Siemens Gamesa Renewable Energy Coding Challenge

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I. Code Description

In order to be able to create all the necessary graphs for analyzing aerodynamic performance, a MATLAB code needs to written that will run XFOIL through it.This was first done by creating an input .csv file. This file will type in all the commands it has into XFOIL and will command XFOIL too. This was done by using fopen and fprintf function in MATLAB. The polar data that is needed is the coefficient of lift (CL), the coefficient of drag (CD) and moment coefficient (CM). The commands needed in XFOIL to retrieve the polar data are NACA, OPER, VISC, PACC, and ASEQ. The NACA command lets you type in the desired airfoil, which in this case is 63(3)-618, the OPER command gives you all the necessary command operations to analyze the airfoil. Once the OPER command was typed in, VISC needs to be typed to retrieve the viscous flow simulation. At this point the wanted Reynolds's number of either 3 million, 10 million, or 15 million would be chosen. After, PACC was used to create the file that will contain all the necessary data for the polar graphs, which will be run by using ASEQ to give all the data for angles of attack from -10 to 20 degrees. Once the polar data was created, the Coefficient of Pressure (CP) data was being created. This is similar to the steps for the polar data, after determining the Reynolds's number that is going to be used, the alfa command is used to pick the angle of attack. Then the CPWR command to save the CP data to a file. Once the file for the polar graphs and the CP graphs were read into MATLAB, the built in textscan function was used to be able to extract the data that is wanted and to ignore the data we don't need. Then using the cell2mat and str2num built in functions were used to get the data into usable numbers. From there is was extracting the data such that each column had its own variable, and then plotted the desired graphs.

For the boundary layer properties data, similar steps were done in MATLAB. However the approach was different since XFOIL is not able to create boundary layer data for a changing angle of attack. First a single MATLAB function was created so that it can be run through a for loop to make parsing the data more efficient. In the function, XFOIL is being called to run by creating an input file to add commands into MATLAB similar to before, however, after you chose a viscous flow and the Reynolds number, the alfa command was used for each angle by the for loop. Then the BL command was done to create the boundary layer data at that angle and then the DUMP command to create the file. After all the data was parsed into its own variable, the data then was plotted.

II. Analysis for Reynolds Number of 3 Million

When picking a Reynolds's Number of 3 Million, there will be more laminar flow than turbulent flow compared to 10 and 15 Million. This is a result of the viscous forces being a bit more dominant than the inertial forces compared to the Reynolds numbers of 10, which would have less turbulent flow compared to an airfoil with a Reynolds number of 15 million. As a result, all the graphs are different at a Reynolds number of 3 million compared to 10 million and 15 million.

A. Polar Graphs

The coefficient of lift is an instrumental property of an airfoil, not only because it changes for each airfoil, but due to the angle, Reynolds's Number, and air speed, it can change. An airfoil is able to generate lift due to the pressure gradient that is created on the top and bottom of the airfoil as air flows over it. When the top has lower pressure than the bottom, the airfoil is able to generate lift in the upwards direction. As seen in figure 1, the coefficient of lift keeps increasing up to a certain angle. At an angle of attack of 18 degrees at a value of roughly 1.9, the most lift is being generated. However, due to the effects of drag, the airfoil will eventually lose lift as the air wont be able to stick onto the airfoil. As a result, the coefficient of drag would then start to decrease and cause a stall eventually.

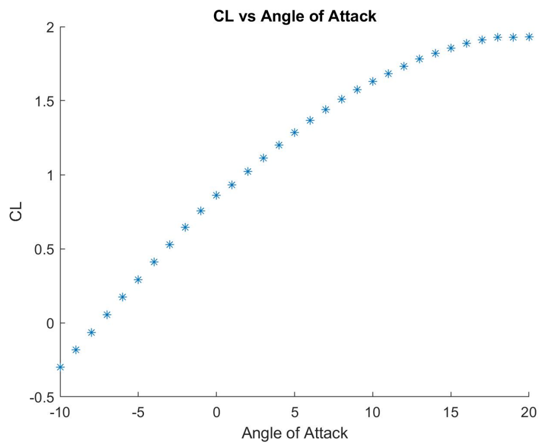

Figure 1: CL vs AoA

It should also be noted that that if an airfoil has a cambered shape to it, it would be able to generate lift at certain negative angles of attack. From figure 1, it can be seen that the angle at which negative lift starts to become positive lift is at an angle of -7 degrees. Meaning at an angle of -6 degrees, the airfoil would be able to still be able to generate lift.

Similar to the coefficient of lift, the coefficient of drag is an instrumental property of an airfoil,as it also changes not only for each airfoil, but due to the angle, Reynolds's Number, and air speed. However drag is a property that is wanted to be as close to zero as possible. In figure 2, it can be seen that the coefficient of drag first starts to decrease and then increases exponentially. The angle at which point the drag is the lowest is at an angle of -2 degrees at a value of roughly 0.008. It should be noted that this is due to pressure drag only and no other types of drag as in skin friction and induced drag from vortices. This is due to the particles being swirled over the airfoil which creates turbulent flow and increases the amount of drag.

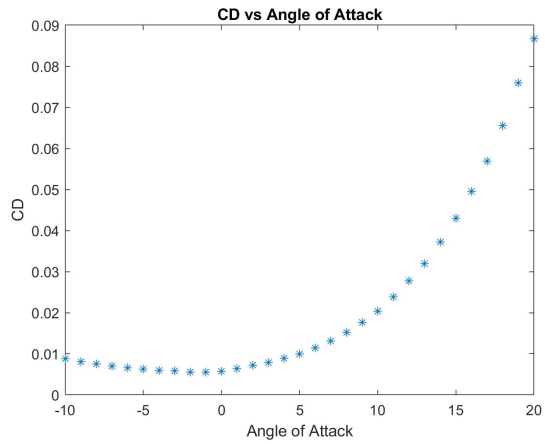

Figure 2: CD vs AoA

The moment coefficient is the moment or torque that is created by the airfoil. This is usually applied at or near the aerodynamic center of the airfoil which is usually at about the quarter chord mark. As seen in figure 3, the values are negative, which would represents stability when pitching. Due to a coefficient of moment being produced, the airfoil would have a tendency to pitch up until no more lift would be produced. In order to prevent this, another moment needs to be created at which it would counter act the moment produced by the airfoil usually done as the tail for an aircraft. According to figure 3, the larger the angle of attack gets, it would produce a higher moment coefficient which could make the system less stable. If the airfoil were to be symmetrical instead of cambered, the moment would be zero. With a Reynolds's number of 3 million, the lowest moment coefficient coefficient is produced at an angle of -2 degrees and its lowest value is -0.218,

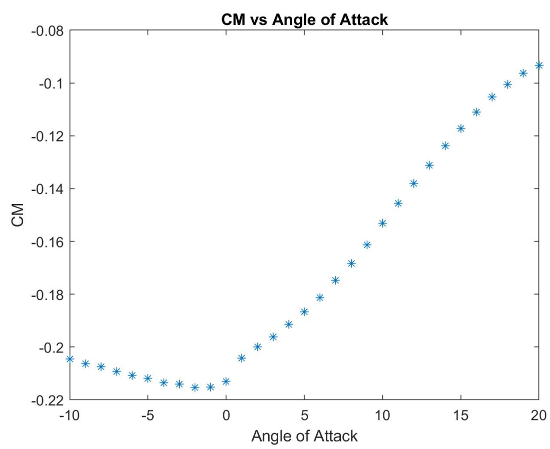

Figure 3: CM vs AoA

The final graph for the polar data is the CL/CD vs angle of attack graph. This graph is used to determine at which angles the airfoil is able to be have its maximum performance. The higher up the graph goes, the better the performance is. In figure 4, the highest point appears at an angle of 2 degrees and has a Cl/Cd of approximately 155.

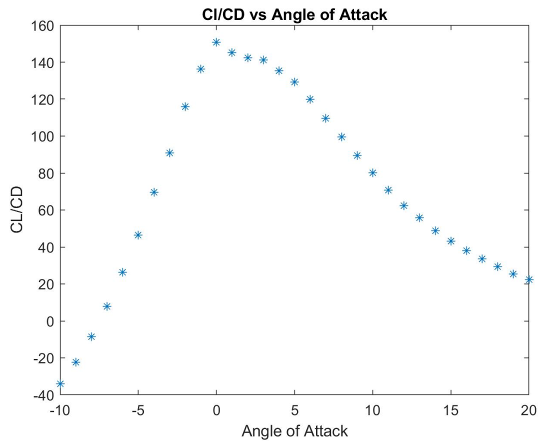

Figure 4: CL/CD vs AoA

B. CP Graphs

The coefficient of pressure distribution graph is a useful graph. This is because by integrating with respect to x/c from 0 to 1, the CL can be found, where x is the position on the chord and c is the chord length. As seen in figure 5, by increasing the angle of attack on an airfoil with a Reynolds number of 3 million, the difference in pressure from the upper and lower halves of the airfoil is increased, thus meaning there is more lift. As the angle is increased, the lowest value for the CP gradually decreases, while the largest value stays roughly the same.

