MAE 424

AERODYNAMICS

PROJECT 2

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<u>Question 1:</u> NACA 2412 Airfoil

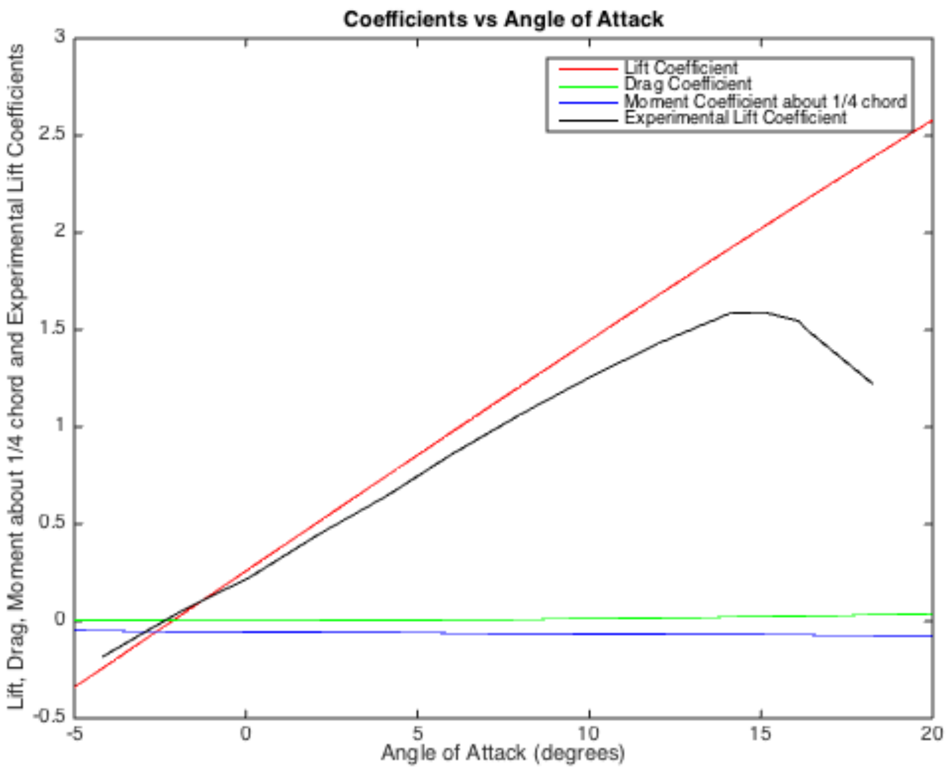

Figure 1: Various Coefficients vs Angle of Attack in degrees

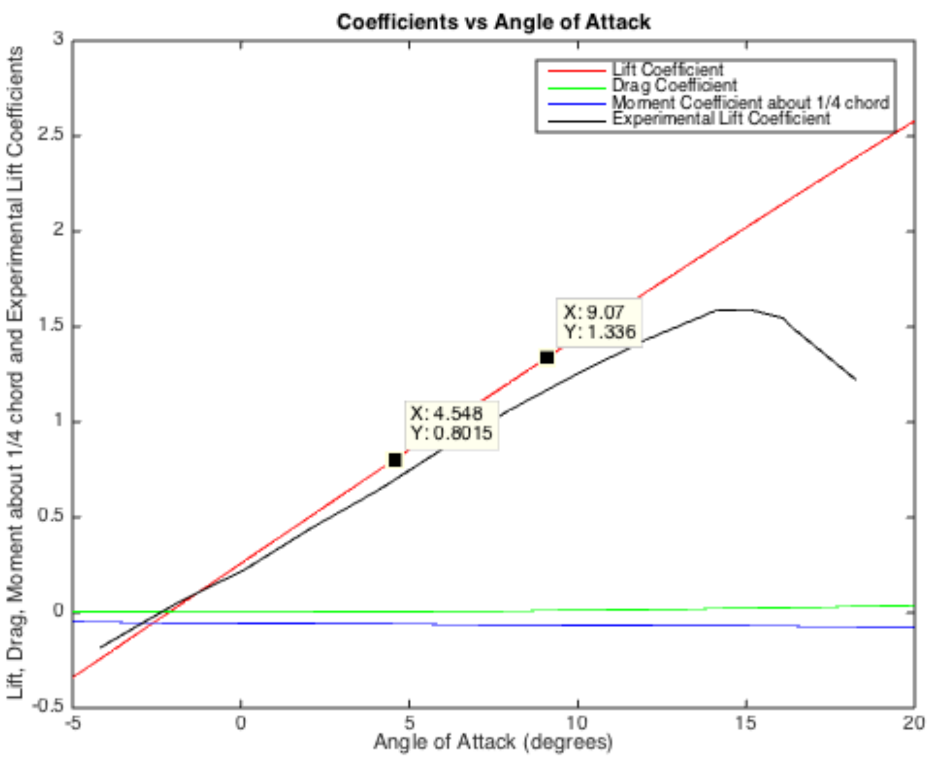

Figure 2: Lift slope for NACA 2412 Airfoil

<u>Part A:</u> Compute the Lift Slope for NACA 2412 Airfoil

Using the points in Figure 2 we can determine the lift slope for a NACA 2412 Airfoil.

<u>Point 1:</u> 4.548 degrees = (4.548 * pi)/180 = 0.079377 rad
<u>Point 2</u>: 9.07 degrees = (9.07 * pi)/180 = 0.158301 rad

Lift Slope = (1.336 - 0.8015) / (0.158301 - 0.079377) = **6.772 rad^{-1}**

Error = (6.772 - 6.283185307) / (6.283185307) = 0.077797

Percent Error = 0.077797 * 100 = **7.78 %**

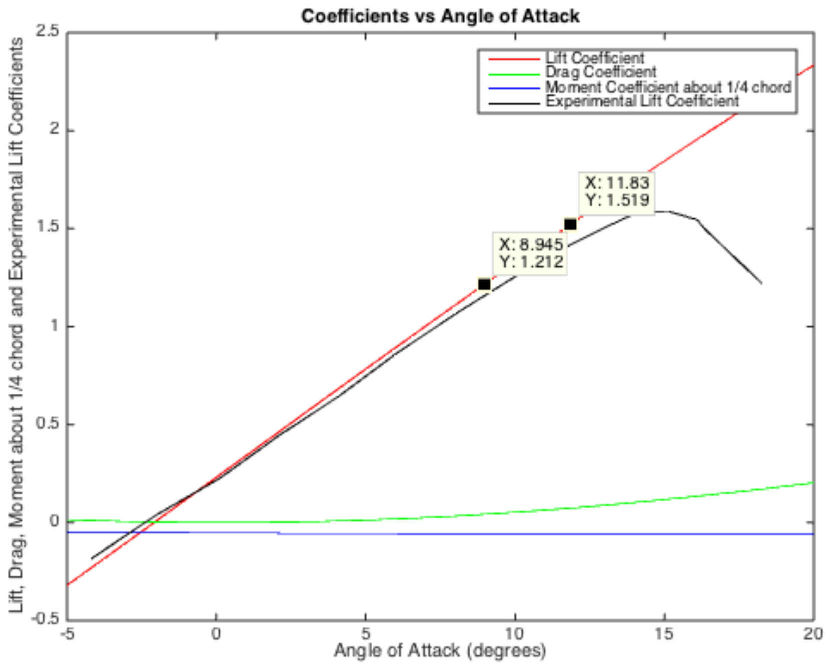

Figure 3: Lift slope for NACA 2402 Airfoil

<u>Part B:</u> Compute the Lift Slope for NACA 2402 Airfoil

Using the points in Figure 3 we can determine the lift slope for a NACA 2402 Airfoil.

<u>Point 1:</u> 8.945 degrees = (8.945 * pi)/180 = 0.1561197 rad
<u>Point 2</u>: 11.83 degrees = (11.83 * pi)/180 = 0.2064724 rad

Lift Slope = (1.519 - 1.212) / (0.2064724 - 0.1561197) = **6.09699 rad^{-1}**

Error = (6.283185307 - 6.09699) / (6.283185307) = 0.0296339

Percent Error = 0.0296339 * 100 = **2.96 %**

The lift slope for NACA 2402 is closer to the reference value than that of the NACA 2412 airfoil.

Part C:

The Lift Coefficient vs Angle of Attack curve for the NACA 2412 airfoil is seen to have a steeper slope than that of the curve produced from the experimental data as can be seen in Figure 1 above. The experimental data is also seen to eventually curve downwards after reaching a peak value of approximately 1.5 for an angle of attack of approximately 15 degrees. However, the same trend is not observed in the curve produced from the MATLAB code, as the curve continues to rise linearly.

