MECH 539: Computational Aerodynamics

Project #4: Investigating Aerodynamic Performance of Airfoils using a Coupled Potential Equation and Integral Boundary Layer Method

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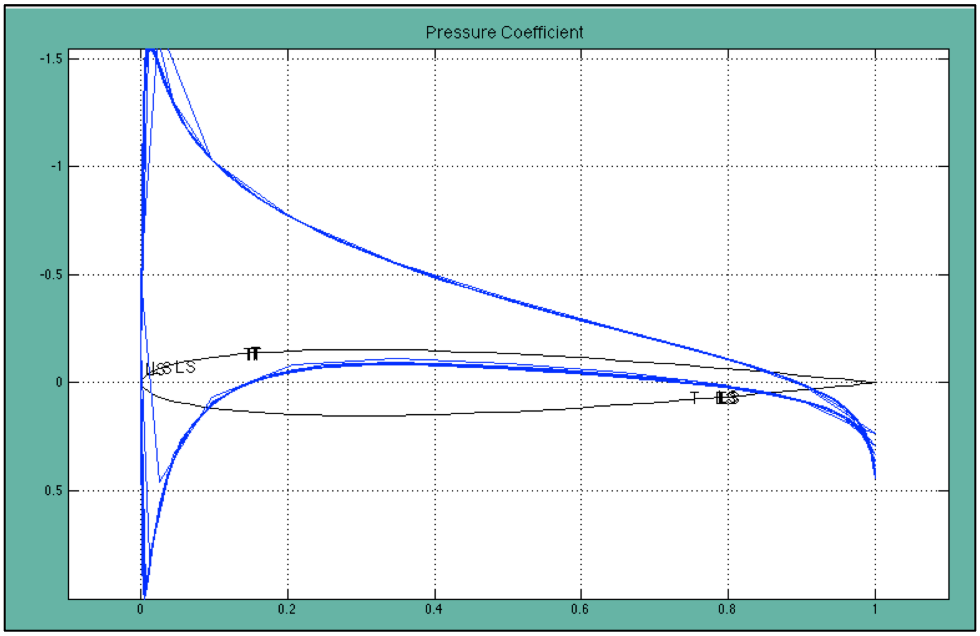
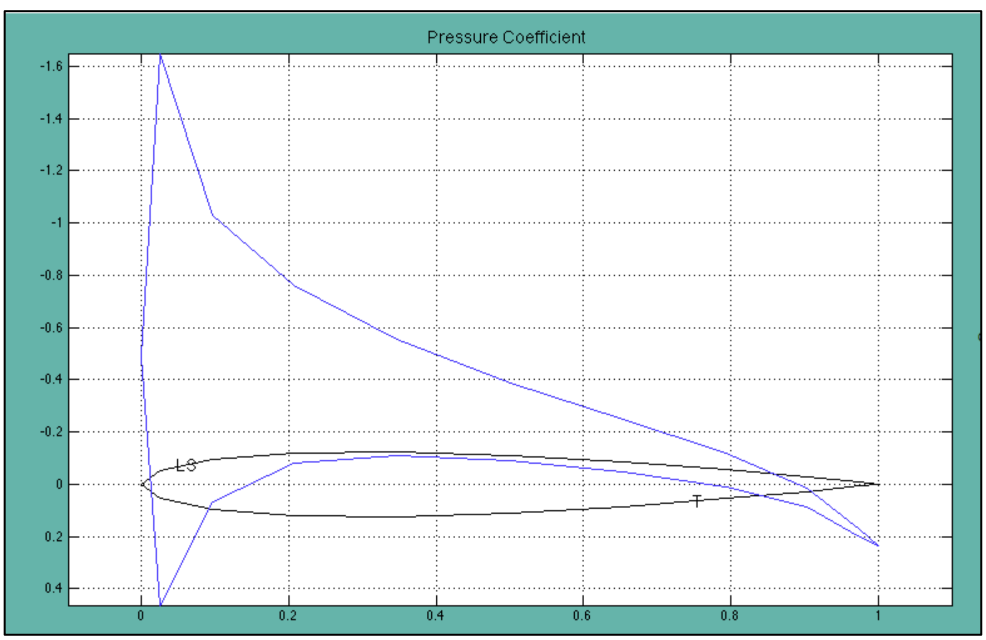
Question 1.

The effect of grid size was studied. NACA 0012 airfoil profile was selected at Reynolds number of 1,000,000 and angle of attack of 4 degrees. The grid size was started from 20 panels to 100 panels by 10.

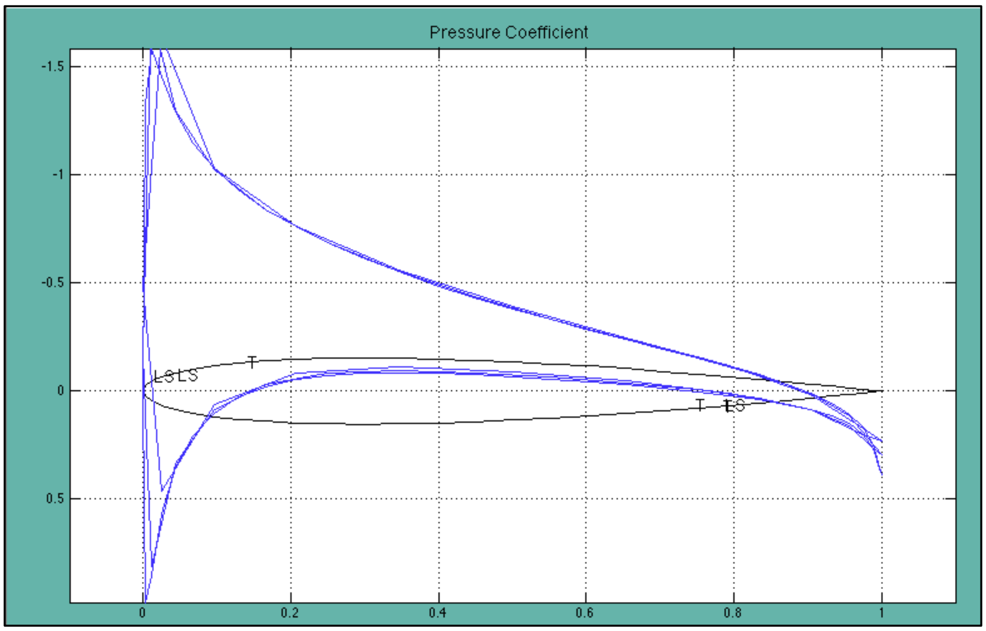


Pressure distribution at different grid sizes



Pressure distribution at 20 panels

At 20 panels, which is the coarsest gird, the pressure distribution at trailing edge was at about pressure coefficient of 0.2. And at the leading edge of airfoil, the pressure distribution was at around pressure coefficient of -1.6 for upper surface of airfoil. The stagnation point is at around pressure coefficient of 0.4 and this is wrong because it should be equal to 1. Also the locations of maximum pressure coefficients and of stagnation point are bit away from the trailing edge of the airfoil.



