configurations were evaluated: The first is a simple, yet reliable method where a circle centered at midchord is drawn, which passes through both leading and trailing edges. The circumference of the circle is divided into arcs of equal length and the projection of these points on the circle gives the boundary points on the surface of the airfoil. In this way, relatively short panels are automatically obtained in the leading- and trailing-edge regions. The second method is a straightforward method of dividing the chord length into equal sections, having each panel be of equal length without regard to the contour of the hydrofoil. The third method is an extension of the second method where the hydrofoil is divided into three regions – nose, middle, and tail – and a higher ratio of panels is used for the nose and tail regions.

As can be seen from the resulting graphs (Appendix B), the number of panels used has a noticeable effect on the data. With an increase in panels, the data plots conform more smoothly to a realistic function of pressure coefficient versus length along the chord. In the case of the DARPA 2 hydrofoil, calculations for a simpler contour are more easily attainable with fewer panels. That is, 48 panels demonstrate a high quality plot of pressure coefficient and any further amount of panels will provide little gain.

The configuration of the panels shows additional consequences. With the latter two configurations, less emphasis is on the locations of large curvature fluctuation. The graphs prove the importance of these areas in the computation of the desired data. The second and third configurations neglect these large surface contour changes and thus do not capture well the effects of these regions in the overall results. For instance, it can be seen that the latter configurations are absent of any smooth curvature around the nose and tail regions for the pressure coefficients and yield a lower lift coefficient and a higher moment coefficient about the leading edge. Moreover, the lack of precision around the nose region of the hydrofoil provides notable differences in the location of the stagnation point (where pressure coefficient is equal to one), with the latter configurations pushing the stagnation point further downstream.

Part 2:

Graphs and tables of pressure coefficient C_p versus x/c for the various paneling methods and nose shape parameters can be found in Appendix B and Appendix C, respectively. As should be expected, the pressure coefficients along the upper and lower surfaces of the hydrofoil are identical at zero angle of attack. Symmetric bodies such as the DARPA 2 should have equal pressure along the top and bottom surfaces as the passing flow will be of equal velocity and thus equal dynamic pressure. As the angle of attack is increased, it is seen that the pressure coefficients along the upper surface begin to decrease while the pressure coefficients along the lower surface begin to increase at lesser rate than that of the upper surface's decreasing coefficient rate. With a net pressure coefficient beginning to form, lift can now be developed from the hydrofoil. It should be noted that this net pressure coefficient increases with angle of attack, thus showing that an increase in angle of attack provides greater lift. As we move along the length of the hydrofoil, the pressure coefficients peak around the first twentieth of the chord length and gradually return toward zero net pressure coefficient following. Again, it should be noted that an increase in angle of attack decreases the rate at which this gradual return occurs.

The rate of change in the values of pressure coefficient along the hydrofoil can be seen in Appendix D. Similarly to the pressure coefficient, the rate of change in pressure coefficient for the zero angle of attack case gives identical values. As this angle is increased, a separation in values is once again observed. The graphs rapidly demonstrate the expected values of pressure gradient along the hydrofoil, with large changes in pressure in the nose region and slower changes further downstream. As with the pressure coefficient plots, the most observable changes occur in the initial tenth of chord along the hydrofoil, including a change from negative to positive dC_p/dx around 0.07c and moving further upstream as the angle of attack is increased. With large positive values of dC_p/dx, the real viscous flow will begin to separate from the hydrofoil. Thus, with larger angle of attack, separation is more likely to occur closer upstream.

Part 3:

The total circulation around the hydrofoil normalized on the chord and the approach free stream velocity as a function of the angle of attack can be referenced by Appendix E. The resulting data demonstrates that the circulation follows a linear relationship with angle of attack. Additionally, with an increase in nose shape parameter, the slope of this linear relationship increases. Thus an increase in nose bluntness generates a higher circulation around the hydrofoil.

Part 4:

The lift coefficient of the hydrofoil, which is the lift per unit span normalized on the chord and the approach free stream velocity, as a function of the angle of attack can be seen in Appendix F. Once again, the resulting data demonstrates that the lift coefficient follows a linear relationship with angle of attack. It should be noted that the normalized lift coefficients are double in value to that of the normalized total circulation around the airfoil. This can be easily shown mathematically by equating the normalized lift coefficient and normalized total circulation as follows:

$$L = \frac{1}{2} C_L \rho_\infty U_\infty^2 c = \rho_\infty U_\infty \Gamma$$

$$C_L = \frac{2\Gamma}{U_\infty c}$$

As with the circulation, an increase in nose bluntness generates a higher lift around the hydrofoil.

Observing the graph, it is easily seen that the zero-lift angle of attack occurs at an angle of attack of zero. This conforms to the previous claim that a symmetric body will produce no lift.

Part 5:

The moment coefficient of the hydrofoil about the leading edge can be found in Appendix B. Additionally, the moment coefficient about the quarter-chord as a function of angle of attack can be located in Appendix G, following from the equation

$$C_{m_b} = C_{m_a} + (C_l \cos\alpha + C_d \sin\alpha)\left(\frac{x_b}{c} - \frac{x_a}{c}\right)$$

where C_{m_b} is the moment about the quarter chord, C_{m_a} is the moment about the leading edge, and $\frac{x_b}{c} - \frac{x_a}{c}$ is the fractional chord length of $1/4$.

From the graph, the aerodynamic center is readily observed. The moment coefficient remains constant as angle of attack changes, thus demonstrating that the aerodynamic center is at the quarter-chord. Integrating the pressure coefficient values aft of the aerodynamic center, it is shown that approximately 30% of the lift loading is downstream of the aerodynamic center at an angle of attack of 10°.

Part 6:

When computing values for any application, it is generally advisable to compare data with that given by theory. In the case of the DARPA 2 hydrofoil, thin airfoil theory can be used to compare lift coefficient values. For thin airfoils, that is airfoils with no thickness, the lift coefficient is given by

$$C_l = 2\pi\alpha$$

where α is the angle of attack in radians. Comparing the computed values for the DARPA 2 hydrofoil, it is observed that the lift coefficient is lesser for the thin airfoil case. More accurately, the rate of increase of the lift coefficient with respect to angle of attack is greater when thickness is incorporated. Thus, finite thickness produces greater lift than that of the thin airfoil case. As remarked upon in the preceding section, an increase in the nose shape parameter causes the rate of increase of the lift coefficient to increase for both the cases of no thickness and finite thickness.

4. Discussion and Conclusion

As with all computational analyses, a certain amount of error is present in the resulting data. Naturally, it necessary to question the accuracy of the results and make conclusions about the usefulness of the computed data. In the case of the DARPA 2 hydrofoil, the parameters which affect the quality of data are the paneling configurations used in the vortex panel method such as the number of panels and the method of boundary control point determination.

Comparing the data for the various paneling configurations, several trends are observed. More panels modeling areas of large curvature such as the nose and tail lead to more consistent results than those paneling methods that do not account for fluctuation in the contour of the body. Observing the differences in the results of the various paneling methods, a stagnation point further from the leading edge is most likely a result of poor paneling in areas of high contour sensitivity. Additionally, it is seen that the number of panels follows diminishing returns, with a smoother body such as the DARPA 2 needing fewer panels to yield accurate results.

Appendix A: DARPA 2 Model Data

(x,y,z) = Cartesian coordinates in Ft

The sail is defined by 3 sections: the forebody, parallel middle body, & afterbody. The sail dimensions and equations follow.

Sail Forebody Length = .325521 Ft (.099m)

Sail Parallel Middle Body Length = .200521 Ft (.061m)

Sail Afterbody Length = .682292 Ft (.208m)

Total Sail Length = 1.208333 Ft (.368m)

Span of Sail with Uniform Profile = .674479 Ft (.206m)

z_{max} = One-Half the Maximum Sail Thickness = 0.109375 (.033m)

<u>SAIL FOREBODY EQUATION</u>

For 3.032986 Ft $\leq$ x $\leq$ 3.358507 Ft

$$z_1 = z_{max}\left[2.094759(A) + .2071781(B) + (C)\right]^{1/N}$$

$$A = 2D\ (D-1)^4$$

$$B = 1/3\ (D^2)\ (D-1)^3$$

$$C = 1-(D-1)^4\ (4D+1)$$

$$D = 3.072000\ (x-3.032986)$$

Bow shape Changes with N

Figure 1 Parameters and Equations for the Model Shape of the DARPA 2 Sail

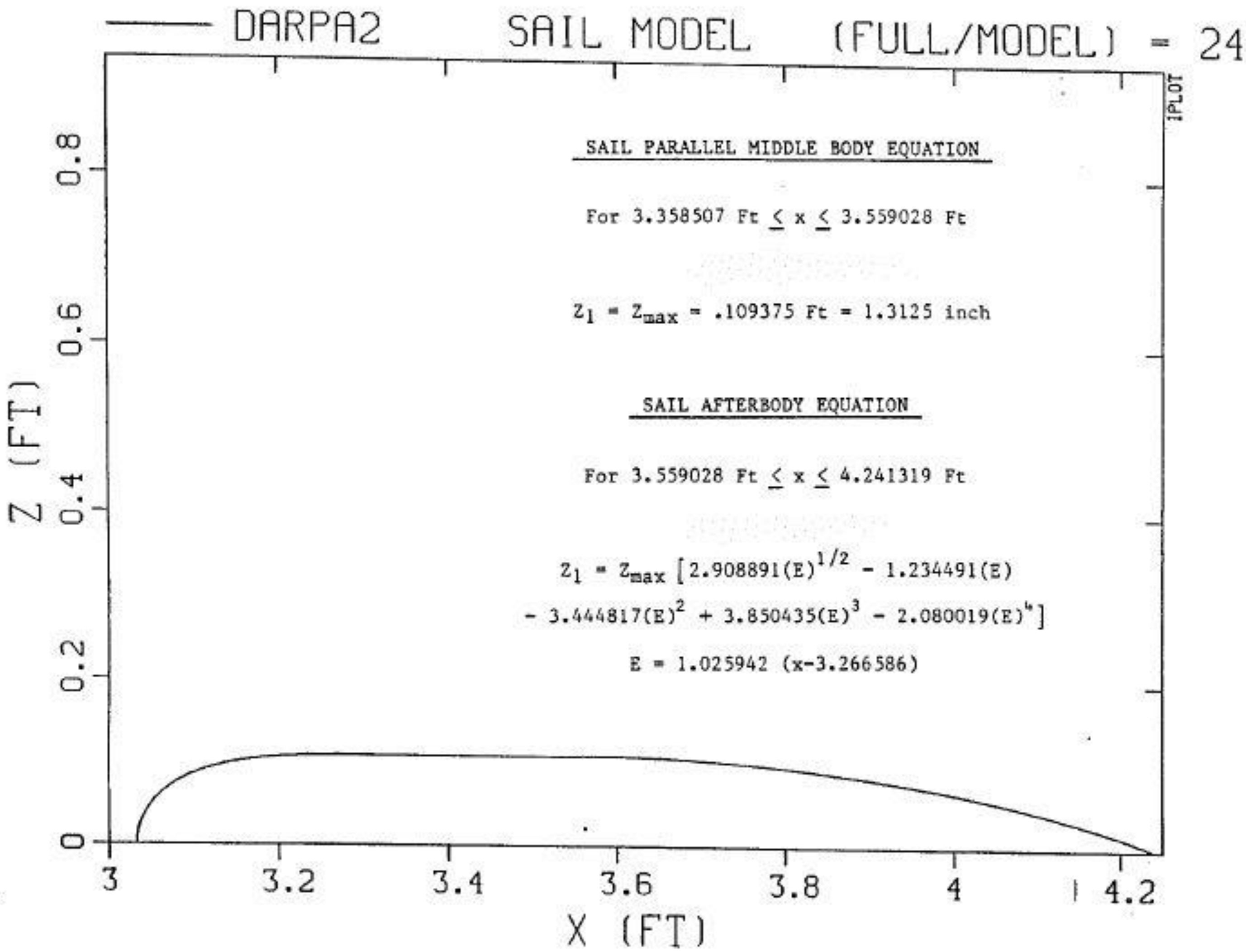

Figure 1 (cont.) Equations for the Model Shape of the DARPA 2 Sail

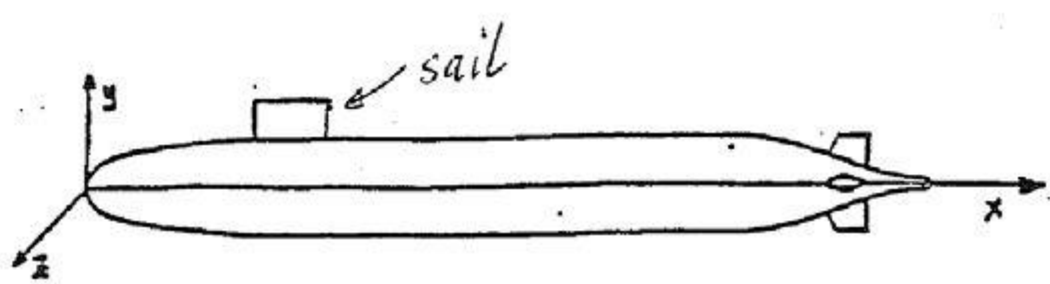

Figure 2 DARPA 2 Submarine and Sail

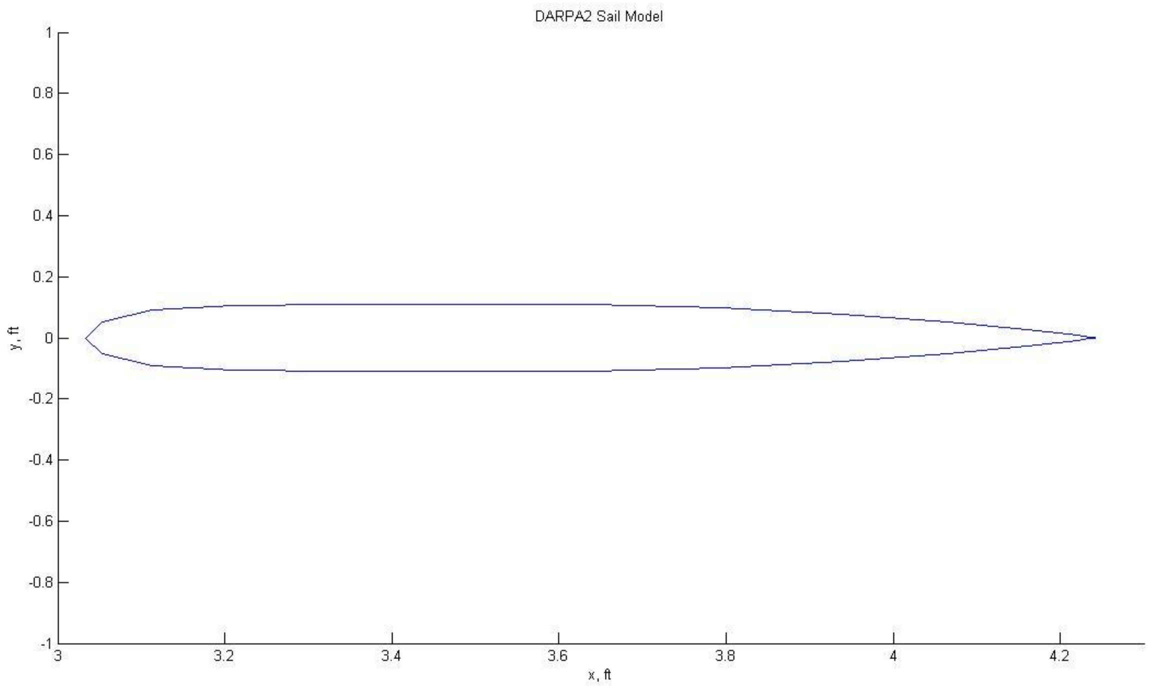

Figure 3 Computed Sail Model with 24 Panels and N = 2

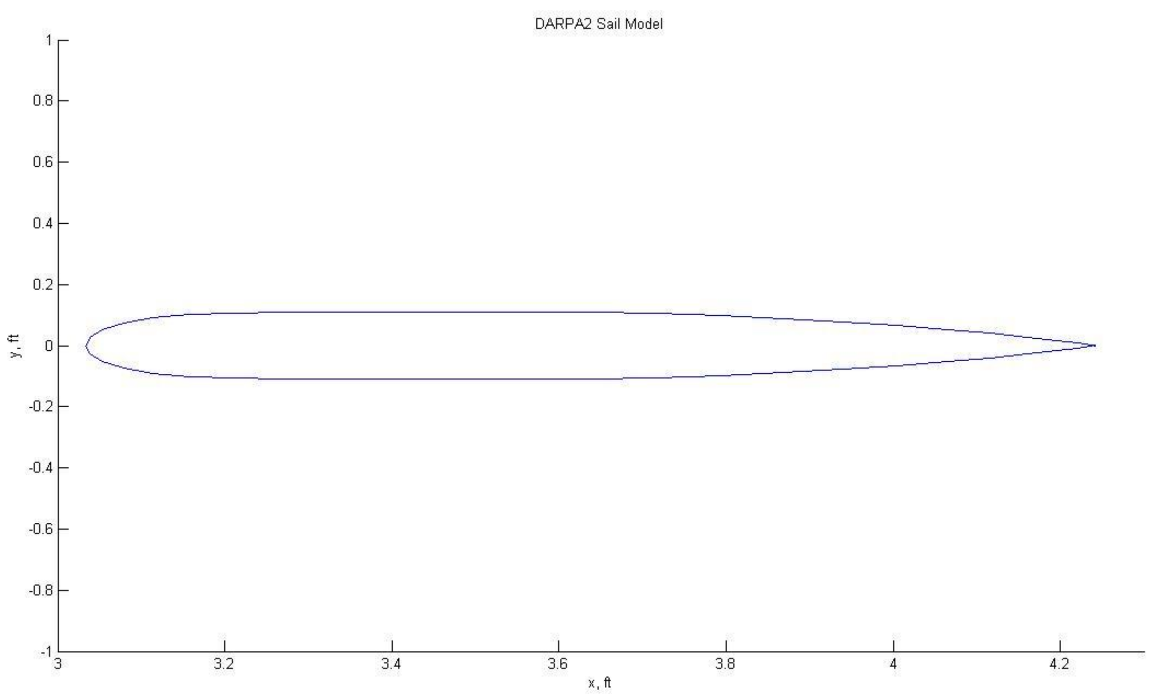

Figure 4 Computed Sail Model with 48 Panels and N = 2

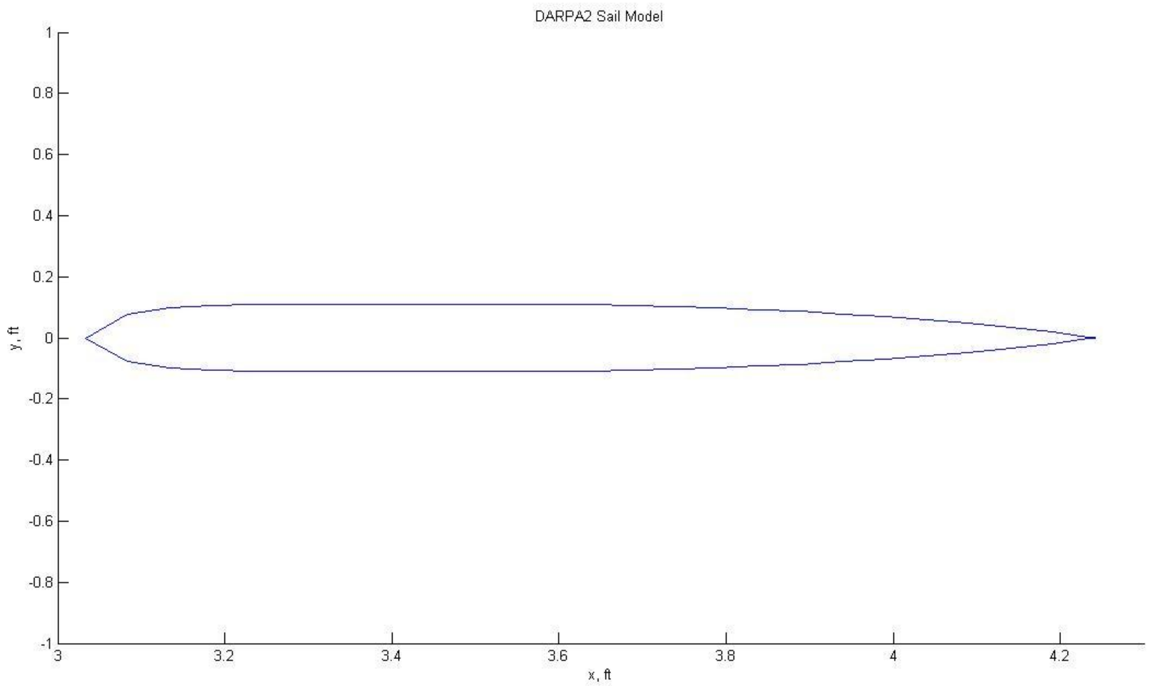

Figure 5 Computed Sail Model (Panel Configuration 2) with 48 Panels and N = 2

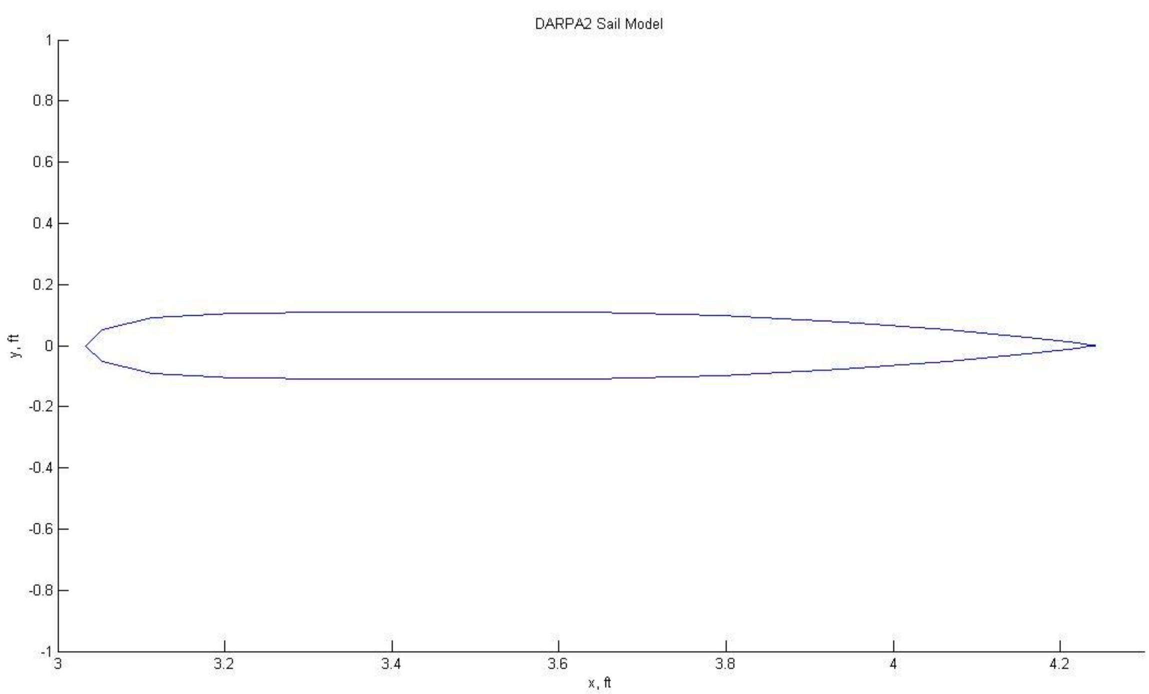

Figure 6 Computed Sail Model (Panel Configuration 3) with 48 Panels and N = 2

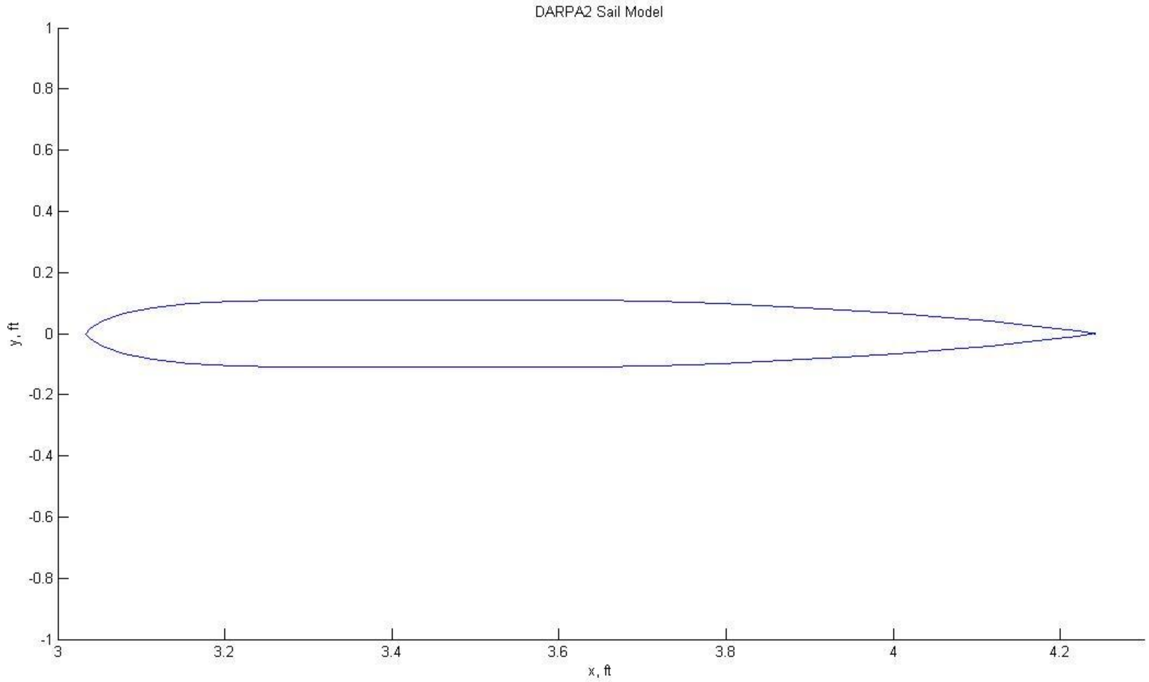

Figure 7 Computed Sail Model with 48 Panels and N = 3/2

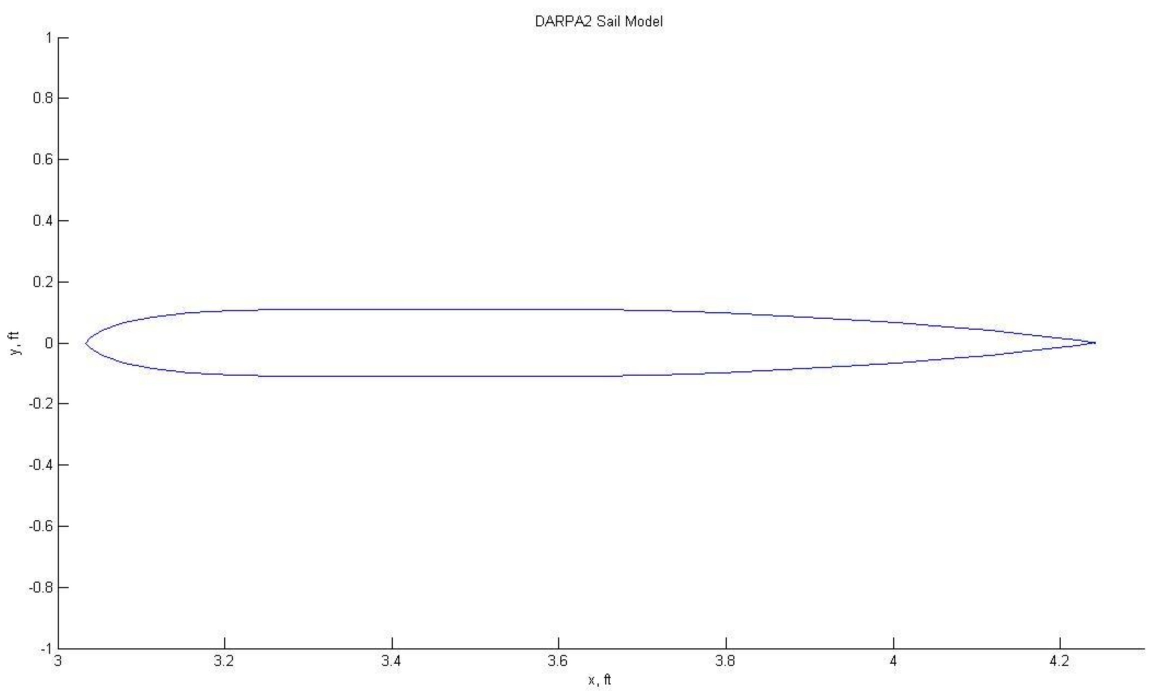

Figure 8 Computed Sail Model with 48 Panels and N = 3

Appendix B: Pressure, Force, and Moment Coefficient Plots

$\boldsymbol{m = 24, N = 2}$:

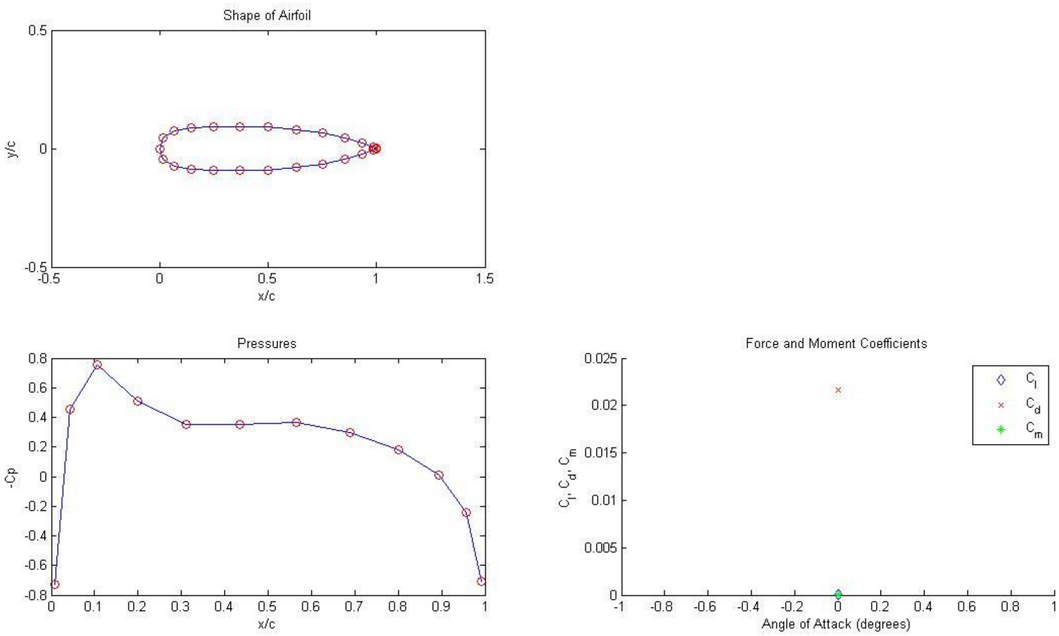

Figure 1 Pressures and Force and Moment Coefficients for 24 Panels, N = 2, $\alpha = 0°$

