Aerodynamics Homework 06

Zakary Steenhoek

November 7, 2024

I. Part 1

The upper and lower surface coordinates for the NLF 1015 Airfoil are found on the course Canvas site. This airfoil has a maximum thickness of 15% at 39.8% chord and a maximum camber at 62.8% chord.

A. Question 1 Problem Statements:

1. *Plot the airfoil section and its camber line on the same plot.*
2. *Find the zero-lift angle and the moment about the aerodynamic center for the NLF1015. Compare your results with the accepted values,* $\alpha_{\mathrm{ZL}} = -6.67°$ *and* $C_{M_{AC}} = -0.19$.
3. *Based on both a) and b) above, discuss some similarities and differences between this airfoil and the NACA 4412.*

Assume: Small disturbances, thin airfoil

Given: Accepted values for $\alpha_{ZL}, C_{M_{AC}}$

Solution: To first plot the airfoil section and its camber line from the data provided, the z-data was plotted against the upper and lower x-dimension data. To determine the camber line, the upper and lower x-dimension at each z-coordinate was added and divided by two to determine the midpoint. This camber line data was then given a symbolic equation using a fourth-order polyfit in MATLAB. This line of fit was also plotted on this graph, and upon visual inspection it fits to an acceptable accuracy. The maximum thickness of this airfoil of 15% is within the range of the thin airfoil theory, so the aerodynamic coefficients were calculated under this assumption. The equation for the camber line was differentiated, and coefficients A_o, A_1, and A_2 were computed using the known thin-airfoil equations, seen below:

$$A_o = \alpha - \frac{1}{\pi}\int_0^{\pi} \frac{dz_c}{dx}\, d\theta \tag{1.1}$$

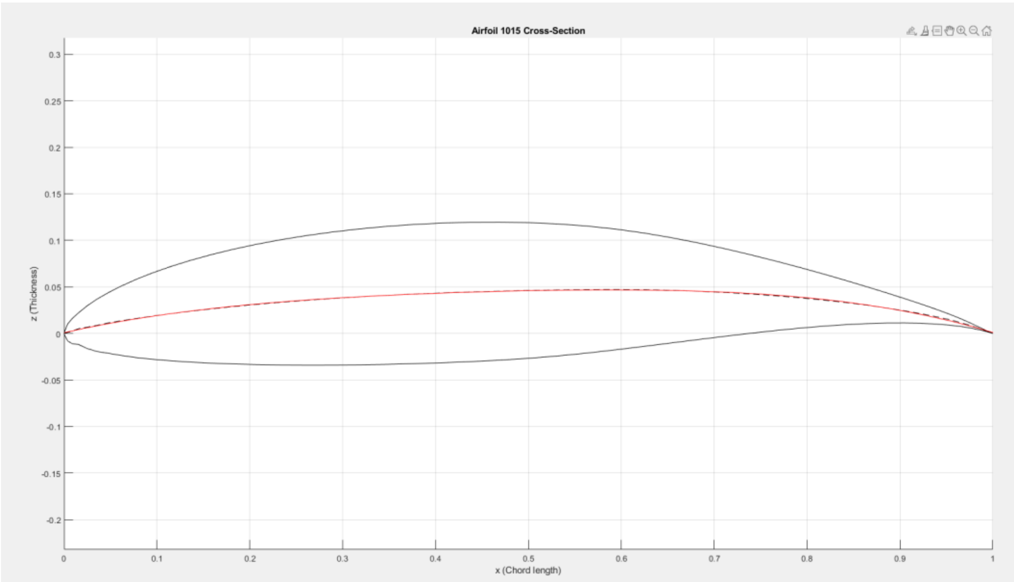
$$A_n = \frac{2}{\pi}\int_0^{\pi} \frac{dz_c}{dx} \cos n\theta \, d\theta \tag{1.2}$$

The zero-lift angle of attack and the moment coefficient were then finally computed, also using the known thin-airfoil equations, seen below:

$$\alpha_{\mathrm{zl}} = -\frac{1}{\pi}\int_0^{\pi} \frac{dz_c}{dx}(\cos\theta - 1)\, d\theta \tag{1.3}$$

$$C_{M_{ac}} = \frac{\pi}{4}(A_2 - A_1) \tag{1.4}$$

Results: The flowing figure shows the airfoil cross section and the camber line. The camber line computed from the average of the upper and lower surface location is shown as the black dotted line, and the MATLAB polyfit is plotted in red over the top. Note the overall similarity, as an extremely minimal amount of the black dotted line can be seen.