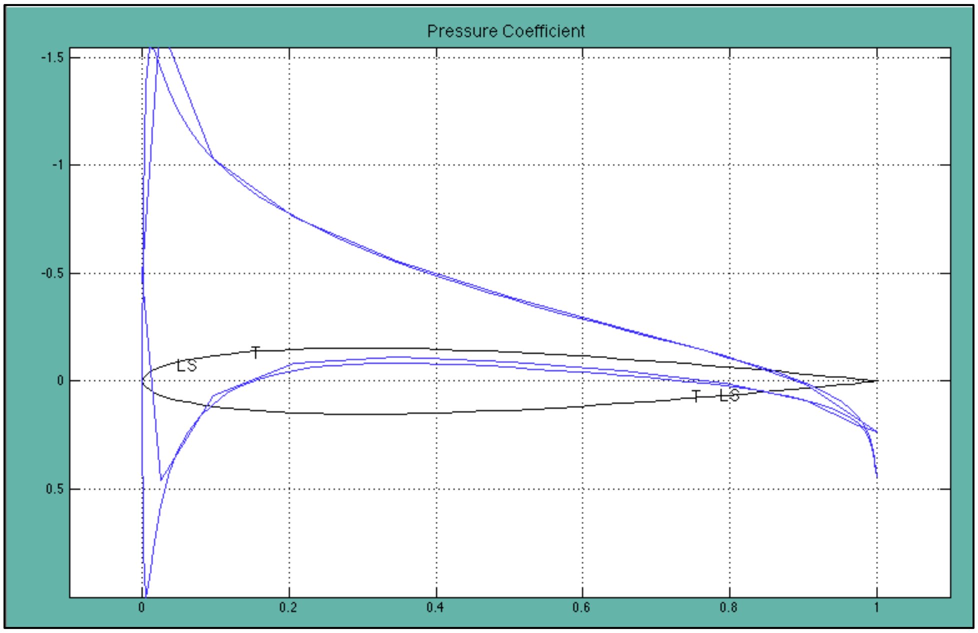
Pressure distribution at 20, 30, and 60 panels

As the number of panels increase, the pressure distribution has been changed. As the panel number is increased, the pressure distribution has shaped in a way that the trailing edge pressure is deflected more downward, stagnation point is shifted more close to the leading edge and its magnitude is increased. Also, the peak pressure at the upper surface is decreased and its location is shifted more close to the leading edge, also. And then, after 70 to 80 panels, the shape of pressure distribution does not change much but almost same shape distribution begins to overlap on the same shape of pressure distributions.

Finally at 100 panels, the stagnation point is at pressure coefficient of 1 and its location is almost at the leading edge. Also, the pressure distribution at the trailing edge is deflected more downward so that the pressure coefficient is close to 0.5. And the peak pressure on the upper surface of airfoil is occurred almost at the leading edge and its magnitude has decreased that the pressure coefficient is -1.5.

Compared between 20 panels and 100 panels, the pressure distribution is quite similar at around the mid-chord of airfoil, but their discrepancy is quite large at both leading edge and trailing edge.

The lower surface of airfoil carries some positive pressure at around both leading edge and trailing edge, but the pressure is mostly in negative, specifically around the mid chord of the airfoil, which means that the force is pulling down the wing downward at around the mid chord of lower surface of the airfoil. These phenomena are the same for both large and small number of panels.

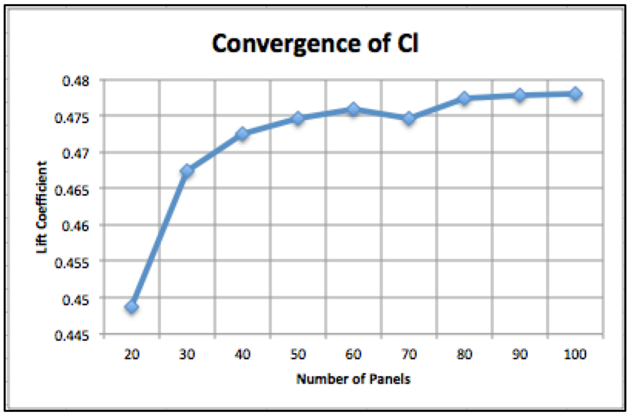


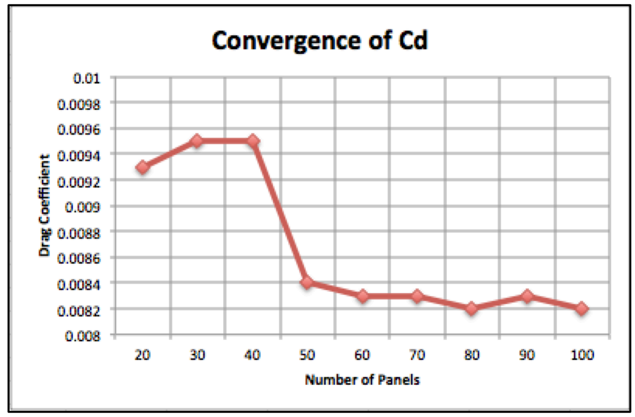
Pressure distribution at 20 and 100 panels

As the number of panels increase, the pressure distribution is changed to certain shaped, but beyond certain number of panels, there is no much difference of the pressure distributions.

The lift coefficient and drag coefficient at different number of panels have been compared, too.

Number of Panels	Cl	Cd
20	0.4488	0.0093
30	0.4673	0.0095
40	0.4724	0.0095
50	0.4746	0.0084
60	0.4759	0.0083
70	0.4746	0.0083
80	0.4773	0.0082
90	0.4778	0.0083
100	0.4781	0.0082





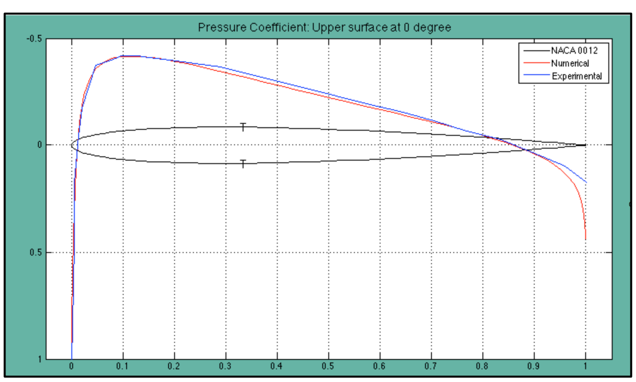
According to the table and graphs, it is noticeable that both lift coefficient and drag coefficient are changed as number of panel increases, but at certain point, they begins to converge to certain values. After 80 panels, lift coefficient and drag coefficient change its value only slightly and begin to converge. Therefore, according to Cl and Cd graphs and pressure distribution plot, it could be confirmed that at least 80 panels can produce the reasonably accurate solution.