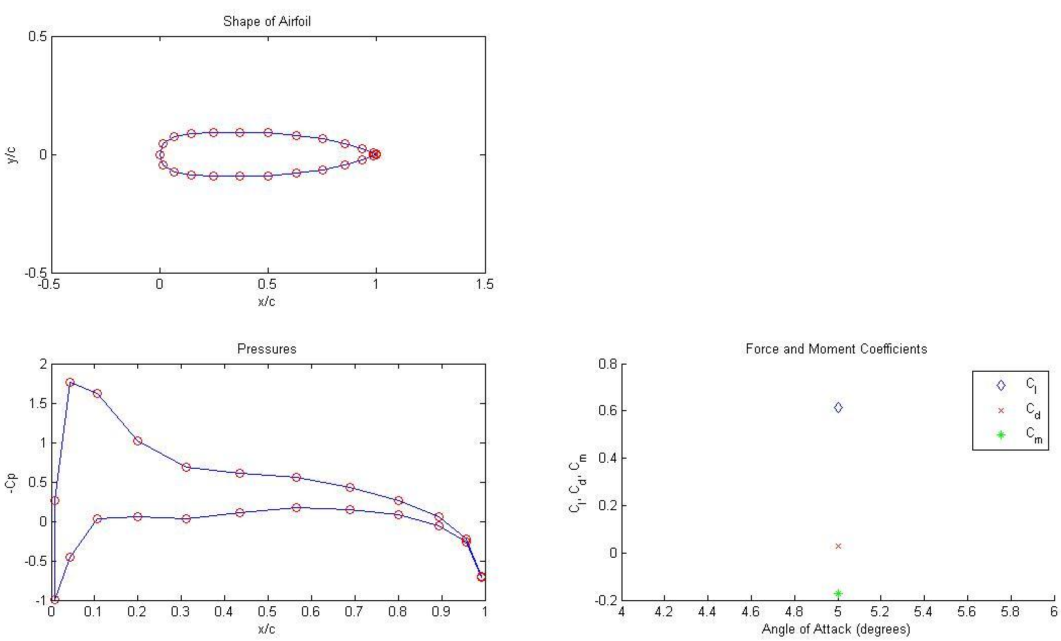

Figure 2 Pressures and Force and Moment Coefficients for 24 Panels, N = 2, $\alpha = 5°$

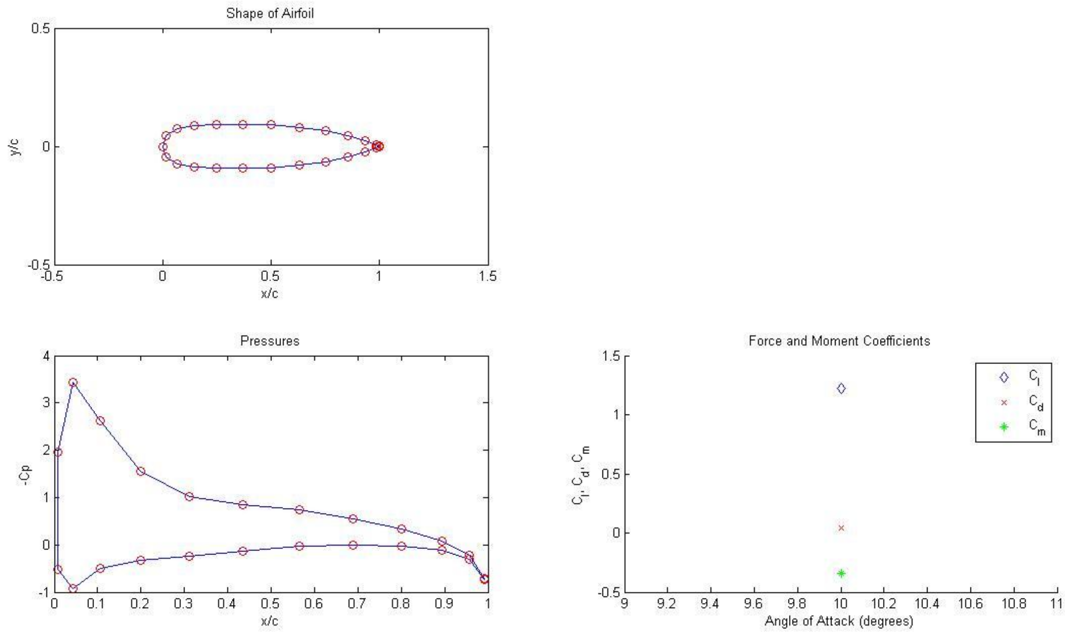

Figure 3 Pressures and Force and Moment Coefficients for 24 Panels, N = 2, $\alpha = 10°$

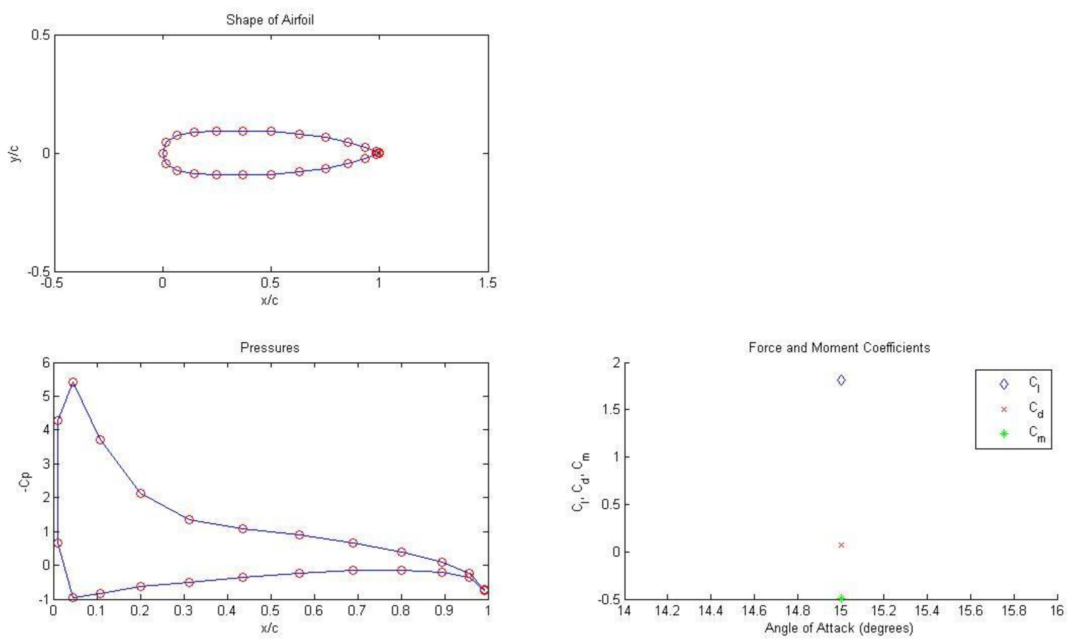

Figure 4 Pressures and Force and Moment Coefficients for 24 Panels, N = 2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 2}$:

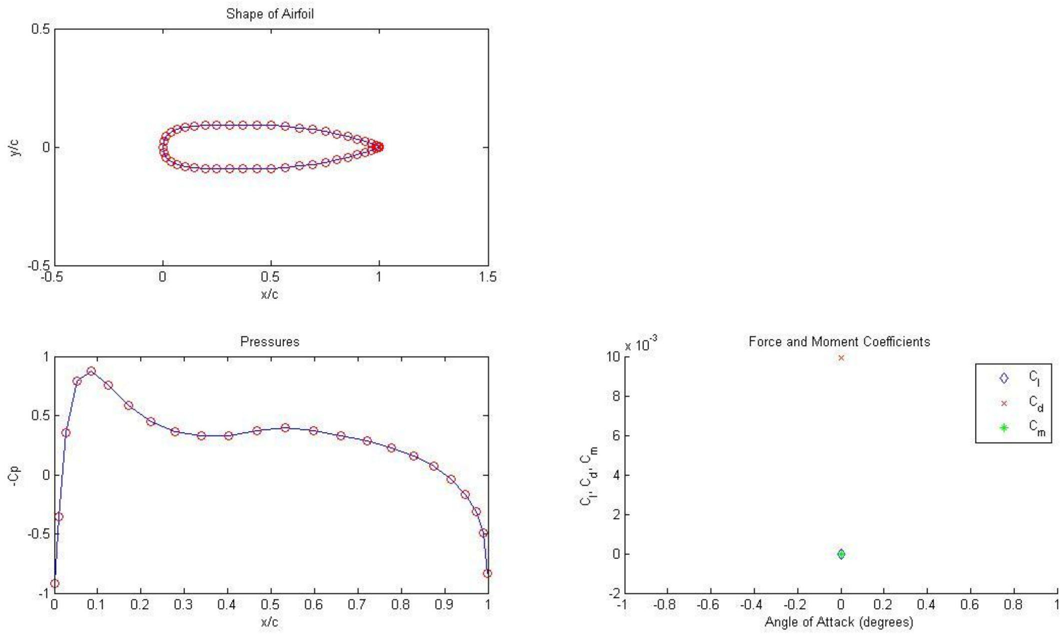

Figure 5 Pressures and Force and Moment Coefficients for 48 Panels, N = 2, $\alpha = 0°$

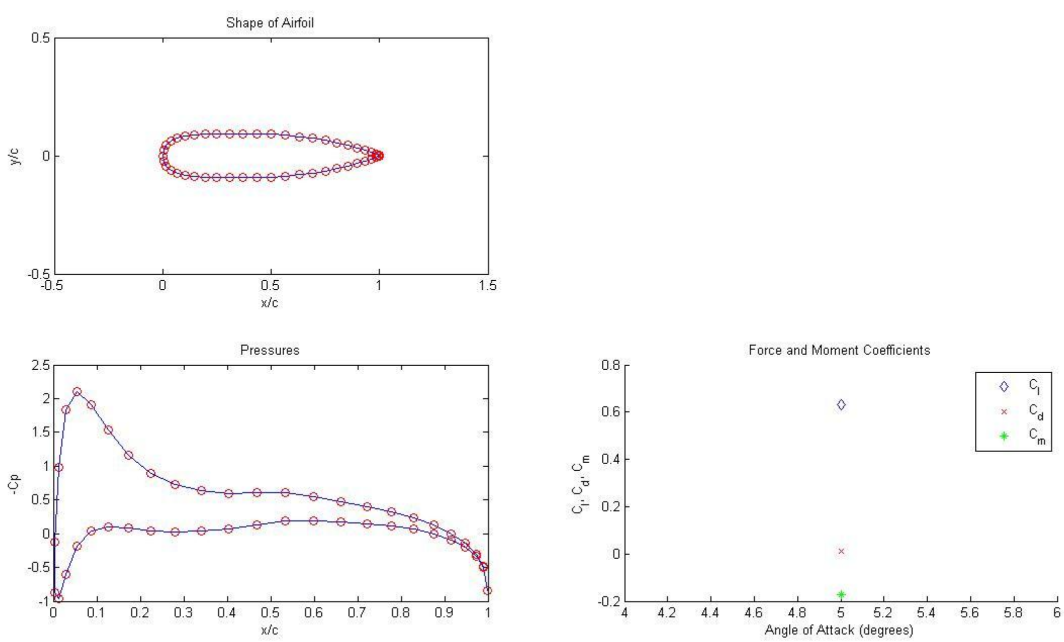

Figure 6 Pressures and Force and Moment Coefficients for 48 Panels, N = 2, $\alpha = 5°$

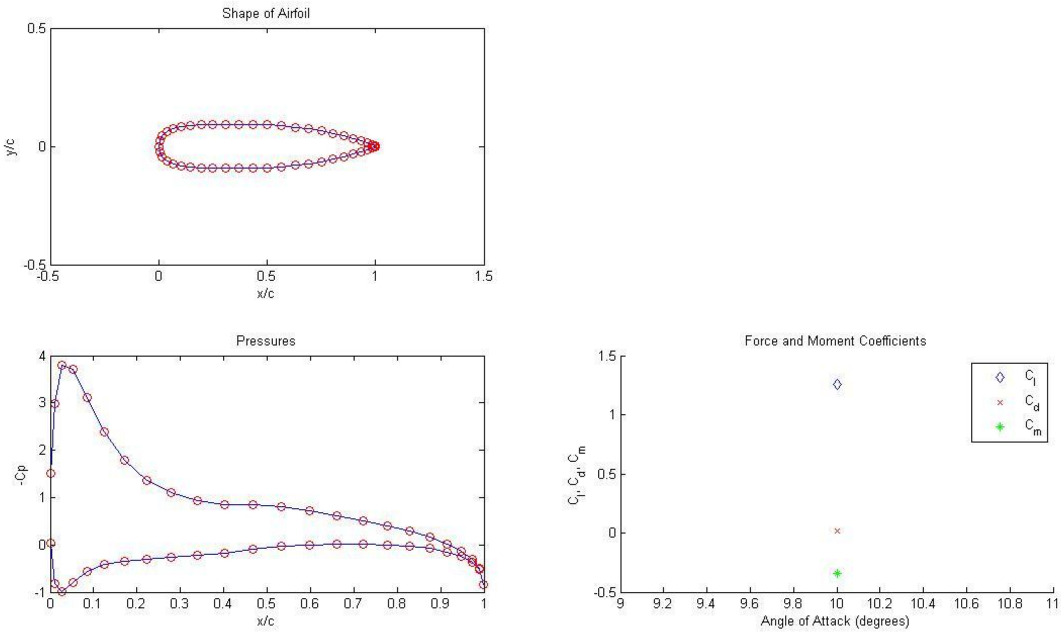

Figure 7 Pressures and Force and Moment Coefficients for 48 Panels, N = 2, $\alpha = 10°$

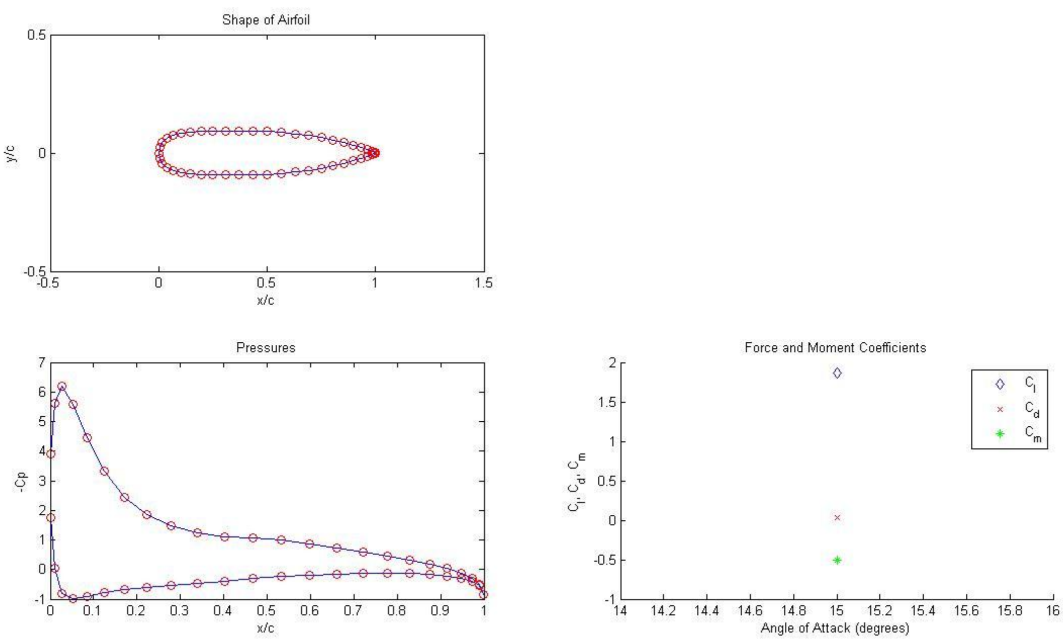

Figure 8 Pressures and Force and Moment Coefficients for 48 Panels, N = 2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 2}$ (panel configuration 2):

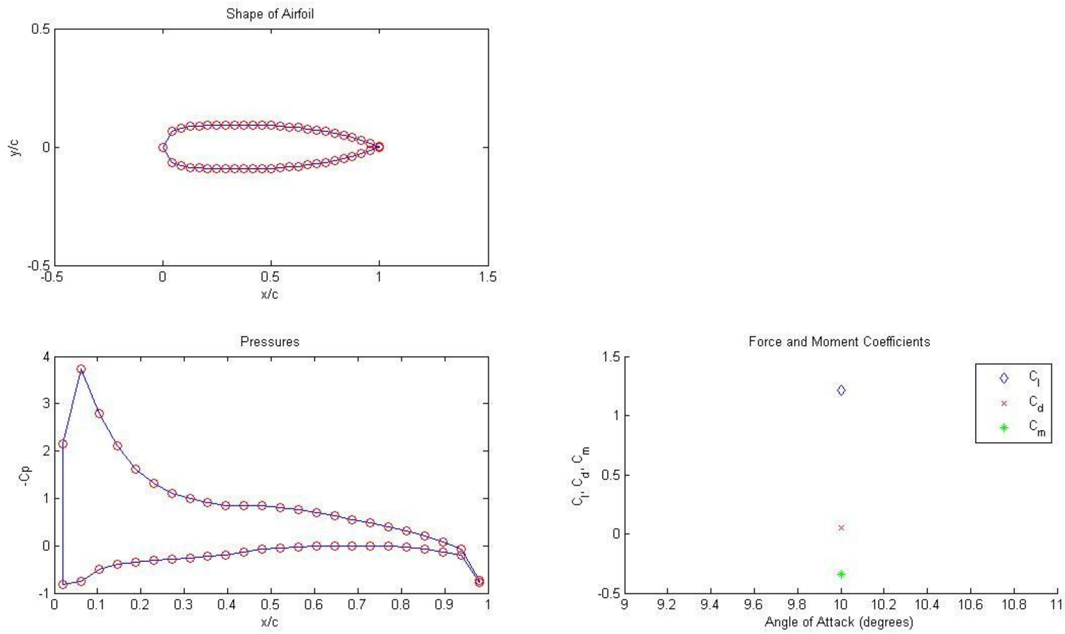

Figure 9 Pressures and Force and Moment Coefficients (Panel Configuration 2) for 48 Panels, N = 2, $\alpha = 10°$

$\boldsymbol{m = 48, N = 2}$ (panel configuration 3):

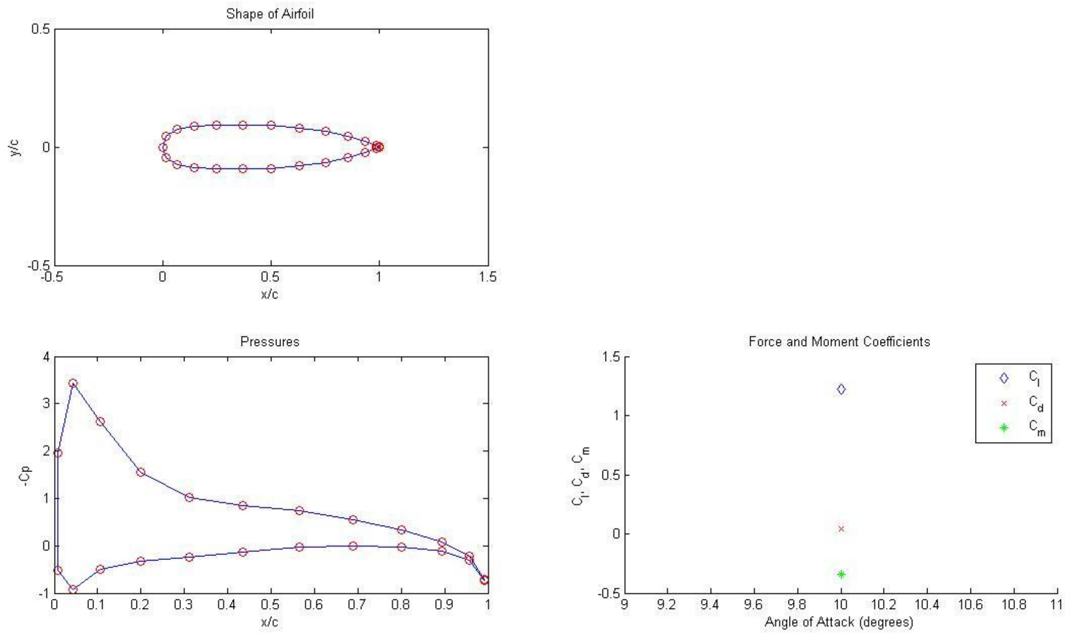

Figure 10 Pressures and Force and Moment Coefficients (Panel Configuration 3) for 48 Panels, N = 2, $\alpha = 10°$

$m = 48, N = 3/2$:

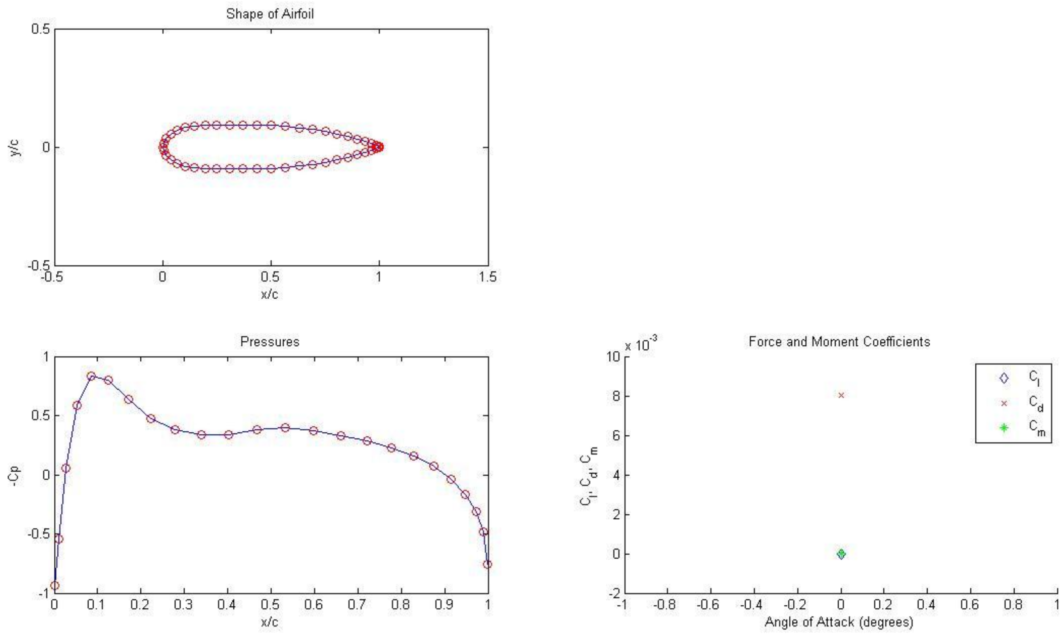

Figure 11 Pressures and Force and Moment Coefficients for 48 Panels, N = 3/2, $\alpha = 0°$

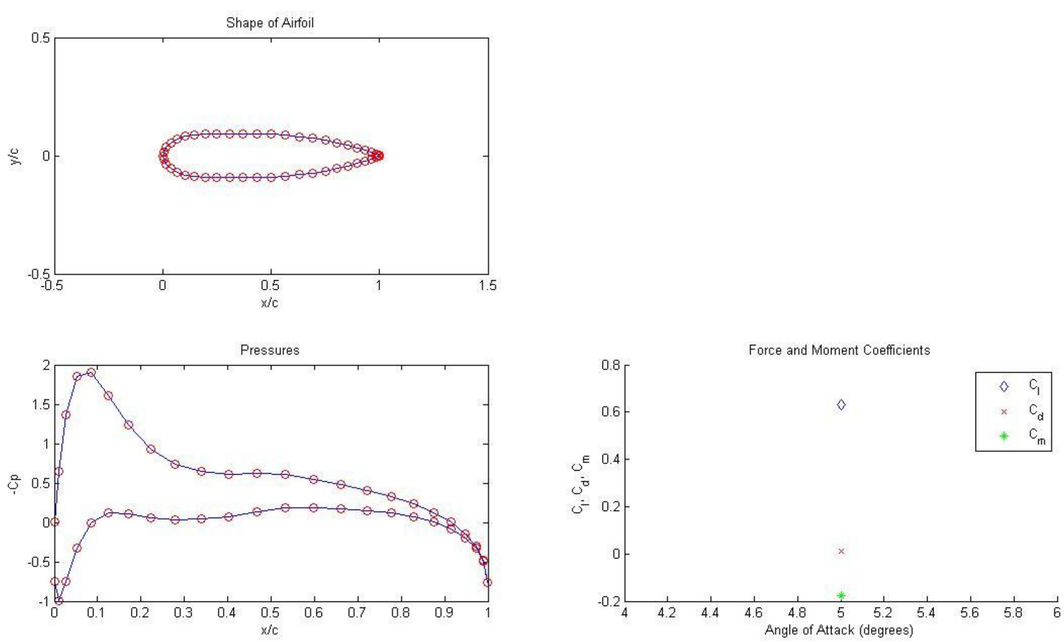

Figure 12 Pressures and Force and Moment Coefficients for 48 Panels, N = 3/2, $\alpha = 5°$

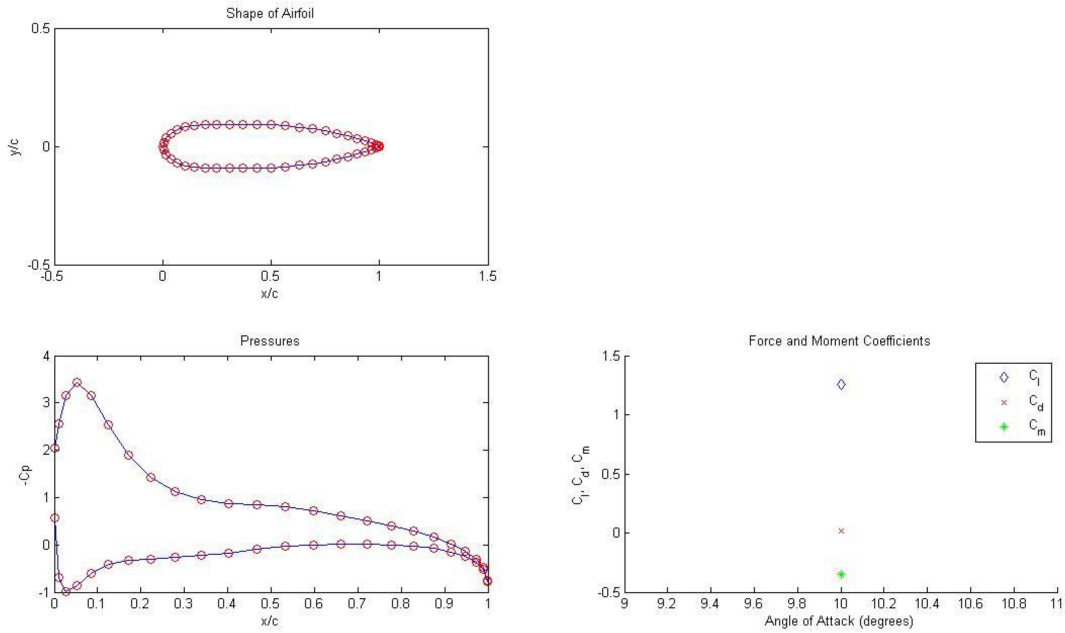

Figure 13 Pressures and Force and Moment Coefficients for 48 Panels, N = 3/2, $\alpha = 10°$

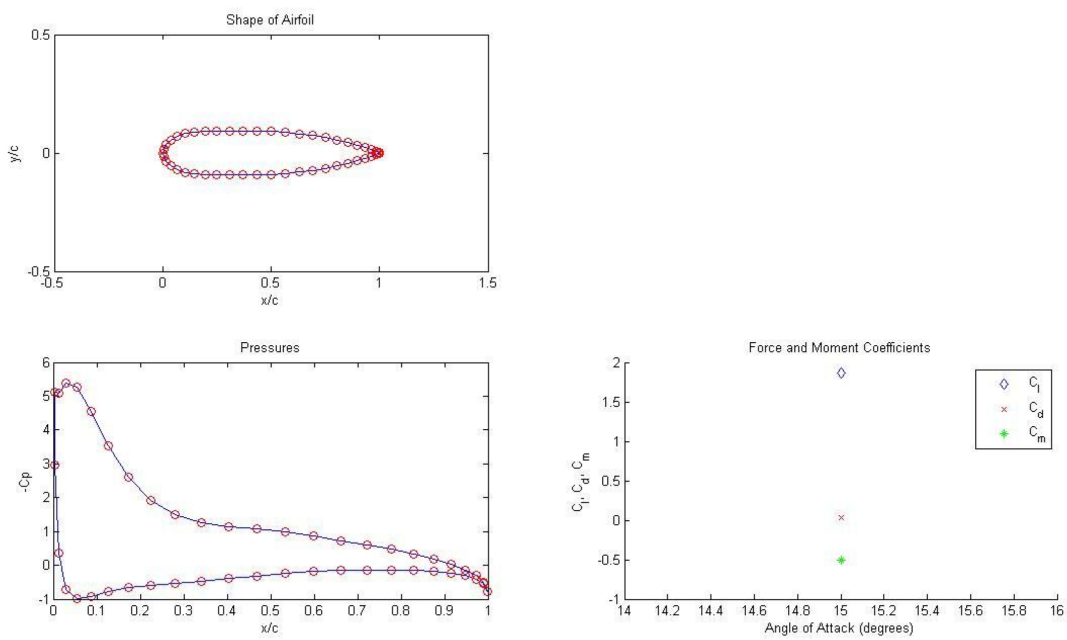

Figure 14 Pressures and Force and Moment Coefficients for 48 Panels, N = 3/2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 3}$:

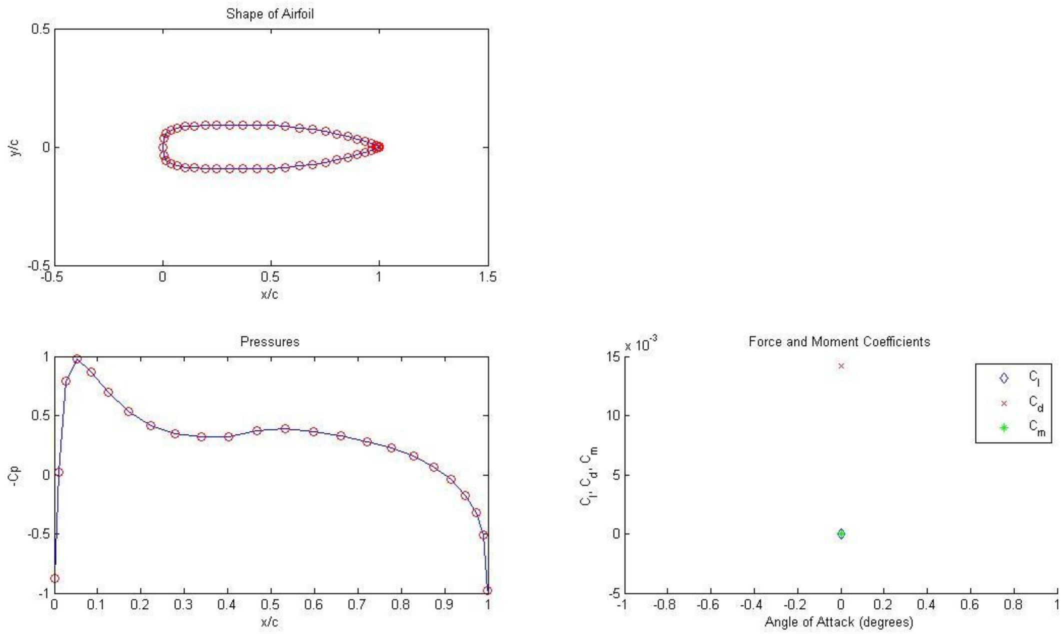

Figure 15 Pressures and Force and Moment Coefficients for 48 Panels, N = 3, $\alpha = 0°$

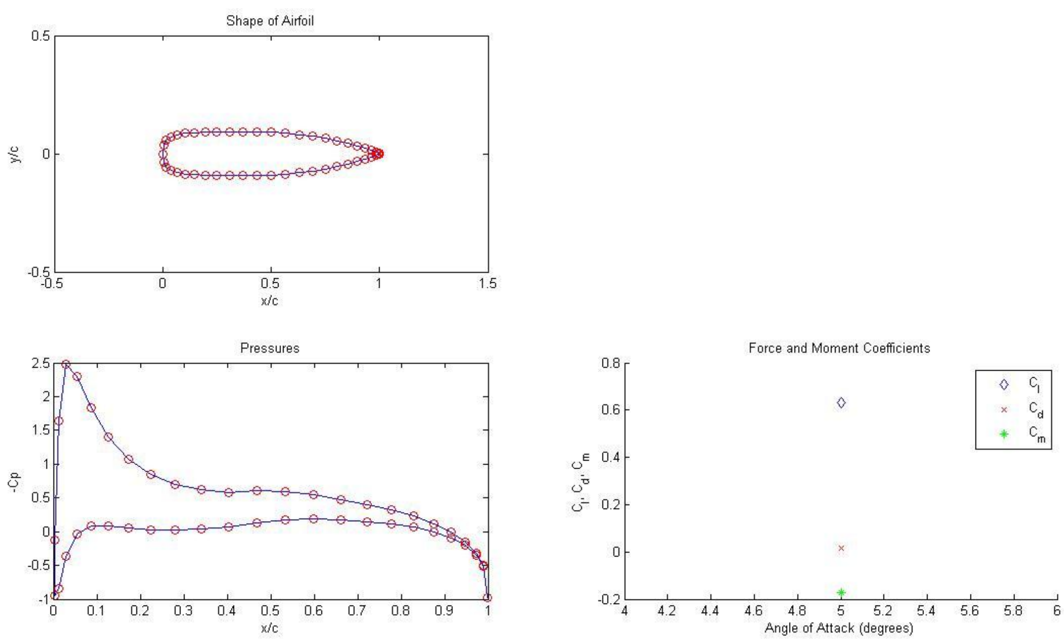

Figure 16 Pressures and Force and Moment Coefficients for 48 Panels, N = 3, $\alpha = 5°$

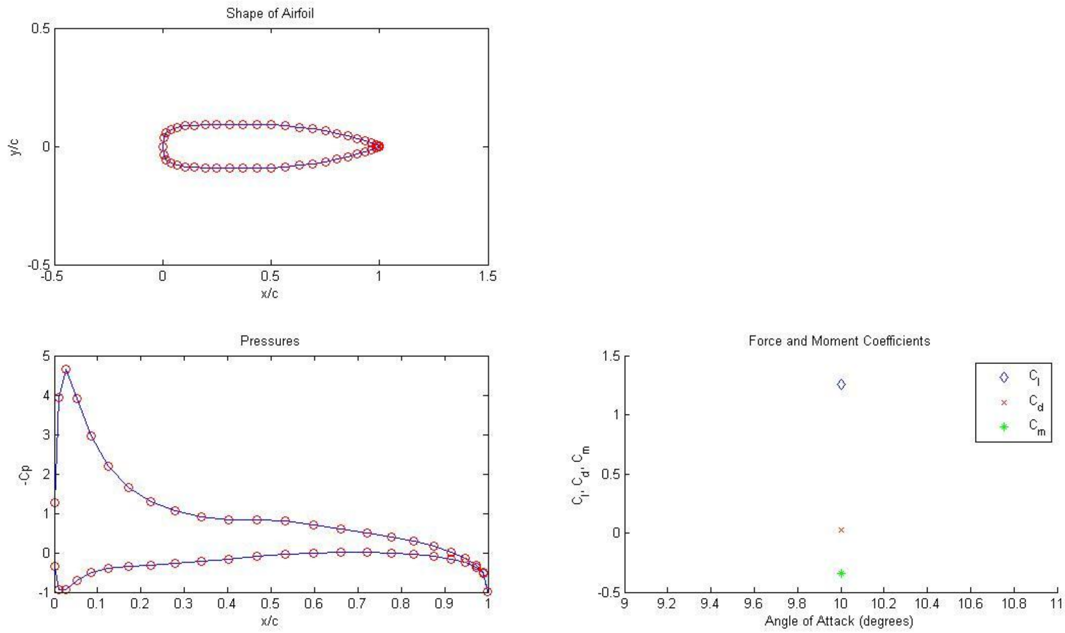

Figure 17 Pressures and Force and Moment Coefficients for 48 Panels, N = 3, $\alpha = 10°$

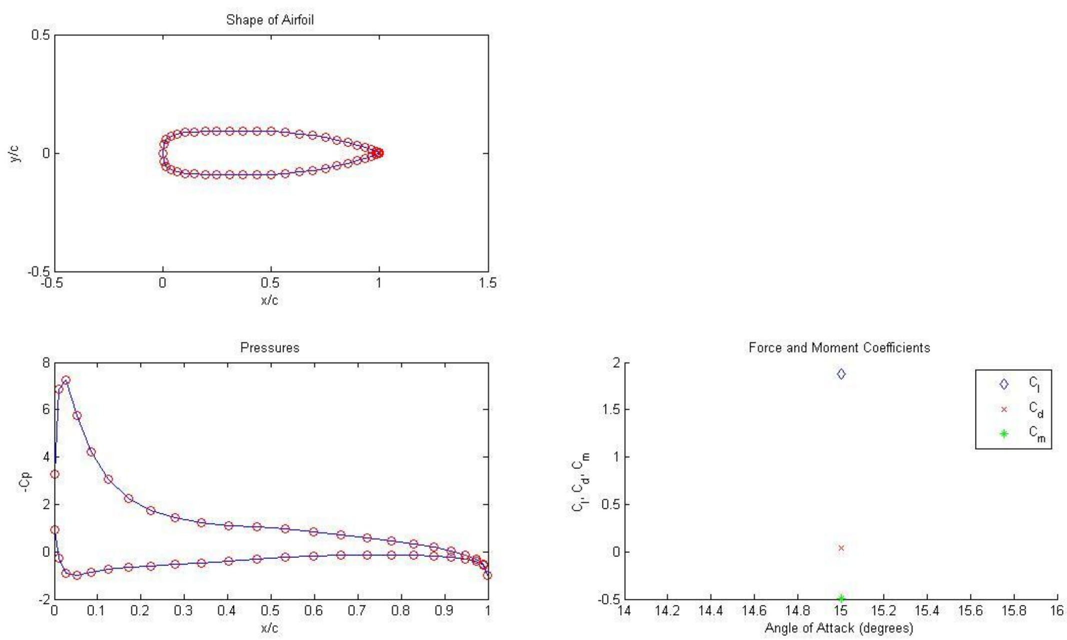

Figure 18 Pressures and Force and Moment Coefficients for 48 Panels, N = 3, $\alpha = 15°$

Appendix C: Tabular Data Around Hydrofoil (γ, v, C_p)

$\boldsymbol{m = 24, N = 2}$:

i	x	y	theta	s	gamma	v	Cp
---	---------	----------	----------	----------	----------	----------	----------
1	0.9915	-0.0032	-2.7830	0.0182	0.1054	-0.5424	0.7058
2	0.9580	-0.0147	-2.8186	0.0527	-0.1113	-0.8713	0.2409
3	0.8933	-0.0338	-2.8788	0.0823	-0.1477	-1.0041	-0.0082
4	0.8018	-0.0547	-2.9462	0.1056	-0.1670	-1.0878	-0.1834
5	0.6897	-0.0730	-3.0090	0.1217	-0.1780	-1.1395	-0.2985
6	0.5647	-0.0855	-3.0724	0.1297	-0.1848	-1.1688	-0.3661
7	0.4353	-0.0903	-3.1377	0.1294	-0.1880	-1.1630	-0.3526
8	0.3103	-0.0905	3.1416	0.1206	-0.1816	-1.1623	-0.3510
9	0.1982	-0.0896	3.1236	0.1036	-0.1866	-1.2296	-0.5119
10	0.1067	-0.0820	2.9768	0.0806	-0.2067	-1.3238	-0.7524
11	0.0420	-0.0598	2.5835	0.0589	-0.2238	-1.2051	-0.4523
12	0.0085	-0.0221	1.9383	0.0474	-0.1693	-0.5229	0.7266
13	0.0085	0.0221	1.2033	0.0474	-0.0000	0.5229	0.7266
14	0.0420	0.0598	0.5581	0.0589	0.1693	1.2051	-0.4523
15	0.1067	0.0820	0.1648	0.0806	0.2238	1.3238	-0.7524
16	0.1982	0.0896	0.0180	0.1036	0.2067	1.2296	-0.5119
17	0.3103	0.0905	0.0000	0.1206	0.1866	1.1623	-0.3510
18	0.4353	0.0903	-0.0039	0.1294	0.1816	1.1630	-0.3526
19	0.5647	0.0855	-0.0692	0.1297	0.1880	1.1688	-0.3661
20	0.6897	0.0730	-0.1326	0.1217	0.1848	1.1395	-0.2985
21	0.8018	0.0547	-0.1954	0.1056	0.1780	1.0878	-0.1834
22	0.8933	0.0338	-0.2628	0.0823	0.1670	1.0041	-0.0082
23	0.9580	0.0147	-0.3230	0.0527	0.1477	0.8713	0.2409
24	0.9915	0.0032	-0.3586	0.0182	0.1113	0.5424	0.7058
25					-0.1054		

C_l = 1.8041e-016
C_d = 0.021607
C_m = 1.2056e-016

Table 1 Resulting Data Around Airfoil for 24 Panels, N = 2, $\alpha = 0°$

i	x	y	theta	s	gamma	v	Cp
---	---------	----------	----------	----------	----------	----------	----------
1	0.9915	-0.0032	-2.7830	0.0182	0.1050	-0.5367	0.7120
2	0.9580	-0.0147	-2.8186	0.0527	-0.1098	-0.8549	0.2691
3	0.8933	-0.0338	-2.8788	0.0823	-0.1441	-0.9738	0.0517
4	0.8018	-0.0547	-2.9462	0.1056	-0.1610	-1.0410	-0.0837
5	0.6897	-0.0730	-3.0090	0.1217	-0.1691	-1.0737	-0.1528
6	0.5647	-0.0855	-3.0724	0.1297	-0.1726	-1.0809	-0.1683
7	0.4353	-0.0903	-3.1377	0.1294	-0.1720	-1.0508	-0.1042
8	0.3103	-0.0905	3.1416	0.1206	-0.1618	-1.0185	-0.0373
9	0.1982	-0.0896	3.1236	0.1036	-0.1605	-1.0293	-0.0595
10	0.1067	-0.0820	2.9768	0.0806	-0.1680	-1.0186	-0.0376
11	0.0420	-0.0598	2.5835	0.0589	-0.1613	-0.7393	0.4535
12	0.0085	-0.0221	1.9383	0.0474	-0.0755	0.0837	0.9930
13	0.0085	0.0221	1.2033	0.0474	0.1089	1.1255	-0.2668
14	0.0420	0.0598	0.5581	0.0589	0.2619	1.6618	-1.7617
15	0.1067	0.0820	0.1648	0.0806	0.2846	1.6189	-1.6208
16	0.1982	0.0896	0.0180	0.1036	0.2438	1.4205	-1.0179
17	0.3103	0.0905	0.0000	0.1206	0.2113	1.2974	-0.6832
18	0.4353	0.0903	-0.0039	0.1294	0.2001	1.2663	-0.6036
19	0.5647	0.0855	-0.0692	0.1297	0.2026	1.2478	-0.5570
20	0.6897	0.0730	-0.1326	0.1217	0.1956	1.1967	-0.4320
21	0.8018	0.0547	-0.1954	0.1056	0.1855	1.1263	-0.2686
22	0.8933	0.0338	-0.2628	0.0823	0.1718	1.0267	-0.0542
23	0.9580	0.0147	-0.3230	0.0527	0.1501	0.8810	0.2238
24	0.9915	0.0032	-0.3586	0.0182	0.1120	0.5439	0.7041
25					-0.1050		

C_l = 0.61355
C_d = 0.027245
C_m = -0.17332

Table 2 Resulting Data Around Airfoil for 24 Panels, N = 2, $\alpha = 5°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9915	-0.0032	-2.7830	0.0182	0.1038	-0.5269	0.7223
2	0.9580	-0.0147	-2.8186	0.0527	-0.1074	-0.8321	0.3076
3	0.8933	-0.0338	-2.8788	0.0823	-0.1394	-0.9361	0.1237
4	0.8018	-0.0547	-2.9462	0.1056	-0.1537	-0.9863	0.0272
5	0.6897	-0.0730	-3.0090	0.1217	-0.1589	-0.9997	0.0006
6	0.5647	-0.0855	-3.0724	0.1297	-0.1591	-0.9848	0.0302
7	0.4353	-0.0903	-3.1377	0.1294	-0.1546	-0.9306	0.1340
8	0.3103	-0.0905	3.1416	0.1206	-0.1408	-0.8668	0.2486
9	0.1982	-0.0896	3.1236	0.1036	-0.1332	-0.8212	0.3256
10	0.1067	-0.0820	2.9768	0.0806	-0.1281	-0.7057	0.5020
11	0.0420	-0.0598	2.5835	0.0589	-0.0976	-0.2678	0.9283
12	0.0085	-0.0221	1.9383	0.0474	0.0190	0.6896	0.5244
13	0.0085	0.0221	1.2033	0.0474	0.2170	1.7196	-1.9569
14	0.0420	0.0598	0.5581	0.0589	0.3525	2.1059	-3.4347
15	0.1067	0.0820	0.1648	0.0806	0.3432	1.9016	-2.6163
16	0.1982	0.0896	0.0180	0.1036	0.2790	1.6006	-1.5620
17	0.3103	0.0905	0.0000	0.1206	0.2343	1.4225	-1.0236
18	0.4353	0.0903	-0.0039	0.1294	0.2170	1.3600	-0.8497
19	0.5647	0.0855	-0.0692	0.1297	0.2156	1.3173	-0.7353
20	0.6897	0.0730	-0.1326	0.1217	0.2049	1.2447	-0.5493
21	0.8018	0.0547	-0.1954	0.1056	0.1917	1.1563	-0.3370
22	0.8933	0.0338	-0.2628	0.0823	0.1753	1.0415	-0.0848
23	0.9580	0.0147	-0.3230	0.0527	0.1515	0.8840	0.2185
24	0.9915	0.0032	-0.3586	0.0182	0.1118	0.5413	0.7069
25					-0.1038		

C_l = 1.2195
C_d = 0.043727
C_m = -0.34137

Table 3 Resulting Data Around Airfoil for 24 Panels, N = 2, $\alpha = 10°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9915	-0.0032	-2.7830	0.0182	0.1018	-0.5132	0.7367
2	0.9580	-0.0147	-2.8186	0.0527	-0.1042	-0.8029	0.3554
3	0.8933	-0.0338	-2.8788	0.0823	-0.1337	-0.8913	0.2055
4	0.8018	-0.0547	-2.9462	0.1056	-0.1452	-0.9241	0.1461
5	0.6897	-0.0730	-3.0090	0.1217	-0.1475	-0.9181	0.1571
6	0.5647	-0.0855	-3.0724	0.1297	-0.1444	-0.8811	0.2236
7	0.4353	-0.0903	-3.1377	0.1294	-0.1361	-0.8033	0.3547
8	0.3103	-0.0905	3.1416	0.1206	-0.1187	-0.7086	0.4979
9	0.1982	-0.0896	3.1236	0.1036	-0.1048	-0.6068	0.6318
10	0.1067	-0.0820	2.9768	0.0806	-0.0871	-0.3874	0.8499
11	0.0420	-0.0598	2.5835	0.0589	-0.0331	0.2058	0.9577
12	0.0085	-0.0221	1.9383	0.0474	0.1133	1.2903	-0.6650
13	0.0085	0.0221	1.2033	0.0474	0.3235	2.3005	-4.2924
14	0.0420	0.0598	0.5581	0.0589	0.4404	2.5339	-5.4206
15	0.1067	0.0820	0.1648	0.0806	0.3993	2.1699	-3.7086
16	0.1982	0.0896	0.0180	0.1036	0.3122	1.7686	-2.1278
17	0.3103	0.0905	0.0000	0.1206	0.2556	1.5369	-1.3619
18	0.4353	0.0903	-0.0039	0.1294	0.2322	1.4434	-1.0834
19	0.5647	0.0855	-0.0692	0.1297	0.2271	1.3768	-0.8956
20	0.6897	0.0730	-0.1326	0.1217	0.2126	1.2833	-0.6467
21	0.8018	0.0547	-0.1954	0.1056	0.1963	1.1774	-0.3864
22	0.8933	0.0338	-0.2628	0.0823	0.1775	1.0484	-0.0992
23	0.9580	0.0147	-0.3230	0.0527	0.1516	0.8803	0.2251
24	0.9915	0.0032	-0.3586	0.0182	0.1108	0.5346	0.7142
25					-0.1018		

C_l = 1.8104
C_d = 0.069783
C_m = -0.49905

Table 4 Resulting Data Around Airfoil for 24 Panels, N = 2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 2}$:

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.1402	-0.4041	0.8367
2	0.9893	-0.0040	-2.7854	0.0136	-0.0872	-0.7142	0.4899
3	0.9725	-0.0101	-2.8038	0.0223	-0.1191	-0.8276	0.3150
4	0.9475	-0.0184	-2.8295	0.0304	-0.1372	-0.9129	0.1667
5	0.9148	-0.0283	-2.8605	0.0378	-0.1503	-0.9806	0.0385
6	0.8751	-0.0390	-2.8944	0.0445	-0.1605	-1.0343	-0.0697
7	0.8290	-0.0498	-2.9288	0.0503	-0.1683	-1.0761	-0.1581
8	0.7772	-0.0600	-2.9621	0.0553	-0.1743	-1.1083	-0.2284
9	0.7207	-0.0693	-2.9936	0.0593	-0.1788	-1.1334	-0.2846
10	0.6604	-0.0774	-3.0238	0.0624	-0.1824	-1.1537	-0.3310
11	0.5973	-0.0838	-3.0547	0.0644	-0.1854	-1.1704	-0.3698
12	0.5326	-0.0883	-3.0898	0.0654	-0.1879	-1.1820	-0.3972
13	0.4674	-0.0903	-3.1339	0.0653	-0.1895	-1.1720	-0.3735
14	0.4027	-0.0905	3.1416	0.0641	-0.1841	-1.1537	-0.3311
15	0.3396	-0.0905	3.1416	0.0619	-0.1827	-1.1531	-0.3296
16	0.2793	-0.0905	3.1416	0.0587	-0.1838	-1.1678	-0.3638
17	0.2228	-0.0904	3.1370	0.0544	-0.1874	-1.2027	-0.4465
18	0.1710	-0.0895	3.1088	0.0492	-0.1956	-1.2600	-0.5876
19	0.1249	-0.0864	3.0370	0.0434	-0.2073	-1.3261	-0.7585
20	0.0852	-0.0798	2.9069	0.0374	-0.2189	-1.3691	-0.8743
21	0.0525	-0.0687	2.7082	0.0319	-0.2235	-1.3366	-0.7866
22	0.0275	-0.0531	2.4390	0.0275	-0.2104	-1.1627	-0.3518
23	0.0107	-0.0336	2.1125	0.0247	-0.1682	-0.8042	0.3532
24	0.0021	-0.0115	1.7543	0.0234	-0.0942	-0.2883	0.9169
25	0.0021	0.0115	1.3873	0.0234	-0.0000	0.2883	0.9169
26	0.0107	0.0336	1.0291	0.0247	0.0942	0.8042	0.3532
27	0.0275	0.0531	0.7026	0.0275	0.1682	1.1627	-0.3518
28	0.0525	0.0687	0.4334	0.0319	0.2104	1.3366	-0.7866
29	0.0852	0.0798	0.2347	0.0374	0.2235	1.3691	-0.8743
30	0.1249	0.0864	0.1046	0.0434	0.2189	1.3261	-0.7585
31	0.1710	0.0895	0.0328	0.0492	0.2073	1.2600	-0.5876
32	0.2228	0.0904	0.0046	0.0544	0.1956	1.2027	-0.4465
33	0.2793	0.0905	0.0000	0.0587	0.1874	1.1678	-0.3638
34	0.3396	0.0905	0.0000	0.0619	0.1838	1.1531	-0.3296
35	0.4027	0.0905	0.0000	0.0641	0.1827	1.1537	-0.3311
36	0.4674	0.0903	-0.0077	0.0653	0.1841	1.1720	-0.3735
37	0.5326	0.0883	-0.0518	0.0654	0.1895	1.1820	-0.3972
38	0.5973	0.0838	-0.0869	0.0644	0.1879	1.1704	-0.3698
39	0.6604	0.0774	-0.1178	0.0624	0.1854	1.1537	-0.3310
40	0.7207	0.0693	-0.1480	0.0593	0.1824	1.1334	-0.2846
41	0.7772	0.0600	-0.1795	0.0553	0.1788	1.1083	-0.2284
42	0.8290	0.0498	-0.2128	0.0503	0.1743	1.0761	-0.1581
43	0.8751	0.0390	-0.2472	0.0445	0.1683	1.0343	-0.0697
44	0.9148	0.0283	-0.2811	0.0378	0.1605	0.9806	0.0385
45	0.9475	0.0184	-0.3121	0.0304	0.1503	0.9129	0.1667
46	0.9725	0.0101	-0.3378	0.0223	0.1372	0.8276	0.3150
47	0.9893	0.0040	-0.3562	0.0136	0.1191	0.7142	0.4899
48	0.9979	0.0008	-0.3658	0.0046	0.0872	0.4041	0.8367
49					-0.1402		

C_l = -2.4286e-017
C_d = 0.0099225
C_m = -3.4694e-018

Table 5 Resulting Data Around Airfoil for 48 Panels, N = 2, $\alpha = 0°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.1397	-0.4012	0.8390
2	0.9893	-0.0040	-2.7854	0.0136	-0.0865	-0.7067	0.5006
3	0.9725	-0.0101	-2.8038	0.0223	-0.1176	-0.8147	0.3362
4	0.9475	-0.0184	-2.8295	0.0304	-0.1347	-0.8937	0.2013
5	0.9148	-0.0283	-2.8605	0.0378	-0.1468	-0.9543	0.0893
6	0.8751	-0.0390	-2.8944	0.0445	-0.1557	-1.0002	-0.0005
7	0.8290	-0.0498	-2.9288	0.0503	-0.1623	-1.0338	-0.0687
8	0.7772	-0.0600	-2.9621	0.0553	-0.1668	-1.0571	-0.1174
9	0.7207	-0.0693	-2.9936	0.0593	-0.1699	-1.0726	-0.1505
10	0.6604	-0.0774	-3.0238	0.0624	-0.1719	-1.0826	-0.1720
11	0.5973	-0.0838	-3.0547	0.0644	-0.1732	-1.0881	-0.1840
12	0.5326	-0.0883	-3.0898	0.0654	-0.1738	-1.0875	-0.1827
13	0.4674	-0.0903	-3.1339	0.0653	-0.1734	-1.0658	-0.1359
14	0.4027	-0.0905	3.1416	0.0641	-0.1663	-1.0355	-0.0723
15	0.3396	-0.0905	3.1416	0.0619	-0.1628	-1.0195	-0.0393
16	0.2793	-0.0905	3.1416	0.0587	-0.1611	-1.0142	-0.0286
17	0.2228	-0.0904	3.1370	0.0544	-0.1612	-1.0217	-0.0439
18	0.1710	-0.0895	3.1088	0.0492	-0.1640	-1.0402	-0.0820
19	0.1249	-0.0864	3.0370	0.0434	-0.1683	-1.0520	-0.1068
20	0.0852	-0.0798	2.9069	0.0374	-0.1695	-1.0230	-0.0465
21	0.0525	-0.0687	2.7082	0.0319	-0.1605	-0.9022	0.1861
22	0.0275	-0.0531	2.4390	0.0275	-0.1315	-0.6339	0.5981
23	0.0107	-0.0336	2.1125	0.0247	-0.0738	-0.1957	0.9617
24	0.0021	-0.0115	1.7543	0.0234	0.0113	0.3644	0.8672
25	0.0021	0.0115	1.3873	0.0234	0.1091	0.9389	0.1184
26	0.0107	0.0336	1.0291	0.0247	0.1989	1.4066	-0.9786
27	0.0275	0.0531	0.7026	0.0275	0.2614	1.6826	-1.8310
28	0.0525	0.0687	0.4334	0.0319	0.2878	1.7609	-2.1008
29	0.0852	0.0798	0.2347	0.0374	0.2849	1.7047	-1.9061
30	0.1249	0.0864	0.1046	0.0434	0.2667	1.5900	-1.5281
31	0.1710	0.0895	0.0328	0.0492	0.2448	1.4703	-1.1616
32	0.2228	0.0904	0.0046	0.0544	0.2256	1.3745	-0.8893
33	0.2793	0.0905	0.0000	0.0587	0.2123	1.3125	-0.7227
34	0.3396	0.0905	0.0000	0.0619	0.2050	1.2779	-0.6331
35	0.4027	0.0905	0.0000	0.0641	0.2012	1.2632	-0.5956
36	0.4674	0.0903	-0.0077	0.0653	0.2004	1.2692	-0.6110
37	0.5326	0.0883	-0.0518	0.0654	0.2042	1.2675	-0.6066
38	0.5973	0.0838	-0.0869	0.0644	0.2006	1.2438	-0.5469
39	0.6604	0.0774	-0.1178	0.0624	0.1962	1.2160	-0.4786
40	0.7207	0.0693	-0.1480	0.0593	0.1915	1.1856	-0.4056
41	0.7772	0.0600	-0.1795	0.0553	0.1864	1.1512	-0.3252
42	0.8290	0.0498	-0.2128	0.0503	0.1804	1.1103	-0.2328
43	0.8751	0.0390	-0.2472	0.0445	0.1731	1.0604	-0.1246
44	0.9148	0.0283	-0.2811	0.0378	0.1641	0.9994	0.0013
45	0.9475	0.0184	-0.3121	0.0304	0.1528	0.9251	0.1442
46	0.9725	0.0101	-0.3378	0.0223	0.1387	0.8342	0.3041
47	0.9893	0.0040	-0.3562	0.0136	0.1198	0.7163	0.4869
48	0.9979	0.0008	-0.3658	0.0046	0.0873	0.4039	0.8368
49					-0.1397		

C_l = 0.63181
C_d = 0.012645
C_m = -0.17427

Table 6 Resulting Data Around Airfoil for 48 Panels, N = 2, $\alpha = 5°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.1381	-0.3953	0.8438
2	0.9893	-0.0040	-2.7854	0.0136	-0.0851	-0.6937	0.5188
3	0.9725	-0.0101	-2.8038	0.0223	-0.1151	-0.7956	0.3670
4	0.9475	-0.0184	-2.8295	0.0304	-0.1311	-0.8677	0.2471
5	0.9148	-0.0283	-2.8605	0.0378	-0.1420	-0.9207	0.1522
6	0.8751	-0.0390	-2.8944	0.0445	-0.1497	-0.9586	0.0811
7	0.8290	-0.0498	-2.9288	0.0503	-0.1550	-0.9835	0.0327
8	0.7772	-0.0600	-2.9621	0.0553	-0.1581	-0.9977	0.0045
9	0.7207	-0.0693	-2.9936	0.0593	-0.1597	-1.0037	-0.0073
10	0.6604	-0.0774	-3.0238	0.0624	-0.1601	-1.0033	-0.0066
11	0.5973	-0.0838	-3.0547	0.0644	-0.1597	-0.9976	0.0049
12	0.5326	-0.0883	-3.0898	0.0654	-0.1584	-0.9848	0.0302
13	0.4674	-0.0903	-3.1339	0.0653	-0.1559	-0.9515	0.0946
14	0.4027	-0.0905	3.1416	0.0641	-0.1473	-0.9094	0.1730
15	0.3396	-0.0905	3.1416	0.0619	-0.1416	-0.8781	0.2290
16	0.2793	-0.0905	3.1416	0.0587	-0.1372	-0.8528	0.2727
17	0.2228	-0.0904	3.1370	0.0544	-0.1336	-0.8330	0.3062
18	0.1710	-0.0895	3.1088	0.0492	-0.1313	-0.8124	0.3400
19	0.1249	-0.0864	3.0370	0.0434	-0.1280	-0.7700	0.4071
20	0.0852	-0.0798	2.9069	0.0374	-0.1188	-0.6691	0.5523
21	0.0525	-0.0687	2.7082	0.0319	-0.0963	-0.4608	0.7876
22	0.0275	-0.0531	2.4390	0.0275	-0.0516	-0.1004	0.9899
23	0.0107	-0.0336	2.1125	0.0247	0.0212	0.4143	0.8284
24	0.0021	-0.0115	1.7543	0.0234	0.1167	1.0144	-0.0291
25	0.0021	0.0115	1.3873	0.0234	0.2174	1.5824	-1.5039
26	0.0107	0.0336	1.0291	0.0247	0.3021	1.9983	-2.9933
27	0.0275	0.0531	0.7026	0.0275	0.3526	2.1896	-3.7945
28	0.0525	0.0687	0.4334	0.0319	0.3629	2.1718	-3.7168
29	0.0852	0.0798	0.2347	0.0374	0.3440	2.0274	-3.1105
30	0.1249	0.0864	0.1046	0.0434	0.3125	1.8418	-2.3924
31	0.1710	0.0895	0.0328	0.0492	0.2803	1.6693	-1.7866
32	0.2228	0.0904	0.0046	0.0544	0.2539	1.5359	-1.3589
33	0.2793	0.0905	0.0000	0.0587	0.2356	1.4473	-1.0946
34	0.3396	0.0905	0.0000	0.0619	0.2247	1.3931	-0.9407
35	0.4027	0.0905	0.0000	0.0641	0.2181	1.3630	-0.8578
36	0.4674	0.0903	-0.0077	0.0653	0.2153	1.3568	-0.8410
37	0.5326	0.0883	-0.0518	0.0654	0.2173	1.3433	-0.8046
38	0.5973	0.0838	-0.0869	0.0644	0.2117	1.3077	-0.7100
39	0.6604	0.0774	-0.1178	0.0624	0.2055	1.2690	-0.6103
40	0.7207	0.0693	-0.1480	0.0593	0.1992	1.2287	-0.5097
41	0.7772	0.0600	-0.1795	0.0553	0.1925	1.1853	-0.4049
42	0.8290	0.0498	-0.2128	0.0503	0.1852	1.1361	-0.2907
43	0.8751	0.0390	-0.2472	0.0445	0.1766	1.0785	-0.1633
44	0.9148	0.0283	-0.2811	0.0378	0.1664	1.0106	-0.0213
45	0.9475	0.0184	-0.3121	0.0304	0.1541	0.9303	0.1345
46	0.9725	0.0101	-0.3378	0.0223	0.1391	0.8345	0.3036
47	0.9893	0.0040	-0.3562	0.0136	0.1196	0.7130	0.4916
48	0.9979	0.0008	-0.3658	0.0046	0.0867	0.4007	0.8395
49					-0.1381		