III. Analysis for Reynolds Number of 10 Million

A. Polar Graphs

The CL graph of the airfoil with a Reynolds number of 10 million, is similar to the one with a Reynolds number of 3 million, except for which the Cl values are much higher. This is a result of there being more turbulent flow on the top of the airfoil which would cause a greater pressure gradient between the top and the bottom. As seen in Figure 6, the highest value is roughly 2.25 at an angle of 18 degrees as well.

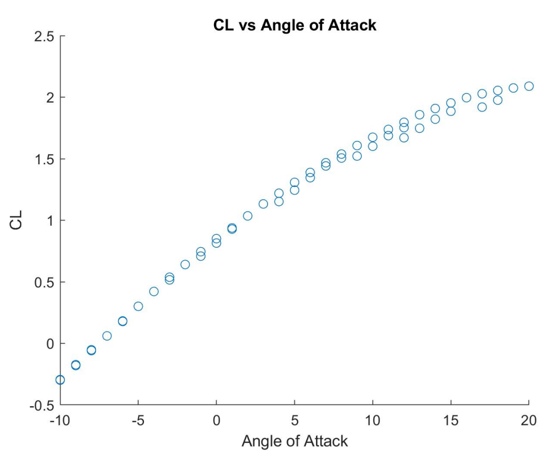

Figure 6: CL vs AoA

The CD graph of the airfoil with a Reynolds number of 10 million is similar to the the airfoil with a Reynolds number of 3 million at the first half, with a similar lowest value of .008 however it happens at an angle of -3 degrees this time as seen in figure 7. However it should be noted that as the angle starts

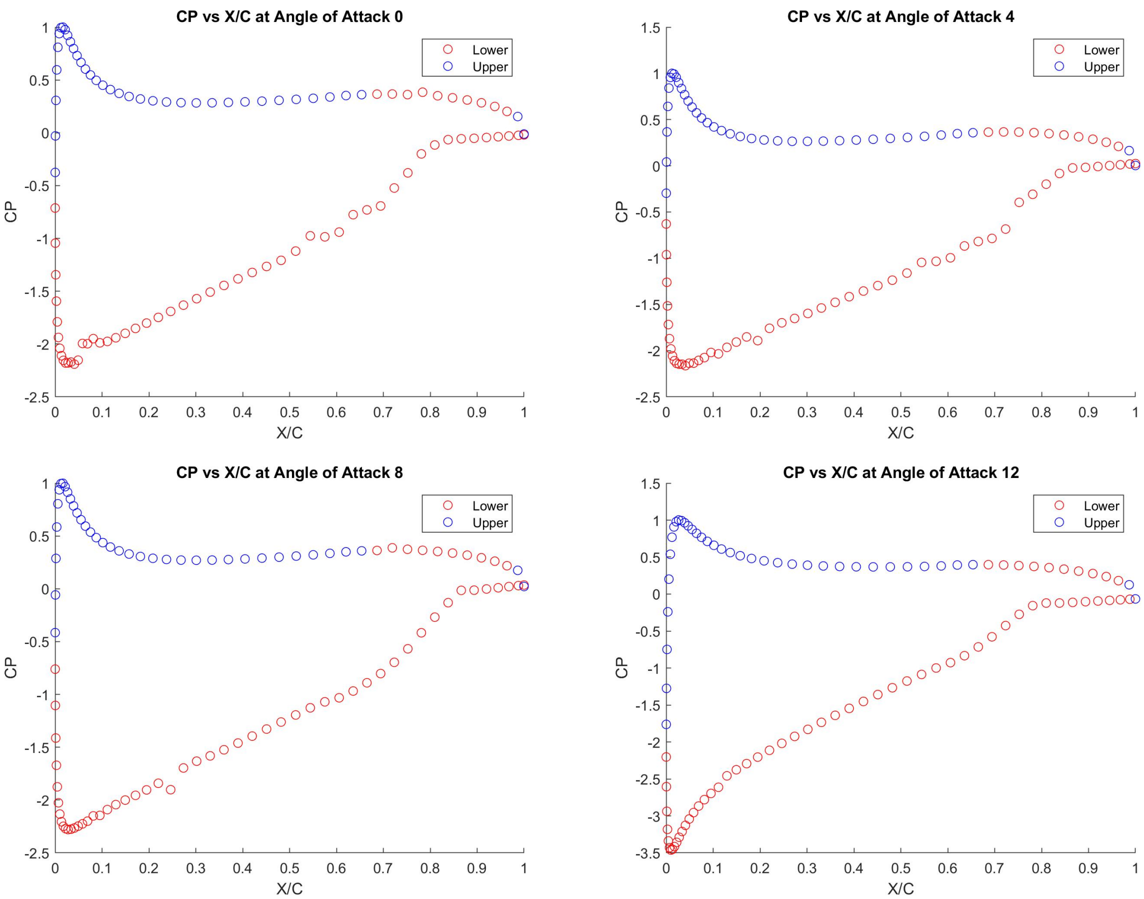

Figure 5: CP at 0, 4, 8, and 12 degrees

to increase, the amount of drag increases as well, but not as fast as compared to the the the airfoil with a Reynolds number of 3 million. This is due to the flow of air being able to stick to the airfoil more and thus creating more of a pressure gradient.

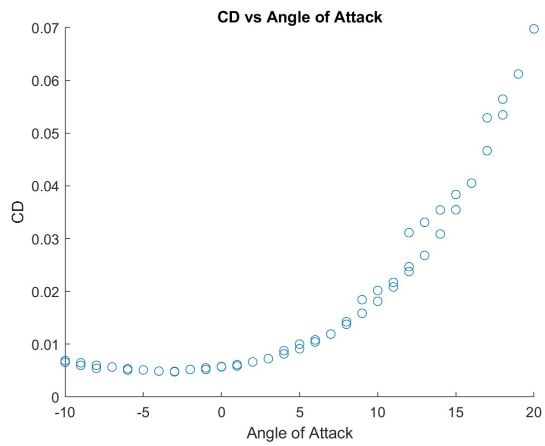

Figure 7: CD vs AoA

When comparing the moment coefficient of the airfoil at a Reynolds number of 3 million and 10 million, the graphs are pretty similar with the lowest value being roughly -0.22. However the angle at which it occurs is different. As seen in figure 8, the lowest moment of coefficient occurs at an angle of -3 degrees compared to -2 from before.As mentioned before, the values are negative, thus confirming that the airfoil has longitudinal stability.

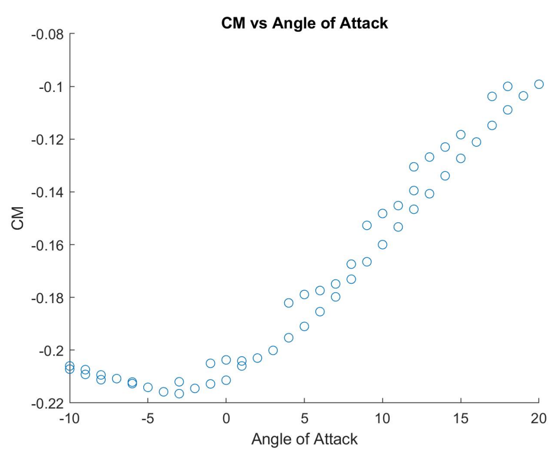

Figure 8: CM vs AoA

The CL/CD graph shows that the airfoil is able to have better performance at the same angles due to its values being higher compared to the airfoil with a Reynolds Number of 3 million. As with the previous airfoil, its highest value is at an angle of 2 degrees, but its value this time is roughly 160, which is higher than the 155 from before.

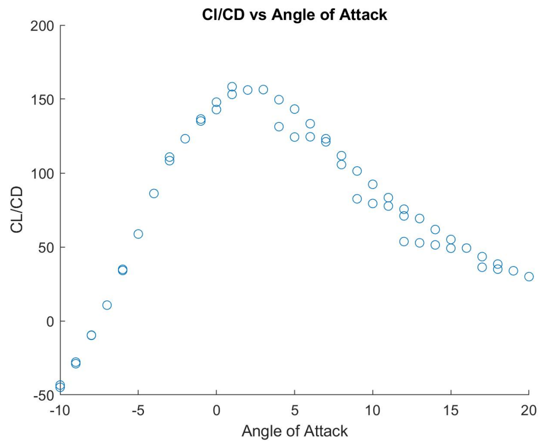

Figure 9: CL/CD vs AoA

B. CP Graphs

As seen in figure 10 the CP graphs tend to have a larger difference of pressure between the top and bottom of the airfoil. As seen in the CP graphs of the previous airfoil, the lowest value decreases as the angle increases, but this time, the value is lower for all four angles of attack compared to the other airfoil, thus result in more lift.