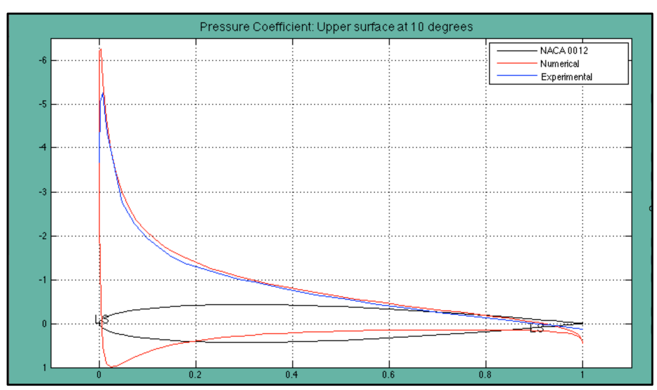
Reference: http://adg.stanford.edu/aa241/airfoils/airfoilpressures.html

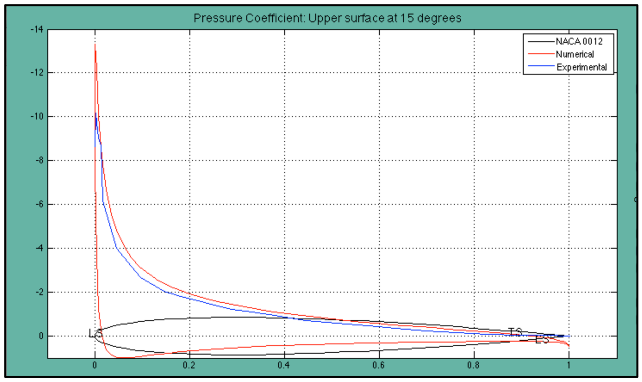
Question 2.

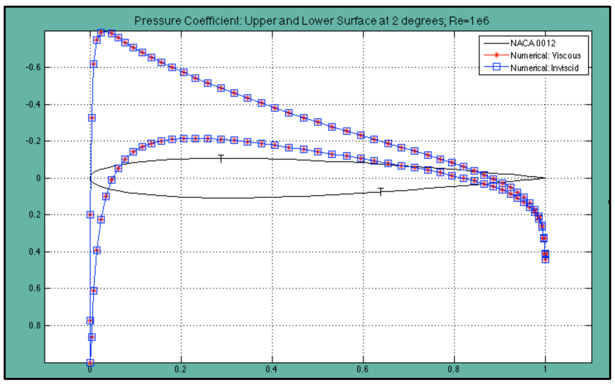
The pressure data of Gregory and O'Reilly is obtained from following website:
http://turbmodels.larc.nasa.gov/NACA0012_validation/CP_Gregory_expdata.dat

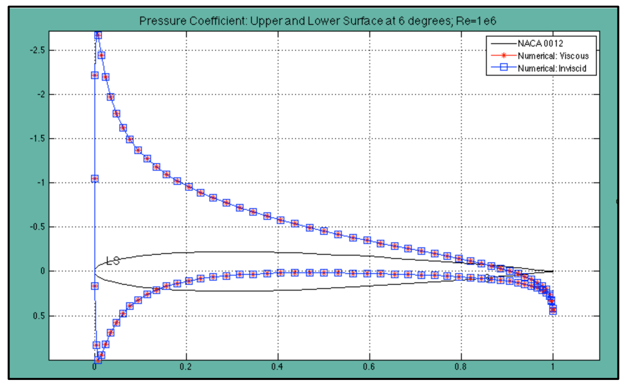
The data is obtained at Reynolds number of 3 million and angles of attack are 0, 10 and 15. Only upper surface data is collected in this website.
Numerical solutions are calculated at Reynolds number of 3 million and at angle of attack of 0, 10 and 15. The flow is set to viscous flow.











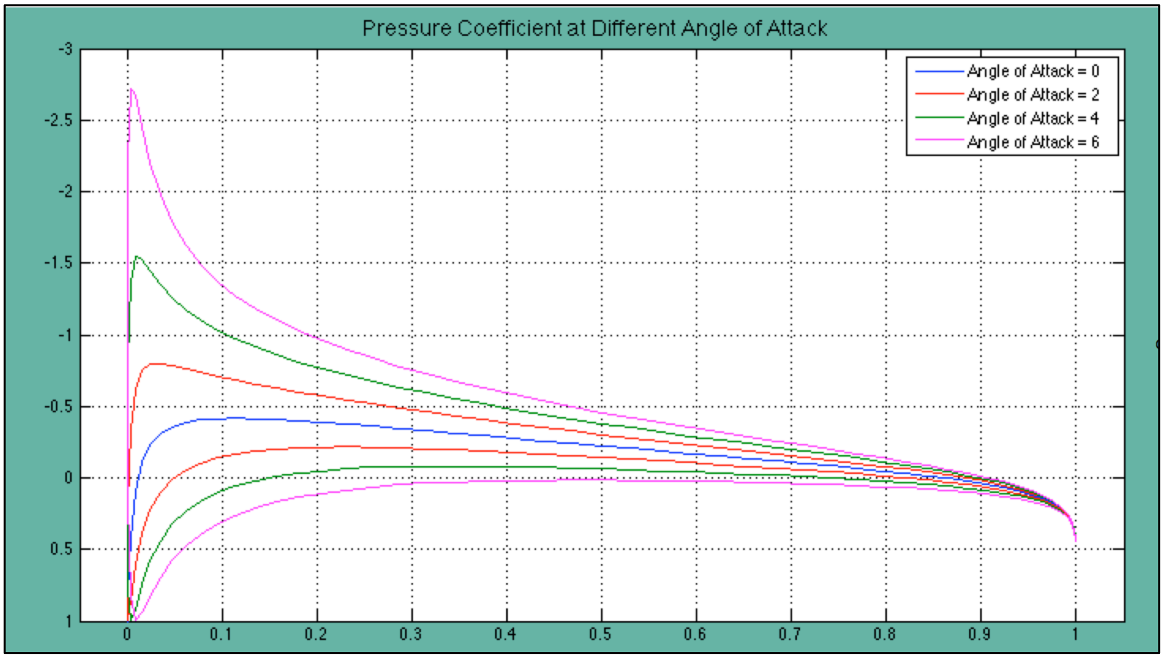
First, for the comparison between experimental data and numerical data, the pressure distributions are quite similar, but experimental pressure coefficient is little bit larger than the numerical pressure coefficient at around the mid chord and there is quite large discrepancy between trailing edge pressures at angle of attack of 0 degree. But, at 10 degrees, numerical pressure distribution is always little bit larger than the experimental pressure distribution, except the numerical trailing edge pressure is always larger in positive than the other. And as the angle of attack increases, the discrepancy between experimental data and numerical data gets larger.

For the comparison between viscous and inviscid flow at each 2 degrees and 6 degrees, the pressure distributions are always exactly the same, but only difference is the existence of drag coefficient. Thus, the drag force is caused by the viscosity of the fluid.

Question 3.