The resulting equation for this camber line and its derivative is seen below:

$$Z_c = -0.3399x^4 + 0.6040x^3 - 0.4913x^2 + 0.2274X + 8.0665e^{-4}$$

$$\frac{d}{dx}Z_c = -1.3596x^3 + 1.8119x^2 - 0.9826x + 0.2274$$

The coefficients A_o, A_1, and A_2 can be seen below:

$$A_o = \alpha + 0.0093$$

$$A_1 = 0.2227$$

$$A_2 = -0.0284$$

Finally, α_{ZL} and $C_{M_{AC}}$ were computed, seen below:

$$\alpha_{ZL} = -6.9135$$

$$C_{M_{AC}} = -0.1972$$

Discussion: First, to compare the results obtained here with the accepted results. On a high note, the moment coefficient about the quarter chord meets the accepted value within the precision provided. This is to say that the thin airfoil theory correctly predicted the moment for this shape airfoil, which is atypical in comparison to the 4-digit NACA series airfoils we have been working with in this class thus far, as the location of maximum camber is much further aft. The zero-lift angle, however, does not quite meet the accepted value. This result was obtained and using two separate methods; the first being Eq. [1.3], and the second was through finding the C_l using aerodynamic coefficients and using solve in MATLAB to determine where the lift coefficient was equal to zero. These two methods produced this exact same result. It is safe to say that, since we obtained the correct moment coefficient, the issue does not lie in the symbolic fit of the camber line. The discrepancy must lie in the fact that the thin airfoil theory breaks down for this airfoil shape at some point.

In comparison to the NACA 4412, This airfoil is unique. It produces a pitching moment almost twice what was found for NACA 4412 in the previous homework, where $C_{M_{AC}} = -0.1062$. This airfoil also produces positive lift through almost negative seven degrees, which is also almost twice the negative AOA range of NACA 4412, where this was found to be -4.1548 degrees. Overall, this airfoil produces more moment and produces lift across a wider range of negative angles of attack when compared to the NACA 4412.

II. Part 2

Files with the data for the NACA 0006, NACA 0012, and NACA 0018 airfoils are found on the course Canvas site. The files are named with the angle of attack. For example, TACAA_NACA0006_3.txt contains NACA 0006 data for $\alpha = 3°$. The letter 'n' or 'm' before the number indicates a negative angle of attack.

A. Problem Statements:

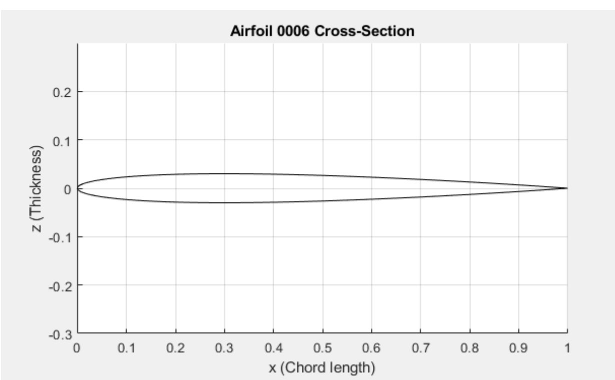
1. *Plot the airfoil profile and C_p vs. x for NACA 0006 and NACA 0018 at an angle of attack equal to 3 degrees. Compare this pressure distribution with that for the NACA 0012 airfoil.*
2. *Using MATLAB, estimate the lift and drag coefficients per unit span on this airfoil for all data sets. In a table, show C_l and C_d for each angle of attack.*
3. *On one graph, plot C_l vs α. for the NACA 0006, the NACA 0012 and the NACA 0018. What is the lift-curve slope for each? What is the zero-lift angle of attack for each? On the same graph, plot the thin-airfoil theory result for a symmetric airfoil.*
4. *On one graph, plot cd vs. for the NACA 0006, the NACA 0012 and the NACA 0018. Comment on the variation of drag coefficient with angle of attack*
5. *Discuss the effect of thickness on the lift-curve slope, the zero-lift angle of attack and the drag curve.*