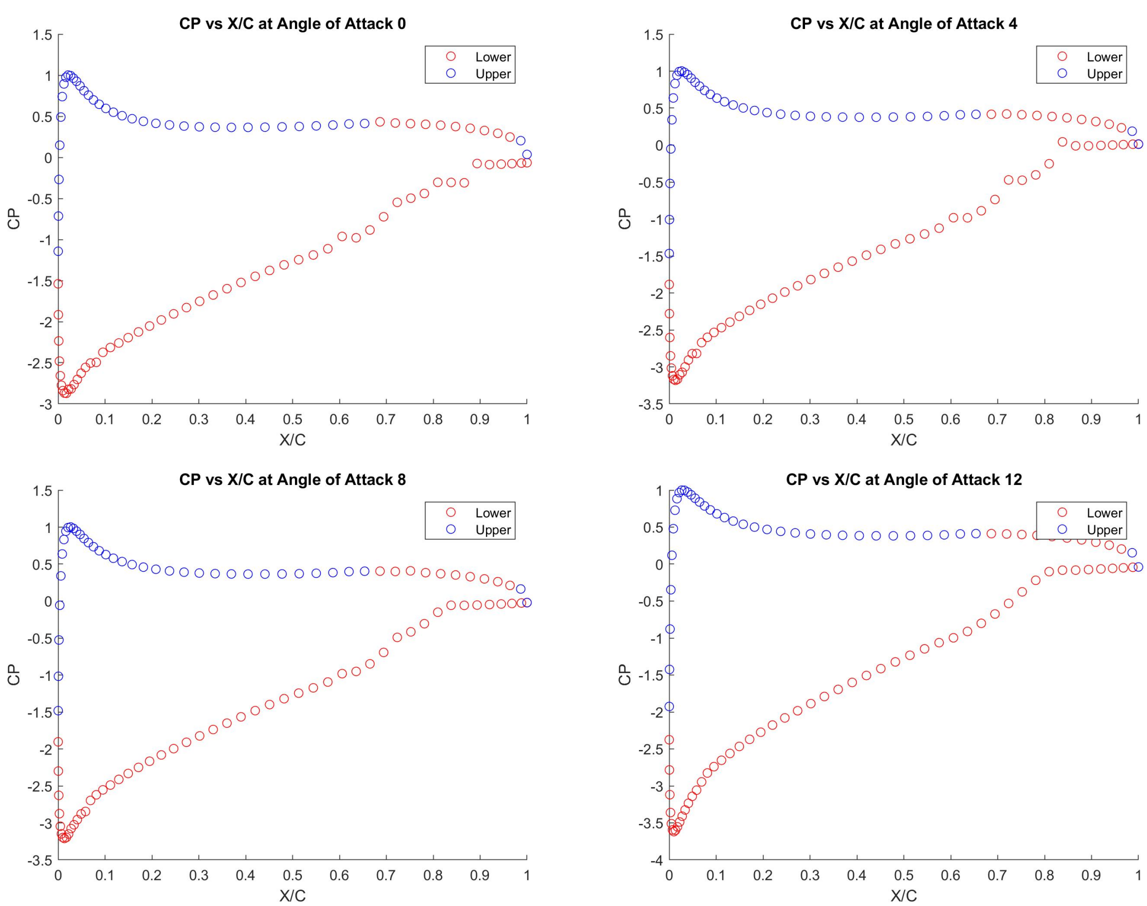

Figure 10: CP at 0, 4, 8, and 12 degrees

IV. Analysis for Reynolds Number of 15 Million

A. Polar Graphs

The CL graph of the airfoil with a Reynolds number of 15 million, with the other airfoils, except for which the Cl values are higher than the first airfoil and lower than the second. As seen in Figure 11, the highest value is roughly 2.1 at an angle of 18 degrees as well.

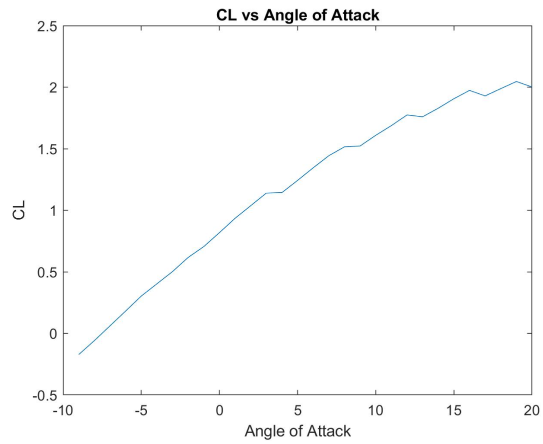

Figure 11: CL vs AoA

the CD graph of the airfoil with a Reynolds number of 15 million is similar to the first two airfoils however, the lowest is still roughly 0.008, but it occurs at an angle of -4 degrees this time as seen in figure 12. It should be noted that as the angle starts to increase, the amount of drag increases as well, but not as fast as compared to the the the airfoil with a Reynolds number of 3 million but faster than the airfoil with a Reynolds number of 10 million.

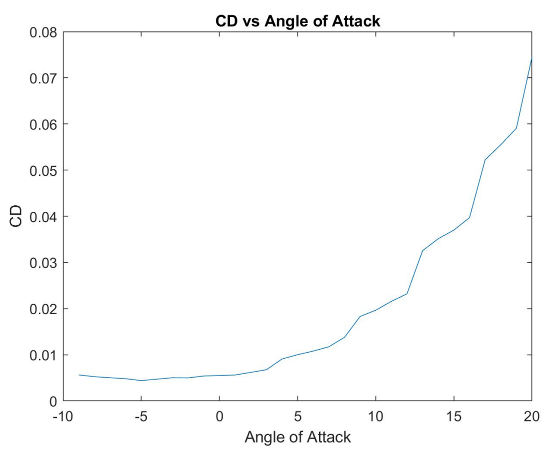

Figure 12: CD vs AoA

When comparing the moment coefficient of the airfoil at a Reynolds number of 3 million,10 million, and 15 million, the graphs are pretty similar with the lowest value being roughly -0.22. However the angle at which it occurs is different. As seen in figure 13, the lowest moment of coefficient occurs at an angle of -5 degrees compared to -2 and -3 degrees.

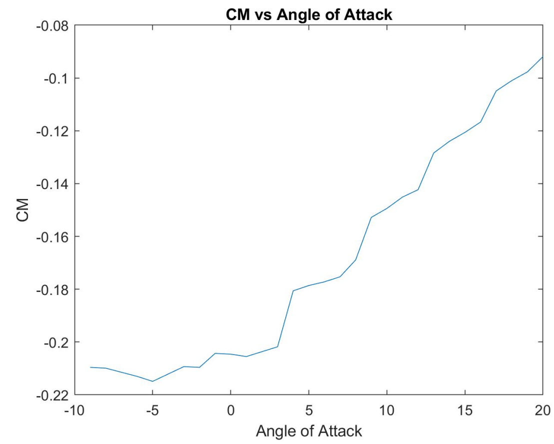

Figure 13: CM vs AoA

The CL/CD graph shows that the airfoil is able to have better performance at the same angles due to its values being higher compared to the airfoils with a Reynolds Number of 3 million and 10 million. As with the previous airfoils, its highest value is at an angle of 2 degrees, but its value this time is roughly at a value 175, which is higher than the 155 and 160 from before.

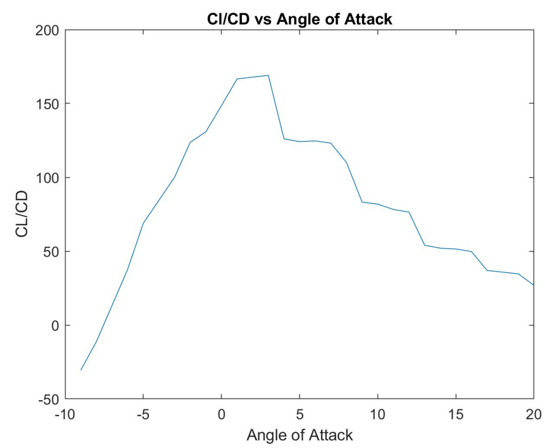

Figure 14: CL/CD vs AoA

B. CP Graphs

As seen in figure 15 the CP graphs tend to have a larger difference of pressure between the top and bottom of the airfoil. As seen in the CP graphs of the previous airfoil, the lowest value decreases as the angle increases, however, the area of the graphs for all the angels are greater than the airfoil with a Reynolds number of 3 million, but is also lower than the airfoil with a Reynolds number of 10 million. As a result the amount of lift generate at the same angles will be greater than the one with a Reynolds number of 3 million but less than the one with 10 million.