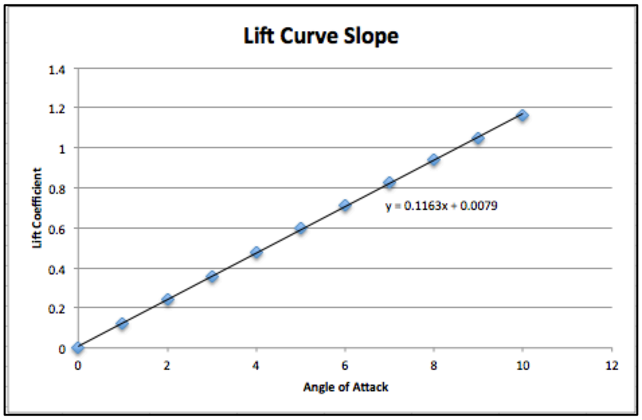
Reynolds number is at 1e6

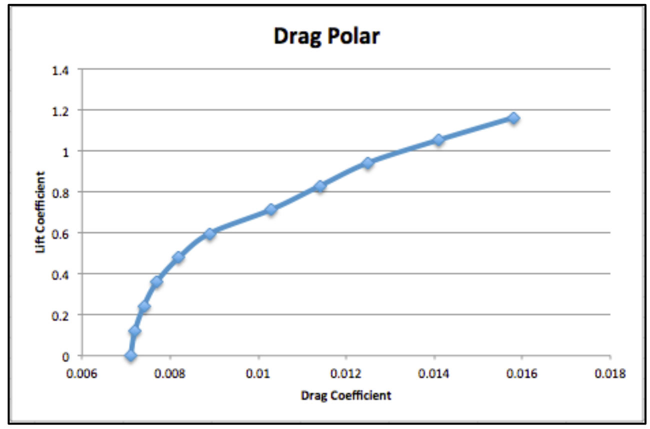
a)



First, the maximum pressure coefficient over the upper surface of airfoil gets large as the angle of attack increases. Although the center of pressure stays constant at $x_{cp} = 0.25$ after 0 degrees due to its symmetrical shape, as the angle of attack increases, airflow would be forced to move along the upper surface faster than would the airflow moves over the lower surface because upper airflow needs to cover more distance than lower airflow due to the airfoil curvature. So, by the Bernoulli's theory, upper surface where the airflow moves the fastest carries the lowest pressure. Thus, as angle of attack increases, since airflow over upper surface needs to cover even more distance, the pressure is decreased even further. And, the positive pressure region at lower surface of the airfoil gets smaller as the angle of attack increase and finally at 6 degrees, lower surface does not carry the positive pressure anymore but maximum pressure coefficient is 0. This phenomenon can be interpreted that the pulling down force exerted on lower surface is diminished as the angle of attack increases. Moreover, The transition location is the same on upper and lower surface, but at 2 degrees, upper surface transition location is shifted to left, while the other is shifted to right. But, at 4 degrees, transition location on upper surface is shifted even further left, while laminar transition is occurred on lower surface near the trailing edge. And, finally, upper surface laminar transition location is now close to the leading edge, and the other laminar transition location is close to the trailing edge. Meanwhile, the trailing edge pressures are constant at different angle of attack.

b)





The relationship between angle of attack and lift coefficient is linearly proportional and its slope is obtained as 0.1163. This represents that the increase in angle of attack g enerates more lift forces without stall up to 10 degrees.
The drag polar represents the relationship between the lift and drag. According to the drag polar graph, lift coefficient exponentially increase as a result of increase in drag coefficient. In this case, C_{D0} is 0.071 and C_{L0} is 0 because NACA 0012 is not cambered wing and it is symmetric. This drag polar reflects the performance of NACA 0012 airfoil that up to 5 degrees of angle of attack, NACA 0012 wing can increase the lift force with little increase in drag coefficient, but beyond that point, drag force is increased in nearly proportional to increase in lift force. In addition, if flap is installed on this wing, then the drag polar plot could have much steeper exponential curve at the low angle of attack, which improves the performance of the wing.

c)
As the angle of attack increases, the drag force also increases. But at low angle of attack, the drag force is increased only slightly, but at high angle of attack, it increases as much as the lift force does.
The source of drag force is skin friction between air molecules and wing surface, which is caused by the viscosity of the fluid.

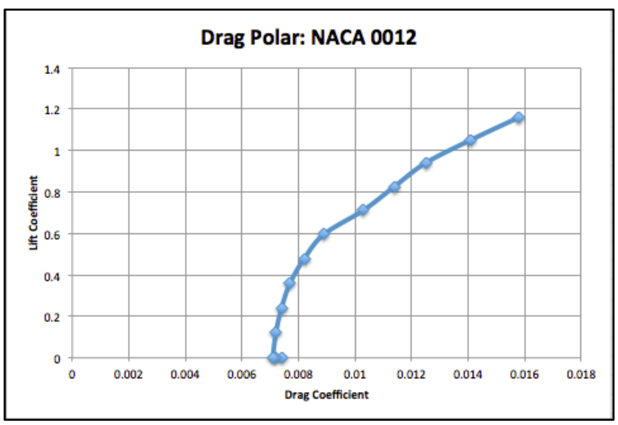
Reference:
http://www.free-online-private-pilot-ground-school.com/aerodynamics.html
http://en.wikipedia.org/wiki/Drag_Polar

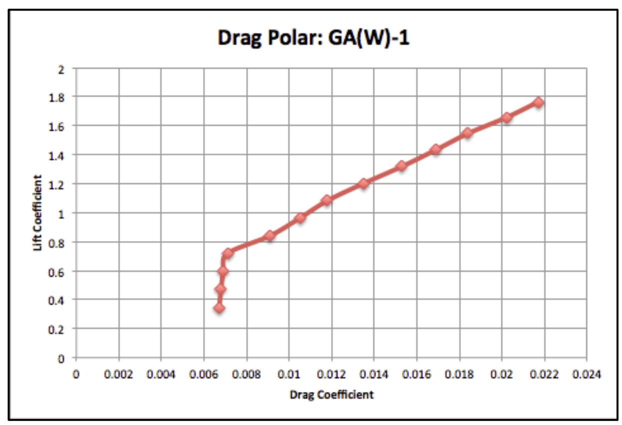
Question 4.