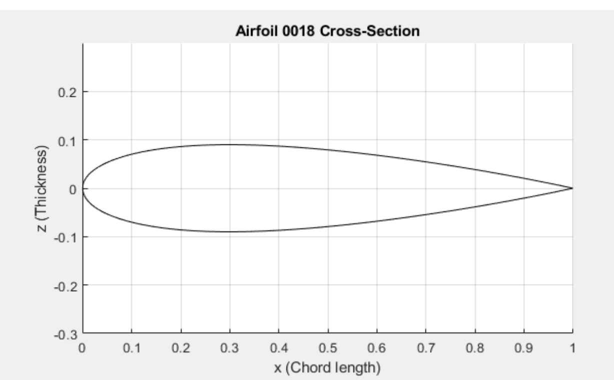
Assume: Small disturbances, thin airfoil

Given: NACA 0006, NACA 0012, and NACA 0018 data sets

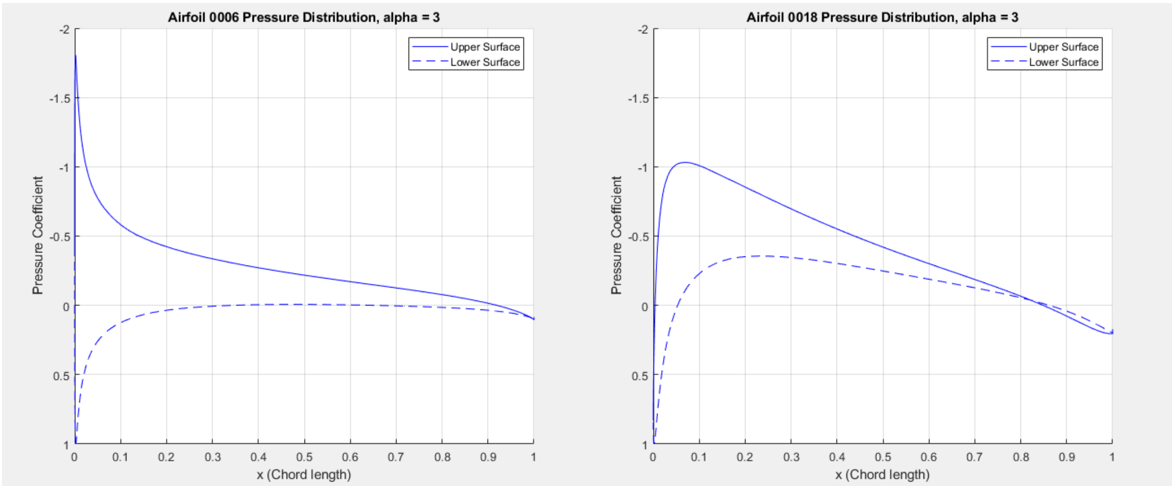
Solution: To generate the plots, code was reused from previous homework. This code is specifically for the TACAA generated data at varying angles of attack and was adapted for the wider range and changes in problem statements. Data was parsed by file, using the name of the file to determine the angle of attack. Standard lift and drag coefficient calculations were performed, and the data was stored for graphing. The file data from when $\alpha = 3$ was also grabbed to plot the airfoil characteristics and pressure distribution. The full code can be found in the appendix.

Results: The cross sections of the NACA 0006 and the NACA 0018 can be Seen below:

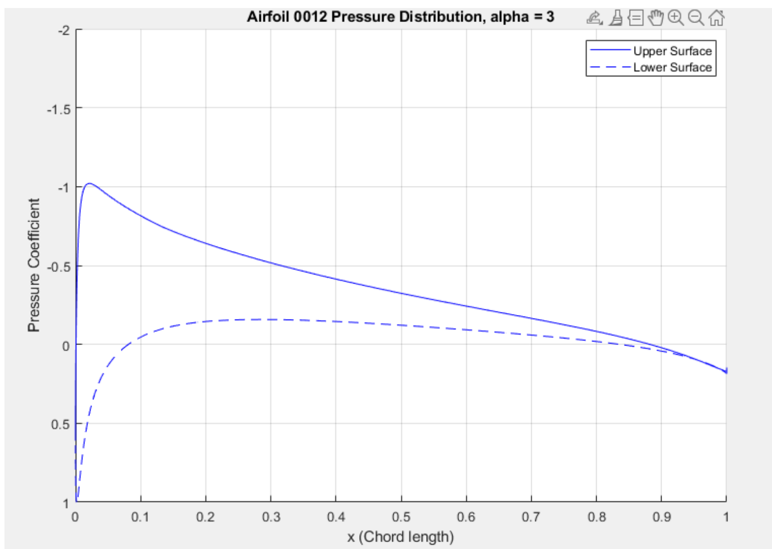




The pressure distributions at $\alpha = 3°$ for NACA 0006 and NACA 0018 can be seen below:



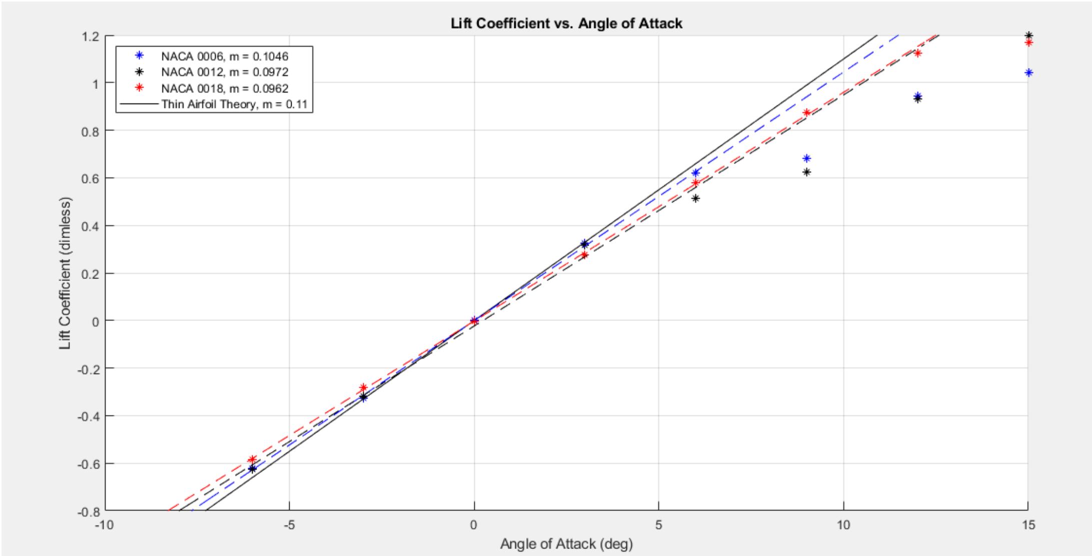
For comparison, here is the graph of C_l vs α for NACA 0012:



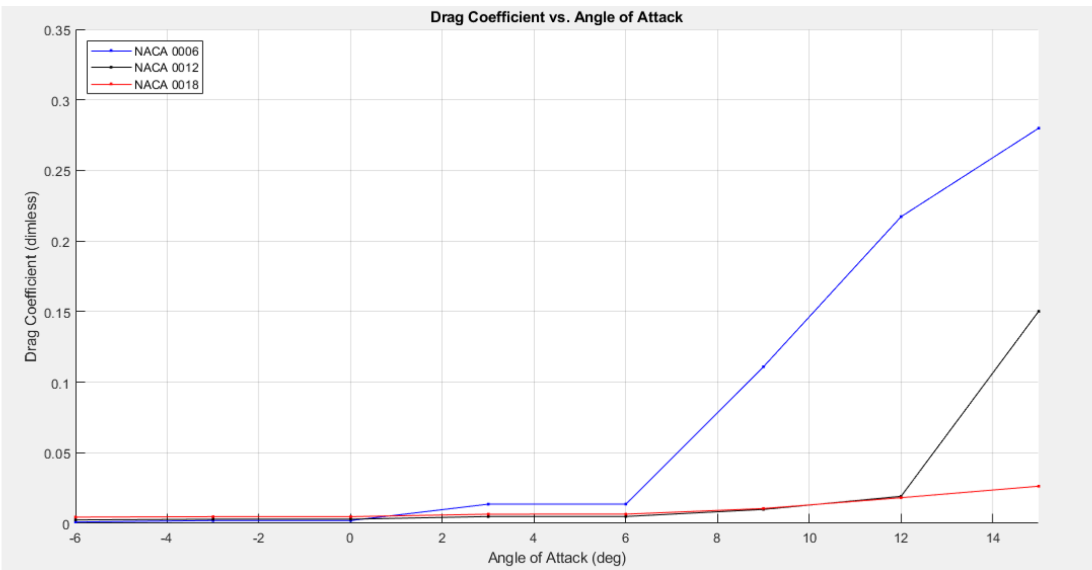
Here are the lift and drag coefficient results from al; three symmetrical airfoils:

NACA 0006			NACA 0012			NACA 0018		
α	C_l	C_d	α	C_l	C_d	α	C_l	C_d
-6	-0.6206	0.0009	-6	-0.6258	0.0024	-6	-0.5831	0.0044
-3	-0.3279	0.0020	-3	-0.3192	0.0029	-3	-0.2828	0.0047
0	-0.0005	0.0020	0	-0.0016	0.0029	0	-0.0030	0.0047
3	0.3268	0.0136	3	0.3155	0.0048	3	0.2776	0.0065
6	0.6203	0.0137	6	0.5151	0.0048	6	0.5800	0.0065
9	0.6836	0.1109	9	0.6227	0.0099	9	0.8747	0.0105
12	0.9429	0.2172	12	0.9319	0.0191	12	1.1249	0.0182
15	1.0407	0.2799	15	1.1979	0.1502	15	1.1705	0.0263

Here is the C_l vs. α plot and slopes for all three airfoils and the thin airfoil theory:



The C_d vs α plot can be seen below:



Discussion: First, a note on the cross sections. As these are all symmetric airfoils, the results are expected. From what we know about the NACA 4-digit series, The only difference between the three airfoils being compared in this part are that the maximum thickness increases by 6% from the 0006 to the 0012 to the 0018. The pressure distributions at a three degree angle of attack provide information on how the airfoil thickness generates lift, but also where it is generated along the airfoil. It appears the thinner airfoil has a more aggressive and rapid pressure drop over the upper surface, which is shown by the large low-pressure spike on the pressure distribution graph. It also looks as though it generates less lift along the rest of the airfoil, indicating that the thinner airfoil will have a more negative moment about the quarter-chord, as more lift is produced left of the quarter chord. Comparing the NACA 0018 airfoil, the opposite seems to be true, and the rise to the lowest pressure us much shorter and the peak is smoothed out. Note that the NACA 0018 produces a very small amount of negative lift toward the very aft of the airfoil, just past 0.82 chord. This airfoil is on the very upper bound, if even within, the range to be considered a thin airfoil, so discrepancies like these can be expected as the theory begins to break down.

The combined lift coefficient vs. angle of attack graph shows that regardless, all these airfoils have relatively similar slopes, all of which are within reason compared to the thin airfoil theory. The graph shows that, when plotted against the theory, all the airfoils fall just short of the expected ~0.11 slope predicted by the thin airfoil theory. This is, as well, to be expected, as the data is not generated according to the thin airfoil theory, but rather using the NASA RANS code. There will always be losses that skew the results due to viscosity, compressibility, and other real-world factors. Here, the thinnest airfoil, the NACA 0006 stands out by being the closest to the thin-airfoil predictions. All three airfoils maintain zero lift at a zero-degree angle of attack, which is as well to be expected, due to the nature of a symmetric airfoil, regardless of thin airfoil theory, as there is no flow differential over the surfaces.