Part D:

The small change in the coefficient for a quarter chord vs the angle of attack makes sense because the airfoil being considered is not entirely a thin airfoil, but instead borders the requirement for being a thin airfoil and therefore the values are not completely accurately calculated. Other reasons could also include external forces and pressures not being taken into account which could present a difference on a cambered airfoil.

The result from the panel method seems to be very similar to the result in the experimental data obtained from Figure 18 in the NACA Report 460. The result seems to variate very less as well but also seems to curve down instantly, which is not clearly reflected in the alpha range when using the panel method.

Part E:

The reason why the coefficient of drag is not exactly zero but instead is a very small value above zero is because viscous effects are not taken into account and these effects can produce some sort of drag. Also the panel method is run using an arbitrary number of panels and therefore the finite amount can result in the coefficient of drag values to not be entirely accurate.

Part F:

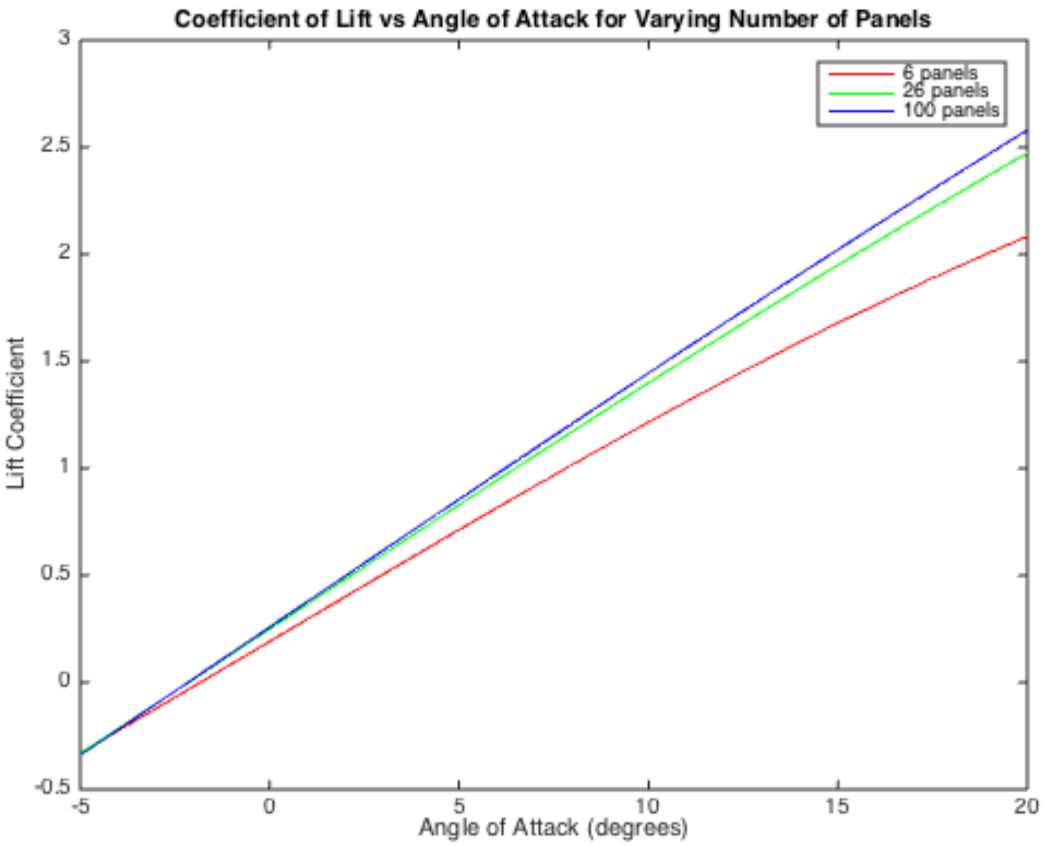

Figure 4: Coefficient of Lift vs Angle of Attack for Number of Panels = 6, 26 100

Assuming that 100 panels produces the best results it is clearly evident that as the number of panels decreases the accuracy seems to decrease. The curves for 6 and 26 panels seem to initially accurate as they align with the curve for a 100 panels, but as the angle of attack increases the curves begin to diverge away from the one produced using 100 panels. They seem to slowly include stall as the angle of attack keeps increasing for a decreasing number of panels.

