NACA Airfoil

EGME 508: Advanced Inviscid Fluid Flow
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Contents

1 Abstract

A NACA MPXX airfoil is defined by four digits relating the Maximum Camber, Position of the Maximum Camber, and the Maximum Thickness. With these four digits, and relating equations, an airfoil can be created through code. Alternatively, data points from a Clark Y airfoil can be used to directly create a model for an airfoil. With information pertaining to the Clark Y airfoil, the information of a 4 digit NACA airfoil can be computed though slightly different means. The Milne-Thomson definitions of what a chord is will also be applied and discussed further.

Certain geometry is important toward understanding an airfoil with the NACA 4 digit standard describing an airfoil based on Maximum Camber, Position of Maximum Camber, and Maximum Thickness. These three dimensions are often enough to accurately describe an airfoil. However there may be times when the dimensions are not given and instead need to be found such as with this Clark-Y airfoil. The camber line splits the airfoil in two between upper and lower portions with maximum camber being the position where this curve is furthest from the chord. Given data for the Clark-Y airfoil experiment by Silverstein, the goal was to recreate the profile of the airfoil and plot it within Matlab while also calculating the NACA dimensions to accurately describe the airfoil.

2 Symbols and Units

Symbol	Measurement	Units
M	Max Camber	%
P	Max Camber Location	%
XX	Thickness	%
C	Chord	Units
α	Angle of Attack	° (degrees)
Y_U	Upper Coordinates	Units
Y_L	Lower Coordinates	Units
Y_{camber}	Camber Coordinates	Units

Note: Data is non-dimensionalized.

3 Results

Using Silverstein's data for the Clark Y Airfoil plots were created to show the profile of the airfoil and other critical information. Maximum Camber, Chord line, Maximum Thickness, and the Leading Edge were all found and plotted using the data provided and computational programs such as MATLAB.

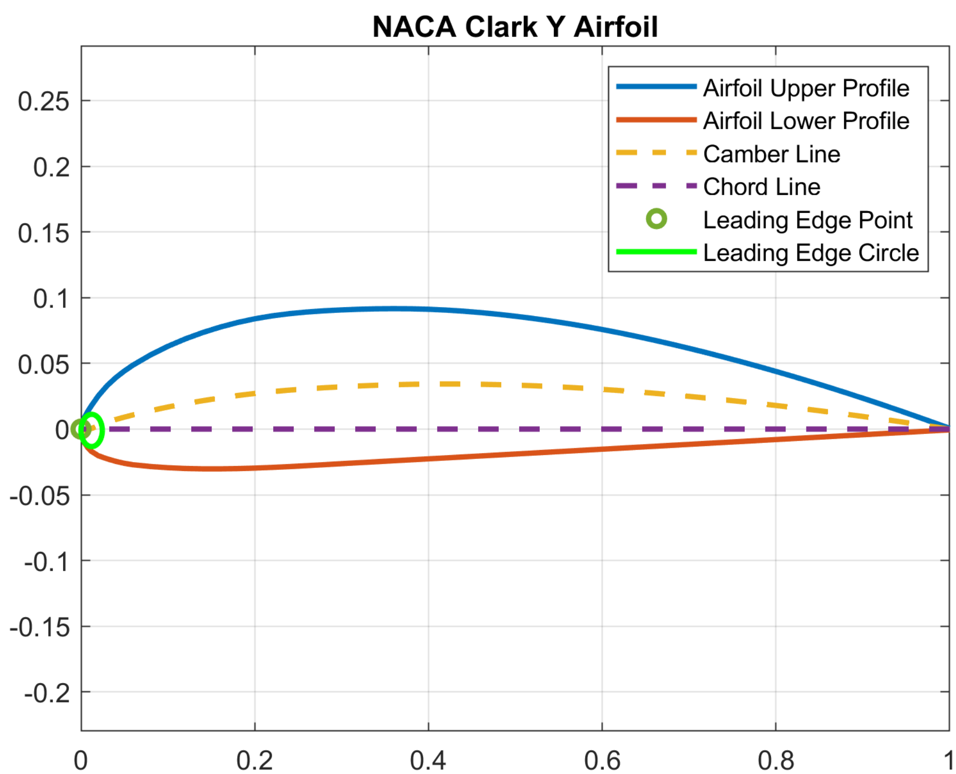

Figure 1. Tip to Tip Chord

Figure 1 shows the airfoil profile that was plotted using the normalized data provided by Silverstein [2]. The leading edge's circle was determined by sampling the first x points from the leading and trailing edge. With these points, circles of minimum radius can be created where their centers can be joined to create the chord line, per one of Milne-Thomson's chord line definitions. These circles were created and plotted using the fitCircle function on MATLAB.