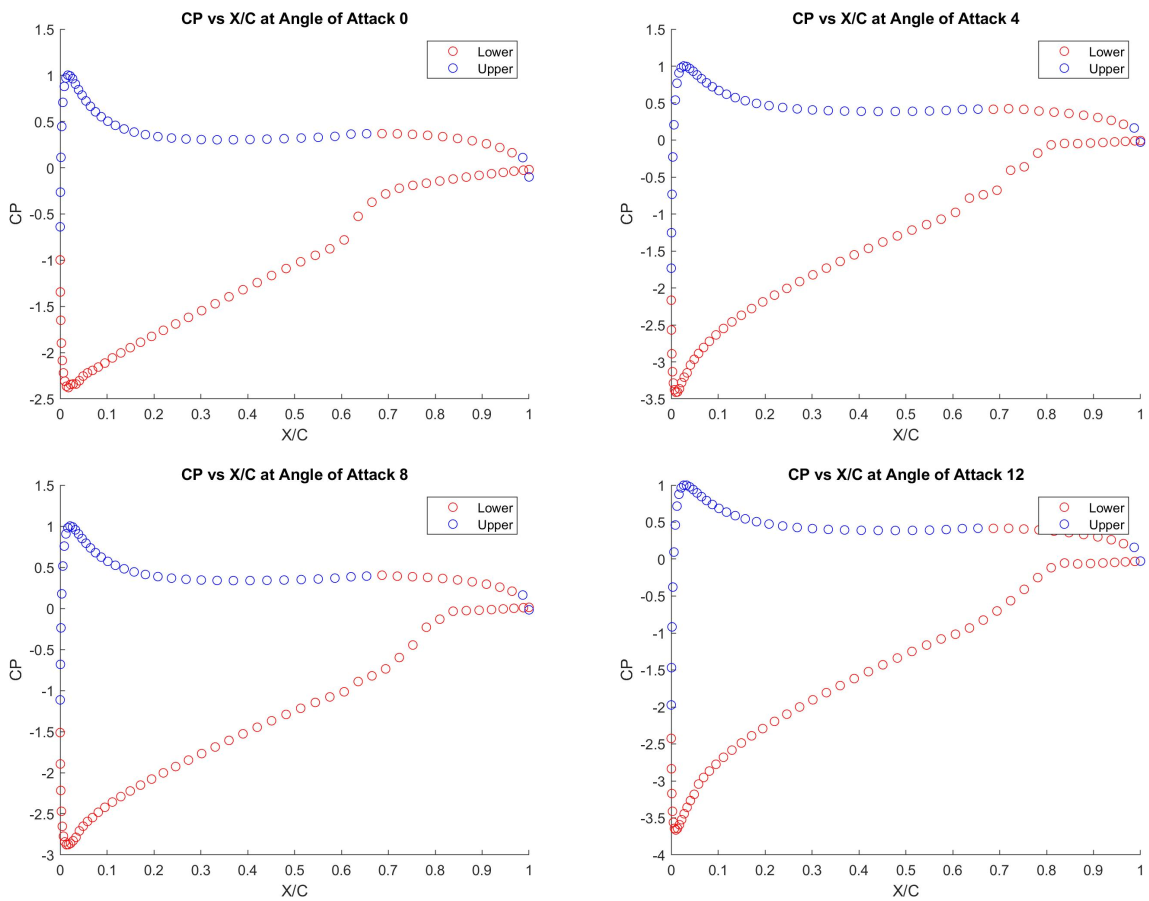

Figure 15: CP at 0, 4, 8, and 12 degrees

V. Boundary Layer

The boundary layer is a thin layer of air flowing over the surface of an airfoil in which the air molecules are sticking on to the airfoil. Due to each layer up on the boundary layer, the air molecules are moving faster, as a result, the top layer of the boundary layer is essentially moving at the same speed of the air. Reynolds number is able to tell us how well the molecules are able to stick onto the airfoil. Withing the boundary layer is where all the aerodynamic forces such as lift and drag take place. As the Reynolds number of an airfoil increases, there tends to be more turbulent flow. Using XFOIL, different properties of the boundary layer can be seen at a certain angle of attack. Figure 16 shows that with a Reynolds number of 3 million, the difference in the thickness and the shape factor at the trailing edge grows up until an angle of attack of 13 degrees and then decreases. However, the momentum thickness is the highest at an angle of 20 degrees but is relatively the same from -10 degrees up until 10 degrees, at which it changes drastically.

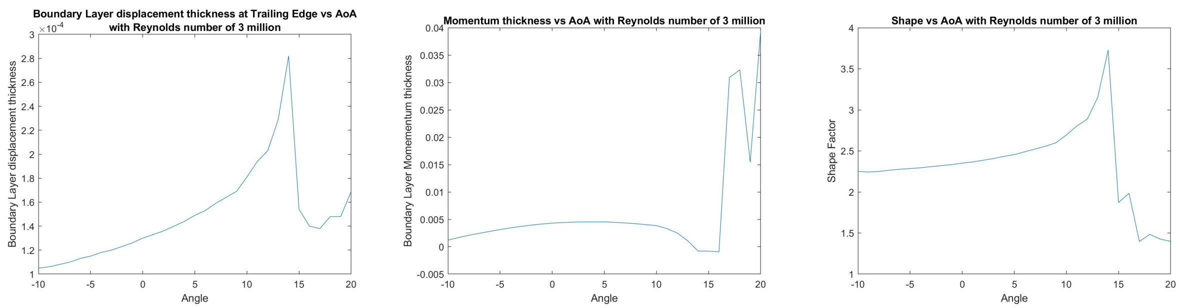

Figure 16: Displacement thickness, Momentum Thickness, and Shape Factor of Boundary layer with Reynolds number of 3 Million

When comparing the graphs of the airfoil with a Reynolds number of 10 million to that of the a Reynolds number of 3 million, it can be seen that the boundary layer thickness is smaller and there is less momentum and the shape factor is a smaller too.

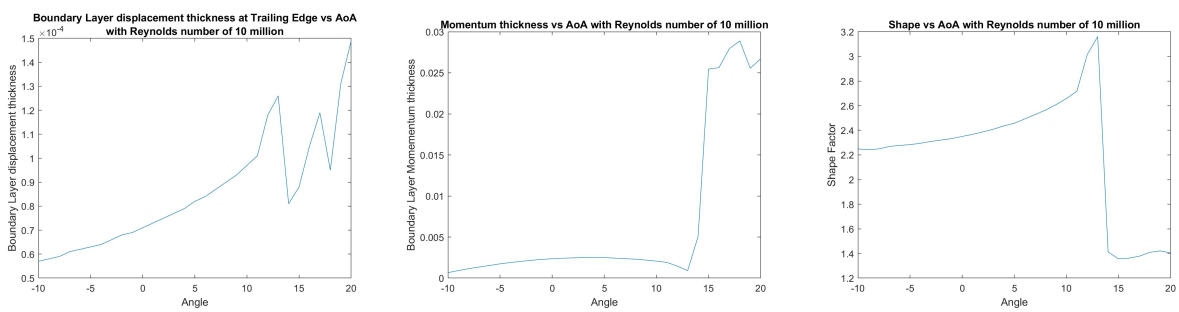

Figure 17: Displacement thickness, Momentum Thickness, and Shape Factor of Boundary layer with Reynolds number of 10 Million

When comparing the graphs of the airfoil with a Reynolds number of 15 million to that of the a Reynolds number of 3 million and 10 million, it can be seen that it has the smallest displacement thickness and momentum thickness, but its shape factor is in between the airfoil with a Reynolds Number of 3 million, which has the biggest value and the airfoil with a Reynolds number of 10 million which is the smallest. It should be noted that the airfoil with the Reynolds number of 3 million is also the only one that has a negative value for its momentum thickness.

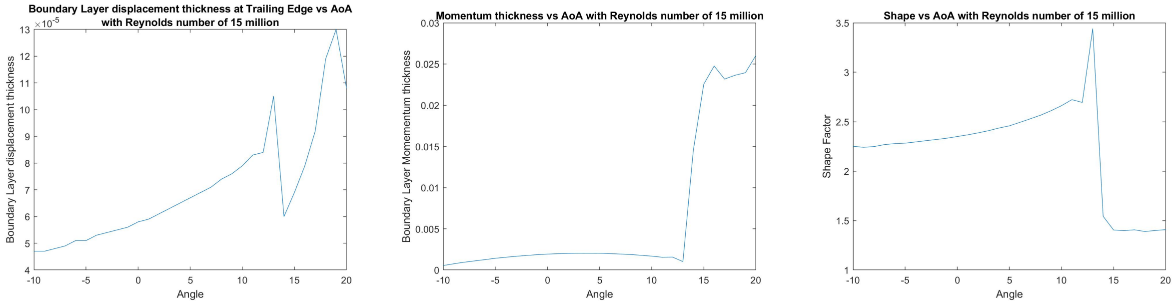

Figure 18: Displacement thickness, Momentum Thickness, and Shape Factor of Boundary layer with Reynolds number of 15 Million

VI. Results

When looking at the three different airfoils, the best one in terms of performance would be the one with the Reynolds number of 15 million. This is due to its Cl/CD values are the largest at low angles of attack. This Reynolds number is also responsible for the momentum thickness graph and displacement thickness that is the smallest, which as a result would mean there is less momentum loss and the smallest amount of potential flow that is lost across the airfoil. Due to the coefficient of drag being roughly the same as the air foil with a Reynolds number of 10 million and lower than the one with 3 million, and its lift being the highest would mean that is does not have to fight against the air as much and would be more efficient as a result as well. However, if cost is much greater to create the airfoil with a Reynolds number of 15 million compared to 10 million, the 10 million one should be picked as the changes in performance are not that drastic.