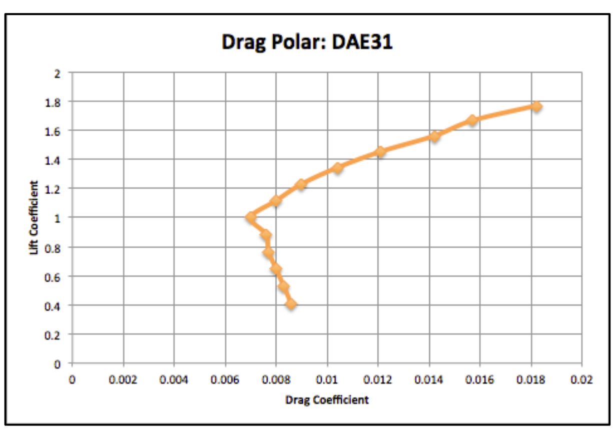
Reynolds number is at 1e6

a)

	NACA 0012		GA(W)-1		DAE31	
Angle of Attack	Cl	Cd	Cl	Cd	Cl	Cd
-2	0	0.0074	0.3457	0.0067	0.4062	0.0086
-1	0	0.0072	0.4702	0.0068	0.5262	0.0083
0	0	0.0071	0.5943	0.0069	0.6459	0.008
1	0.1201	0.0072	0.718	0.0071	0.765	0.0077
2	0.2401	0.0074	0.8408	0.0091	0.8831	0.0076
3	0.3595	0.0077	0.9625	0.0105	1.0002	0.007
4	0.4781	0.0082	1.0829	0.0118	1.1159	0.008
5	0.5957	0.0089	1.2018	0.0135	1.2299	0.009
6	0.7121	0.0103	1.3188	0.0153	1.3421	0.0104
7	0.8269	0.0114	1.4337	0.0169	1.4521	0.0121
8	0.94	0.0125	1.5463	0.0184	1.5598	0.0142
9	1.0511	0.0141	1.6563	0.0202	1.665	0.0157
10	1.16	0.0158	1.7635	0.0217	1.7674	0.0182







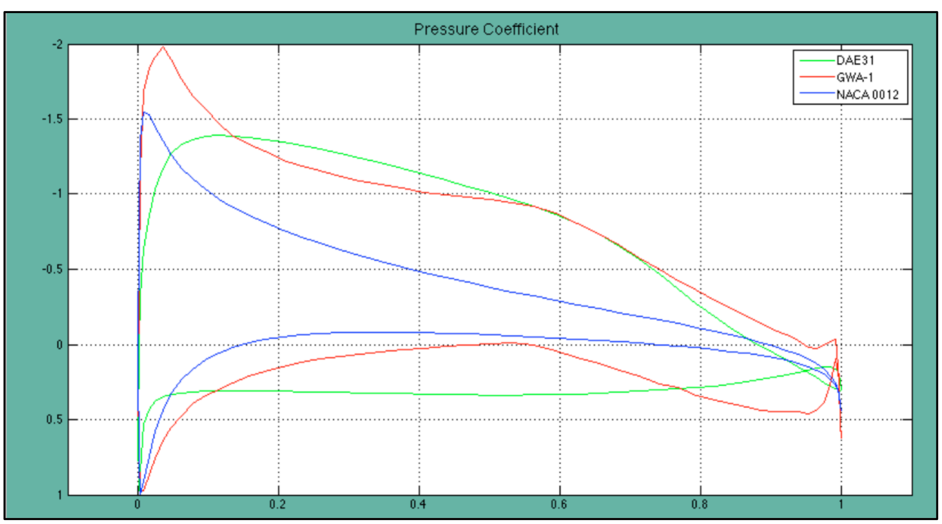
For all three airfoils, lift coefficient exponentially increase as a result of increase in drag coefficient after C_{L0}. For NACA 0012, from 0 to negative angles, the airfoil does not generate any lift force, but only the drag force. The drag coefficient is decreased from -2 degrees to 0 degree, but after, it is increased for positive angle of attack. And, its C_{D0} is 0.071 and C_{L0} is 0 because NACA 0012 is not cambered wing and it is symmetric. For GA(W)-1 airfoil, from -2 degrees to 1 degree, the lift coefficient is increased almost vertically as the drag coefficient increase only a little. The reason that drag polar plot is not smooth polynomial shape but quite discontinuous is that GA(W)-1 wing is cambered and is asymmetrical wing. Lastly, for DAE31 airfoil, from -2 degrees to 3 degrees, the lift coefficient is decreased as the drag coefficient is increased so that its C_{L0} is 1.0002. This is because DAE31 is cambered wing and it is not symmetrical wing.

b)
The values of drag coefficient at an equivalent lift coefficient of 1.0 are calculated by linear interpolation method.

Cl	Cd		
	NACA 0012	GA(W)-1	DAE31
1	0.0134	0.0109	0.0070

The table above shows the corresponding drag coefficient at lift coefficient of 1.0 for three different airfoils. The highest values is 0.0134 of NACA 0012 airfoil, next is 0.0109 of GA(W)-1, and lowest is 0.0070 of DAE31. This can be interpreted as the performance of airfoil. The lower drag coefficient for the same lift coefficient means that for the same lift force, the less drag force is exerted on the wing. Therefore, DAE31 airfoil is the most efficient, next is GA(W)-1 airfoil, and the least efficient is NACA 0012 airfoil.

c)

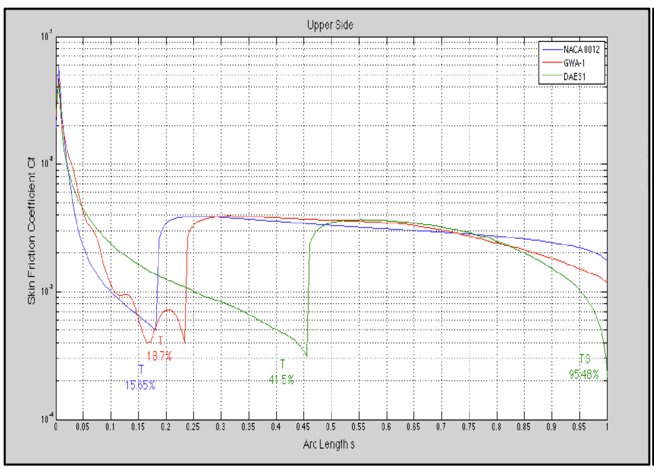


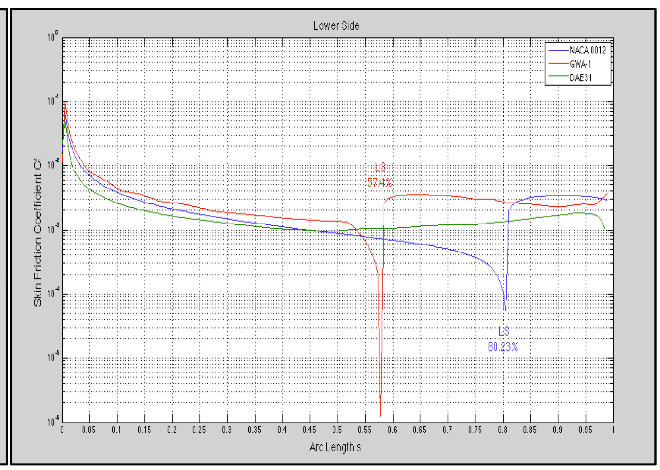
Pressure distribution for three airfoils at angle of attack of 4 degrees

A favorable pressure gradient is a pressure field that decreases in the flow direction, which is caused by the free stream velocity accelerating in the flow direction. But, according to the plot above, the favorable pressure gradient exists only near front half of the chord on the lower surface of all three airfoils. It is all adverse pressure gradients for rest of regions. It is reasonable results because accelerated airflow on the upper surface near the leading edge of the airfoil needs to be brought back down to freestream velocity as it travels over the airfoil. So, as the airflow velocity decreases, the pressure over the surface increases as it goes to trailing edge. Furthermore, it is obvious that the upper surface pressure area is the largest area of the total pressure distribution for all three airfoils. This means that the upper surface of the airfoil that carries high negative pressure mainly generates the lift force. Thus, the lift is mainly generated by first three quarters of chord on the upper surface. But, in addition, the lower surface of DAE31 carries more positive pressure area than other two airfoils. So, for the DAE31 airfoil, although upper surface generates most of lift force, the lower surface also quite contributes to generate the lift force.

d)

The y-axes are log scaled.