The camber line was determined by averaging the position of the upper and lower coordinates of the airfoil, which are normalized by the chord length, and plotting the resultant coordinates.

$$Y_{camber} = \frac{Y_U + Y_L}{2}$$

It follows the curve of the airfoil profile and is an accurate representation of the camber line. The chord line was found by connecting the leading edge and trailing edge of the airfoil with a straight line. The airfoil profile begins and ends at the 0 mark on the x-axis.

Additionally, the position and value of maximum camber were determined with MATLAB. By utilizing the Camber Line as well as MATLAB's "max" function, the maximum camber percentage and its location can be determined. It was calculated that the Clark Y airfoil has a Maximum Camber of 3.4331% located at 42% of the way from the leading edge.

Furthermore, the maximum Thickness was calculated with the following equation:

$$Maximum \quad Thickness = Y_U - Y_L$$

Once more, MATLAB's "max" function was used on the results to obtain the value for the Maximum Thickness. The Clark Y airfoil has a Maximum Thickness of 11.7% located at 28% of the way from the leading edge.

Assuming inviscid and incompressible flow, it was determined that the NACA 3411 is identical to the Clark Y airfoil [3]. This NACA airfoil supports the findings that MATLAB calculated on the Clark Y data entries. The NACA 3411 airfoil has a 3% maximum camber located 40% from the leading edge with am 11% maximum thickness. All of which are similar to the data calculated via MATLAB,