VII. Appendix

A. Plots

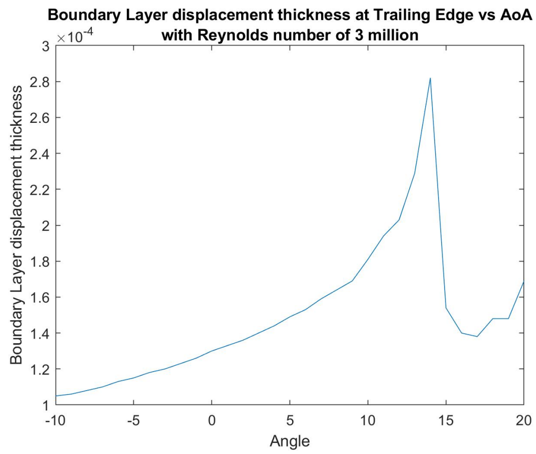

Figure 19: Displacement Thickness

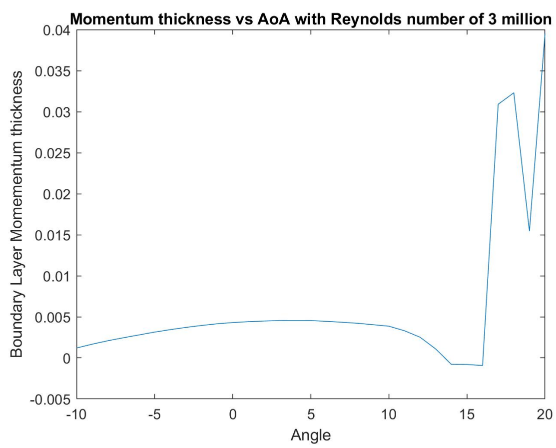

Figure 20: Momentum of Boundary Layer

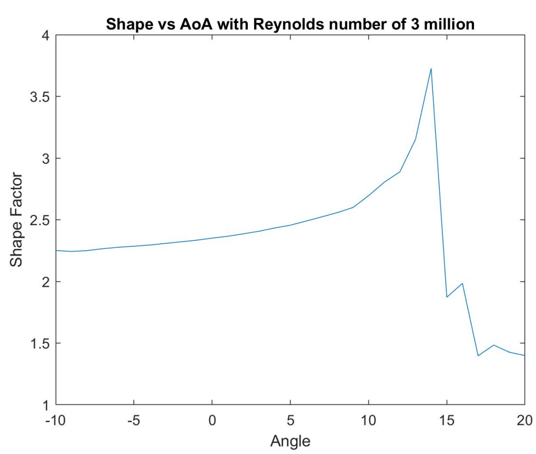

Figure 21: Shape Factor of Boundary Layer

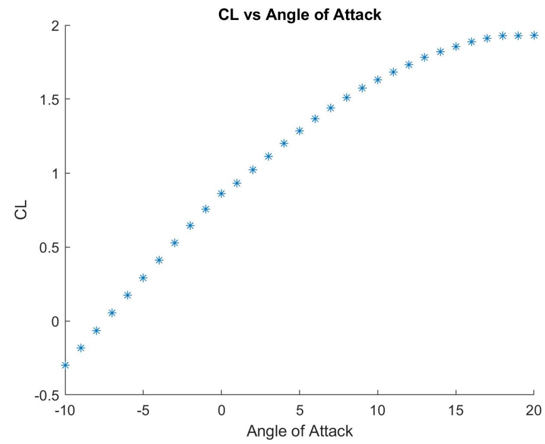

Figure 22: CL vs AoA

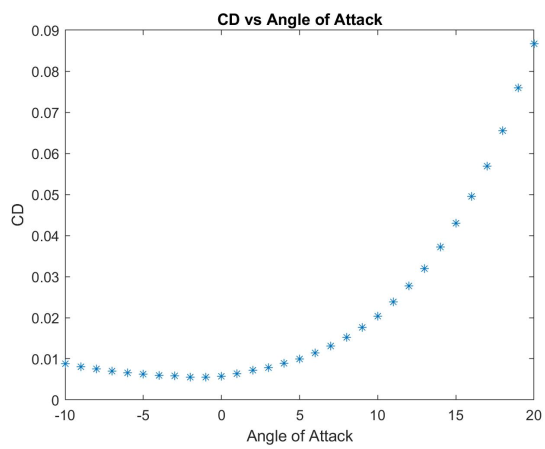

Figure 23: CD vs AoA

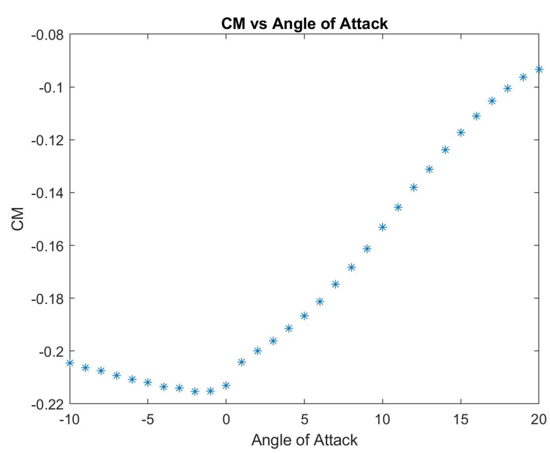

Figure 24: CM vs AoA

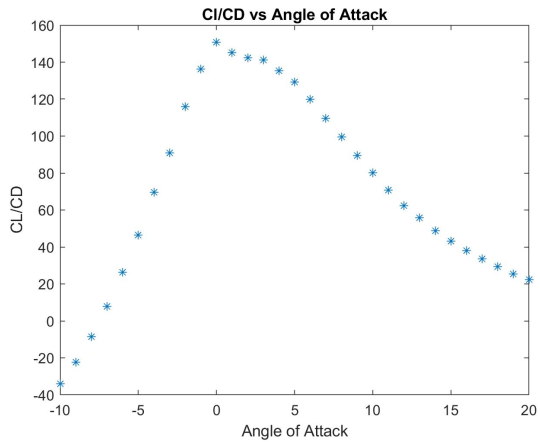

Figure 25: CL/CD vs AoA

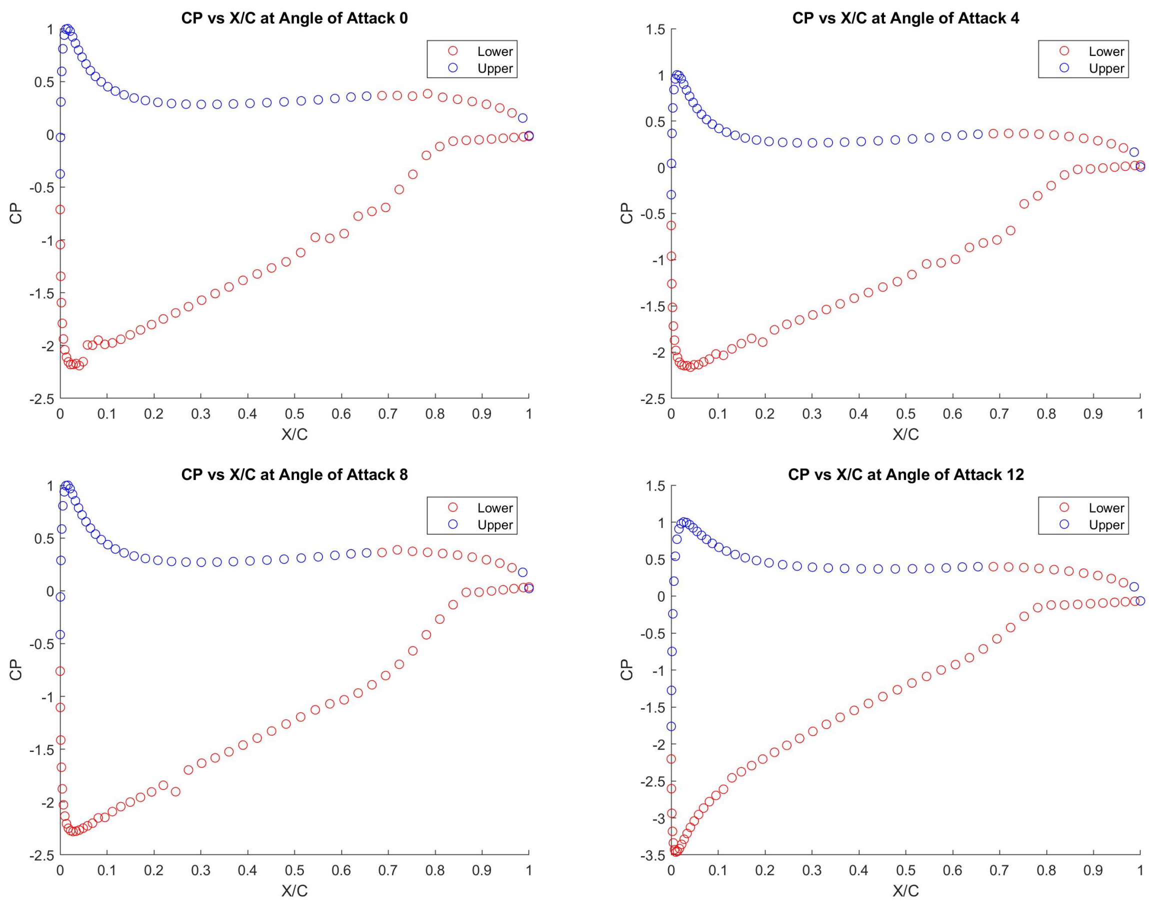

Figure 26: CP at 0, 4, 8, and 12 degrees

1. Reynolds number of 10 million

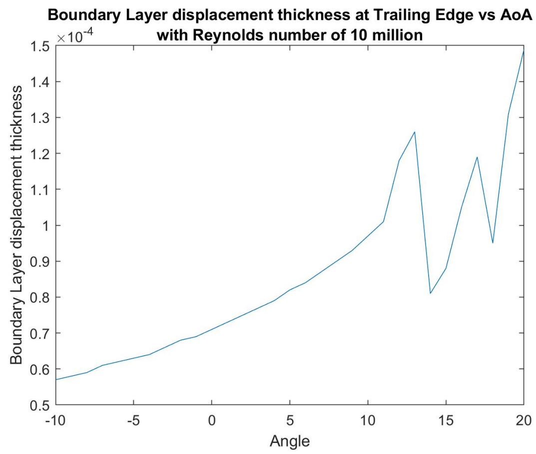