Next, for the graph of drag coefficient vs. angle of attack, over the operation angles of attack, i.e. the linear range of lift production before stall, the NACA 0018 produces the most drag, but by a fractional amount. All airfoils produce relatively low amounts of drag, and there is not much difference between them. However, the thickest airfoil produces the smaller dramatic change in drag as the angle of attack approaches the stall range, while the NACA 0006, the thinnest airfoil, experiences a massive amount of drag in comparison. Looking back on the pressure distributions, recall that the NACA 0018 produces negative lift toward the aft section of the airfoil. From what we know about reflex airfoils, this behavior has the tendency to reduce flow separation and keep the flow ‘attached’ to the body of the foil for longer as the AoA increases. In combination with the larger thickness, it can be assumed that these airfoil characteristics have the tendency to reduce the effects and increase the angle of attack at which flow separation occurs, which dramatically reduces bluff body drag and large turbulent wake zones. The thinner airfoil seems as though it keeps flow attached for a smaller range of AoA, which results in sudden and dramatic bluff body drag then this flow *does* separate, and a huge spike in drag coefficient.

III. Part 3

A straight-line vortex of strength Γ extends from 1 < y_1 < 1 along the y-axis. The direction of the circulation is such that $d\vec{s} = +dy_1\hat{\jmath}$. Let the control point (the point where you want to find the effect of the line vortex) be in the $x - z$ plane ($y = 0$).

A. Question 3 Problem Statements:

1. *Find the velocity at the control point,* $(x, 0, z)$
2. *Show that this is the same as that induced by a 'point' vortex (strength* Γ*) at the origin in 2 dimensions*

Assume: Small disturbances

Given: Solution parameters

Solution: To determine the velocity at the control point $(x, 0, z)$ and show that this is the same as a point vortex, we will work with the Biot-Savart law, Eq. [3.1]:

$$d\vec{V} = \frac{\Gamma}{4\pi}\frac{d\vec{s} \times \vec{r}}{|\vec{\mathrm{r}}|^3} \tag{3.1}$$

This related the induced velocity to the cross product between the circulation direction and the position vector, and the magnitude of the position vector. The cross product and magnitude in question are evaluated:

$$d\vec{s} \times \vec{r} = (dy_1\hat{\jmath}) \times \left(x\hat{\imath} - y_1\hat{\jmath} + z\hat{k}\right) = dy_1\left(x\hat{k} - z\hat{\imath}\right)$$

$$|\vec{r}| = \sqrt{x^2 + y_1^2 + z^2}$$

Plugging these into Eq. [3.1] yields Eq. [3.2]:

$$d\vec{V} = \frac{\Gamma}{4\pi}\frac{dy_1\left(x\hat{k} - z\hat{\imath}\right)}{(x^2 + y_1^2 + z^2)^{3/2}} \tag{3.2}$$

Integrating to determine the velocity:

$$\vec{V} = \frac{\Gamma}{4\pi}\int_{-\infty}^{\infty}\frac{\left(x\hat{k} - z\hat{\imath}\right)}{(x^2 + y_1^2 + z^2)^{3/2}}\, dy_1$$

Note that this integral is non-zero only when integrating in the direction of $\hat{\imath}$. The integration simplifies to the following:

$$\vec{V} = -\frac{\Gamma \mathrm{z}}{4\pi}\int_{-\infty}^{\infty}\frac{1}{(x^2 + y_1^2 + z^2)^{3/2}}\, dy_1 = -\frac{\Gamma}{2\pi}\frac{z}{x^2 + z^2}\hat{\imath}$$

To compare this with the result of a point vortex with identical strength in 2 dimensions, the following standard point vortex calculations that have been used extensively previously are shown, in cartesian form, to maintain consistency:

$$V = \frac{\Gamma}{2\pi}\frac{1}{x^2 + z^2}$$

$$V_x = -\frac{\Gamma}{2\pi}\frac{z}{x^2 + z^2}\hat{\imath} + \frac{\Gamma}{2\pi}\frac{\mathrm{x}}{x^2 + z^2}\hat{k}$$

Note that the x-component of the 2-dimensional version of this standard point vortex function matches the result obtained from the above integration, proving equivalence for a point vortex and an infinite line vortex in the same 2-dimensional plane.