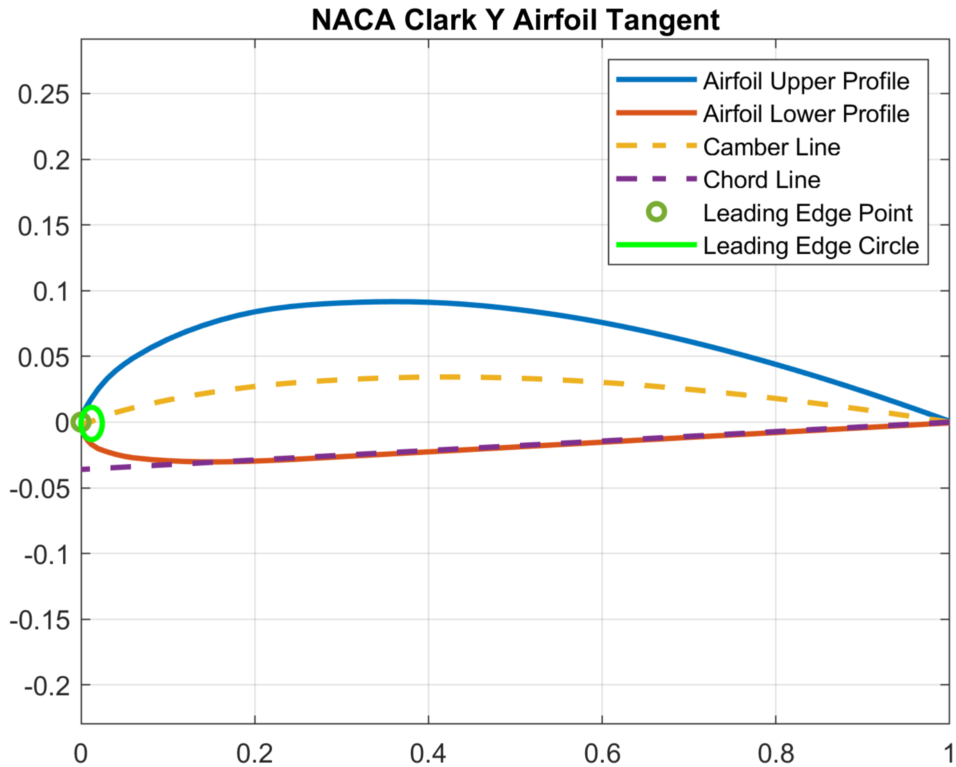

Figure 2. Tangent Line Chord

Figure 2 displays the airfoil where a different definition of chord line has been used. The definition states a chord line should be a double tangent line on the lower edge of the airfoil. Interestingly, the double tangent line would need to be plotted on two curved edges. However, the trailing edge of the airfoil is a point with this data set. For the double tangent to be applied, one curve is used with the trailing edge point. The result yields a chord that differs from that of Figure 1. The differences between the chord in Figures 1 and 2 can be closely seen in Figure 3.

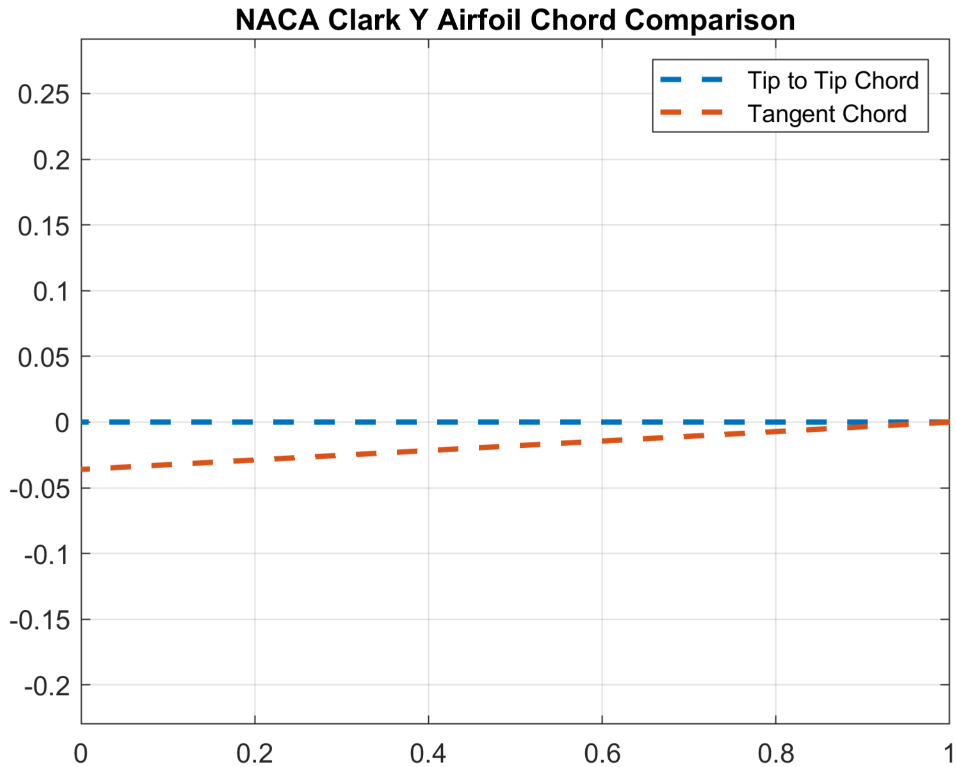

Figure 3. Chord Comparison

In Figure 3, the different definitions on what a chord is, by Milne-Thomson, can be observed [4]. The blue line represents to the longest line that can be created within the context of an airfoil. It can otherwise be denoted as the line connecting the leading and trailing edges. Both yield the exact same chord. The orange line refers to the double tangent. The Clark Y airfoil profile can be defined with this method. In this case, a double tangent line is created to define the chord. The tangent chord yields a different chord that our first definition of leading to trailing edges. Because there is a discrepancy between both chords, the geometric incidence changes. Incidence is another term for the angle of attack, α [4]. When the chord is parallel to the direction of motion, there is no angle of attack that can be applied, i.e. $\alpha = 0$. This means that the incidence is the same at each segment or section of the airfoil. However, in the case where $\alpha \neq 0$, incidence is present, This is referred to as twist [4]. Furthermore, if a different chord is used, incidence may differ as well. Typically the definition of a chord falls with one of the three methods discussed, but in some cases a different definition may be applied. Since there is no definite definition, as long as consistency is present, any line could be denoted as a chord.