C_l = 1.2574
C_d = 0.020606
C_m = -0.34324

Table 7 Resulting Data Around Airfoil for 48 Panels, N = 2, $\alpha = 10°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.1354	-0.3863	0.8508
2	0.9893	-0.0040	-2.7854	0.0136	-0.0830	-0.6755	0.5437
3	0.9725	-0.0101	-2.8038	0.0223	-0.1118	-0.7705	0.4064
4	0.9475	-0.0184	-2.8295	0.0304	-0.1266	-0.8351	0.3026
5	0.9148	-0.0283	-2.8605	0.0378	-0.1363	-0.8802	0.2253
6	0.8751	-0.0390	-2.8944	0.0445	-0.1426	-0.9096	0.1725
7	0.8290	-0.0498	-2.9288	0.0503	-0.1465	-0.9258	0.1429
8	0.7772	-0.0600	-2.9621	0.0553	-0.1482	-0.9308	0.1336
9	0.7207	-0.0693	-2.9936	0.0593	-0.1483	-0.9271	0.1405
10	0.6604	-0.0774	-3.0238	0.0624	-0.1471	-0.9164	0.1603
11	0.5973	-0.0838	-3.0547	0.0644	-0.1449	-0.8994	0.1911
12	0.5326	-0.0883	-3.0898	0.0654	-0.1418	-0.8745	0.2352
13	0.4674	-0.0903	-3.1339	0.0653	-0.1373	-0.8300	0.3111
14	0.4027	-0.0905	3.1416	0.0641	-0.1272	-0.7764	0.3972
15	0.3396	-0.0905	3.1416	0.0619	-0.1194	-0.7300	0.4671
16	0.2793	-0.0905	3.1416	0.0587	-0.1123	-0.6850	0.5308
17	0.2228	-0.0904	3.1370	0.0544	-0.1051	-0.6379	0.5931
18	0.1710	-0.0895	3.1088	0.0492	-0.0975	-0.5785	0.6654
19	0.1249	-0.0864	3.0370	0.0434	-0.0867	-0.4821	0.7676
20	0.0852	-0.0798	2.9069	0.0374	-0.0671	-0.3101	0.9038
21	0.0525	-0.0687	2.7082	0.0319	-0.0313	-0.0160	0.9997
22	0.0275	-0.0531	2.4390	0.0275	0.0288	0.4339	0.8117
23	0.0107	-0.0336	2.1125	0.0247	0.1161	1.0211	-0.0427
24	0.0021	-0.0115	1.7543	0.0234	0.2211	1.6567	-1.7447
25	0.0021	0.0115	1.3873	0.0234	0.3240	2.2138	-3.9007
26	0.0107	0.0336	1.0291	0.0247	0.4030	2.5748	-5.6296
27	0.0275	0.0531	0.7026	0.0275	0.4411	2.6801	-6.1827
28	0.0525	0.0687	0.4334	0.0319	0.4353	2.5662	-5.5853
29	0.0852	0.0798	0.2347	0.0374	0.4005	2.3347	-4.4508
30	0.1249	0.0864	0.1046	0.0434	0.3558	2.0797	-3.3250
31	0.1710	0.0895	0.0328	0.0492	0.3138	1.8557	-2.4435
32	0.2228	0.0904	0.0046	0.0544	0.2802	1.6855	-1.8411
33	0.2793	0.0905	0.0000	0.0587	0.2570	1.5710	-1.4680
34	0.3396	0.0905	0.0000	0.0619	0.2427	1.4976	-1.2428
35	0.4027	0.0905	0.0000	0.0641	0.2335	1.4525	-1.1097
36	0.4674	0.0903	-0.0077	0.0653	0.2285	1.4341	-1.0567
37	0.5326	0.0883	-0.0518	0.0654	0.2288	1.4090	-0.9852
38	0.5973	0.0838	-0.0869	0.0644	0.2212	1.3616	-0.8540
39	0.6604	0.0774	-0.1178	0.0624	0.2132	1.3124	-0.7223
40	0.7207	0.0693	-0.1480	0.0593	0.2053	1.2625	-0.5939
41	0.7772	0.0600	-0.1795	0.0553	0.1972	1.2103	-0.4649
42	0.8290	0.0498	-0.2128	0.0503	0.1885	1.1532	-0.3298
43	0.8751	0.0390	-0.2472	0.0445	0.1788	1.0884	-0.1847
44	0.9148	0.0283	-0.2811	0.0378	0.1674	1.0141	-0.0284
45	0.9475	0.0184	-0.3121	0.0304	0.1542	0.9284	0.1380
46	0.9725	0.0101	-0.3378	0.0223	0.1384	0.8284	0.3137
47	0.9893	0.0040	-0.3562	0.0136	0.1184	0.7043	0.5040
48	0.9979	0.0008	-0.3658	0.0046	0.0855	0.3944	0.8445
49					-0.1354		

C_l = 1.8706
C_d = 0.033191
C_m = -0.50178

Table 8 Resulting Data Around Airfoil for 48 Panels, N = 2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 2}$ (panel configuration 2):

i	x	y	theta	s	gamma	v	Cp
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1	0.9792	-0.0075	-2.7963	0.0443	0.2566	-0.4822	0.7675
2	0.9375	-0.0215	-2.8392	0.0436	-0.1083	-0.8909	0.2064
3	0.8958	-0.0336	-2.8772	0.0432	-0.1244	-0.9309	0.1334
4	0.8542	-0.0442	-2.9106	0.0428	-0.1372	-0.9647	0.0694
5	0.8125	-0.0533	-2.9400	0.0425	-0.1439	-0.9836	0.0325
6	0.7708	-0.0613	-2.9660	0.0423	-0.1478	-0.9941	0.0117
7	0.7292	-0.0682	-2.9892	0.0422	-0.1500	-0.9991	0.0018
8	0.6875	-0.0741	-3.0105	0.0420	-0.1512	-1.0004	-0.0008
9	0.6458	-0.0792	-3.0308	0.0419	-0.1518	-0.9990	0.0021
10	0.6042	-0.0834	-3.0512	0.0418	-0.1518	-0.9952	0.0095
11	0.5625	-0.0867	-3.0727	0.0418	-0.1514	-0.9888	0.0222
12	0.5208	-0.0891	-3.0967	0.0417	-0.1504	-0.9785	0.0426
13	0.4792	-0.0904	-3.1248	0.0417	-0.1486	-0.9593	0.0797
14	0.4375	-0.0906	3.1367	0.0417	-0.1444	-0.9275	0.1397
15	0.3958	-0.0905	3.1416	0.0417	-0.1378	-0.8986	0.1926
16	0.3542	-0.0905	3.1416	0.0417	-0.1347	-0.8790	0.2274
17	0.3125	-0.0905	3.1416	0.0417	-0.1317	-0.8602	0.2601
18	0.2708	-0.0905	3.1415	0.0417	-0.1288	-0.8428	0.2898
19	0.2292	-0.0905	3.1387	0.0417	-0.1262	-0.8275	0.3152
20	0.1875	-0.0900	3.1226	0.0417	-0.1244	-0.8120	0.3406
21	0.1458	-0.0883	3.0777	0.0418	-0.1222	-0.7842	0.3850
22	0.1042	-0.0835	2.9796	0.0422	-0.1170	-0.7118	0.4934
23	0.0625	-0.0722	2.7769	0.0446	-0.1011	-0.4964	0.7535
24	0.0208	-0.0321	2.1463	0.0766	-0.0490	0.4142	0.8284
25	0.0208	0.0321	0.9952	0.0766	0.2189	1.7734	-2.1451
26	0.0625	0.0722	0.3647	0.0446	0.3714	2.1775	-3.7416
27	0.1042	0.0835	0.1620	0.0422	0.3242	1.9477	-2.7936
28	0.1458	0.0883	0.0639	0.0418	0.2892	1.7621	-2.1050
29	0.1875	0.0900	0.0190	0.0417	0.2618	1.6207	-1.6265
30	0.2292	0.0905	0.0029	0.0417	0.2421	1.5207	-1.3124
31	0.2708	0.0905	0.0000	0.0417	0.2289	1.4540	-1.1140
32	0.3125	0.0905	0.0000	0.0417	0.2205	1.4100	-0.9881
33	0.3542	0.0905	0.0000	0.0417	0.2149	1.3798	-0.9039
34	0.3958	0.0905	0.0000	0.0417	0.2109	1.3585	-0.8455
35	0.4375	0.0906	0.0049	0.0417	0.2079	1.3556	-0.8377
36	0.4792	0.0904	-0.0168	0.0417	0.2107	1.3593	-0.8476
37	0.5208	0.0891	-0.0449	0.0417	0.2100	1.3470	-0.8145
38	0.5625	0.0867	-0.0689	0.0418	0.2066	1.3253	-0.7563
39	0.6042	0.0834	-0.0904	0.0418	0.2025	1.3007	-0.6918
40	0.6458	0.0792	-0.1108	0.0419	0.1981	1.2747	-0.6249
41	0.6875	0.0741	-0.1311	0.0420	0.1934	1.2475	-0.5561
42	0.7292	0.0682	-0.1524	0.0422	0.1884	1.2181	-0.4838
43	0.7708	0.0613	-0.1756	0.0423	0.1827	1.1853	-0.4049
44	0.8125	0.0533	-0.2016	0.0425	0.1761	1.1466	-0.3147
45	0.8542	0.0442	-0.2310	0.0428	0.1676	1.0985	-0.2068
46	0.8958	0.0336	-0.2644	0.0432	0.1562	1.0339	-0.0690
47	0.9375	0.0215	-0.3024	0.0436	0.1383	0.9601	0.0781
48	0.9792	0.0075	-0.3452	0.0443	0.1166	0.5089	0.7410
49					-0.2566		

C_l = 1.2162
C_d = 0.053711
C_m = -0.33954

Table 9 Resulting Data Around Airfoil (Panel Configuration 2) for 48 Panels, N = 2, $\alpha = 10°$

$\boldsymbol{m = 48, N = 2}$ (panel configuration 3):

i	x	y	theta	s	gamma	v	Cp
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1	0.9915	-0.0032	-2.7830	0.0182	0.1038	-0.5269	0.7223
2	0.9580	-0.0147	-2.8186	0.0527	-0.1074	-0.8321	0.3076
3	0.8933	-0.0338	-2.8788	0.0823	-0.1394	-0.9361	0.1237
4	0.8018	-0.0547	-2.9462	0.1056	-0.1537	-0.9863	0.0272
5	0.6897	-0.0730	-3.0090	0.1217	-0.1589	-0.9997	0.0006
6	0.5647	-0.0855	-3.0724	0.1297	-0.1591	-0.9848	0.0302
7	0.4353	-0.0903	-3.1377	0.1294	-0.1546	-0.9306	0.1340
8	0.3103	-0.0905	3.1416	0.1206	-0.1408	-0.8668	0.2486
9	0.1982	-0.0896	3.1236	0.1036	-0.1332	-0.8212	0.3256
10	0.1067	-0.0820	2.9768	0.0806	-0.1281	-0.7057	0.5020
11	0.0420	-0.0598	2.5835	0.0589	-0.0976	-0.2678	0.9283
12	0.0085	-0.0221	1.9383	0.0474	0.0190	0.6896	0.5244
13	0.0085	0.0221	1.2033	0.0474	0.2170	1.7196	-1.9569
14	0.0420	0.0598	0.5581	0.0589	0.3525	2.1059	-3.4347
15	0.1067	0.0820	0.1648	0.0806	0.3432	1.9016	-2.6163
16	0.1982	0.0896	0.0180	0.1036	0.2790	1.6006	-1.5620
17	0.3103	0.0905	0.0000	0.1206	0.2343	1.4225	-1.0236
18	0.4353	0.0903	-0.0039	0.1294	0.2170	1.3600	-0.8497
19	0.5647	0.0855	-0.0692	0.1297	0.2156	1.3173	-0.7353
20	0.6897	0.0730	-0.1326	0.1217	0.2049	1.2447	-0.5493
21	0.8018	0.0547	-0.1954	0.1056	0.1917	1.1563	-0.3370
22	0.8933	0.0338	-0.2628	0.0823	0.1753	1.0415	-0.0848
23	0.9580	0.0147	-0.3230	0.0527	0.1515	0.8840	0.2185
24	0.9915	0.0032	-0.3586	0.0182	0.1118	0.5413	0.7069
25					-0.1038		

C_l = 1.2195
C_d = 0.043727
C_m = -0.34137

Table 10 Resulting Data Around Airfoil (Panel Configuration 3) for 48 Panels, N = 2, $\alpha = 10°$

$\boldsymbol{m = 48, N = 3/2}$:

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.0304	-0.4917	0.7583
2	0.9893	-0.0040	-2.7854	0.0136	-0.0959	-0.7187	0.4834
3	0.9725	-0.0101	-2.8038	0.0223	-0.1221	-0.8295	0.3120
4	0.9475	-0.0184	-2.8295	0.0304	-0.1387	-0.9139	0.1647
5	0.9148	-0.0283	-2.8605	0.0378	-0.1513	-0.9814	0.0369
6	0.8751	-0.0390	-2.8944	0.0445	-0.1612	-1.0350	-0.0712
7	0.8290	-0.0498	-2.9288	0.0503	-0.1689	-1.0768	-0.1595
8	0.7772	-0.0600	-2.9621	0.0553	-0.1747	-1.1090	-0.2299
9	0.7207	-0.0693	-2.9936	0.0593	-0.1792	-1.1341	-0.2863
10	0.6604	-0.0774	-3.0238	0.0624	-0.1828	-1.1545	-0.3329
11	0.5973	-0.0838	-3.0547	0.0644	-0.1858	-1.1714	-0.3722
12	0.5326	-0.0883	-3.0898	0.0654	-0.1883	-1.1833	-0.4002
13	0.4674	-0.0903	-3.1339	0.0653	-0.1899	-1.1736	-0.3773
14	0.4027	-0.0905	3.1416	0.0641	-0.1846	-1.1560	-0.3362
15	0.3396	-0.0905	3.1416	0.0619	-0.1833	-1.1564	-0.3372
16	0.2793	-0.0905	3.1415	0.0587	-0.1846	-1.1734	-0.3770
17	0.2228	-0.0903	3.1354	0.0544	-0.1889	-1.2138	-0.4733
18	0.1710	-0.0891	3.0980	0.0492	-0.1984	-1.2782	-0.6339
19	0.1249	-0.0851	3.0047	0.0435	-0.2114	-1.3423	-0.8017
20	0.0852	-0.0765	2.8450	0.0380	-0.2215	-1.3554	-0.8372
21	0.0525	-0.0629	2.6291	0.0332	-0.2178	-1.2594	-0.5862
22	0.0275	-0.0448	2.3848	0.0289	-0.1914	-1.0268	-0.0543
23	0.0107	-0.0247	2.1330	0.0239	-0.1423	-0.6764	0.5425
24	0.0021	-0.0073	1.8557	0.0152	-0.0769	-0.2513	0.9369
25	0.0021	0.0073	1.2859	0.0152	-0.0000	0.2513	0.9369
26	0.0107	0.0247	1.0085	0.0239	0.0769	0.6764	0.5425
27	0.0275	0.0448	0.7568	0.0289	0.1423	1.0268	-0.0543
28	0.0525	0.0629	0.5125	0.0332	0.1914	1.2594	-0.5862
29	0.0852	0.0765	0.2966	0.0380	0.2178	1.3554	-0.8372
30	0.1249	0.0851	0.1369	0.0435	0.2215	1.3423	-0.8017
31	0.1710	0.0891	0.0436	0.0492	0.2114	1.2782	-0.6339
32	0.2228	0.0903	0.0062	0.0544	0.1984	1.2138	-0.4733
33	0.2793	0.0905	0.0000	0.0587	0.1889	1.1734	-0.3770
34	0.3396	0.0905	0.0000	0.0619	0.1846	1.1564	-0.3372
35	0.4027	0.0905	0.0000	0.0641	0.1833	1.1560	-0.3362
36	0.4674	0.0903	-0.0077	0.0653	0.1846	1.1736	-0.3773
37	0.5326	0.0883	-0.0518	0.0654	0.1899	1.1833	-0.4002
38	0.5973	0.0838	-0.0869	0.0644	0.1883	1.1714	-0.3722
39	0.6604	0.0774	-0.1178	0.0624	0.1858	1.1545	-0.3329
40	0.7207	0.0693	-0.1480	0.0593	0.1828	1.1341	-0.2863
41	0.7772	0.0600	-0.1795	0.0553	0.1792	1.1090	-0.2299
42	0.8290	0.0498	-0.2128	0.0503	0.1747	1.0768	-0.1595
43	0.8751	0.0390	-0.2472	0.0445	0.1689	1.0350	-0.0712
44	0.9148	0.0283	-0.2811	0.0378	0.1612	0.9814	0.0369
45	0.9475	0.0184	-0.3121	0.0304	0.1513	0.9139	0.1647
46	0.9725	0.0101	-0.3378	0.0223	0.1387	0.8295	0.3120
47	0.9893	0.0040	-0.3562	0.0136	0.1221	0.7187	0.4834
48	0.9979	0.0008	-0.3658	0.0046	0.0959	0.4917	0.7583
49					-0.0304		

C_l = -7.1818e-016
C_d = 0.008025
C_m = 3.9899e-016

Table 11 Resulting Data Around Airfoil for 48 Panels, N = 3/2, $\alpha = 0°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.0303	-0.4884	0.7614
2	0.9893	-0.0040	-2.7854	0.0136	-0.0951	-0.7111	0.4943
3	0.9725	-0.0101	-2.8038	0.0223	-0.1205	-0.8165	0.3333
4	0.9475	-0.0184	-2.8295	0.0304	-0.1361	-0.8947	0.1995
5	0.9148	-0.0283	-2.8605	0.0378	-0.1477	-0.9550	0.0880
6	0.8751	-0.0390	-2.8944	0.0445	-0.1564	-1.0008	-0.0017
7	0.8290	-0.0498	-2.9288	0.0503	-0.1628	-1.0343	-0.0697
8	0.7772	-0.0600	-2.9621	0.0553	-0.1673	-1.0576	-0.1184
9	0.7207	-0.0693	-2.9936	0.0593	-0.1703	-1.0731	-0.1516
10	0.6604	-0.0774	-3.0238	0.0624	-0.1722	-1.0831	-0.1732
11	0.5973	-0.0838	-3.0547	0.0644	-0.1735	-1.0887	-0.1853
12	0.5326	-0.0883	-3.0898	0.0654	-0.1741	-1.0882	-0.1843
13	0.4674	-0.0903	-3.1339	0.0653	-0.1737	-1.0667	-0.1378
14	0.4027	-0.0905	3.1416	0.0641	-0.1667	-1.0367	-0.0748
15	0.3396	-0.0905	3.1416	0.0619	-0.1632	-1.0213	-0.0431
16	0.2793	-0.0905	3.1415	0.0587	-0.1617	-1.0176	-0.0355
17	0.2228	-0.0903	3.1354	0.0544	-0.1621	-1.0288	-0.0584
18	0.1710	-0.0891	3.0980	0.0492	-0.1660	-1.0511	-0.1048
19	0.1249	-0.0851	3.0047	0.0435	-0.1708	-1.0571	-0.1175
20	0.0852	-0.0765	2.8450	0.0380	-0.1698	-0.9978	0.0045
21	0.0525	-0.0629	2.6291	0.0332	-0.1531	-0.8214	0.3252
22	0.0275	-0.0448	2.3848	0.0289	-0.1132	-0.5064	0.7436
23	0.0107	-0.0247	2.1330	0.0239	-0.0505	-0.0620	0.9962
24	0.0021	-0.0073	1.8557	0.0152	0.0326	0.5017	0.7483
25	0.0021	0.0073	1.2859	0.0152	0.1381	1.0023	-0.0045
26	0.0107	0.0247	1.0085	0.0239	0.1858	1.2857	-0.6529
27	0.0275	0.0448	0.7568	0.0289	0.2331	1.5394	-1.3697
28	0.0525	0.0629	0.5125	0.0332	0.2682	1.6879	-1.8489
29	0.0852	0.0765	0.2966	0.0380	0.2809	1.7028	-1.8996
30	0.1249	0.0851	0.1369	0.0435	0.2715	1.6172	-1.6154
31	0.1710	0.0891	0.0436	0.0492	0.2505	1.4956	-1.2369
32	0.2228	0.0903	0.0062	0.0544	0.2292	1.3896	-0.9309
33	0.2793	0.0905	0.0000	0.0587	0.2142	1.3204	-0.7434
34	0.3396	0.0905	0.0000	0.0619	0.2061	1.2826	-0.6451
35	0.4027	0.0905	0.0000	0.0641	0.2020	1.2664	-0.6037
36	0.4674	0.0903	-0.0077	0.0653	0.2011	1.2716	-0.6169
37	0.5326	0.0883	-0.0518	0.0654	0.2047	1.2693	-0.6111
38	0.5973	0.0838	-0.0869	0.0644	0.2010	1.2452	-0.5505
39	0.6604	0.0774	-0.1178	0.0624	0.1966	1.2171	-0.4814
40	0.7207	0.0693	-0.1480	0.0593	0.1919	1.1865	-0.4079
41	0.7772	0.0600	-0.1795	0.0553	0.1868	1.1521	-0.3272
42	0.8290	0.0498	-0.2128	0.0503	0.1809	1.1111	-0.2346
43	0.8751	0.0390	-0.2472	0.0445	0.1737	1.0612	-0.1262
44	0.9148	0.0283	-0.2811	0.0378	0.1647	1.0002	-0.0005
45	0.9475	0.0184	-0.3121	0.0304	0.1537	0.9262	0.1421
46	0.9725	0.0101	-0.3378	0.0223	0.1401	0.8361	0.3009
47	0.9893	0.0040	-0.3562	0.0136	0.1227	0.7209	0.4804
48	0.9979	0.0008	-0.3658	0.0046	0.0960	0.4912	0.7588
49					-0.0303		

C_l = 0.62991
C_d = 0.010966
C_m = -0.1759

Table 12 Resulting Data Around Airfoil for 48 Panels, N = 3/2, $\alpha = 5°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.0300	-0.4815	0.7682
2	0.9893	-0.0040	-2.7854	0.0136	-0.0936	-0.6981	0.5126
3	0.9725	-0.0101	-2.8038	0.0223	-0.1180	-0.7974	0.3642
4	0.9475	-0.0184	-2.8295	0.0304	-0.1326	-0.8686	0.2454
5	0.9148	-0.0283	-2.8605	0.0378	-0.1429	-0.9214	0.1510
6	0.8751	-0.0390	-2.8944	0.0445	-0.1504	-0.9591	0.0802
7	0.8290	-0.0498	-2.9288	0.0503	-0.1554	-0.9839	0.0320
8	0.7772	-0.0600	-2.9621	0.0553	-0.1585	-0.9980	0.0039
9	0.7207	-0.0693	-2.9936	0.0593	-0.1600	-1.0039	-0.0078
10	0.6604	-0.0774	-3.0238	0.0624	-0.1604	-1.0035	-0.0070
11	0.5973	-0.0838	-3.0547	0.0644	-0.1599	-0.9977	0.0046
12	0.5326	-0.0883	-3.0898	0.0654	-0.1586	-0.9849	0.0299
13	0.4674	-0.0903	-3.1339	0.0653	-0.1561	-0.9517	0.0944
14	0.4027	-0.0905	3.1416	0.0641	-0.1476	-0.9096	0.1726
15	0.3396	-0.0905	3.1416	0.0619	-0.1419	-0.8785	0.2283
16	0.2793	-0.0905	3.1415	0.0587	-0.1375	-0.8540	0.2708
17	0.2228	-0.0903	3.1354	0.0544	-0.1341	-0.8359	0.3012
18	0.1710	-0.0891	3.0980	0.0492	-0.1323	-0.8160	0.3342
19	0.1249	-0.0851	3.0047	0.0435	-0.1289	-0.7639	0.4164
20	0.0852	-0.0765	2.8450	0.0380	-0.1168	-0.6325	0.6000
21	0.0525	-0.0629	2.6291	0.0332	-0.0871	-0.3772	0.8577
22	0.0275	-0.0448	2.3848	0.0289	-0.0341	0.0179	0.9997
23	0.0107	-0.0247	2.1330	0.0239	0.0417	0.5529	0.6943
24	0.0021	-0.0073	1.8557	0.0152	0.1418	1.2507	-0.5644
25	0.0021	0.0073	1.2859	0.0152	0.2751	1.7456	-2.0472
26	0.0107	0.0247	1.0085	0.0239	0.2932	1.8851	-2.5537
27	0.0275	0.0448	0.7568	0.0289	0.3221	2.0403	-3.1627
28	0.0525	0.0629	0.5125	0.0332	0.3429	2.1034	-3.4244
29	0.0852	0.0765	0.2966	0.0380	0.3419	2.0372	-3.1504
30	0.1249	0.0851	0.1369	0.0435	0.3194	1.8798	-2.5338
31	0.1710	0.0891	0.0436	0.0492	0.2876	1.7017	-1.8957
32	0.2228	0.0903	0.0062	0.0544	0.2584	1.5548	-1.4173
33	0.2793	0.0905	0.0000	0.0587	0.2378	1.4573	-1.1236
34	0.3396	0.0905	0.0000	0.0619	0.2261	1.3991	-0.9576
35	0.4027	0.0905	0.0000	0.0641	0.2191	1.3672	-0.8692
36	0.4674	0.0903	-0.0077	0.0653	0.2160	1.3599	-0.8493
37	0.5326	0.0883	-0.0518	0.0654	0.2180	1.3457	-0.8108
38	0.5973	0.0838	-0.0869	0.0644	0.2122	1.3095	-0.7147
39	0.6604	0.0774	-0.1178	0.0624	0.2060	1.2704	-0.6140
40	0.7207	0.0693	-0.1480	0.0593	0.1996	1.2299	-0.5127
41	0.7772	0.0600	-0.1795	0.0553	0.1930	1.1863	-0.4073
42	0.8290	0.0498	-0.2128	0.0503	0.1857	1.1370	-0.2928
43	0.8751	0.0390	-0.2472	0.0445	0.1772	1.0794	-0.1652
44	0.9148	0.0283	-0.2811	0.0378	0.1671	1.0115	-0.0232
45	0.9475	0.0184	-0.3121	0.0304	0.1550	0.9315	0.1324
46	0.9725	0.0101	-0.3378	0.0223	0.1405	0.8364	0.3005
47	0.9893	0.0040	-0.3562	0.0136	0.1224	0.7175	0.4852
48	0.9979	0.0008	-0.3658	0.0046	0.0953	0.4869	0.7629
49					-0.0300		