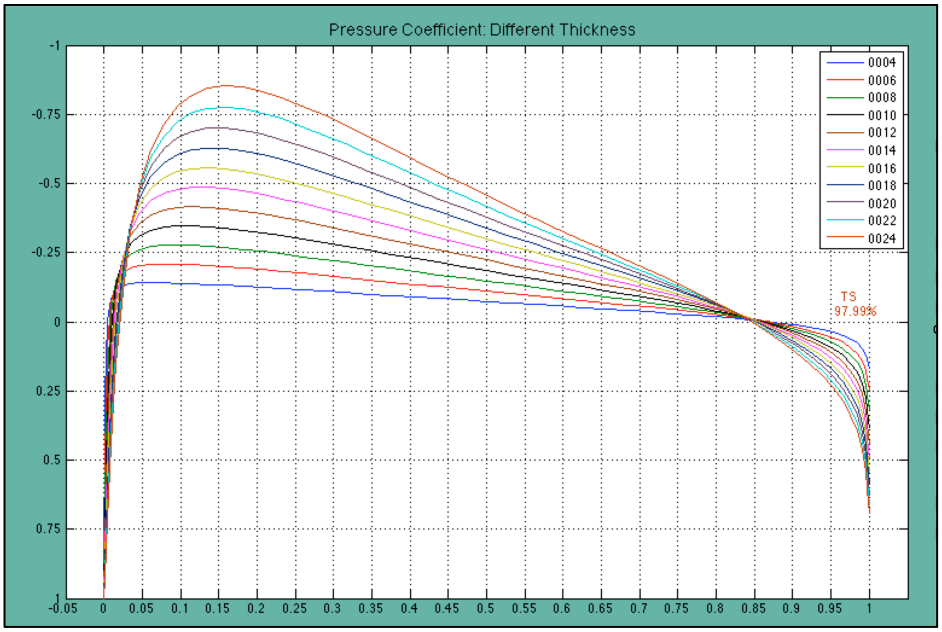


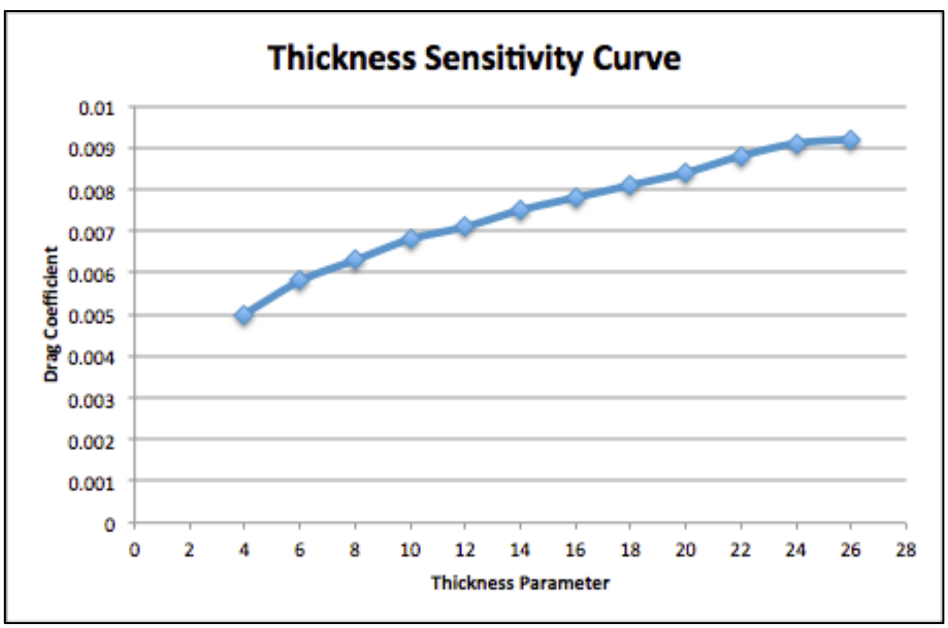
According to the plots above, for all three airfoils, the transition on the upper surface of airfoil is occurred about 5% of chord behind the sudden increase of skin friction coefficient and after the turbulent separation, the skin friction coefficient decreases exponentially. Also, the laminar separation on the lower surface is occurred right at the location where sudden drop and increase of skin friction coefficient occurs. Since the lower surface of DAE31 airfoil is fully laminar, there is no label of laminar separation. These phenomena can be interpreted that the transition from laminar-turbulent on the upper surface induces more skin frictions, while turbulent separation decreases the skin frictions. So, the laminar-turbulent transition is the prediction of the increase of skin friction coefficient. Also, the laminar separation on lower surface produces the unstable airflow and it dramatically changes the skin friction. Then, ultimately, the skin friction increases slightly more. Therefore, the turbulent airflow from the transition and unstable flow from laminar separation produce more skin friction forces.

Question 5.

Reynolds number is at 1e6

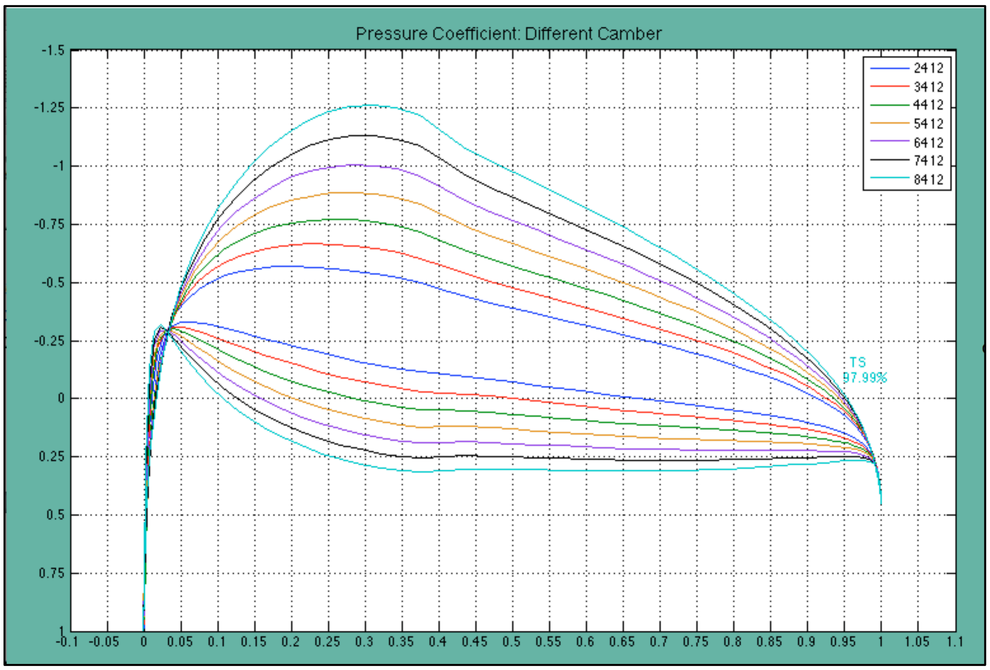
a) Thickness parameter (NACA 00XX)