4 Discussion and Conclusion

Airfoils are the cornerstone structure of aerodynamic analysis. Wind tunnel tests on different airfoils give an insight into phenomena such as turbulence, boundary layer theory, and other vital fluid dynamic concepts. Therefore, a standardized method to construct and identify airfoils is paramount in the pursuit of aerodynamic research. NASA achieved this first by normalizing all variables in the NACA airfoil series. In this way, sets of airfoils can be looked at and tested whether in a scaled wind tunnel or a computational simulation for a Cessna or a Boeing 747.

This project demonstrated the potential for both modeling airfoil from given dimensions and using given dimensions for the upper and lower edges to then find the key properties of an NACA 4 digit airfoil. This means that just from the raw geometry, we can determine the exact classification of an airfoil, in the case it is not known. The computer model also showed how different interpretations of the chord produce different angles (to balance the changed perspective) so that the measurements would come out the same, as changing the definition of the chord does not change the physical situation.

5 References

[1] "Circle_fit." File Exchange - MATLAB Central, www.mathworks.com/matlabcentral/fileexchange/36361-circle_fit. Accessed 7 Sept. 2024.

[2] Silverstein, Abe. "Scale Effect on Clark y Airfoil Characteristics from NACA Full-Scale Wind-Tunnel Tests - NASA Technical Reports Server (NTRS)." NASA, NASA, 1 Jan. 1935, ntrs.nasa.gov/citations/19930091575.

[3] Chakraborty, Manash. (2015). A Computational Study on two horizontally close sequential airfoils to determine conjoined pressure distribution and aerodynamic influences on each other. 10.13140/RG.2.1.5041.4562.

[4] Milne-Thomson, Louis Melville. "Theoretical Aerodynamics.Pdf." Theoretical Aerodynamics, Dover Publications, Inc., 20 Jan. 2011, aerostarsolutions.wordpress.com/wp-content/uploads /2011/10/theoretical-aerodynamics.pdf.