C_l = 1.2535
C_d = 0.019566
C_m = -0.34646

Table 13 Resulting Data Around Airfoil for 48 Panels, N = 3/2, $\alpha = 10°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.0294	-0.4709	0.7783
2	0.9893	-0.0040	-2.7854	0.0136	-0.0914	-0.6798	0.5378
3	0.9725	-0.0101	-2.8038	0.0223	-0.1146	-0.7721	0.4038
4	0.9475	-0.0184	-2.8295	0.0304	-0.1280	-0.8360	0.3011
5	0.9148	-0.0283	-2.8605	0.0378	-0.1371	-0.8808	0.2243
6	0.8751	-0.0390	-2.8944	0.0445	-0.1432	-0.9100	0.1719
7	0.8290	-0.0498	-2.9288	0.0503	-0.1469	-0.9260	0.1425
8	0.7772	-0.0600	-2.9621	0.0553	-0.1485	-0.9309	0.1334
9	0.7207	-0.0693	-2.9936	0.0593	-0.1486	-0.9271	0.1405
10	0.6604	-0.0774	-3.0238	0.0624	-0.1473	-0.9163	0.1605
11	0.5973	-0.0838	-3.0547	0.0644	-0.1451	-0.8991	0.1915
12	0.5326	-0.0883	-3.0898	0.0654	-0.1420	-0.8741	0.2359
13	0.4674	-0.0903	-3.1339	0.0653	-0.1374	-0.8294	0.3121
14	0.4027	-0.0905	3.1416	0.0641	-0.1273	-0.7756	0.3985
15	0.3396	-0.0905	3.1416	0.0619	-0.1195	-0.7289	0.4686
16	0.2793	-0.0905	3.1415	0.0587	-0.1123	-0.6838	0.5324
17	0.2228	-0.0903	3.1354	0.0544	-0.1051	-0.6367	0.5946
18	0.1710	-0.0891	3.0980	0.0492	-0.0976	-0.5746	0.6698
19	0.1249	-0.0851	3.0047	0.0435	-0.0860	-0.4649	0.7839
20	0.0852	-0.0765	2.8450	0.0380	-0.0629	-0.2624	0.9312
21	0.0525	-0.0629	2.6291	0.0332	-0.0206	0.0699	0.9951
22	0.0275	-0.0448	2.3848	0.0289	0.0452	0.5421	0.7062
23	0.0107	-0.0247	2.1330	0.0239	0.1336	1.1636	-0.3539
24	0.0021	-0.0073	1.8557	0.0152	0.2499	1.9903	-2.9614
25	0.0021	0.0073	1.2859	0.0152	0.4101	2.4757	-5.1291
26	0.0107	0.0247	1.0085	0.0239	0.3985	2.4702	-5.1021
27	0.0275	0.0448	0.7568	0.0289	0.4086	2.5256	-5.3789
28	0.0525	0.0629	0.5125	0.0332	0.4150	2.5030	-5.2650
29	0.0852	0.0765	0.2966	0.0380	0.4002	2.3562	-4.5515
30	0.1249	0.0851	0.1369	0.0435	0.3649	2.1282	-3.5291
31	0.1710	0.0891	0.0436	0.0492	0.3225	1.8947	-2.5900
32	0.2228	0.0903	0.0062	0.0544	0.2855	1.7081	-1.9177
33	0.2793	0.0905	0.0000	0.0587	0.2597	1.5831	-1.5061
34	0.3396	0.0905	0.0000	0.0619	0.2443	1.5050	-1.2650
35	0.4027	0.0905	0.0000	0.0641	0.2346	1.4576	-1.1245
36	0.4674	0.0903	-0.0077	0.0653	0.2293	1.4379	-1.0674
37	0.5326	0.0883	-0.0518	0.0654	0.2295	1.4118	-0.9932
38	0.5973	0.0838	-0.0869	0.0644	0.2218	1.3638	-0.8600
39	0.6604	0.0774	-0.1178	0.0624	0.2138	1.3141	-0.7269
40	0.7207	0.0693	-0.1480	0.0593	0.2058	1.2639	-0.5975
41	0.7772	0.0600	-0.1795	0.0553	0.1977	1.2115	-0.4678
42	0.8290	0.0498	-0.2128	0.0503	0.1890	1.1542	-0.3322
43	0.8751	0.0390	-0.2472	0.0445	0.1793	1.0894	-0.1868
44	0.9148	0.0283	-0.2811	0.0378	0.1681	1.0151	-0.0304
45	0.9475	0.0184	-0.3121	0.0304	0.1551	0.9296	0.1358
46	0.9725	0.0101	-0.3378	0.0223	0.1399	0.8303	0.3106
47	0.9893	0.0040	-0.3562	0.0136	0.1212	0.7087	0.4978
48	0.9979	0.0008	-0.3658	0.0046	0.0939	0.4790	0.7706
49					-0.0294		

C_l = 1.8646
C_d = 0.033165
C_m = -0.50649

Table 14 Resulting Data Around Airfoil for 48 Panels, N = 3/2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 3}$:

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.4494	-0.1576	0.9752
2	0.9893	-0.0040	-2.7854	0.0136	-0.0629	-0.7018	0.5075
3	0.9725	-0.0101	-2.8038	0.0223	-0.1109	-0.8228	0.3230
4	0.9475	-0.0184	-2.8295	0.0304	-0.1331	-0.9102	0.1715
5	0.9148	-0.0283	-2.8605	0.0378	-0.1478	-0.9787	0.0421
6	0.8751	-0.0390	-2.8944	0.0445	-0.1587	-1.0328	-0.0668
7	0.8290	-0.0498	-2.9288	0.0503	-0.1670	-1.0749	-0.1553
8	0.7772	-0.0600	-2.9621	0.0553	-0.1732	-1.1071	-0.2257
9	0.7207	-0.0693	-2.9936	0.0593	-0.1778	-1.1322	-0.2818
10	0.6604	-0.0774	-3.0238	0.0624	-0.1815	-1.1523	-0.3279
11	0.5973	-0.0838	-3.0547	0.0644	-0.1845	-1.1689	-0.3662
12	0.5326	-0.0883	-3.0898	0.0654	-0.1870	-1.1802	-0.3929
13	0.4674	-0.0903	-3.1339	0.0653	-0.1886	-1.1697	-0.3682
14	0.4027	-0.0905	3.1416	0.0641	-0.1831	-1.1507	-0.3242
15	0.3396	-0.0905	3.1416	0.0619	-0.1815	-1.1488	-0.3197
16	0.2793	-0.0905	3.1416	0.0587	-0.1824	-1.1607	-0.3473
17	0.2228	-0.0904	3.1385	0.0544	-0.1854	-1.1894	-0.4146
18	0.1710	-0.0898	3.1196	0.0492	-0.1919	-1.2377	-0.5320
19	0.1249	-0.0877	3.0706	0.0432	-0.2019	-1.3011	-0.6929
20	0.0852	-0.0832	2.9768	0.0368	-0.2139	-1.3669	-0.8684
21	0.0525	-0.0753	2.8153	0.0305	-0.2253	-1.4055	-0.9754
22	0.0275	-0.0633	2.5475	0.0254	-0.2294	-1.3376	-0.7891
23	0.0107	-0.0463	2.1430	0.0236	-0.2058	-1.0126	-0.0253
24	0.0021	-0.0182	1.6879	0.0366	-0.1235	-0.3551	0.8739
25	0.0021	0.0182	1.4537	0.0366	0.0000	0.3551	0.8739
26	0.0107	0.0463	0.9986	0.0236	0.1235	1.0126	-0.0253
27	0.0275	0.0633	0.5941	0.0254	0.2058	1.3376	-0.7891
28	0.0525	0.0753	0.3263	0.0305	0.2294	1.4055	-0.9754
29	0.0852	0.0832	0.1648	0.0368	0.2253	1.3669	-0.8684
30	0.1249	0.0877	0.0710	0.0432	0.2139	1.3011	-0.6929
31	0.1710	0.0898	0.0220	0.0492	0.2019	1.2377	-0.5320
32	0.2228	0.0904	0.0031	0.0544	0.1919	1.1894	-0.4146
33	0.2793	0.0905	0.0000	0.0587	0.1854	1.1607	-0.3473
34	0.3396	0.0905	0.0000	0.0619	0.1824	1.1488	-0.3197
35	0.4027	0.0905	0.0000	0.0641	0.1815	1.1507	-0.3242
36	0.4674	0.0903	-0.0077	0.0653	0.1831	1.1697	-0.3682
37	0.5326	0.0883	-0.0518	0.0654	0.1886	1.1802	-0.3929
38	0.5973	0.0838	-0.0869	0.0644	0.1870	1.1689	-0.3662
39	0.6604	0.0774	-0.1178	0.0624	0.1845	1.1523	-0.3279
40	0.7207	0.0693	-0.1480	0.0593	0.1815	1.1322	-0.2818
41	0.7772	0.0600	-0.1795	0.0553	0.1778	1.1071	-0.2257
42	0.8290	0.0498	-0.2128	0.0503	0.1732	1.0749	-0.1553
43	0.8751	0.0390	-0.2472	0.0445	0.1670	1.0328	-0.0668
44	0.9148	0.0283	-0.2811	0.0378	0.1587	0.9787	0.0421
45	0.9475	0.0184	-0.3121	0.0304	0.1478	0.9102	0.1715
46	0.9725	0.0101	-0.3378	0.0223	0.1331	0.8228	0.3230
47	0.9893	0.0040	-0.3562	0.0136	0.1109	0.7018	0.5075
48	0.9979	0.0008	-0.3658	0.0046	0.0629	0.1576	0.9752
49					-0.4494		

C_l = -4.8225e-016
C_d = 0.014212
C_m = 3.4694e-017

Table 15 Resulting Data Around Airfoil for 48 Panels, N = 3, $\alpha = 0°$

i	x	y	theta	s	gamma	v	Cp

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1	0.9979	-0.0008	-2.7758	0.0046	0.4477	-0.1557	0.9758
2	0.9893	-0.0040	-2.7854	0.0136	-0.0622	-0.6943	0.5179
3	0.9725	-0.0101	-2.8038	0.0223	-0.1094	-0.8099	0.3440
4	0.9475	-0.0184	-2.8295	0.0304	-0.1306	-0.8911	0.2059
5	0.9148	-0.0283	-2.8605	0.0378	-0.1443	-0.9526	0.0926
6	0.8751	-0.0390	-2.8944	0.0445	-0.1540	-0.9989	0.0021
7	0.8290	-0.0498	-2.9288	0.0503	-0.1609	-1.0327	-0.0664
8	0.7772	-0.0600	-2.9621	0.0553	-0.1657	-1.0561	-0.1153
9	0.7207	-0.0693	-2.9936	0.0593	-0.1690	-1.0717	-0.1485
10	0.6604	-0.0774	-3.0238	0.0624	-0.1711	-1.0817	-0.1700
11	0.5973	-0.0838	-3.0547	0.0644	-0.1724	-1.0871	-0.1818
12	0.5326	-0.0883	-3.0898	0.0654	-0.1730	-1.0864	-0.1803
13	0.4674	-0.0903	-3.1339	0.0653	-0.1726	-1.0644	-0.1330
14	0.4027	-0.0905	3.1416	0.0641	-0.1655	-1.0337	-0.0686
15	0.3396	-0.0905	3.1416	0.0619	-0.1619	-1.0169	-0.0341
16	0.2793	-0.0905	3.1416	0.0587	-0.1601	-1.0098	-0.0197
17	0.2228	-0.0904	3.1385	0.0544	-0.1597	-1.0131	-0.0264
18	0.1710	-0.0898	3.1196	0.0492	-0.1615	-1.0264	-0.0535
19	0.1249	-0.0877	3.0706	0.0432	-0.1648	-1.0407	-0.0831
20	0.0852	-0.0832	2.9768	0.0368	-0.1674	-1.0380	-0.0775
21	0.0525	-0.0753	2.8153	0.0305	-0.1655	-0.9826	0.0344
22	0.0275	-0.0633	2.5475	0.0254	-0.1513	-0.8004	0.3593
23	0.0107	-0.0463	2.1430	0.0236	-0.1075	-0.3942	0.8446
24	0.0021	-0.0182	1.6879	0.0366	-0.0193	0.2274	0.9483
25	0.0021	0.0182	1.4537	0.0366	0.0896	0.9348	0.1261
26	0.0107	0.0463	0.9986	0.0236	0.2266	1.6232	-1.6348
27	0.0275	0.0633	0.5941	0.0254	0.3026	1.8645	-2.4764
28	0.0525	0.0753	0.3263	0.0305	0.3056	1.8176	-2.3038
29	0.0852	0.0832	0.1648	0.0368	0.2834	1.6854	-1.8405
30	0.1249	0.0877	0.0710	0.0432	0.2588	1.5516	-1.4075
31	0.1710	0.0898	0.0220	0.0492	0.2375	1.4396	-1.0725
32	0.2228	0.0904	0.0031	0.0544	0.2210	1.3565	-0.8402
33	0.2793	0.0905	0.0000	0.0587	0.2097	1.3028	-0.6972
34	0.3396	0.0905	0.0000	0.0619	0.2033	1.2719	-0.6178
35	0.4027	0.0905	0.0000	0.0641	0.1998	1.2590	-0.5850
36	0.4674	0.0903	-0.0077	0.0653	0.1993	1.2660	-0.6029
37	0.5326	0.0883	-0.0518	0.0654	0.2032	1.2650	-0.6002
38	0.5973	0.0838	-0.0869	0.0644	0.1996	1.2417	-0.5419
39	0.6604	0.0774	-0.1178	0.0624	0.1953	1.2142	-0.4744
40	0.7207	0.0693	-0.1480	0.0593	0.1905	1.1840	-0.4019
41	0.7772	0.0600	-0.1795	0.0553	0.1854	1.1497	-0.3219
42	0.8290	0.0498	-0.2128	0.0503	0.1793	1.1089	-0.2296
43	0.8751	0.0390	-0.2472	0.0445	0.1718	1.0589	-0.1212
44	0.9148	0.0283	-0.2811	0.0378	0.1623	0.9975	0.0051
45	0.9475	0.0184	-0.3121	0.0304	0.1503	0.9224	0.1491
46	0.9725	0.0101	-0.3378	0.0223	0.1346	0.8294	0.3122
47	0.9893	0.0040	-0.3562	0.0136	0.1116	0.7039	0.5045
48	0.9979	0.0008	-0.3658	0.0046	0.0631	0.1584	0.9749
49					-0.4477		

C_l = 0.63302
C_d = 0.016964
C_m = -0.17199

Table 16 Resulting Data Around Airfoil for 48 Panels, N = 3, $\alpha = 5°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.4426	-0.1526	0.9767
2	0.9893	-0.0040	-2.7854	0.0136	-0.0611	-0.6815	0.5355
3	0.9725	-0.0101	-2.8038	0.0223	-0.1070	-0.7909	0.3745
4	0.9475	-0.0184	-2.8295	0.0304	-0.1272	-0.8652	0.2514
5	0.9148	-0.0283	-2.8605	0.0378	-0.1396	-0.9191	0.1552
6	0.8751	-0.0390	-2.8944	0.0445	-0.1480	-0.9574	0.0833
7	0.8290	-0.0498	-2.9288	0.0503	-0.1536	-0.9826	0.0345
8	0.7772	-0.0600	-2.9621	0.0553	-0.1571	-0.9970	0.0060
9	0.7207	-0.0693	-2.9936	0.0593	-0.1588	-1.0030	-0.0061
10	0.6604	-0.0774	-3.0238	0.0624	-0.1593	-1.0028	-0.0056
11	0.5973	-0.0838	-3.0547	0.0644	-0.1590	-0.9971	0.0058
12	0.5326	-0.0883	-3.0898	0.0654	-0.1577	-0.9843	0.0311
13	0.4674	-0.0903	-3.1339	0.0653	-0.1553	-0.9511	0.0955
14	0.4027	-0.0905	3.1416	0.0641	-0.1467	-0.9089	0.1739
15	0.3396	-0.0905	3.1416	0.0619	-0.1410	-0.8773	0.2303
16	0.2793	-0.0905	3.1416	0.0587	-0.1365	-0.8512	0.2754
17	0.2228	-0.0904	3.1385	0.0544	-0.1327	-0.8292	0.3125
18	0.1710	-0.0898	3.1196	0.0492	-0.1298	-0.8072	0.3484
19	0.1249	-0.0877	3.0706	0.0432	-0.1264	-0.7724	0.4033
20	0.0852	-0.0832	2.9768	0.0368	-0.1195	-0.7012	0.5083
21	0.0525	-0.0753	2.8153	0.0305	-0.1045	-0.5523	0.6949
22	0.0275	-0.0633	2.5475	0.0254	-0.0722	-0.2572	0.9338
23	0.0107	-0.0463	2.1430	0.0236	-0.0083	0.2271	0.9484
24	0.0021	-0.0182	1.6879	0.0366	0.0849	0.8081	0.3470
25	0.0021	0.0182	1.4537	0.0366	0.1784	1.5075	-1.2725
26	0.0107	0.0463	0.9986	0.0236	0.3281	2.2215	-3.9350
27	0.0275	0.0633	0.5941	0.0254	0.3971	2.3773	-4.6515
28	0.0525	0.0753	0.3263	0.0305	0.3796	2.2159	-3.9104
29	0.0852	0.0832	0.1648	0.0368	0.3394	1.9910	-2.9642
30	0.1249	0.0877	0.0710	0.0432	0.3018	1.7903	-2.2051
31	0.1710	0.0898	0.0220	0.0492	0.2713	1.6306	-1.6588
32	0.2228	0.0904	0.0031	0.0544	0.2483	1.5134	-1.2903
33	0.2793	0.0905	0.0000	0.0587	0.2325	1.4349	-1.0590
34	0.3396	0.0905	0.0000	0.0619	0.2227	1.3854	-0.9192
35	0.4027	0.0905	0.0000	0.0641	0.2166	1.3576	-0.8431
36	0.4674	0.0903	-0.0077	0.0653	0.2140	1.3528	-0.8300
37	0.5326	0.0883	-0.0518	0.0654	0.2162	1.3402	-0.7961
38	0.5973	0.0838	-0.0869	0.0644	0.2106	1.3051	-0.7034
39	0.6604	0.0774	-0.1178	0.0624	0.2045	1.2669	-0.6050
40	0.7207	0.0693	-0.1480	0.0593	0.1981	1.2269	-0.5053
41	0.7772	0.0600	-0.1795	0.0553	0.1915	1.1836	-0.4009
42	0.8290	0.0498	-0.2128	0.0503	0.1840	1.1345	-0.2870
43	0.8751	0.0390	-0.2472	0.0445	0.1752	1.0769	-0.1596
44	0.9148	0.0283	-0.2811	0.0378	0.1646	1.0086	-0.0173
45	0.9475	0.0184	-0.3121	0.0304	0.1516	0.9276	0.1396
46	0.9725	0.0101	-0.3378	0.0223	0.1351	0.8296	0.3117
47	0.9893	0.0040	-0.3562	0.0136	0.1114	0.7007	0.5090
48	0.9979	0.0008	-0.3658	0.0046	0.0628	0.1579	0.9751
49					-0.4426		

C_l = 1.2598
C_d = 0.025005
C_m = -0.33876

Table 17 Resulting Data Around Airfoil for 48 Panels, N = 3, $\alpha = 10°$

i	x	y	theta	s	gamma	v	Cp
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1	0.9979	-0.0008	-2.7758	0.0046	0.4341	-0.1483	0.9780
2	0.9893	-0.0040	-2.7854	0.0136	-0.0595	-0.6636	0.5597
3	0.9725	-0.0101	-2.8038	0.0223	-0.1038	-0.7659	0.4134
4	0.9475	-0.0184	-2.8295	0.0304	-0.1227	-0.8327	0.3066
5	0.9148	-0.0283	-2.8605	0.0378	-0.1339	-0.8787	0.2279
6	0.8751	-0.0390	-2.8944	0.0445	-0.1410	-0.9086	0.1744
7	0.8290	-0.0498	-2.9288	0.0503	-0.1452	-0.9251	0.1443
8	0.7772	-0.0600	-2.9621	0.0553	-0.1472	-0.9303	0.1345
9	0.7207	-0.0693	-2.9936	0.0593	-0.1474	-0.9268	0.1411
10	0.6604	-0.0774	-3.0238	0.0624	-0.1464	-0.9162	0.1605
11	0.5973	-0.0838	-3.0547	0.0644	-0.1443	-0.8995	0.1910
12	0.5326	-0.0883	-3.0898	0.0654	-0.1412	-0.8748	0.2347
13	0.4674	-0.0903	-3.1339	0.0653	-0.1368	-0.8305	0.3103
14	0.4027	-0.0905	3.1416	0.0641	-0.1267	-0.7771	0.3961
15	0.3396	-0.0905	3.1416	0.0619	-0.1190	-0.7310	0.4656
16	0.2793	-0.0905	3.1416	0.0587	-0.1119	-0.6862	0.5292
17	0.2228	-0.0904	3.1385	0.0544	-0.1047	-0.6389	0.5918
18	0.1710	-0.0898	3.1196	0.0492	-0.0971	-0.5820	0.6613
19	0.1249	-0.0877	3.0706	0.0432	-0.0870	-0.4983	0.7517
20	0.0852	-0.0832	2.9768	0.0368	-0.0708	-0.3591	0.8710
21	0.0525	-0.0753	2.8153	0.0305	-0.0426	-0.1178	0.9861
22	0.0275	-0.0633	2.5475	0.0254	0.0075	0.2880	0.9171
23	0.0107	-0.0463	2.1430	0.0236	0.0909	0.8467	0.2830
24	0.0021	-0.0182	1.6879	0.0366	0.1885	1.3826	-0.9117
25	0.0021	0.0182	1.4537	0.0366	0.2659	2.0686	-3.2792
26	0.0107	0.0463	0.9986	0.0236	0.4270	2.8029	-6.8561
27	0.0275	0.0633	0.5941	0.0254	0.4885	2.8719	-7.2481
28	0.0525	0.0753	0.3263	0.0305	0.4506	2.5974	-5.7464
29	0.0852	0.0832	0.1648	0.0368	0.3927	2.2815	-4.2054
30	0.1249	0.0877	0.0710	0.0432	0.3424	2.0153	-3.0615
31	0.1710	0.0898	0.0220	0.0492	0.3031	1.8091	-2.2730
32	0.2228	0.0904	0.0031	0.0544	0.2737	1.6587	-1.7514
33	0.2793	0.0905	0.0000	0.0587	0.2534	1.5562	-1.4216
34	0.3396	0.0905	0.0000	0.0619	0.2404	1.4883	-1.2149
35	0.4027	0.0905	0.0000	0.0641	0.2317	1.4459	-1.0907
36	0.4674	0.0903	-0.0077	0.0653	0.2270	1.4292	-1.0426
37	0.5326	0.0883	-0.0518	0.0654	0.2276	1.4052	-0.9745
38	0.5973	0.0838	-0.0869	0.0644	0.2201	1.3586	-0.8458
39	0.6604	0.0774	-0.1178	0.0624	0.2122	1.3099	-0.7158
40	0.7207	0.0693	-0.1480	0.0593	0.2042	1.2604	-0.5886
41	0.7772	0.0600	-0.1795	0.0553	0.1961	1.2085	-0.4604
42	0.8290	0.0498	-0.2128	0.0503	0.1873	1.1514	-0.3258
43	0.8751	0.0390	-0.2472	0.0445	0.1774	1.0867	-0.1808
44	0.9148	0.0283	-0.2811	0.0378	0.1657	1.0121	-0.0243
45	0.9475	0.0184	-0.3121	0.0304	0.1517	0.9257	0.1431
46	0.9725	0.0101	-0.3378	0.0223	0.1345	0.8236	0.3217
47	0.9893	0.0040	-0.3562	0.0136	0.1104	0.6922	0.5209
48	0.9979	0.0008	-0.3658	0.0046	0.0620	0.1563	0.9756
49					-0.4341		

C_l = 1.8741
C_d = 0.037716
C_m = -0.49523

Table 18 Resulting Data Around Airfoil for 48 Panels, N = 3, $\alpha = 15°$

Appendix C: Change in Pressure Around Hydrofoil

$\boldsymbol{m = 24, N = 2}$:

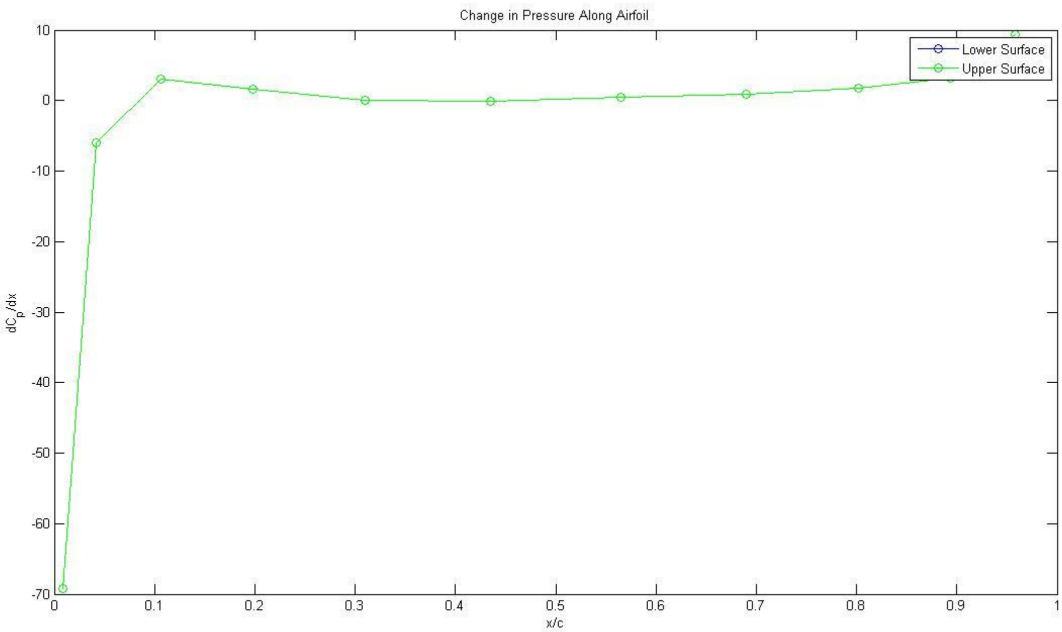

Figure 1 Change in Pressure Coefficient for 24 Panels, N = 2, $\alpha = 0°$

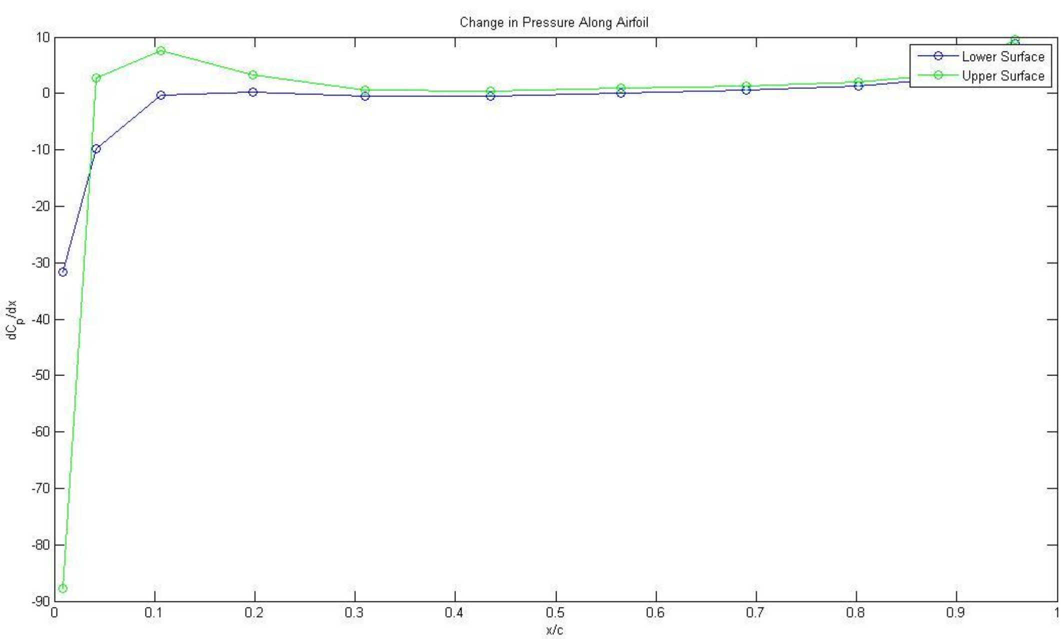

Figure 2 Change in Pressure Coefficient for 24 Panels, N = 2, $\alpha = 5°$

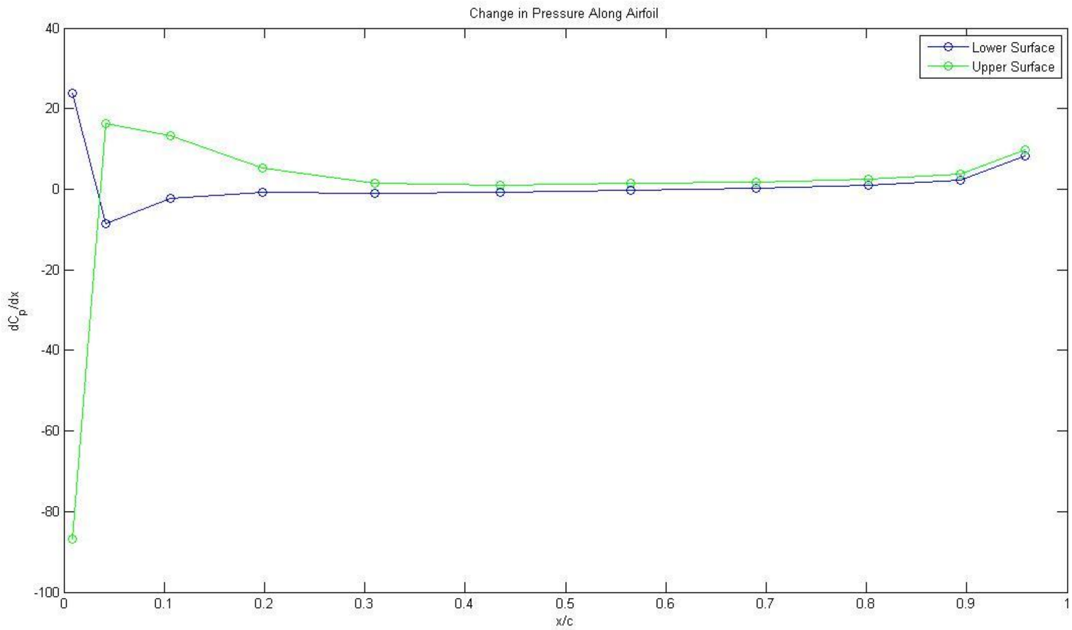

Figure 3 Change in Pressure Coefficient for 24 Panels, N = 2, $\alpha = 10°$

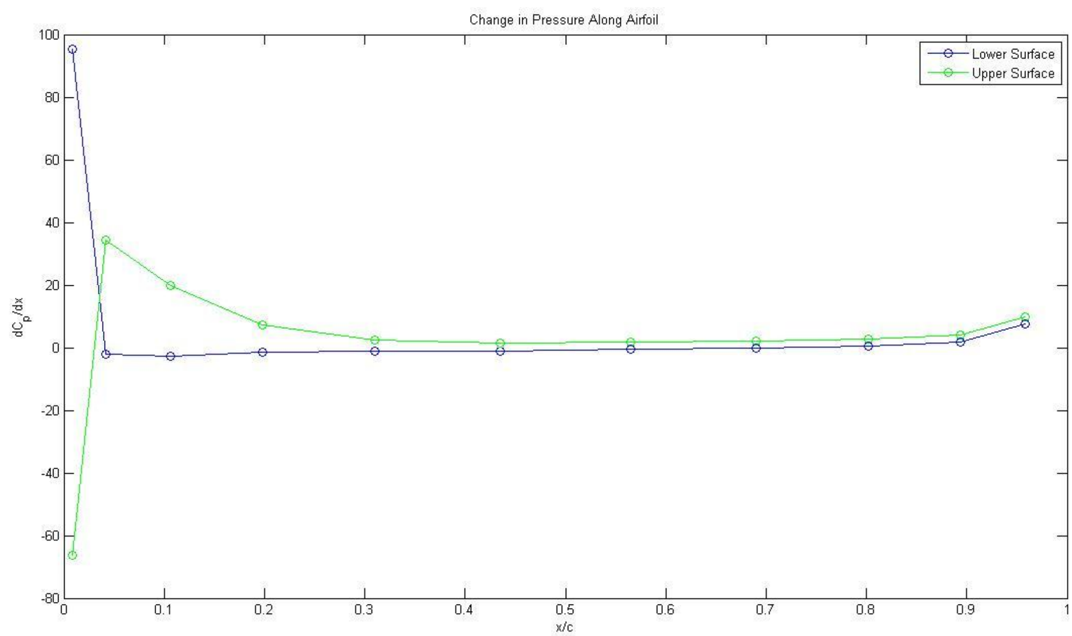

Figure 4 Change in Pressure Coefficient for 24 Panels, N = 2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 2}$:

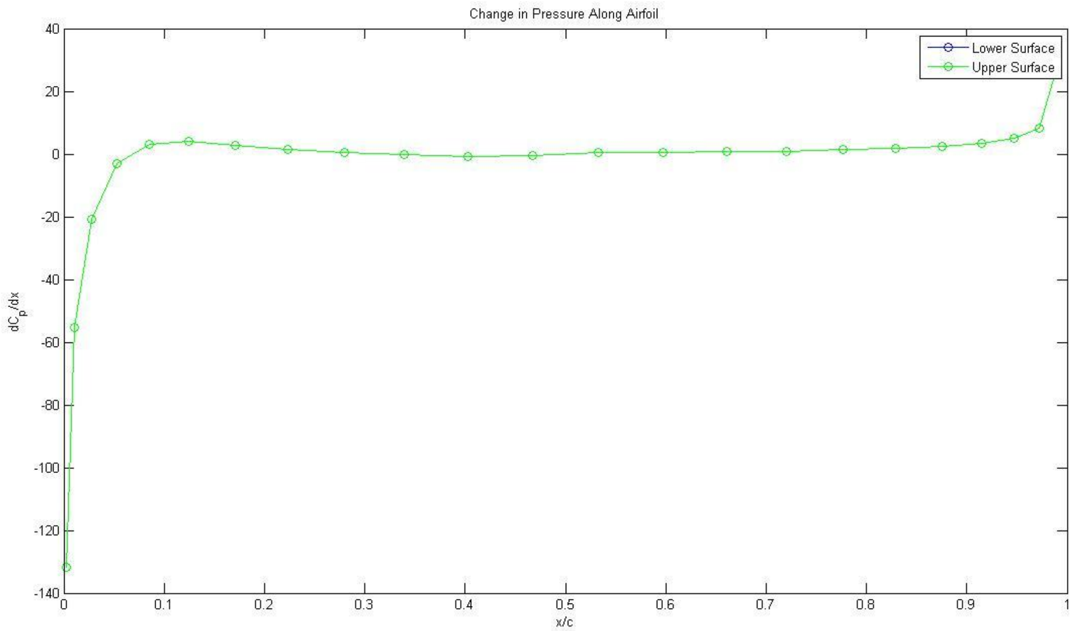

Figure 5 Change in Pressure Coefficient for 48 Panels, N = 2, $\alpha = 0°$

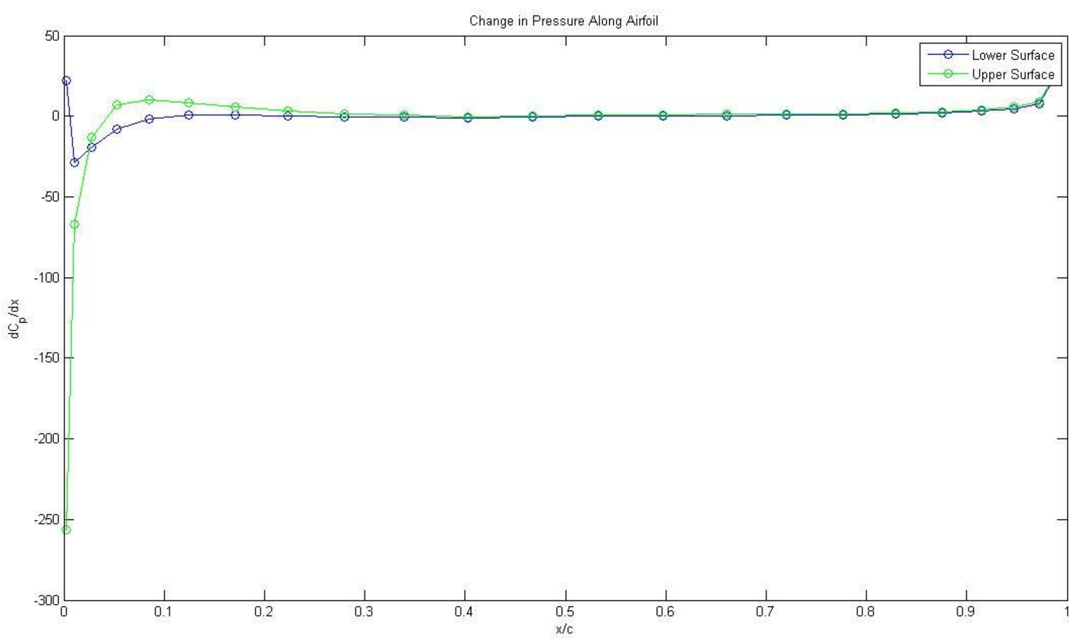

Figure 6 Change in Pressure Coefficient for 48 Panels, N = 2, $\alpha = 5°$

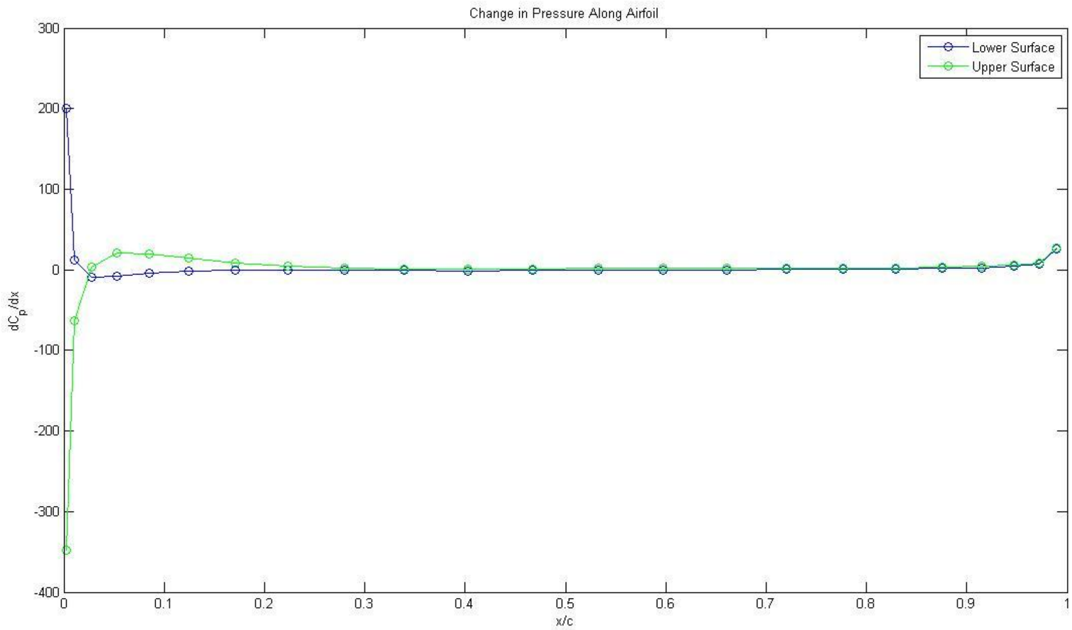

Figure 7 Change in Pressure Coefficient for 48 Panels, N = 2, $\alpha = 10°$

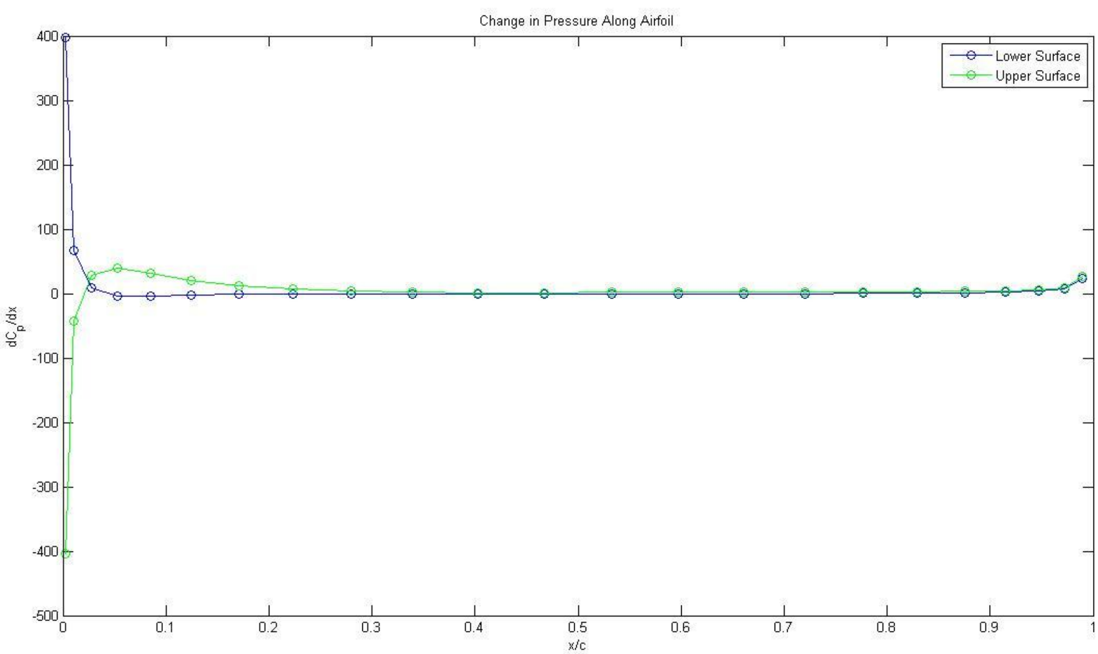

Figure 8 Change in Pressure Coefficient for 48 Panels, N = 2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 2}$: (panel configuration 2 & 3):

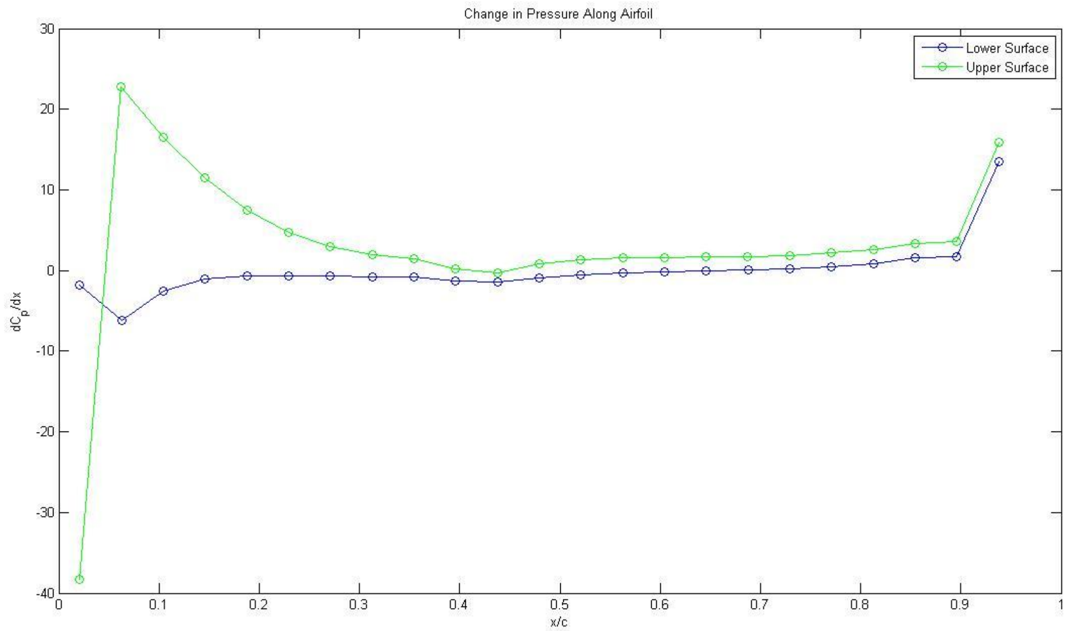

Figure 9 Change in Pressure Coefficient (Panel Configuration 2) for 48 Panels, N = 2, $\alpha = 10°$

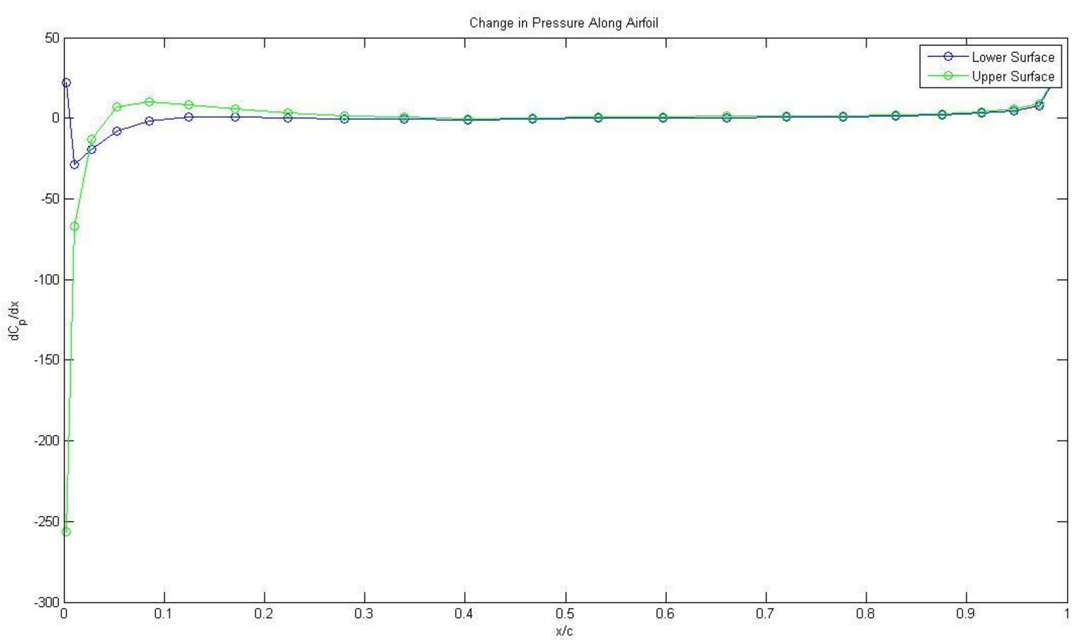

Figure 10 Change in Pressure Coefficient (Configuration 3) for 48 Panels, N = 2, $\alpha = 10°$

$\boldsymbol{m = 48, N = 3/2}$:

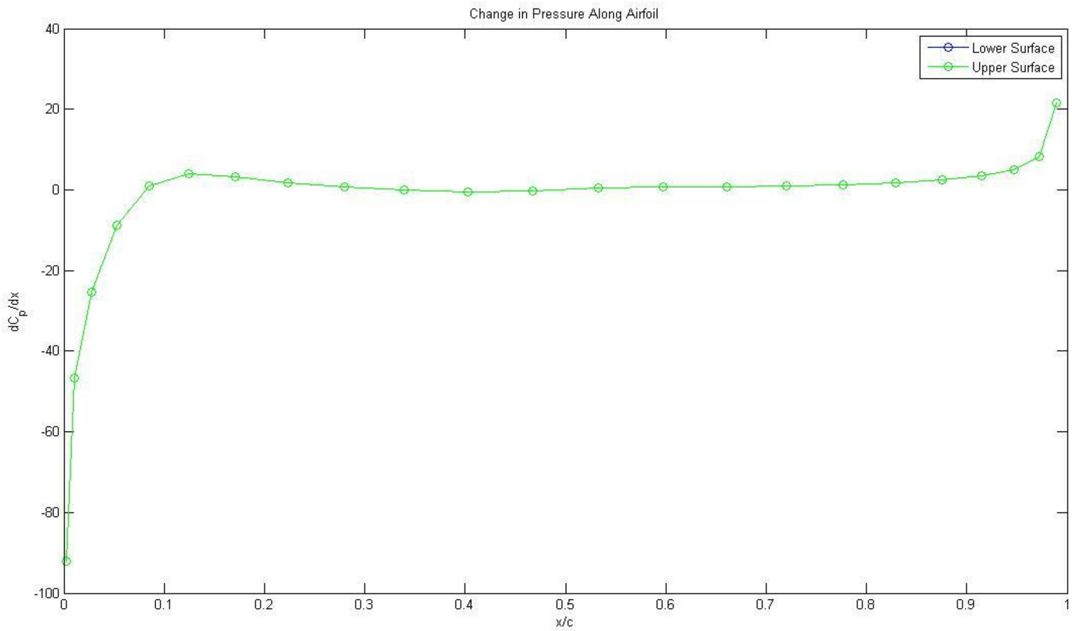

Figure 11 Change in Pressure Coefficient for 48 Panels, N = 3/2, $\alpha = 0°$

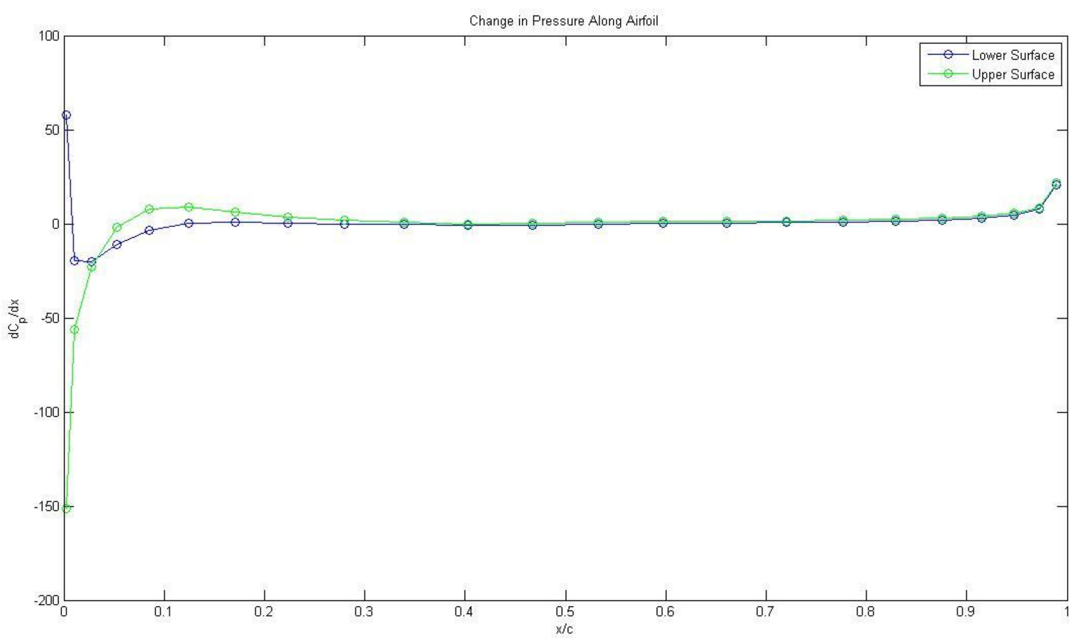

Figure 12 Change in Pressure Coefficient for 48 Panels, N = 3/2, $\alpha = 5°$

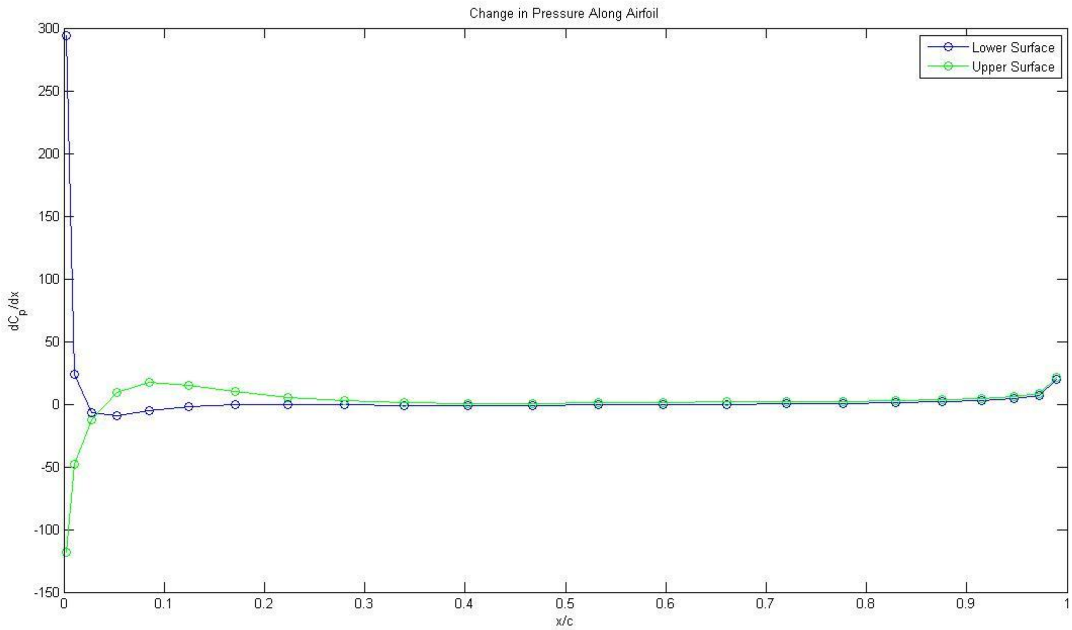

Figure 13 Change in Pressure Coefficient for 48 Panels, N = 3/2, $\alpha = 10°$

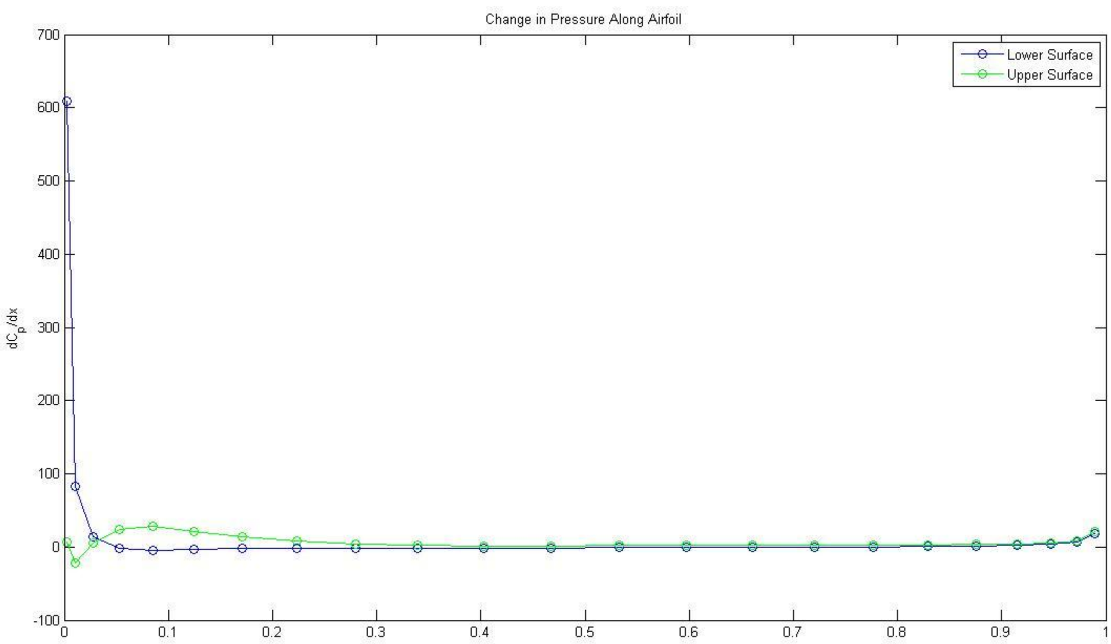

Figure 14 Change in Pressure Coefficient for 48 Panels, N = 3/2, $\alpha = 15°$

$\boldsymbol{m = 48, N = 3}$:

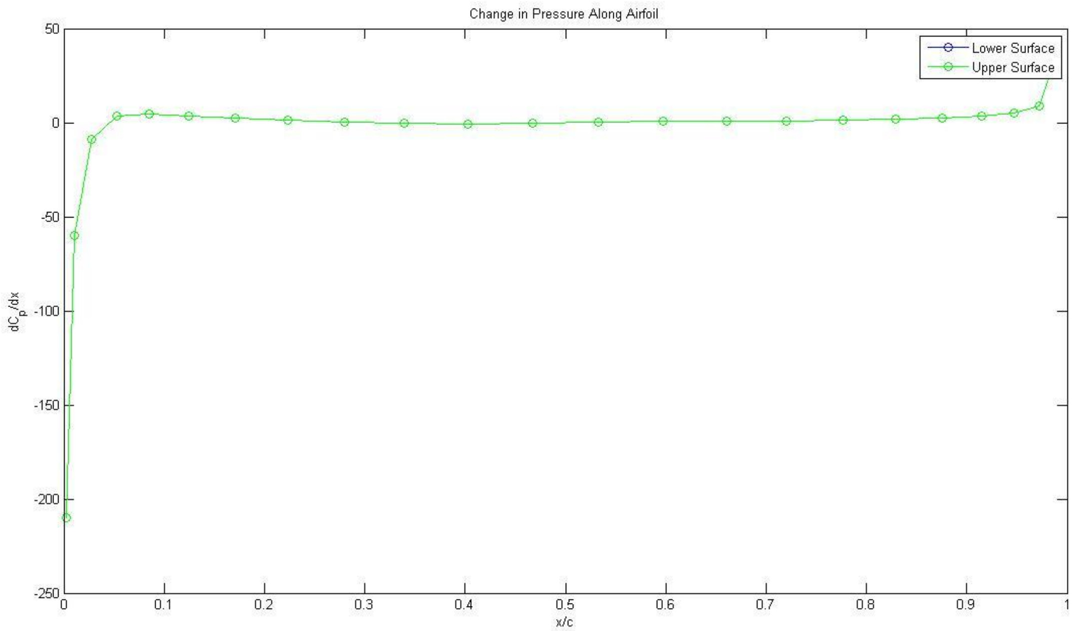

Figure 15 Change in Pressure Coefficient for 48 Panels, N = 3, $\alpha = 0°$

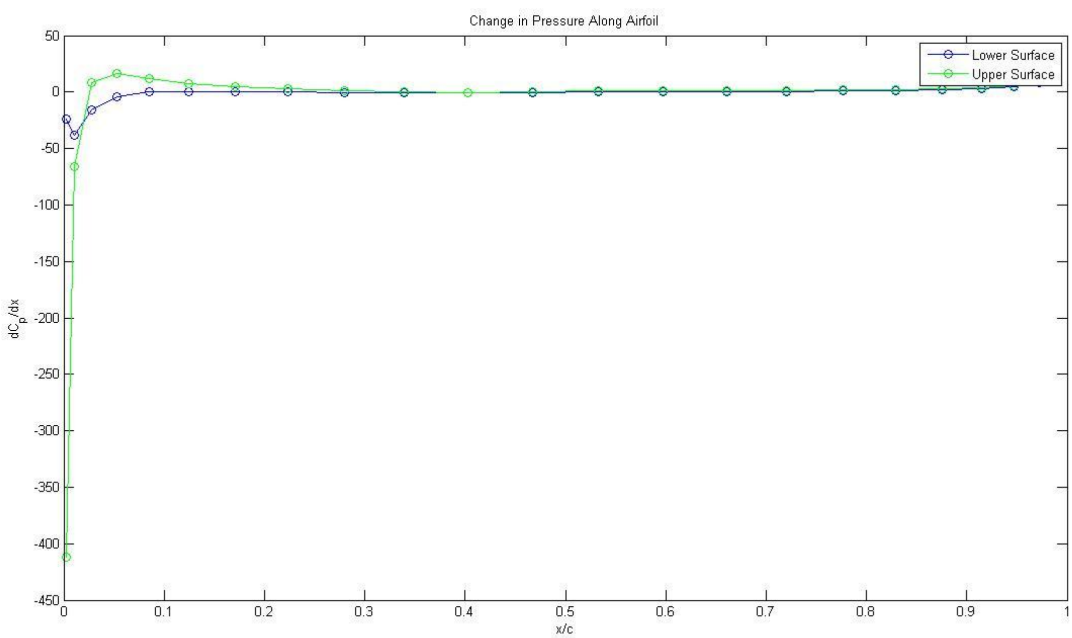

Figure 16 Change in Pressure Coefficient for 48 Panels, N = 3, $\alpha = 5°$

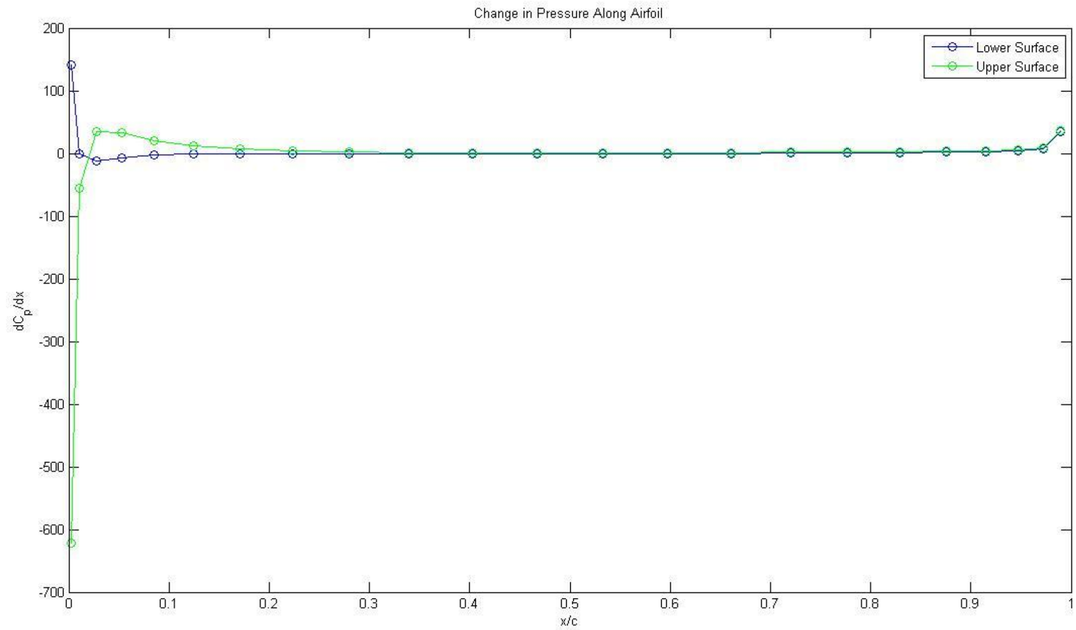

Figure 17 Change in Pressure Coefficient for 48 Panels, N = 3, $\alpha = 10°$

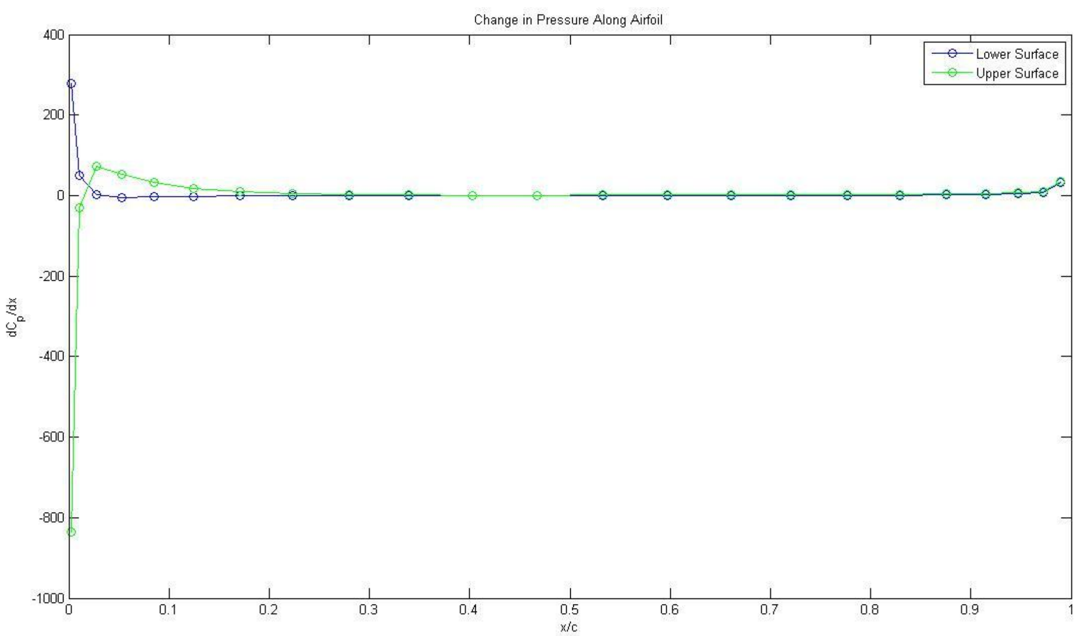

Figure 18 Change in Pressure Coefficient for 48 Panels, N = 3, $\alpha = 15°$

Appendix E: Normalized Total Circulation versus Angle of Attack

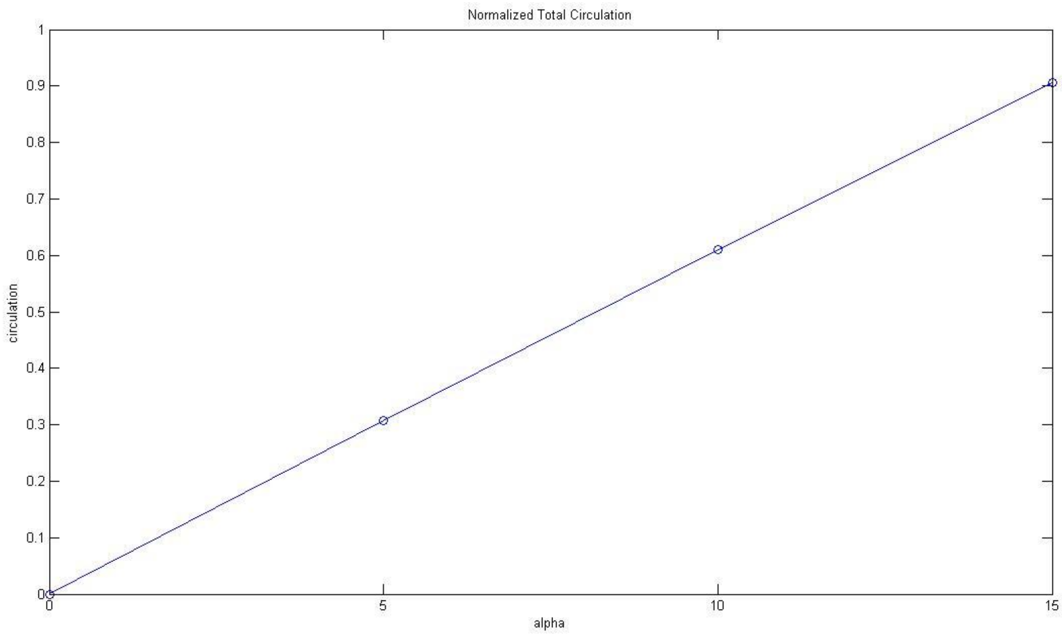

Figure 1 Normalized Total Circulation versus Angle of Attack for 24 Panels, N = 2