Question 2:

Part A:

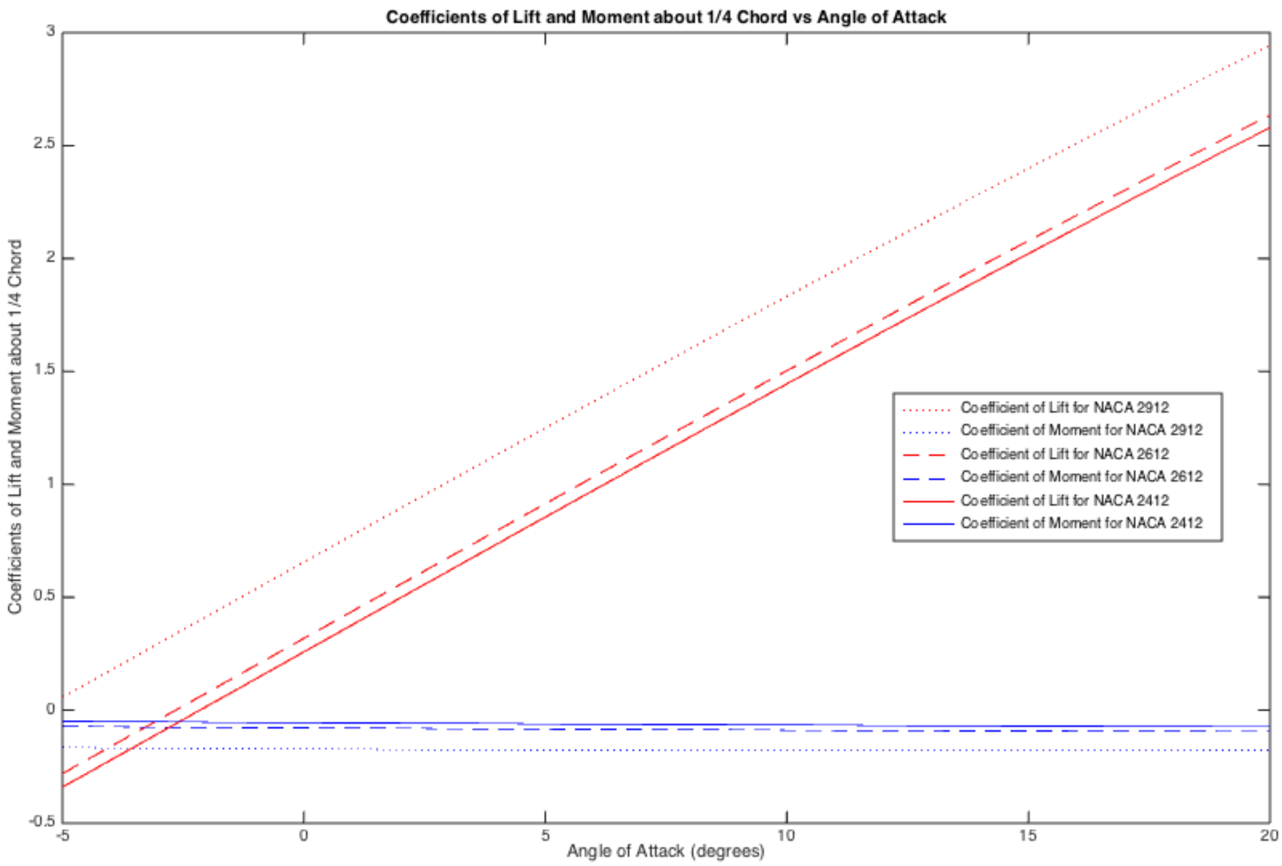

Figure 5: Coefficients of Lift and Moment about 1/4 Chord vs Angle of Attack for Varying NACA

All of the plots above are for airfoils of equal thickness and max camber, however the location of the max camber value seems to vary. As the location of the max camber increases the values for the coefficient of lift seem to increase as the curves are translated upwards. The values of the coefficient of lift seem to proportionally increase. However, the location of the max camber seems to effect the coefficient of moment about the 1/4 chord to a smaller degree, however the same but opposite effect is seen, as the coefficient of moment about the 1/4 chord decreases as the location of the max camber increases. The trailing edge shape of the NACA 2912 airfoil seems to be similar to that of a corresponding flapped airfoil.