6 Appendix A

6.1 MATLAB Code

```
clc, close all, clear, clear variables;

T=importdata("Clark Y Airfoil.txt");
x=T.data(:,1); % X Data set Export
Yup=T.data(:,2); % Y Upper Data set Export
Ylow=T.data(:,3); % Y Lower Data set Export
Ycamber=(Yup+Ylow)/2; % Camber at each point in the data set
[Ycmax,index_max_camb]= max(Ycamber); % Y Position of Max Camber and Position in data set
xmax=x(index_max_camb); % X Position of Max Camber
g = min(x);

% Finding Leading Edge Radius Thingy
[x_le, index_leading_edge] = min(x);
yup_le = Yup(index_leading_edge);
ylow_le = Ylow(index_leading_edge);
y_le = (yup_le+ylow_le)/2;

% Getting Data Set from Leading Edge (CAN CHANGE)
flipped_x = flipud(x);
flipped_yup = flipud(Yup);
flipped_ylow = flipud(Ylow);

% Getting Data from nose of airfoil
Sample_set = 4;
Sample_set_x = flipped_x(1:Sample_set);
Sample_set_yup = flipped_yup(1:Sample_set);
Sample_set_ylow = flipped_ylow(1:Sample_set);

% Use fitcircle function to uhhhh fit a circle :P
[xc_up, yc_up, r_le_up] = fitCircle(Sample_set_x,Sample_set_yup);
[xc_low, yc_low, r_le_low] = fitCircle(Sample_set_x,Sample_set_ylow);
% Average Lower and Upper Radii
r_le_avg = (r_le_up+r_le_low)/2;

% Circle data
theta = linspace(0, 2*pi, 100);
% Upper (ended up being useless)
x_circle_up = xc_up+r_le_up*cos(theta);
y_circle_up = yc_up+r_le_up*sin(theta);
% Lower
x_circle_low = xc_low+r_le_low*cos(theta);
y_circle_low = yc_low+r_le_low*sin(theta);

%% Plot 1
figure(1)
set(groot,'DefaultLineLineWidth',2)
plot(x,Yup) % Plotting Upper Half of Airfoil Profile
hold on
plot(x,Ylow) % Plotting Lower Half of Airfoil Profile
plot(x,Ycamber, '--') % Plotting Camber Line
plot(x,g+x*0, '--') % Plotting Chord Line
plot(x_le,y_le, 'o') % Plotting Leading Edge
plot(x_circle_low, y_circle_low, 'g'); % Plotting circle
hold off
legend('Airfoil Upper Profile', 'Airfoil Lower Profile', 'Camber Line', 'Chord Line', 'Leading
    Edge Point', 'Leading Edge Circle')
xlim([0,1]) % Adjusting Aspect Ratio of plot
ylim([(min(Ylow)-.2) (max(Yup)+.2)])
title('NACA Clark Y Airfoil')
grid on
```

```
%% Plot 2 Information
[Low_Point_Ycoord, Low_Point_Xcoord] = min(Ylow);
Low_Point_Xcoord = x(Low_Point_Xcoord);
End_Xcoord = max(x);
End_Ycoord = Ylow(end);
Slope = (End_Ycoord - Low_Point_Ycoord) / (End_Xcoord - Low_Point_Xcoord);
m = Slope;
b1 = Low_Point_Ycoord - m * Low_Point_Xcoord;
b2 = End_Ycoord - m * End_Xcoord;

if b1 == b2
    disp('Slope-Intercept Equation can be formed.')
else
    disp('Fix your code')
end
Tangent = m * x + b1;

%% Plot 2
figure(2)
set(groot,'DefaultLineLineWidth',2)
plot(x,Yup) % Plotting Upper Half of Airfoil Profile
hold on
plot(x,Ylow) % Plotting Lower Half of Airfoil Profile
plot(x,Ycamber, '--') % Plotting Camber Line
plot(x,Tangent, '--') % Plotting Chord Line
plot(x_le,y_le, 'o') % Plotting Leading Edge
plot(x_circle_low, y_circle_low, 'g'); % Plotting circle
hold off
legend('Airfoil Upper Profile', 'Airfoil Lower Profile', 'Camber Line', 'Chord Line', 'Leading
    Edge Point', 'Leading Edge Circle')
xlim([0,1]) % Adjusting Aspect Ratio of plot
ylim([(min(Ylow)-.2) (max(Yup)+.2)])
title('NACA Clark Y Airfoil Tangent')
grid on

%% Plot 3
figure(3)
set(groot,'DefaultLineLineWidth',2)
plot(x,g+x*0, '--') % Plotting Chord Line % Plotting Tip to Tip Chord
hold on
plot(x,Tangent, '--') % Plotting Tangent Chord
hold off
legend('Tip to Tip Chord','Tangent Chord')
xlim([0,1]) % Adjusting Aspect Ratio of plot
ylim([(min(Ylow)-.2) (max(Yup)+.2)])
title('NACA Clark Y Airfoil Chord Comparison')
grid on

%% Results We Need
% does ycamber max give the max camber as a percentage
Ycmaxper=Ycmax*100; % getting a % for max camber

% Max thickness
Thicc = Yup - Ylow;
[Thicc_max, Thicc_pos] = max(Thicc);
Thicc_max_per = Thicc_max * 100;
Thicc_max_loc = x(Thicc_pos);

z1 = ['Max Camber as a Percentage of the Chord Length: ' num2str(Ycmaxper) ' Percent'];
z2 = ['Location of Max Camber From Leading Edge: ' num2str(xmax) ' Units'];
z3 = ['Max Thickness of Airfoil: ' num2str(Thicc_max_per) ' Percent'];
z4 = ['Location of Max Thickness From Leading Edge: ' num2str(Thicc_max_loc) ' Units'];
disp(z1)
disp(z2)
disp(z3)
disp(z4)

%% Helper function to fit a circle to a set of points
```

```
function [xc, yc, R] = fitCircle(x, y)
    % This function fits a circle to the given x and y points.
    % Output: (xc, yc) center of the circle, R is the radius.

    % Create the matrix A and vector b to solve Ax = b
    A = [2*x, 2*y, ones(size(x))];
    b = x.^2 + y.^2;

    % Solve the system
    params = A \ b;

    % Extract the circle parameters
    xc = params(1);
    yc = params(2);
    R = sqrt(xc^2 + yc^2 + params(3));
end
```