Figure 27: Displacement Thickness

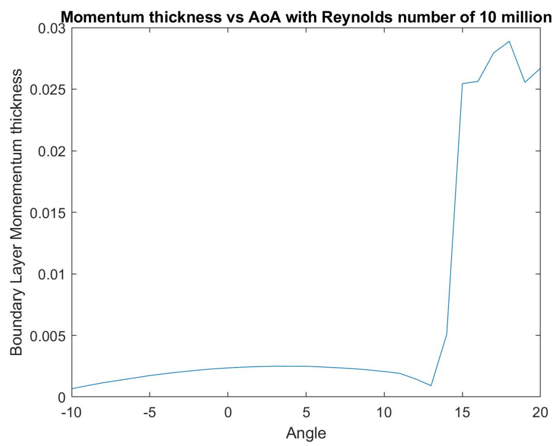

Figure 28: Momentum of Boundary Layer

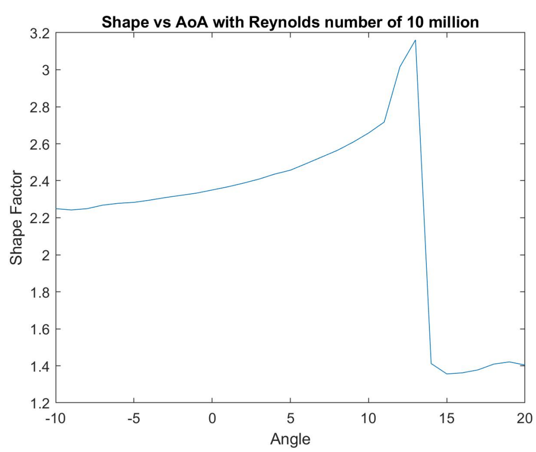

Figure 29: Shape Factor of Boundary Layer

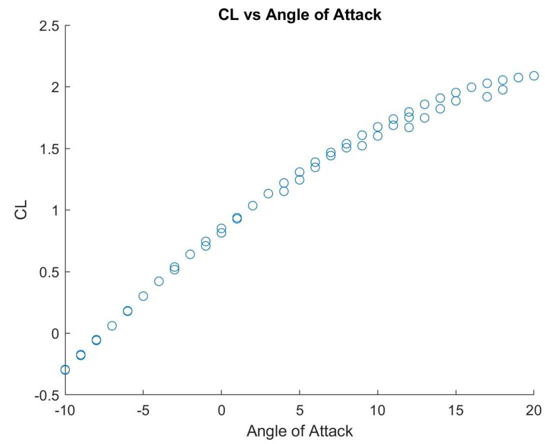

Figure 30: CL vs AoA

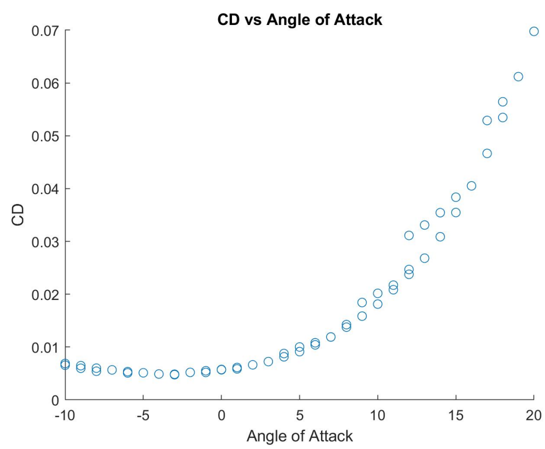

Figure 31: CD vs AoA

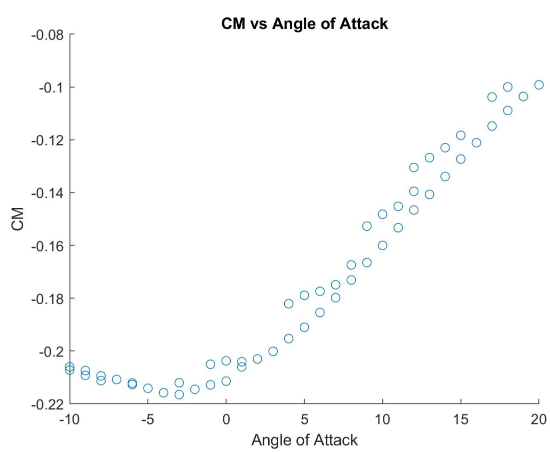

Figure 32: CM vs AoA

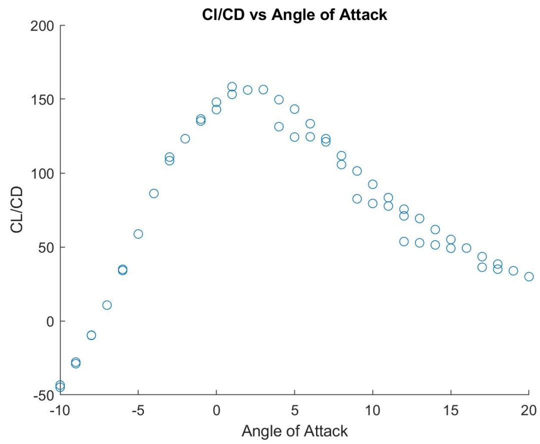

Figure 33: CL/CD vs AoA

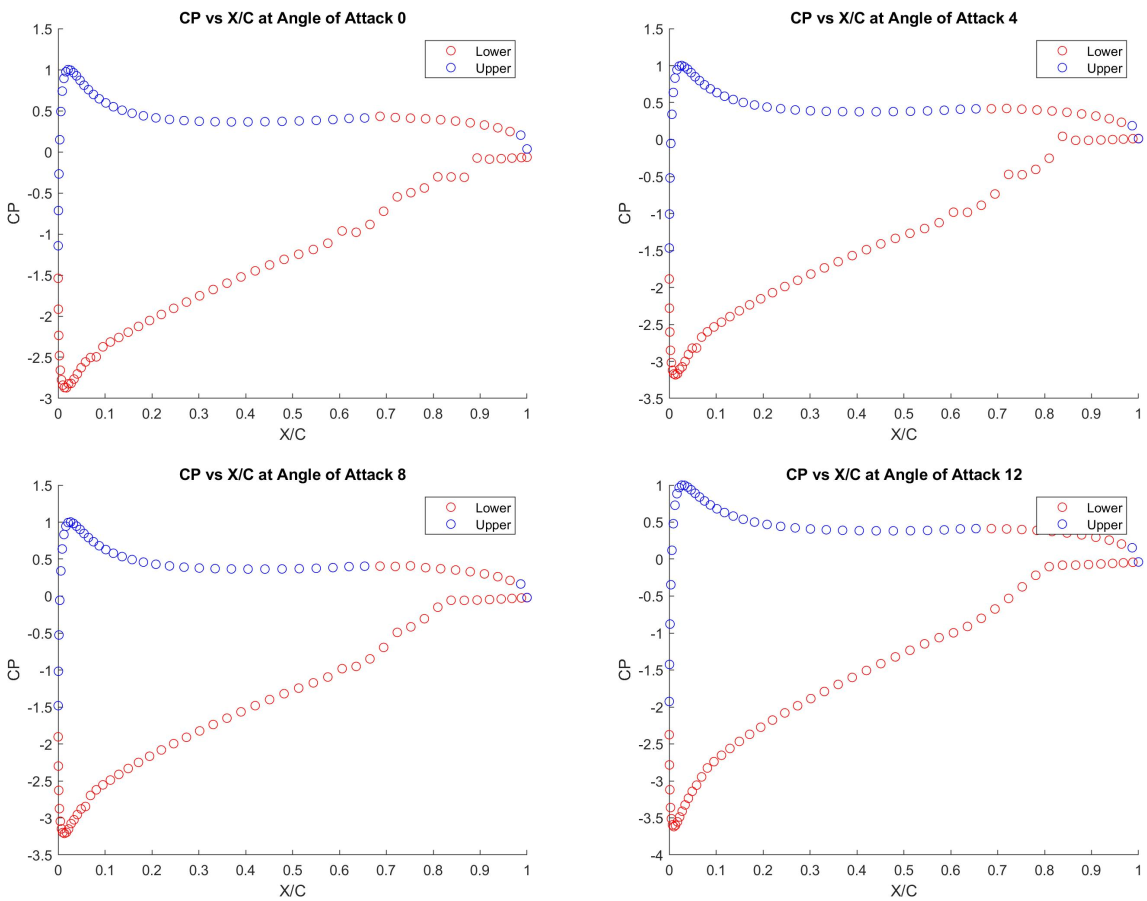

Figure 34: CP at 0, 4, 8, and 12 degrees

2. Reynolds number of 15 million

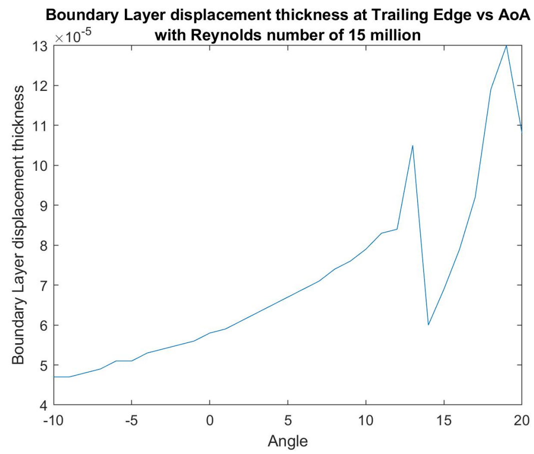

Figure 35: Displacement Thickness

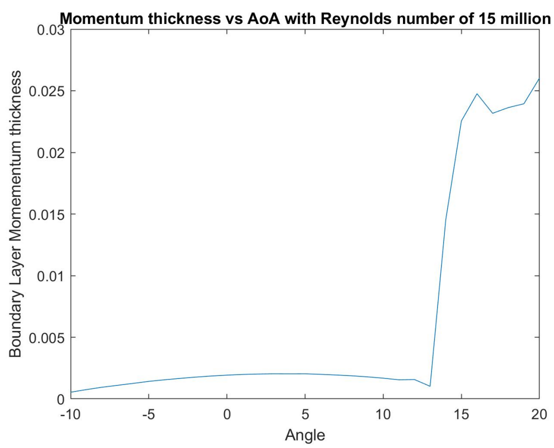