Part B:

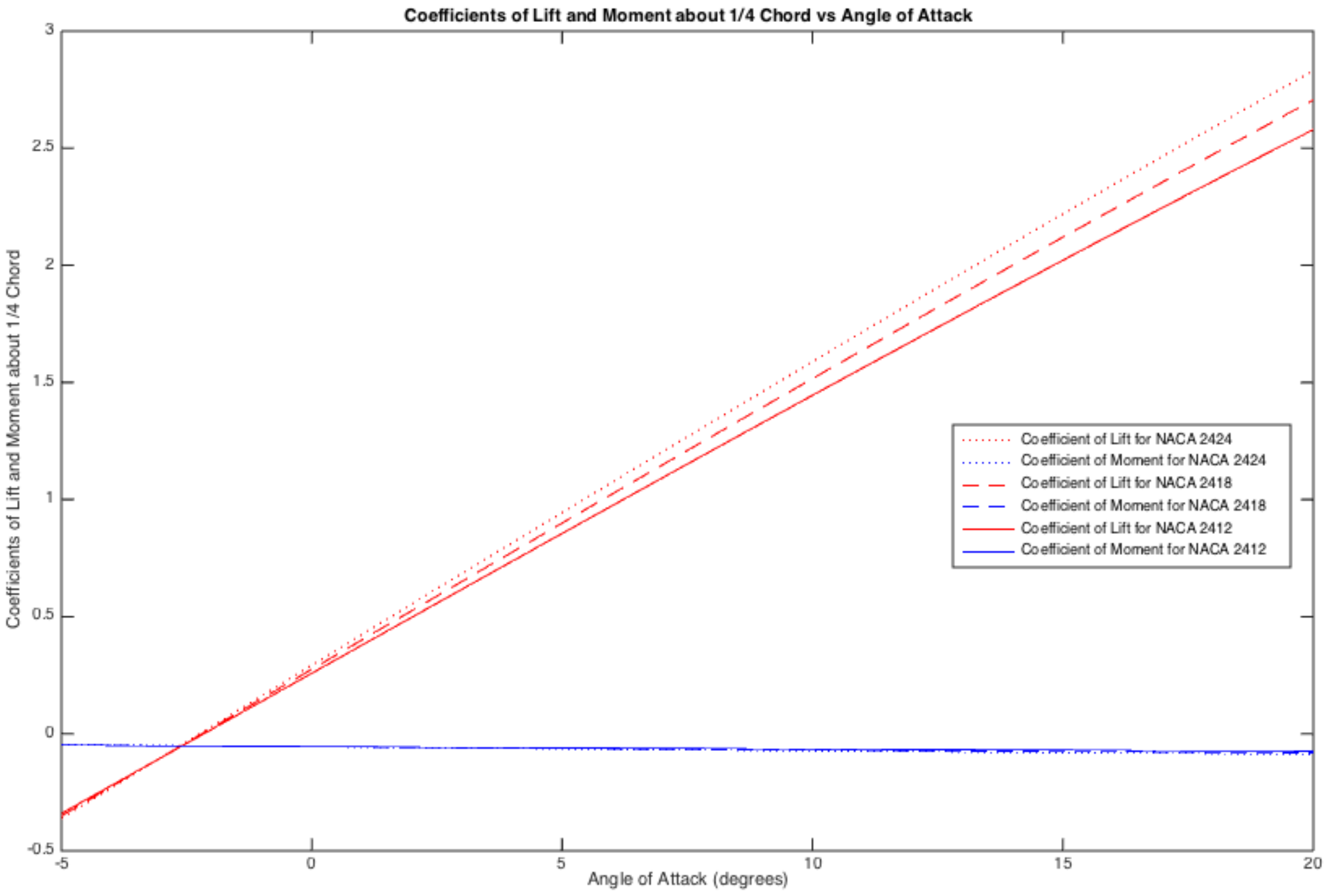

Figure 5: Coefficients of Lift and Moment about 1/4 Chord vs Angle of Attack for Varying NACA

All of the plots above are for airfoils of equal max camber and location of max camber, however the thickness value seems to vary. As the thickness increases the values for the coefficient of lift seem to increase as the curves are translated about a pivot point around (-2.5,0). The values of the coefficient of lift seem to initially be the same but eventually diverge away from each other. However, the thickness seems to effect the coefficient of moment about the 1/4 chord to a very small degree as the curves seem to overlap each other.