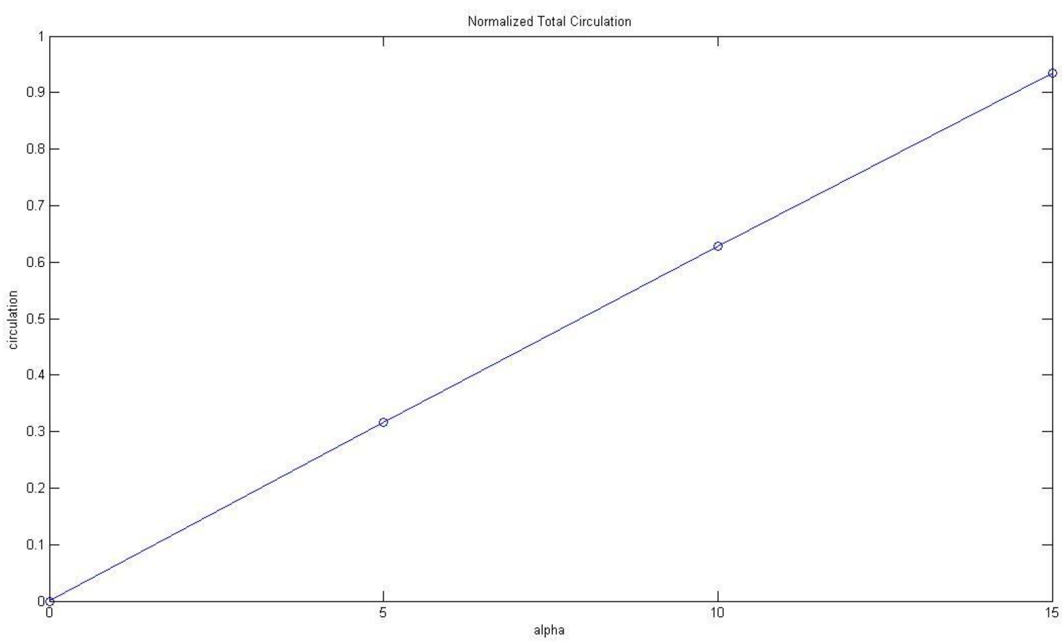

Figure 2 Normalized Total Circulation versus Angle of Attack for 48 Panels, N = 2

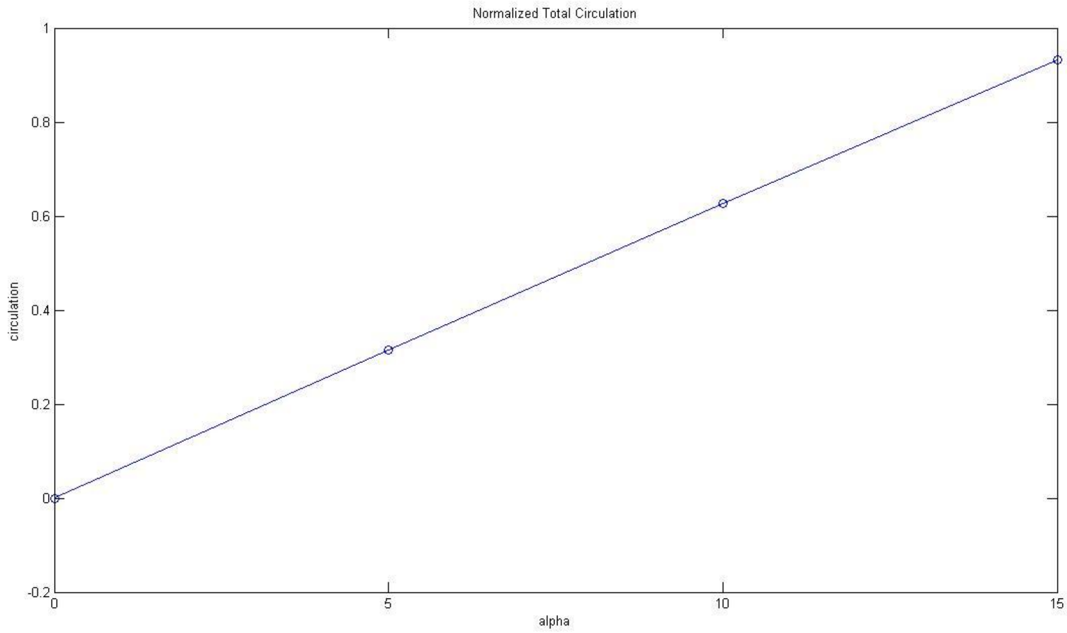

Figure 3 Normalized Total Circulation versus Angle of Attack for 48 Panels, N = 3/2

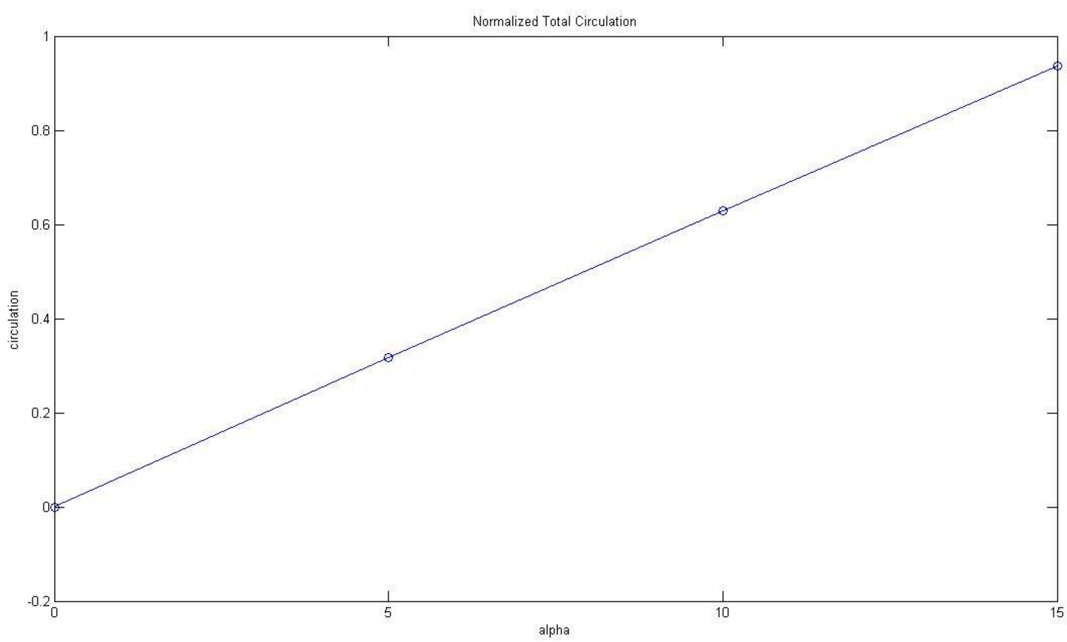

Figure 4 Normalized Total Circulation versus Angle of Attack for 48 Panels, N = 3

Appendix F: Lift Coefficient versus Angle of Attack

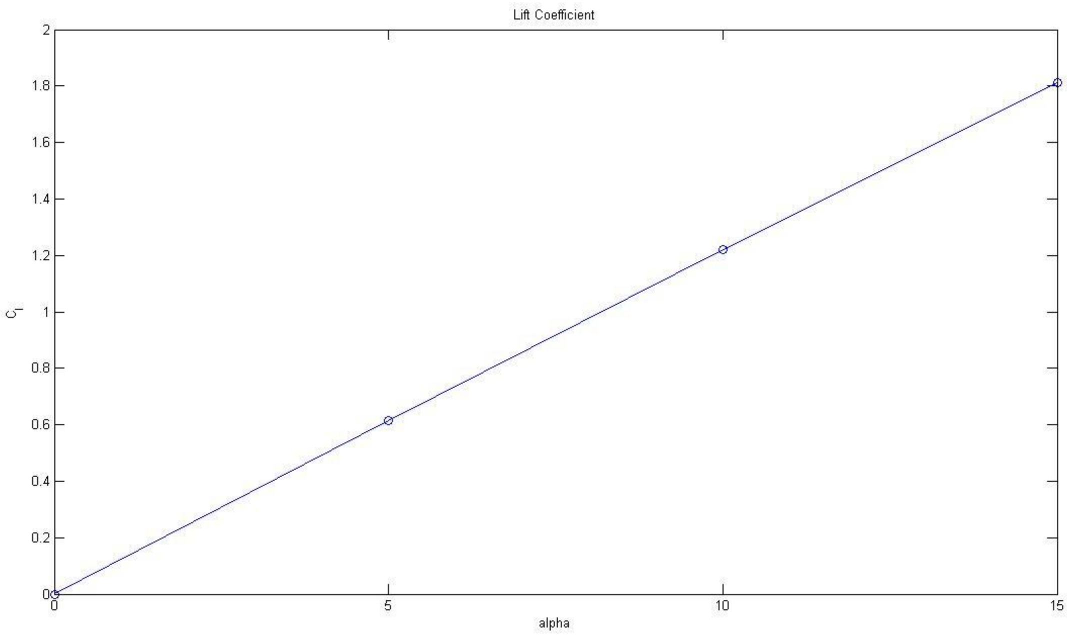

Figure 1 Lift Coefficient versus Angle of Attack for 24 Panels, N = 2

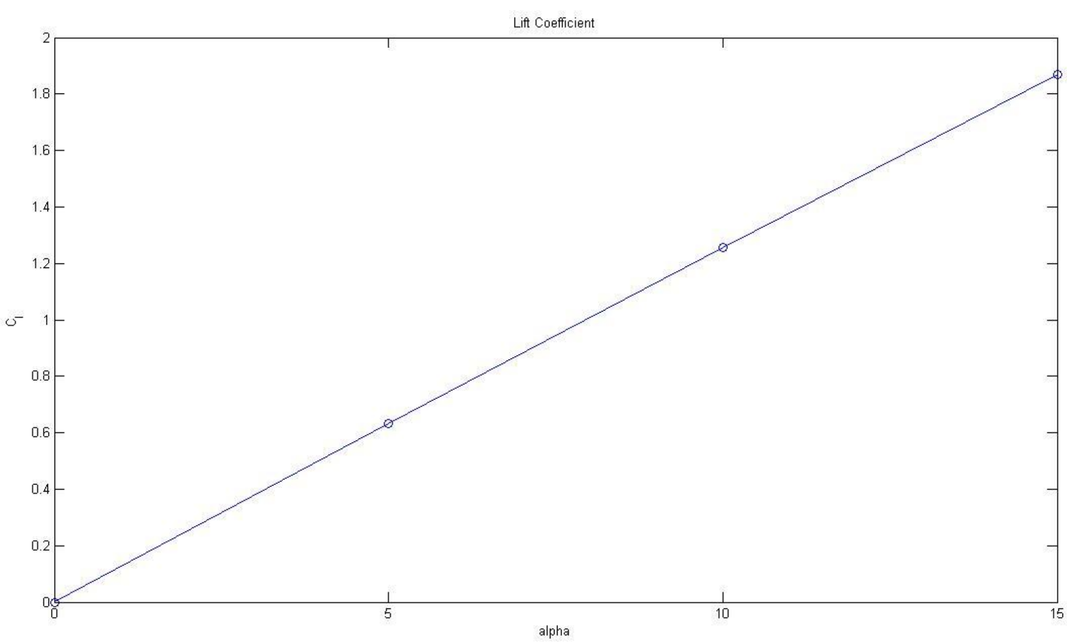

Figure 2 Lift Coefficient versus Angle of Attack for 48 Panels, N = 2

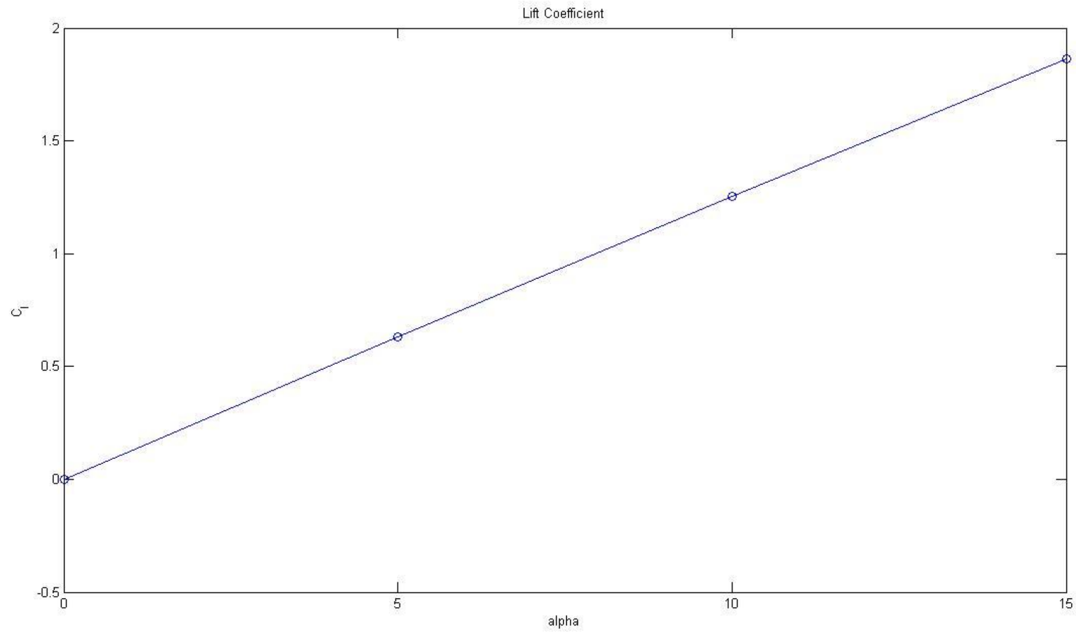

Figure 3 Lift Coefficient versus Angle of Attack for 48 Panels, N = 3/2

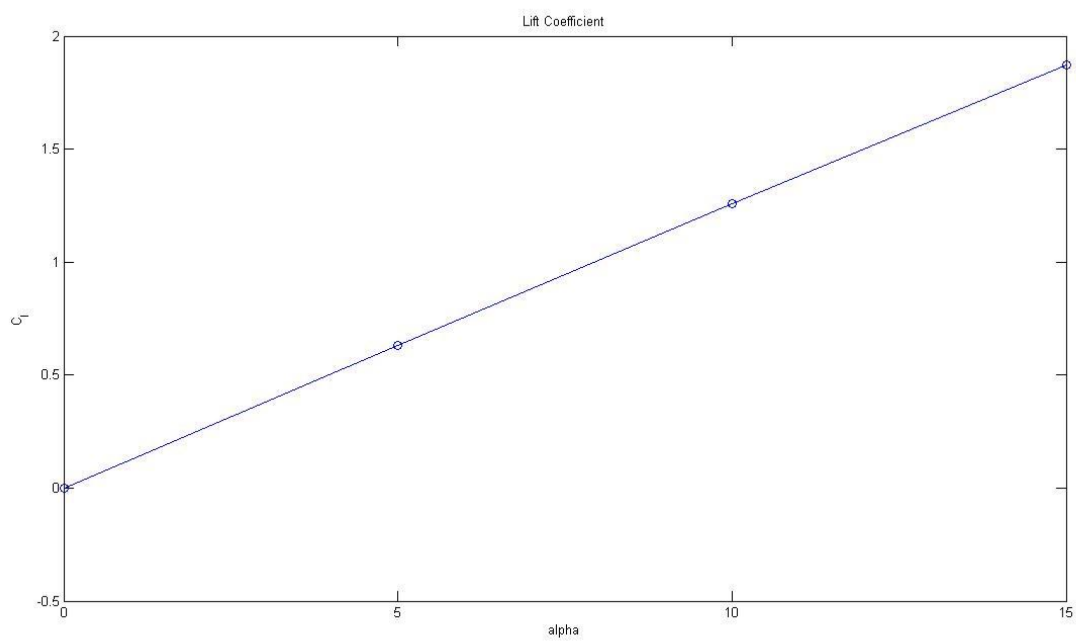

Figure 4 Lift Coefficient versus Angle of Attack for 48 Panels, N = 3

Appendix G: Moment Coefficient about ¼-Chord versus Angle of Attack

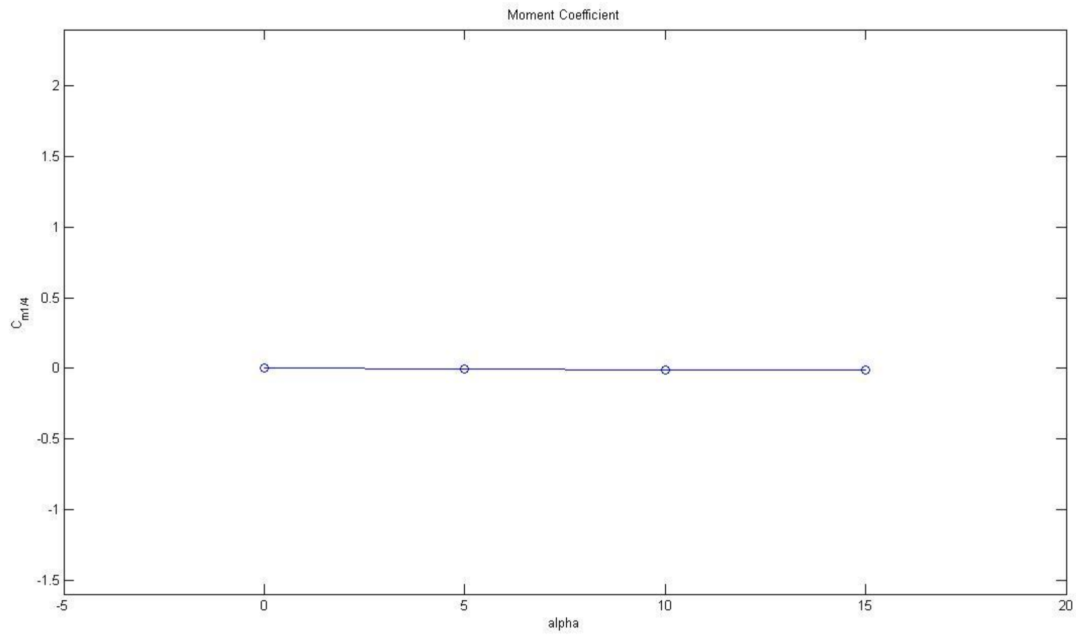

Figure 1 Moment Coefficient about ¼-Chord versus Angle of Attack for 24 Panels, N = 2

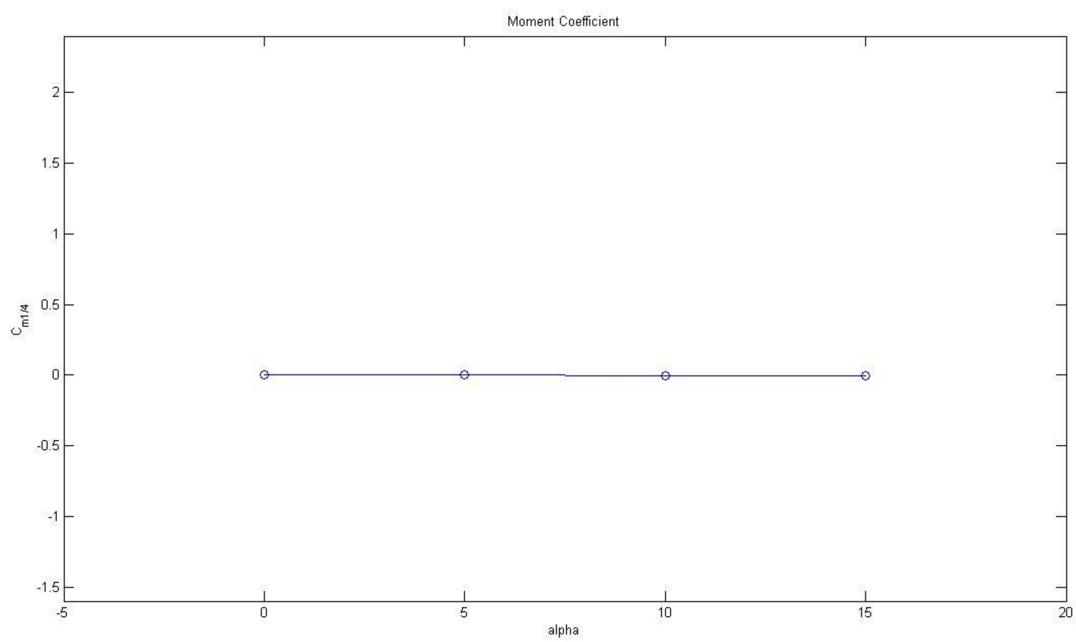

Figure 2 Moment Coefficient about ¼-Chord versus Angle of Attack for 48 Panels, N = 2

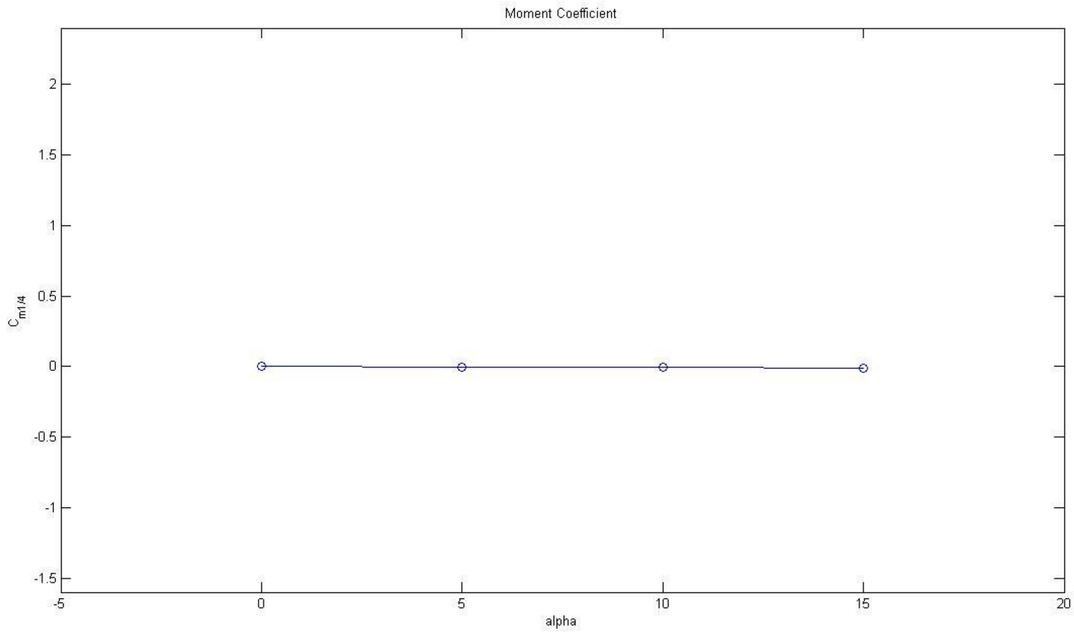

Figure 3 Moment Coefficient about ¼-Chord versus Angle of Attack for 48 Panels, N = 3/2

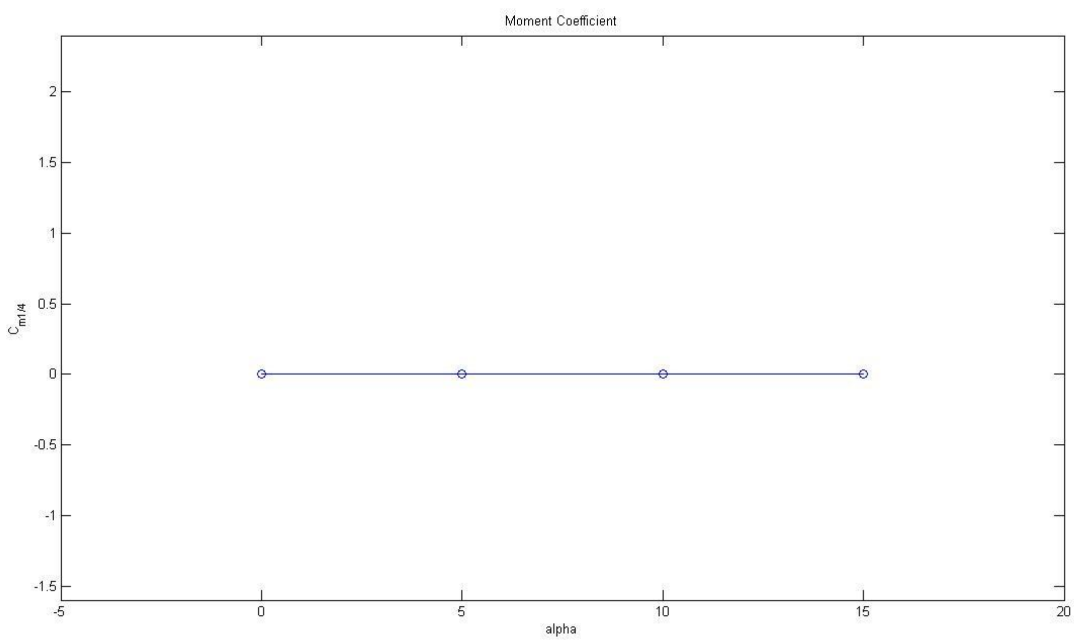

Figure 4 Moment Coefficient about ¼-Chord versus Angle of Attack for 48 Panels, N = 3