Figure 36: Momentum of Boundary Layer

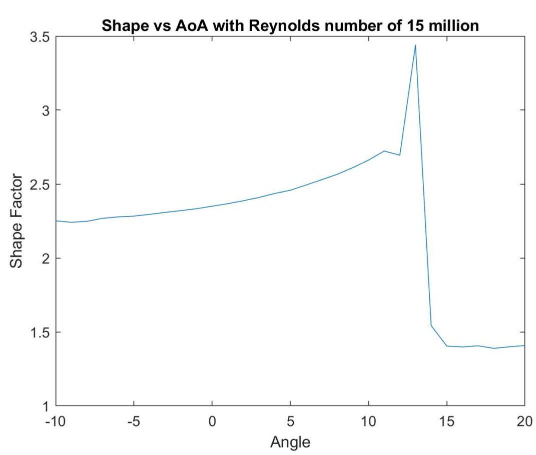

Figure 37: Shape Factor of Boundary Layer

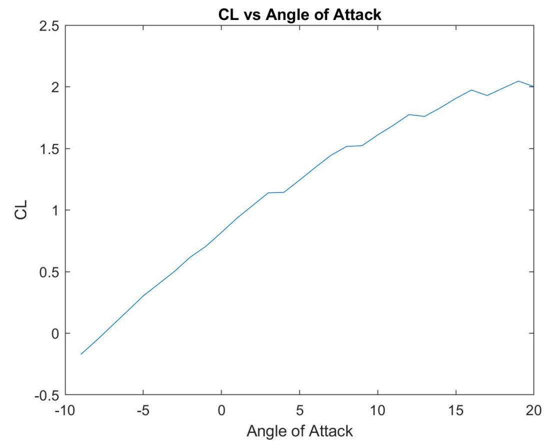

Figure 38: CL vs AoA

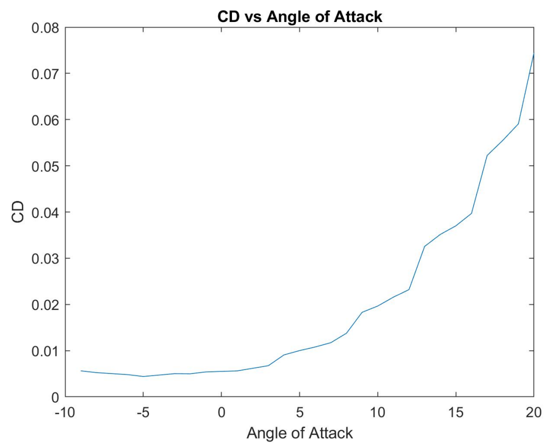

Figure 39: CD vs AoA

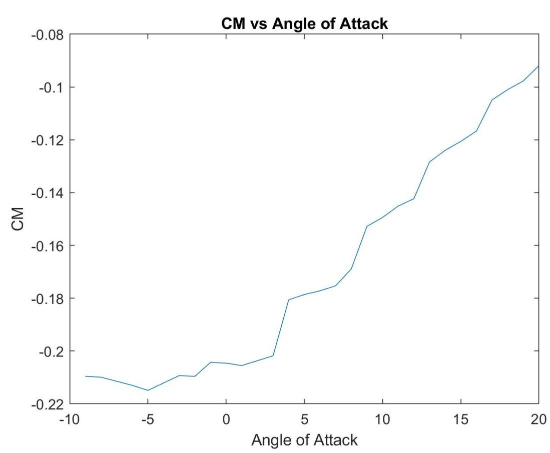

Figure 40: CM vs AoA

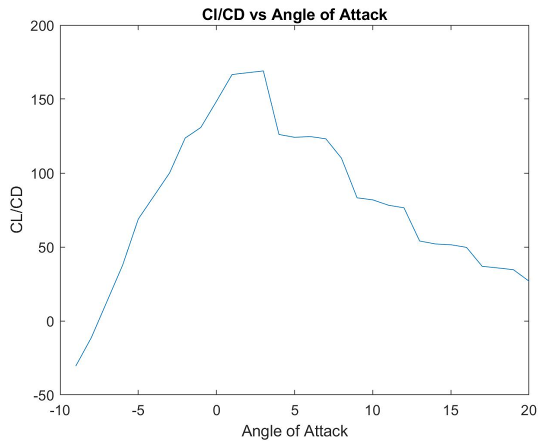

Figure 41: CL/CD vs AoA

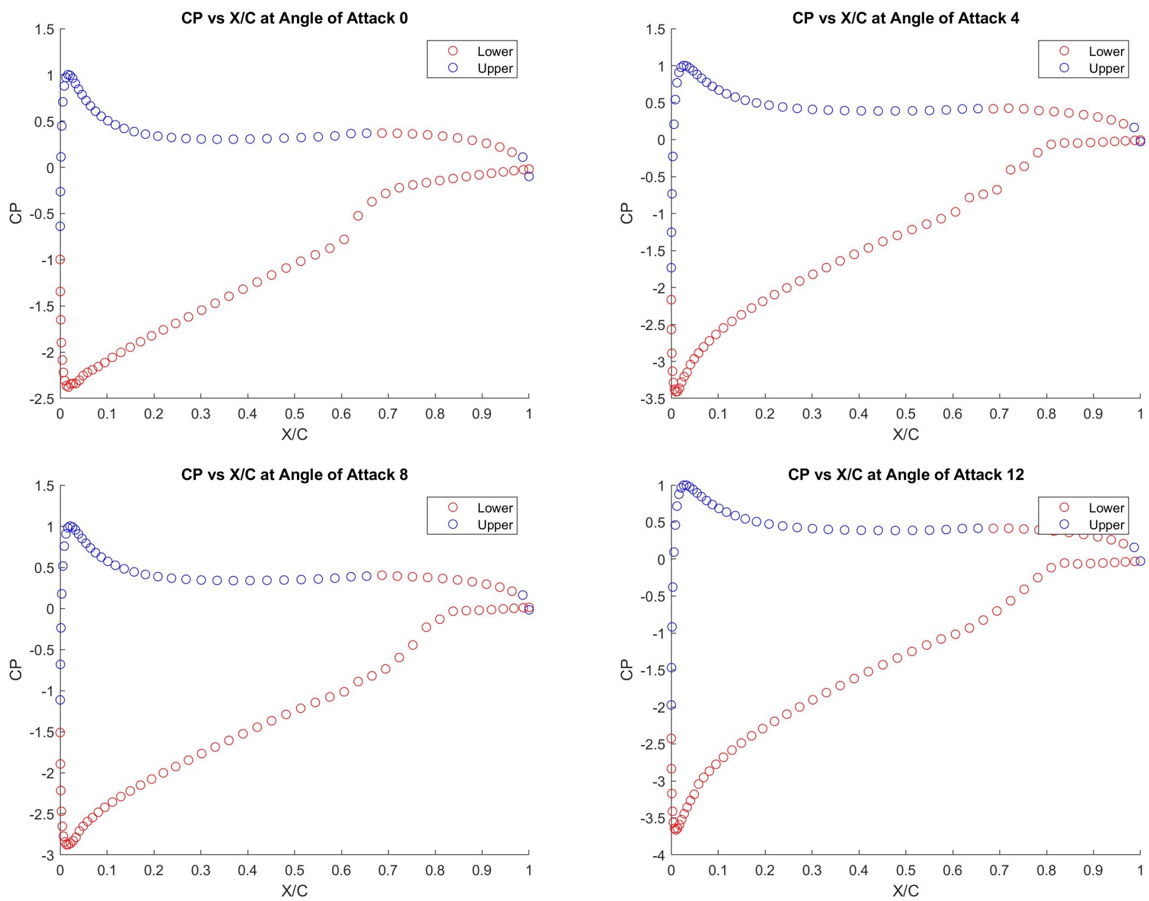

Figure 42: CP at 0, 4, 8, and 12 degrees

B. Code

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%Coding Challenge for Internship Position at
% Siemens Gamesa Renewable Energy
%created By Angelo Hristopoulos on June 04 2020
%Code used for aerodyanamic analysis for NACA 64-618 airfoil