<u>Question 3:</u> NACA 2412 Airfoil

<u>Part A:</u>

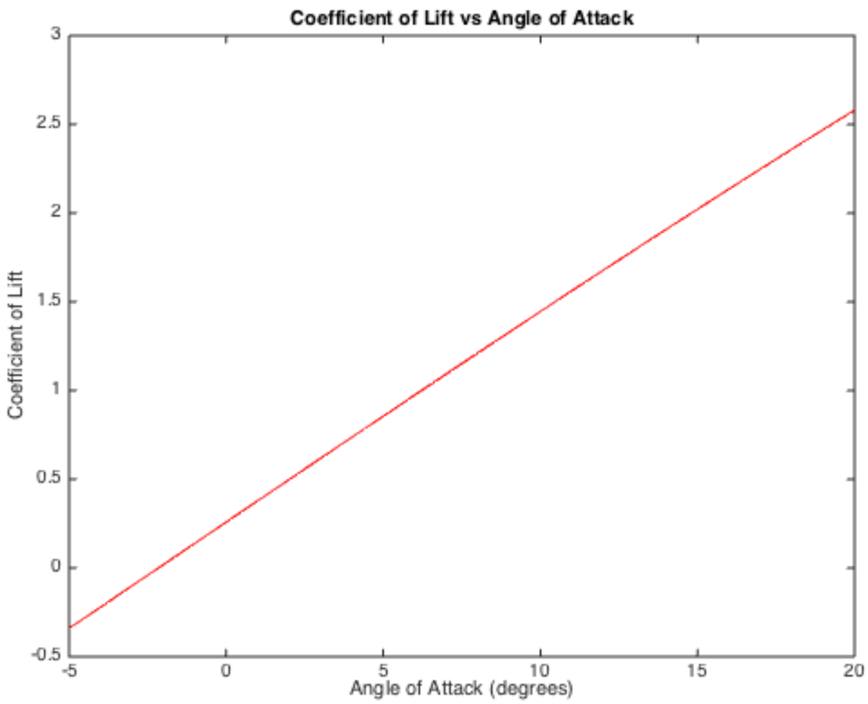

Figure 7: Coefficient of Lift vs Angle of Attack for NACA 2412 Airfoil

<u>Part B:</u>

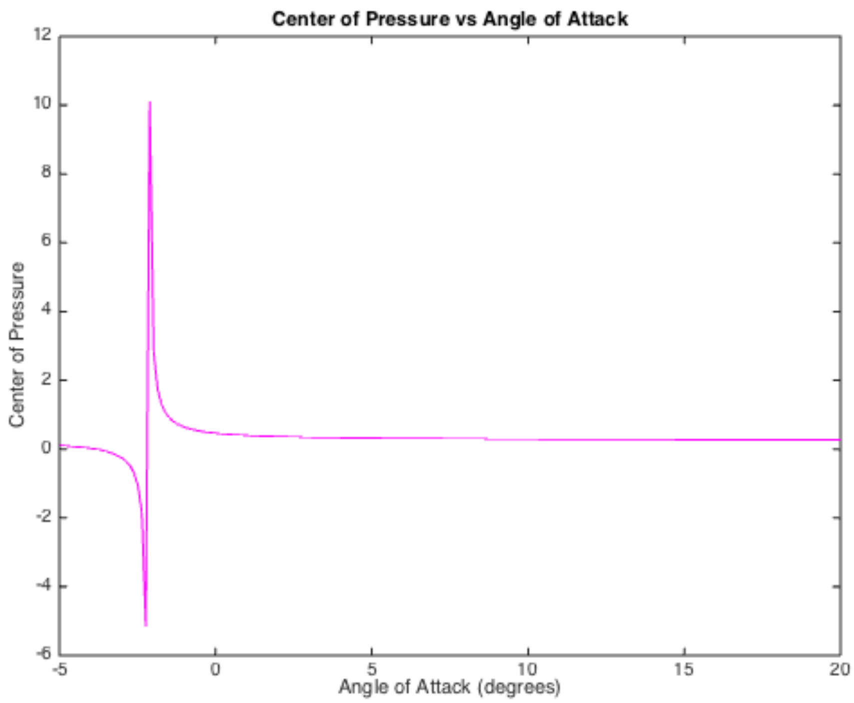

Figure 8: Center of Pressure vs Angle of Attack for NACA 2412 Airfoil

The behaviour of the center of pressure vs angle of attack curve can be observed in Figure 8 above. It can be seen that the curve initially starts from 0 for an angle of attack of -5 degrees and curves down as a parabola for small changes in the angle of attack after which it spikes up to a high positive value at an angle of attack of approximately -3 degrees. The curve then drops very fast and slowly approaches a value of zero as the angle of attack continues to increase.