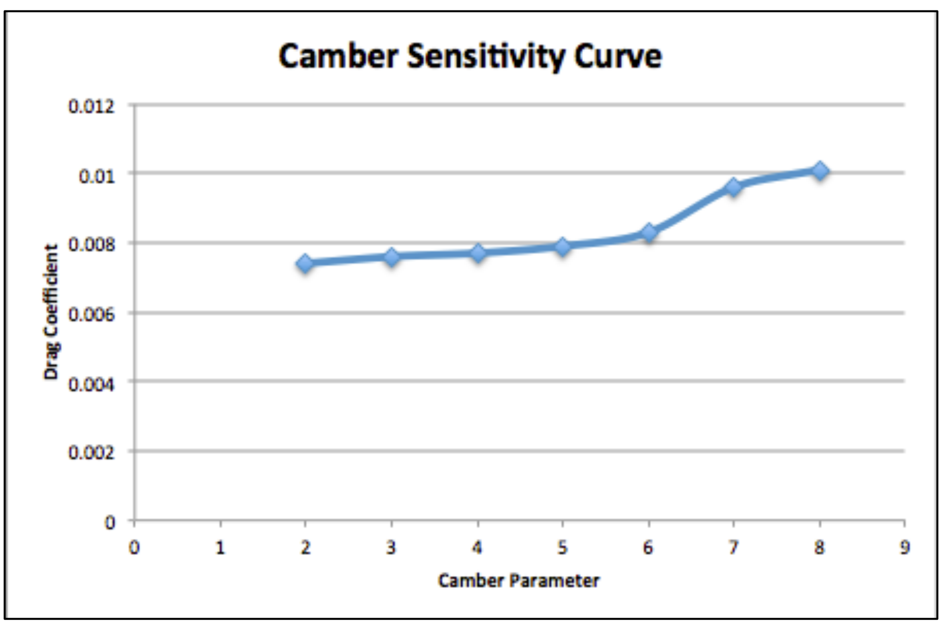




As the thickness parameter increases, drag coefficient increases slightly exponentially and tends to converge to certain value of drag coefficient after parameter of 22. Also as the thickness of airfoil increases, the larger negative pressure is carried by the airfoil so that it generates more lift force. And, the turbulent separation is only occurred in the case of highest thickness parameter at near the trailing edge.

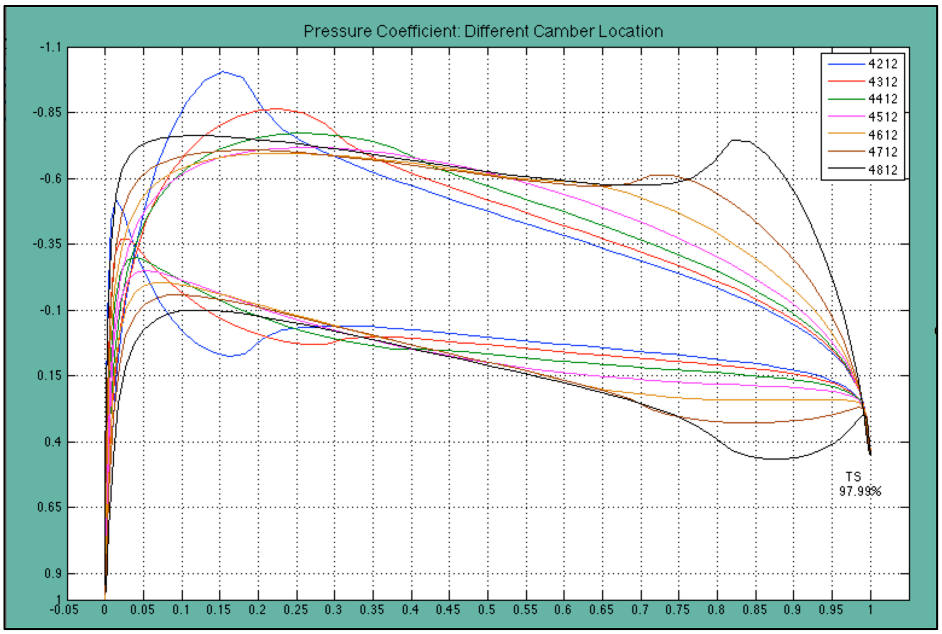
b) Camber Parameter (NACA X412)

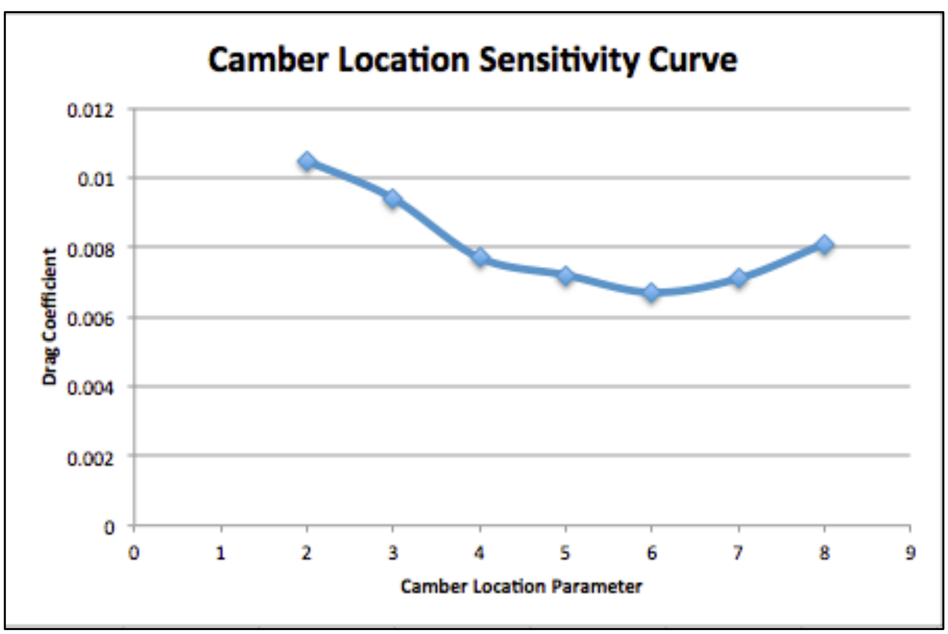




The slope between camber parameter and drag coefficient is fairly flat up the parameter of 5, but from the parameter of 6, the slope between those gets much steeper. Also the camber parameter increases, not only the upper surface carries larger negative pressure, but also the lower surface tends to carry larger positive pressure. Thus, the higher the camber parameter, the more lift force is generated by the wing. And, the turbulent separation is only occurred in the case of highest camber parameter at near the trailing edge.