Appendix H: Comparison with Thin Airfoil Theory Lift Coefficient

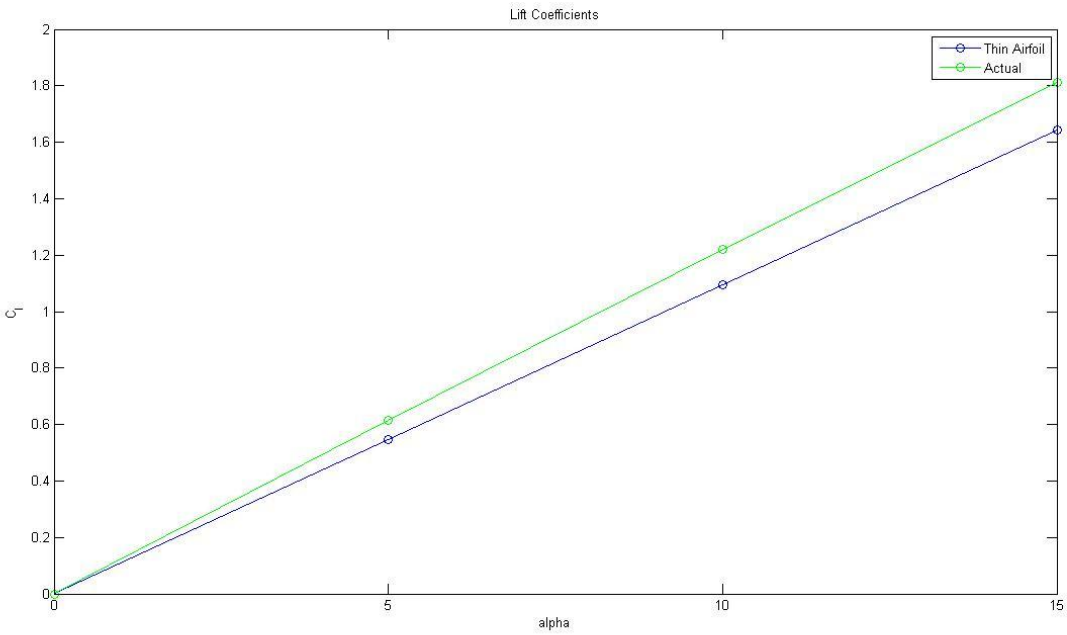

Figure 1 Lift Coefficient Comparisons versus Angle of Attack for 24 Panels, N = 2

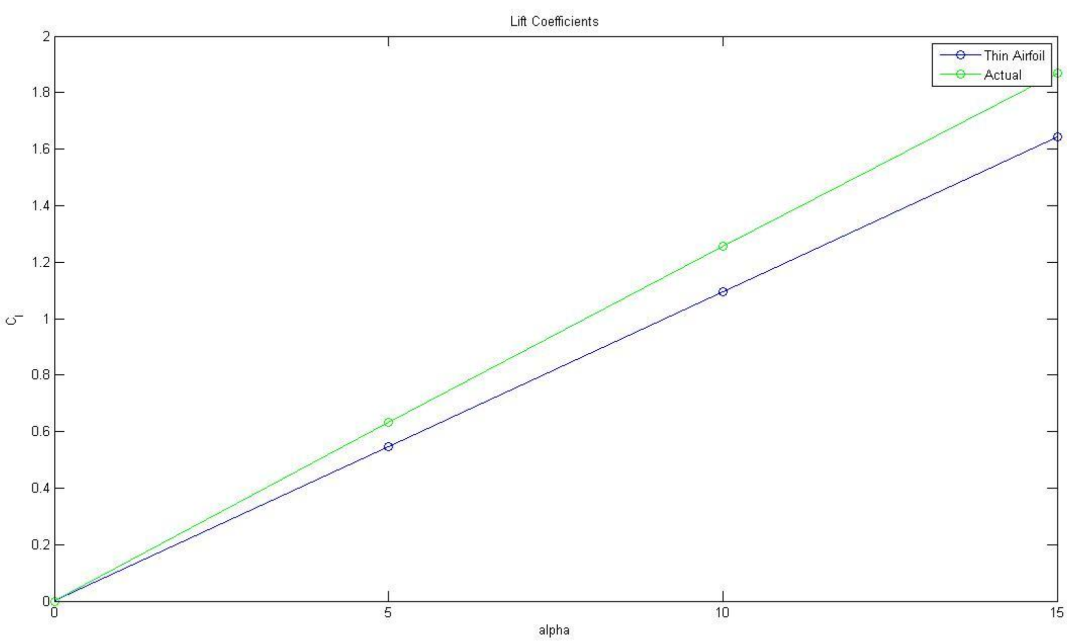

Figure 2 Lift Coefficient Comparisons versus Angle of Attack for 48 Panels, N = 2

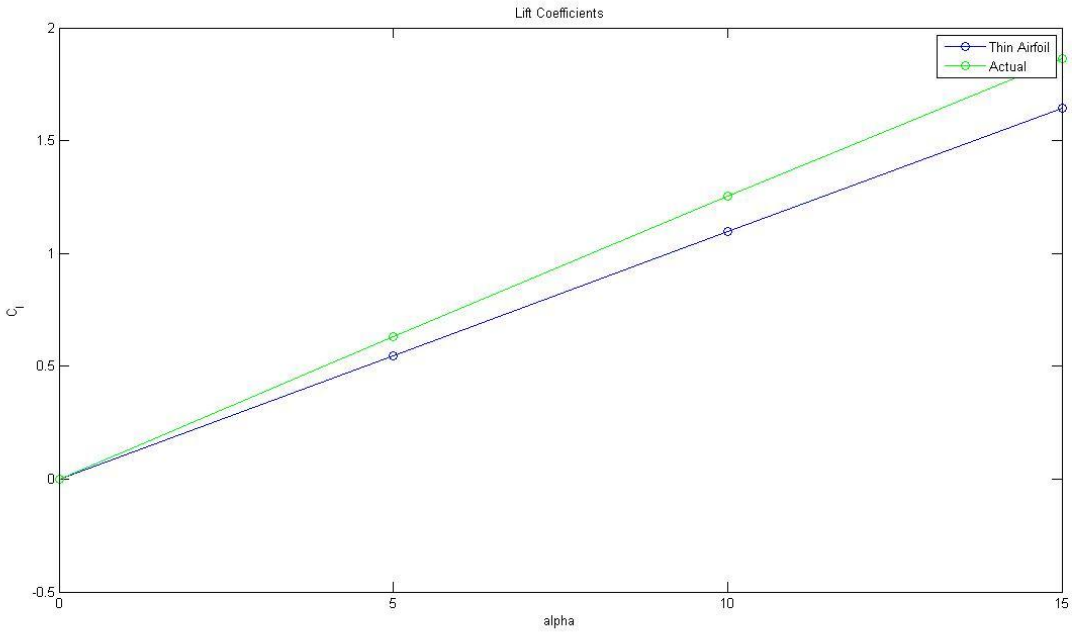

Figure 3 Lift Coefficient Comparisons versus Angle of Attack for 48 Panels, N = 3/2

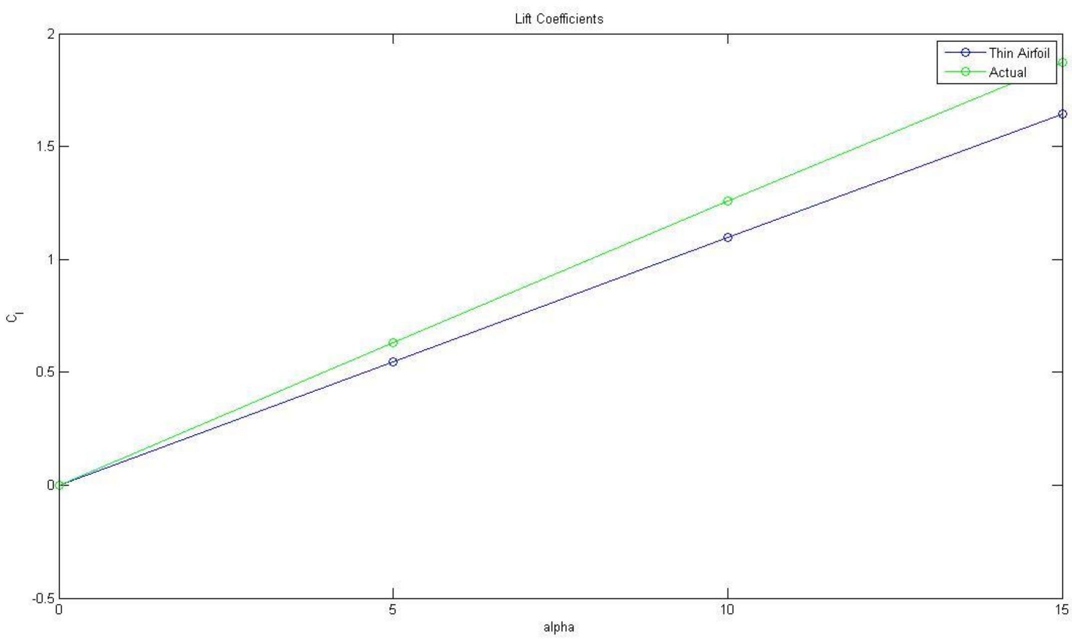

Figure 4 Lift Coefficient Comparisons versus Angle of Attack for 48 Panels, N = 3

Appendix I: MATLAB Code

DARPA 2 Contour (Configuration 1)

```
% ------------------------------------------------------------------------
% AOE 3014 (CRN: 90320)
% Aero/Hydrodynamics
% Computer Project
%
% @author   Michael D. Naper, Jr. (VT PID: NaperMD)
% @version  2010.11.05
% ------------------------------------------------------------------------

function [] = darpa2(m, N)
% Calculates the x, y, x/c, and y/c values for the contour of a DARPA 2
% submarine sail.
%
% m is the number of panels to be used when approximating the shape of the
% sail. N is the nose shape parameter.


% ------------------------------------------------------------------------
% Argument Checks

if (nargin < 2)
    error('Too few input arguments. Function takes two arguments.');
end

if (nargin > 2)
    error('Too many input arguments. Function takes two arguments.');
end

if (length(m) ~= 1)
    error('Vector of x-values must not be an empty set.');
end

if (length(N) ~= 1)
    error('Vector of y-values must not be an empty set.');
end


% ------------------------------------------------------------------------
% Input Parameters

x_lead = 3.032986;       % Location of Leading Edge (ft)
L_fore = 0.325521;       % Sail Forebody Length (ft)
L_mid = 0.200521;        % Sail Parallel Middle Body Length (ft)
L_aft = 0.682292;        % Sail Afterbody Length (ft)
c = 1.208333;            % Total Sail Length, i.e., Chord (ft)
b = 0.674479;            % Span of Sail with Uniform Profile (ft)
y_max = 0.109375;        % One-Half the Maximum Sail Thickness (ft)


% ------------------------------------------------------------------------
% Calculations

X = zeros(m + 1, 1);     % Vector of x-values
Y = zeros(m + 1, 1);     % Vector of corresponding y-values

x_mid = x_lead + c / 2;
deltaDeg = 2 * pi / m;

for arc = 1 : m + 1
    theta = (arc - 1) * -deltaDeg;

    X(arc) = x_mid + c / 2 * cos(theta);

    if (x_lead <= X(arc) && X(arc) < x_lead + L_fore)  % Sail Forebody Equation
        D = 3.072000 * (X(arc) - 3.032986);
        A = 2 * D * (D - 1)^4;
```

```
        B = 1 / 3 * D^2 * (D - 1)^3;
        C = 1 - (D - 1)^4 * (4 * D + 1);

        Y(arc) = y_max * (2.094759 * A + 0.2071781 * B + C)^(1 / N);
    elseif (x_lead + L_fore <= X(arc) && X(arc) < x_lead + L_fore + L_mid) % Sail Parallel Middle
Body Equation
        Y(arc) = y_max;
    elseif (x_lead + L_fore + L_mid <= X(arc) && X(arc) <= x_lead + L_fore + L_mid + L_aft)    % Sail
Afterbody Equation
        E = 1.025942 * (X(arc) - 3.266586);

        Y(arc) = y_max * (2.908891 * E^(1/2) - 1.234491 * E - 3.444817 * E^2 + 3.850435 * E^3 -
2.080019 * E^4);
    else
        error('Out of bounds error.');
    end

    if (theta > -pi)
            Y(arc) = -Y(arc);
    end
end


X_C = zeros(m + 1, 1);  % Vector of x/c values
Y_C = zeros(m + 1, 1);  % Vector of y/c values

for pos = 1 : m + 1
    X_C(pos) = (X(pos) - x_lead) / c;
    Y_C(pos) = Y(pos) / c;
end

disp('DARPA2 Contour Calculated:');
disp('   x values:   X');
disp('   y values:   Y');
disp('   x/c values: X_C');
disp('   y/c values: Y_C');


% -------------------------------------------------------------------------
% Plot

figure(1);
hold on;
title('DARPA2 Sail Model');
xlabel('x, ft');
ylabel('y, ft');
axis ([3, 4.3, -1, 1]);
plot(X, Y);
hold off;


% -------------------------------------------------------------------------
% Calculate Velocities and Pressure Coefficients

vortexPanel(X_C, Y_C);
```

DARPA 2 Contour (Configuration 2)

```
% -------------------------------------------------------------------------
% AOE 3014 (CRN: 90320)
% Aero/Hydrodynamics
% Computer Project
%
% @author   Michael D. Naper, Jr. (VT PID: NaperMD)
% @version  2010.11.05
% -------------------------------------------------------------------------

function [] = darpa2_config2(m, N)
% Calculates the x, y, x/c, and y/c values for the contour of a DARPA 2
% submarine sail.
%
% m is the number of panels to be used when approximating the shape of the
% sail. N is the nose shape parameter.


% -------------------------------------------------------------------------
% Argument Checks

if (nargin < 2)
    error('Too few input arguments. Function takes two arguments.');
end

if (nargin > 2)
    error('Too many input arguments. Function takes two arguments.');
end

if (length(m) ~= 1)
    error('Vector of x-values must not be an empty set.');
end

if (length(N) ~= 1)
    error('Vector of y-values must not be an empty set.');
end

% -------------------------------------------------------------------------
% Input Parameters

x_lead = 3.032986;       % Location of Leading Edge (ft)
L_fore = 0.325521;       % Sail Forebody Length (ft)
L_mid = 0.200521;        % Sail Parallel Middle Body Length (ft)
L_aft = 0.682292;        % Sail Afterbody Length (ft)
c = 1.208333;            % Total Sail Length, i.e., Chord (ft)
b = 0.674479;            % Span of Sail with Uniform Profile (ft)
y_max = 0.109375;        % One-Half the Maximum Sail Thickness (ft)


% -------------------------------------------------------------------------
% Calculations

X = zeros(m + 1, 1);     % Vector of x-values
Y = zeros(m + 1, 1);     % Vector of corresponding y-values

x_tail = x_lead + c;
deltaX = 2 * c / m;

for pos = 1 : m / 2
    X(pos) = x_tail - (pos - 1) * deltaX;

    if (x_lead <= X(pos) && X(pos) < x_lead + L_fore)  % Sail Forebody Equation
        D = 3.072000 * (X(pos) - 3.032986);
        A = 2 * D * (D - 1)^4;
        B = 1 / 3 * D^2 * (D - 1)^3;
        C = 1 - (D - 1)^4 * (4 * D + 1);

        Y(pos) = -y_max * (2.094759 * A + 0.2071781 * B + C)^(1 / N);
    elseif (x_lead + L_fore <= X(pos) && X(pos) < x_lead + L_fore + L_mid) % Sail Parallel Middle
Body Equation
```

```
            Y(pos) = -y_max;
        elseif (x_lead + L_fore + L_mid <= X(pos) && X(pos) <= x_lead + c)     % Sail Afterbody Equation
            E = 1.025942 * (X(pos) - 3.266586);

            Y(pos) = -y_max * (2.908891 * E^(1/2) - 1.234491 * E - 3.444817 * E^2 + 3.850435 * E^3 -
    2.080019 * E^4);
        else
            error('Out of bounds error.');
        end
    end

    for pos = m / 2 + 1 : m + 1
        X(pos) = x_lead + (pos - m / 2 - 1) * deltaX;

        if (x_lead <= X(pos) && X(pos) < x_lead + L_fore)  % Sail Forebody Equation
            D = 3.072000 * (X(pos) - 3.032986);
            A = 2 * D * (D - 1)^4;
            B = 1 / 3 * D^2 * (D - 1)^3;
            C = 1 - (D - 1)^4 * (4 * D + 1);

            Y(pos) = y_max * (2.094759 * A + 0.2071781 * B + C)^(1 / N);
        elseif (x_lead + L_fore <= X(pos) && X(pos) < x_lead + L_fore + L_mid) % Sail Parallel Middle
    Body Equation
            Y(pos) = y_max;
        elseif (x_lead + L_fore + L_mid <= X(pos) && X(pos) <= x_lead + L_fore + L_mid + L_aft)    % Sail
    Afterbody Equation
            E = 1.025942 * (X(pos) - 3.266586);

            Y(pos) = y_max * (2.908891 * E^(1/2) - 1.234491 * E - 3.444817 * E^2 + 3.850435 * E^3 -
    2.080019 * E^4);
        else
            error('Out of bounds error.');
        end
    end


    X_C = zeros(m + 1, 1);  % Vector of x/c values
    Y_C = zeros(m + 1, 1);  % Vector of y/c values

    for pos = 1 : m + 1
        X_C(pos) = (X(pos) - x_lead) / c;
        Y_C(pos) = Y(pos) / c;
    end

    disp('DARPA2 Contour Calculated:');
    disp('   x values:   X');
    disp('   y values:   Y');
    disp('   x/c values: X_C');
    disp('   y/c values: Y_C');


    % -------------------------------------------------------------------------
    % Plot

    figure(1);
    hold on;
    title('DARPA2 Sail Model');
    xlabel('x, ft');
    ylabel('y, ft');
    axis ([3, 4.3, -1, 1]);
    plot(X, Y);
    hold off;


    % -------------------------------------------------------------------------
    % Calculate Velocities and Pressure Coefficients

vortexPanel(X_C, Y_C);
```

DARPA 2 Contour (Configuration 3)

```
% -------------------------------------------------------------------------
% AOE 3014 (CRN: 90320)
% Aero/Hydrodynamics
% Computer Project
%
% @author   Michael D. Naper, Jr. (VT PID: NaperMD)
% @version  2010.11.05
% -------------------------------------------------------------------------

function [] = darpa2_config3(m, N)
% Calculates the x, y, x/c, and y/c values for the contour of a DARPA 2
% submarine sail.
%
% m is the number of panels to be used when approximating the shape of the
% sail. N is the nose shape parameter.


% -------------------------------------------------------------------------
% Argument Checks

if (nargin < 2)
    error('Too few input arguments. Function takes two arguments.');
end

if (nargin > 2)
    error('Too many input arguments. Function takes two arguments.');
end

if (length(m) ~= 1)
    error('Vector of x-values must not be an empty set.');
end

if (length(N) ~= 1)
    error('Vector of y-values must not be an empty set.');
end

% -------------------------------------------------------------------------
% Input Parameters

x_lead = 3.032986;       % Location of Leading Edge (ft)
L_fore = 0.325521;       % Sail Forebody Length (ft)
L_mid = 0.200521;        % Sail Parallel Middle Body Length (ft)
L_aft = 0.682292;        % Sail Afterbody Length (ft)
c = 1.208333;            % Total Sail Length, i.e., Chord (ft)
b = 0.674479;            % Span of Sail with Uniform Profile (ft)
y_max = 0.109375;        % One-Half the Maximum Sail Thickness (ft)


% -------------------------------------------------------------------------
% Calculations

Y = zeros(m + 1, 1);     % Vector of corresponding y-values

x_tail = x_lead + L_fore + L_mid + L_aft;

X = [x_tail - (0 : 5 * L_aft / (m - 3) : L_aft)';
    x_tail - L_aft - (0 : 10 * L_mid / (m - 3) : L_mid)';
    x_tail - L_aft - L_mid - (0 : 5 * L_fore / (m - 3) : L_fore)';
    x_lead + (0 : 5 * L_fore / (m - 4) : L_fore)';
    x_lead + L_fore + (0 : 10 * L_mid / (m - 4) : L_mid)';
    x_lead + L_fore + L_mid + (0 : 5 * L_aft / (m - 4) : L_aft)';
    x_tail];

sign = -1;

for pos = 1 : length(X)
    if (sign ~= 1 && pos > 1 && X(pos) > X(pos - 1))
            sign = 1;
    end
```

```
    if (x_lead <= X(pos) && X(pos) < x_lead + L_fore)  % Sail Forebody Equation
        D = 3.072000 * (X(pos) - 3.032986);
        A = 2 * D * (D - 1)^4;
        B = 1 / 3 * D^2 * (D - 1)^3;
        C = 1 - (D - 1)^4 * (4 * D + 1);

        Y(pos) = sign * y_max * (2.094759 * A + 0.2071781 * B + C)^(1 / N);
    elseif (x_lead + L_fore <= X(pos) && X(pos) < x_lead + L_fore + L_mid) % Sail Parallel Middle
Body Equation
        Y(pos) = sign * y_max;
    elseif (x_lead + L_fore + L_mid <= X(pos) && X(pos) <= x_lead + L_fore + L_mid + L_aft)    % Sail
Afterbody Equation
        E = 1.025942 * (X(pos) - 3.266586);

        Y(pos) = sign * y_max * (2.908891 * E^(1/2) - 1.234491 * E - 3.444817 * E^2 + 3.850435 * E^3
- 2.080019 * E^4);
    else
        error('Out of bounds error.');
    end
end


X_C = zeros(m + 1, 1);  % Vector of x/c values
Y_C = zeros(m + 1, 1);  % Vector of y/c values

for pos = 1 : m + 1
    X_C(pos) = (X(pos) - x_lead) / c;
    Y_C(pos) = Y(pos) / c;
end

disp('DARPA2 Contour Calculated:');
disp('   x values:   X');
disp('   y values:   Y');
disp('   x/c values: X_C');
disp('   y/c values: Y_C');


% ----------------------------------------------------------------------
% Plot

figure(1);
hold on;
title('DARPA2 Sail Model');
xlabel('x, ft');
ylabel('y, ft');
axis ([3, 4.3, -1, 1]);
plot(X, Y);
hold off;


% ----------------------------------------------------------------------
% Calculate Velocities and Pressure Coefficients

vortexPanel(X_C, Y_C);
```

Vortex Panel Method

```
% -----------------------------------------------------------------------
% AOE 3014 (CRN: 90320)
% Aero/Hydrodynamics
% Computer Project
%
% @author   Michael D. Naper, Jr. (VT PID: NaperMD)
% @version  2010.11.05
% -----------------------------------------------------------------------

function [] = vortexPanel(X, Y)
% Calculates the velocity and pressure coefficients around an airfoil whose
% contour is approximated by vortex panels of linearly varying strengths.
%
% X is the vector of x/c values of the contour from tail to nose to tail. Y
% is the vector of y/c values of the contour from tail to nose to tail.


% -----------------------------------------------------------------------
% Argument Checks

if (nargin < 2)
    error('Too few input arguments. Function takes two arguments.');
end

if (nargin > 2)
    error('Too many input arguments. Function takes two arguments.');
end

if (isempty(X))
    error('Vector of x-values must not be an empty set.');
end

if (isempty(Y))
    error('Vector of y-values must not be an empty set.');
end

[m_x, n_x] = size(X);
[m_y, n_y] = size(Y);

if (m_x ~= 1 && n_x ~= 1)
    error('Vector of x-values must be a one-dimensional array.');
end

if (m_y ~= 1 && n_y ~= 1)
    error('Vector of y-values must be a one-dimensional array.');
end

if (length(X) ~= length(Y))
    error('Vector of x-values and vector of y-values must have the same number of elements.');
end


% -----------------------------------------------------------------------
% Initialize Values:

if (n_x ~= 1)
    X = X';
end

if (n_y ~= 1)
    Y = Y';
end

numPoints = length(X);  % Number of points that approximate airfoil

A_n = zeros(numPoints);
A_t = zeros(numPoints);

C_n1 = zeros(numPoints - 1);
```

```
C_n2 = zeros(numPoints - 1);
C_t1 = zeros(numPoints - 1);
C_t2 = zeros(numPoints - 1);

V = zeros(numPoints - 1, 1);    % Vector of velocities
C_p = zeros(numPoints - 1, 1);  % Vector of pressure coefficients


% -------------------------------------------------------------------------
% Calculations:

X_mid = (X(2 : numPoints) + X(1 : numPoints - 1)) / 2;
Y_mid = (Y(2 : numPoints) + Y(1 : numPoints - 1)) / 2;

dX = X(2 : numPoints) - X(1 : numPoints - 1);
dY = Y(2 : numPoints) - Y(1 : numPoints - 1);

dS = sqrt(dX .* dX + dY .* dY);

THETA = atan2(Y(2 : numPoints) - Y(1 : numPoints - 1), X(2 : numPoints) - X(1 : numPoints - 1));

for i = 1 : numPoints - 1
    for j = 1 : numPoints - 1
        if (i == j)
            C_n1(i, j) = -1.0;
            C_n2(i, j) = 1.0;
            C_t1(i, j) = 0.5 * pi;
            C_t2(i, j) = 0.5 * pi;
        else
            A = -(X_mid(i) - X(j)) * cos(THETA(j)) - (Y_mid(i) - Y(j)) * sin(THETA(j));
            B = (X_mid(i) - X(j))^2 + (Y_mid(i) - Y(j))^2;
            C = sin(THETA(i) - THETA(j));
            D = cos(THETA(i) - THETA(j));
            E = (X_mid(i) - X(j)) * sin(THETA(j)) - (Y_mid(i) - Y(j)) * cos(THETA(j));
            F = log(1 + dS(j) * (dS(j) + 2 * A) / B);
            G = atan2(E * dS(j), B + A * dS(j));

            P = (X_mid(i) - X(j)) * sin(THETA(i) - 2 * THETA(j)) + (Y_mid(i) - Y(j)) * cos(THETA(i) -
2 * THETA(j));
            Q = (X_mid(i) - X(j)) * cos(THETA(i) - 2 * THETA(j)) - (Y_mid(i) - Y(j)) * sin(THETA(i) -
2 * THETA(j));

            C_n2(i, j) = D + 0.5 * Q * F / dS(j) - (A * C + D * E) * G / dS(j);
            C_n1(i, j) = 0.5 * D * F + C * G - C_n2(i, j);
            C_t2(i, j) = C + 0.5 * P * F / dS(j) + (A * D - C * E) * G / dS(j);
            C_t1(i, j) = 0.5 * C * F - D * G - C_t2(i, j);
        end
    end
end

for i = 1 : numPoints - 1
    A_n(i, 1) = C_n1(i, 1);
    A_t(i, 1) = C_t1(i, 1);

    for j = 2 : numPoints - 1
        A_n(i, j) = C_n1(i, j) + C_n2(i, j - 1);
        A_t(i, j) = C_t1(i, j) + C_t2(i, j - 1);
    end

    A_n(i, numPoints) = C_n2(i, numPoints - 1);
    A_t(i, numPoints) = C_t2(i, numPoints - 1);
end

A_n(numPoints, 1) = 1;
A_n(numPoints, numPoints) = 1;


% -------------------------------------------------------------------------
% Input and Output:

alpha = input('Enter an angle of attack between -90 and 90 degrees: ');
```

```
while (alpha >= -90 && alpha <= 90)
    alpha_rad = alpha * pi / 180;

    RHS = [sin(THETA - alpha_rad); 0];

    GAMMA = A_n \ RHS;

    for i = 1 : numPoints - 1
        V(i) = cos(THETA(i) - alpha_rad);

        for j = 1 : numPoints
            V(i) = V(i) + A_t(i, j) * GAMMA(j);
        end
    end

    C_p = 1 - V .* V;

    C_fx = C_p' * dY;
    C_fy = -C_p' * dX;

    C_m = C_p' * (dX .* X_mid + dY .* Y_mid);

    C_d = C_fx * cos(alpha_rad) + C_fy * sin(alpha_rad);
    C_l = C_fy * cos(alpha_rad) - C_fx * sin(alpha_rad);

    disp('    ');
    disp('                    Resulting Data Around Airfoil');
    disp('    ');
    disp('     i     x        y     theta       s     gamma       v       Cp');
    disp('    --- -------- --------- --------  -------- --------- --------- --------');

    for i = 1 : numPoints
        if i == numPoints
            outString = sprintf('    %2d                                     %7.4f', i, GAMMA(i));

            disp(outString);
        else
            outString = sprintf('    %2d  %.4f  %7.4f  %7.4f  %7.4f  %7.4f  %7.4f  %7.4f', i,
X_mid(i), Y_mid(i), THETA(i), dS(i), GAMMA(i), V(i), C_p(i));

            disp(outString);
        end
    end

    disp('     ');
    disp(['    C_l = ', num2str(C_l)]);
    disp(['    C_d = ', num2str(C_d)]);
    disp(['    C_m = ', num2str(C_m)]);


    filename = input('Enter filename of where to write calculated data: ', 's');
    filename = strcat(filename, '.txt');

    outStream = fopen(filename, 'w');

    fprintf(outStream, '\n');
    fprintf(outStream, '\t\t\t\tResulting Data Around Airfoil\n');
    fprintf(outStream, '\n');
    fprintf(outStream, '\t i\t   x\t   y\t theta\t\ts\t gamma\t   v\t   Cp\n');
    fprintf(outStream, '\t---\t------\t-------\t-------\t-------\t-------\t-------\t-------\n');

    for i = 1 : numPoints
        if i == numPoints
            fprintf(outStream, '\t%2d\t\t\t\t\t\t\t\t\t%7.4f\n', i, GAMMA(i));
        else
            fprintf(outStream, '\t%2d\t%.4f\t%7.4f\t%7.4f\t%7.4f\t%7.4f\t%7.4f\t%7.4f\n', i,
X_mid(i), Y_mid(i), THETA(i), dS(i), GAMMA(i), V(i), C_p(i));
        end
    end

    fprintf(outStream, '\n');
    fprintf(outStream, '\tC_l = %s\n', num2str(C_l));
```

```
    fprintf(outStream, '\tC_d = %s\n', num2str(C_d));
    fprintf(outStream, '\tC_m = %s', num2str(C_m));

    fclose(outStream);


    % ----------------------------------------------------------------------
    % Plot:

    figure(2);

    subplot(2, 2, 1), plot(X, Y, 'b-', X, Y, 'ro');
    subplot(2, 2, 1), axis([-.5, 1.5, -.5, .5]);
    subplot(2, 2, 1), title('Shape of Airfoil');
    subplot(2, 2, 1), xlabel('x/c');
    subplot(2, 2, 1), ylabel('y/c');
    subplot(2, 2, 4), hold on;

    subplot(2, 2, 3), plot([X_mid; X_mid(1)], -[C_p; C_p(1)], 'b-', [X_mid; X_mid(1)], -[C_p;
C_p(1)], 'ro');
    subplot(2, 2, 3), title('Pressures');
    subplot(2, 2, 3), xlabel('x/c');
    subplot(2, 2, 3), ylabel('-Cp');

    subplot(2, 2, 4), plot(alpha, C_l, 'bd');
    subplot(2, 2, 4), plot(alpha, C_d, 'rx');
    subplot(2, 2, 4), plot(alpha, C_m, 'g*');
    subplot(2, 2, 4), title('Force and Moment Coefficients');
    subplot(2, 2, 4), xlabel('Angle of Attack (degrees)');
    subplot(2, 2, 4), ylabel('C_l, C_d, C_m');
    legend('C_l', 'C_d', 'C_m');

    alpha = input('\nEnter another angle of attack between -90 and 90 degrees:\n(Enter an angle of
attack outside this range to quit.)    ');
end
```