%% Create Xfoil files
clear;
clc;
close all;
% Must delete all
%COnstants
NACA = '6618'; %Type of Airfoil
AoA = '-10'; %First angle of attack
AoA2 = '20'; %Last angle of attack
AoA3 = '1'; %STEP Size between angles of attack

Rey = '3000000'; %Reynolds Number

%Angles of CP data
CP_Alpha1  = '0';
CP_Alpha2  = '4';
CP_Alpha3  = '8';
CP_Alpha4  = '12';

%Number of nodes
numNodes = '100';

%File Names
Airfoil_File = 'Air.csv';
Data_File = 'Data.csv';
CP_File = 'CP.csv';


% Create the airfoil
fid = fopen('xfoil_input.csv','w');
fprintf(fid,['NACA ' NACA '\n']);
fprintf(fid,'PPAR\n');
fprintf(fid,['N ' numNodes '\n']);
fprintf(fid,'\n\n');

% Save the airfoil data points
fprintf(fid,['PSAV ' Airfoil_File '\n']);

% Find the POLAR by creating script for XFOIL
fprintf(fid,'OPER\n');
fprintf(fid,'Visc\n');
fprintf(fid,[Rey '\n']);
fprintf(fid,'M\n');
fprintf(fid,'.1\n');
fprintf(fid,'Pacc\n');
fprintf(fid, [Data_File '\n\n']);
fprintf(fid,['ASEQ' AoA '\n']);
fprintf(fid,[AoA2 '\n']);
fprintf(fid,[AoA3 '\n']);

% Find CP by creating script for XFOIL
fprintf(fid,['Alfa ' CP_Alpha1 '\n']);  %Change for the wanted Angle
fprintf(fid,['CPWR ' CP_File]);

% Close file
fclose(fid);
% Run XFoil using input file
cmd = 'xfoil.exe < xfoil_input.csv';
[status,result] = system(cmd);
%% READ DATA FILE: AIRFOIL
 %Load in and extract the data for CL,CD,CP,CDP
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Airfoil_File = 'Air.csv';
fidAirfoil = fopen(Airfoil_File);
CP_Data = textscan(fidAirfoil,'%f %f','CollectOutput',1,...
'Delimiter','','HeaderLines',0);
fclose(fidAirfoil);
delete(Airfoil_File);

% Separate boundary points
XB = CP_Data{1}(:,1);
YB = CP_Data{1}(:,2);
%% READ DATA FILE and plotting: AOA,CL,CD,CM,CDp for chosen Reynolds Number
Data_File = 'Data.csv';
fidData = fopen(Data_File);
[¬,¬,data] = xlsread('Data.csv');

fclose(fidData);
delete(Data_File);
%Extract Data
data = data(13:end,:);
x = cell2mat(textscan(char(data),'%f'));
data = cell2mat(data);
data = str2num(data);

%AOA
Alpha = data(:,1);
%CL
CL = data(:,2);
%CD
CD = data(:,3);
%CDp
CDp = data(:,4);
%CM
CM = data(:,5);
%CL/CD
CLD = CL./CD;

% Plotting Data
figure
scatter(Alpha,CL,'*')
xlim([-10,20])
title('CL vs Angle of Attack')
ylabel('CL')
xlabel('Angle of Attack')

figure
plot(Alpha,CD,'*')
xlim([-10,20])
title('CD vs Angle of Attack')
ylabel('CD')
xlabel('Angle of Attack')

figure
plot(Alpha,CM,'*')
xlim([-10,20])
title('CM vs Angle of Attack')
ylabel('CM')
xlabel('Angle of Attack')

figure
plot(Alpha,CLD,'*')
xlim([-10,20])
title('Cl/CD vs Angle of Attack')
ylabel('CL/CD')
xlabel('Angle of Attack')
 %% PLOT DATA CP
 %Load in and extract the data
 CP_file = 'CP.csv';
fidCP = fopen(CP_file);
CP_Data = textscan(fidCP,'%f %f %f','HeaderLines',3,...
                            'CollectOutput',1,...
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```
                              'Delimiter','');
fclose(fidCP);
delete(CP_file);
% Separate Cp data
X  = CP_Data{1,1}(:,1);
Y  = CP_Data{1,1}(:,2);
Cp = CP_Data{1,1}(:,3);

% Split Xfoil results into (U)pper and (L)ower
Cp_Upper = Cp(YB > 0);
Cp_Lower = Cp(YB < 0);
X_Upper  = X(YB > 0);
X_Lower  = X(YB < 0);

figure;
hold on
scatter(X_Upper,Cp_Upper,'r');
scatter(X_Lower,Cp_Lower,'b');
xlabel('X Coordinate');
ylabel('Cp');
ylim('auto');
title('CP vs X/C at Angle of Attack 0')
ylabel('CP')
xlabel('X/C')
legend('Lower','Upper')

%% Boundary Layer
%Reynolds numbers
 RN3 = 3000000;
 RN10 = 10000000;
 RN15 = 15000000;

 %Retrieving data for the Boundary layer
 AoA = -10:20;
 for i =1:length(AoA)
[dstar(i),Momentum(i),Shape(i)] = Boundary(RN3,AoA(i));
 end

  for i =1:length(AoA)
[dstar2(i),Momentum2(i),Shape2(i)] = Boundary(RN10,AoA(i));
  end

   for i =1:length(AoA)
[dstar3(i),Momentum3(i),Shape3(i)] = Boundary(RN15,AoA(i));
 end
 %Plotting graphs for the Boundary layer properties of displacement,
 %momentum and shape factor
 figure
plot(AoA,dstar)
title({'Boundary Layer displacement thickness at Trailing Edge vs AoA','with Reynolds ...
    number of 3 million'})
xlabel('Angle')
ylabel('Boundary Layer displacement thickness')

 figure
plot(AoA,Momentum)
title('Momentum thickness vs AoA with Reynolds number of 3 million' )
xlabel('Angle')
ylabel('Boundary Layer Momementum thickness')
 figure
plot(AoA,Shape)
title('Shape vs AoA with Reynolds number of 3 million')
xlabel('Angle')
ylabel('Shape Factor')

figure
plot(AoA,dstar2)
title({'Boundary Layer displacement thickness at Trailing Edge vs AoA','with Reynolds ...
    number of 10 million'})
xlabel('Angle')
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ylabel('Boundary Layer displacement thickness')

 figure
plot(AoA,Momentum2)
title('Momentum thickness vs AoA with Reynolds number of 10 million' )
xlabel('Angle')
ylabel('Boundary Layer Momementum thickness')
 figure
plot(AoA,Shape2)
title('Shape vs AoA with Reynolds number of 10 million')
xlabel('Angle')
ylabel('Shape Factor')


figure
plot(AoA,dstar3)
title({'Boundary Layer displacement thickness at Trailing Edge vs AoA','with Reynolds ...
    number of 15 million'})
xlabel('Angle')
ylabel('Boundary Layer displacement thickness')

 figure
plot(AoA,Momentum3)
title('Momentum thickness vs AoA with Reynolds number of 15 million' )
xlabel('Angle')
ylabel('Boundary Layer Momementum thickness')
 figure
plot(AoA,Shape3)
title('Shape vs AoA with Reynolds number of 15 million')
xlabel('Angle')
ylabel('Shape Factor')
```

```
  function [dstar,rtheta,HK] = Boundary(Rn,AoA)

  %Function will be used to find the boundary layer properties for all
  %reynolds numbers
NACA = '6618'; %Type of Airfoil
Rey = num2str(Rn); %Reynolds Number
AoA = num2str(AoA);
%File Name
Bound_File = 'bound.csv';

% Create the airfoil
fid = fopen('xfoil_input.csv','w');
fprintf(fid,['NACA ' NACA '\n']);
fprintf(fid,'PPAR\n');
fprintf(fid,'\n\n');

% Find the Boundary layer data in XFoil
fprintf(fid,'OPER\n');
fprintf(fid,'Visc\n');
fprintf(fid,[Rey '\n']);
fprintf(fid,'Alfa\n');
fprintf(fid,[AoA '\n']);
fprintf(fid,'BL \n');
fprintf(fid,'Dump \n');
fprintf(fid, 'bound.csv \n\n');
% Close file
fclose(fid);
% Run XFoil using input file
cmd = 'xfoil.exe < xfoil_input.csv';
[¬,¬] = system(cmd);
Bound_File = 'bound.csv';
fidData = fopen(Bound_File);
[qww,qw,data] = xlsread('bound.csv');

fclose(fidData);
 delete(Bound_File);
```

```
%extracting data for the trailing edge only
data = data(2:end,:);
x = cell2mat(textscan(char(data),'%f'));
data = cell2mat(data);
data = str2num(data);
chord = data(66,1);
thick = data(66,2);
dstar = data(66,5);
rtheta = data(66,7);
HK = data(66,end);
[dstar,rtheta,HK];
  end
```