c) Camber Location Parameter (NACA 4X12)





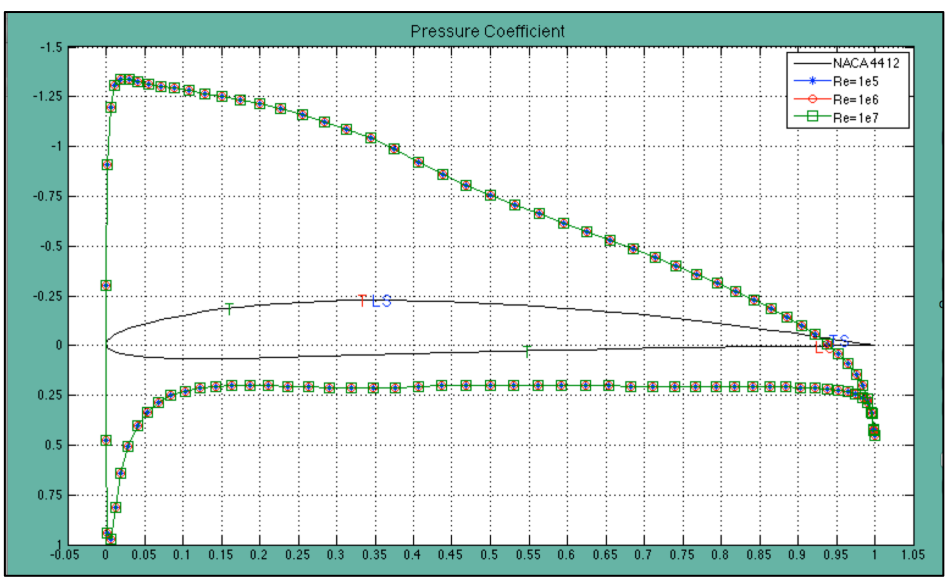
The relationship between camber location parameter is similar to the polynomial function. The drag coefficient decreases as the camber location decreases up to parameter of 6, but after that, the drag coefficient increases again. Meanwhile, according to the pressure distribution plot, as the camber location parameter increases, the peak pressure coefficient of upper and lower surface shifts from leading edge to the trailing edge that at lower camber location parameter, the pressure area is larger around front half of the chord, while at higher camber location parameter, the pressure area around rear half of the chord is larger. In other words, camber location parameter determines the region of airfoil where the lift force is mainly produced.

Each NACA airfoil parameter carries different sensitivities to the drag coefficient. Also, each parameter affects on pressure distribution over the airfoil differently. For thickness parameter, higher parameter produces more pressure distribution area, which is more lift force, but at the same time, the drag coefficient increases. Also, higher camber parameter produces much larger pressure distribution area from both upper and lower surfaces, but also the drag force increases. Lastly, the camber location parameter of 6 results the least drag coefficient, but it produces the second smallest pressure distribution area, compared to other parameters. Thus, such phenomena imply that there is no best value of parameters, but in depth study is required to fin the optimal value of each parameter that produces large pressure distribution area, while keeping the drag coefficient small.

Question 6.

NACA 4412 airfoil. Angle of attack is at 4 degrees.

a)



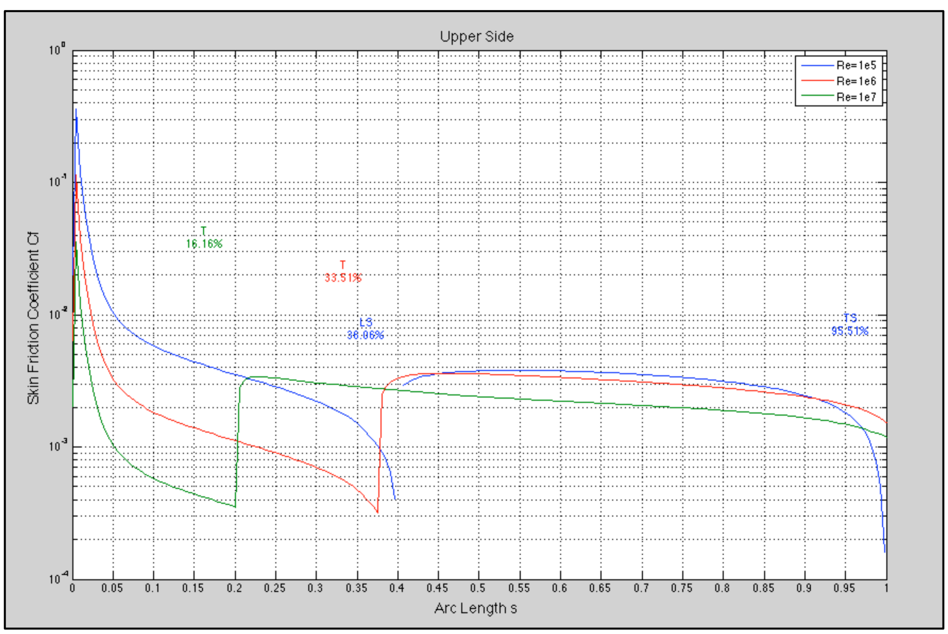
The increase in Reynolds number does not change the pressure distribution at all, but it only changes the laminar-turbulent transition location, and laminar separation locations. According to the plot, at high airflow speed, the transition is occurred at both upper and lower surfaces, and at medium airflow speed, transition is occurred at upper surface, while laminar separation is occurred at lower surface. Lastly, at low airflow speed, the laminar separation is occurred at both upper and lower surfaces. This could be concluded that if fast laminar airflow travels along the airfoil, it experiences the laminar-turbulent transition, and if slow laminar airflow travels along the airfoil, it experiences the laminar separation. In addition to this, at Re = 1e6, the upper surface airflow is faster than the airflow on lower surface, upper surface airflow experiences the laminar-turbulent transition, while the lower surface airflow experiences the laminar separation.

b)

Reynolds Number	Cl	Cd
1.00E+05	0.9636	0.0174
1.00E+06	0.9636	0.0083
1.00E+07	0.9636	0.0069

As the Reynolds number increases, the drag coefficient decreases, but the lift coefficient stays the same all the time. For the lift coefficient, it is predictable result from the pressure coefficient plot, because the pressure distribution area corresponds to the lift force, and they are the same on the pressure coefficient plot.

c)



If the Reynolds number is higher, the transition tends to occur and its location is closer to the leading edge of the airfoil. And, if the Reynolds number is lower, the laminar separation tends to occur and turbulent separation is occurred at near the trailing edge. After the laminar-turbulent transition and laminar separation, the skin friction coefficient tends to increase much higher, but after the turbulent separation, the skin friction coefficient tends to drops dramatically. So, the at laminar region, the skin friction force is fairly small, while at the turbulent region, the skin friction coefficient is larger. The phenomena are the same as the result obtained in question 4.