The coefficient of lift vs angle of attack plot in Figure 7 is seen to increase linearly as the angle of attack increases. The coefficient of lift also approaches 0 as the angle of attack approaches a value of approximately -3 degrees, matching the point of an instant spike in the center of pressure vs angle of attack plot.

Part C:

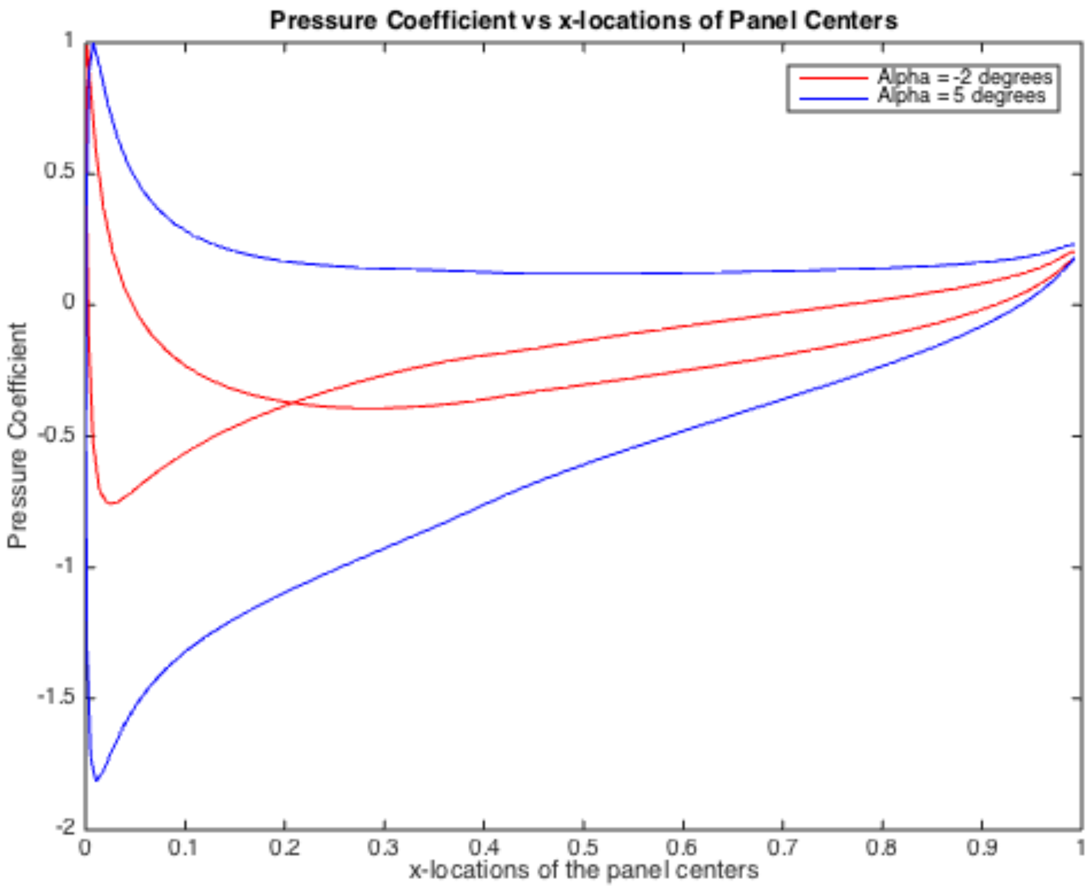

Figure 9: Pressure Coefficient vs x-location of Panel Centers for different Angles of Attack

The pressure distribution of the coefficients with the x-location of the panel centre seems to make sense, because as the angle of attack increases the coefficient of lift also increases. Near an angle of attack of -2 degrees, we can see from Figure 7 that the coefficient of lift approaches 0 while the centre of pressure seems to spike to a max value at this angle of attack.