I. Appendix A

```
%% Header
% Author: Zakary Steenhoek
% Date: 7 October 2024
% AEE 360 HW06

clc; clear; clf; %close all;

%% Declare

% Syms and substitution
syms X T a
sympref('FloatingPointOutput',true);
X_TERMS_THETA = 0.5*(1-cos(T));

% Load data from the text file
dataFile = importdata("data/NLF1015ul.txt");

% Extract x and z data
dimlessX = dataFile.data(:, 1);
dimlessZu = dataFile.data(:, 2);
dimlessZl = dataFile.data(:, 3);
dimlessCamber = (dimlessZu+dimlessZl)/2;

%% Math

% Camber line and deriv
Zc = poly2sym(polyfit(dimlessX, dimlessCamber, 3), X);
dZc = diff(Zc, X);
dZc = subs(dZc, X, X_TERMS_THETA);

% Ao,1,2
A_o = a-(1/pi)*(int(dZc, T, 0, pi));
A_1 = (2/pi)*(int(dZc.*cos(T), T, 0, pi));
A_2 = (2/pi)*(int(dZc.*cos(2*T), T, 0, pi));

% Compute C_l, a_zl, Cm_ac
C_l = 2*pi*(A_o+0.5*A_1);
a_zl1 = rad2deg(solve(C_l == 0,a));
a_zl2 = rad2deg(-(1/pi)*int(dZc*(cos(T)-1), T, 0, pi));
Cm_ac = (pi/4)*(A_2-A_1);

%% Plots

% Plot the x-z cross section and camber line
figure(1); hold on;
xlabel('x (Chord length)'); ylabel('z (Thickness)');
title('Airfoil 1015 Cross-Section');
grid on; axis equal; xlim([0,1]);
plot(dimlessX, dimlessZu, 'k-'); plot(dimlessX, dimlessZl, 'k-');
plot(dimlessX, dimlessCamber, 'k--');
fplot(Zc, 'r-');
hold off;
%% Header
```

```
% Author: Zakary Steenhoek
% Date: 7 October 2024
% AEE 360 HW06

clc; clear; clf; %close all;

%% Find data files for NACA 0012 & 4412

% Load data from the text files for both airfoils (personal full path is redacted)
dataPath_0006 = 'C:\<...>\MATLAB\AEE360\HW06\data\NACA0006';
dataPath_0012 = 'C:\<…>\MATLAB\AEE360\HW03\0012_data';
dataPath_0018 = 'C:\<…>\MATLAB\AEE360\HW06\data\NACA0018';
dataFiles_0006 = dir(fullfile(dataPath_0006));
dataFiles_0012 = dir(fullfile(dataPath_0012));
dataFiles_0018 = dir(fullfile(dataPath_0018));

% Get rid of standard listed dirs
dataFiles_0006(1:2)=[]; dataFiles_0012(1:2)=[]; dataFiles_0018(1:2)=[];

% To record the data for each airfoil
all_0006 = zeros(length(dataFiles_0006), 3);
all_0012 = zeros(length(dataFiles_0012), 3);
all_0018 = zeros(length(dataFiles_0018), 3);

%% Math for NACA 0006

% For the number data files
for itr = 1:length(dataFiles_0006)
    % Build path to the file
    filePath = fullfile(dataPath_0006, dataFiles_0006(itr).name);
    dataFile = importdata(filePath);

    % Parse the filename to get the angle of attack
    subs = split(dataFiles_0006(itr).name, '_'); lastSub = subs{end};
    alphadeg = erase(lastSub, '.txt');

    % Determine if alpha is negative and convert to int
    if contains(lastSub, 'n')
        alphadeg = erase(alphadeg, 'n');
        alphadeg = -1*str2double(alphadeg);
    else
        alphadeg = str2double(alphadeg);
    end

    % Convert to rad
    alpha = alphadeg*pi/180;

    % Extract x and z columns and ipc column
    dimlessX_0006 = flipud(dataFile.data(:, 1));
    dimlessZ_0006 = flipud(dataFile.data(:, 3));
    Cp_0006 = flipud(dataFile.data(:, 4));

    % Find force coefficients for x and z
    cfx = trapz(dimlessZ_0006, Cp_0006); cfz = trapz(dimlessX_0006, -Cp_0006);

    % Lift and drag coefficient calculations
```

```
    cl = cfz*cos(alpha)-cfx*sin(alpha);
    cd = cfz*sin(alpha)+cfx*cos(alpha);

    all_0006(itr,1) = alphadeg;
    all_0006(itr,2) = cl; all_0006(itr,3) = cd;

    if (alphadeg == 3)
        dimlessX_0006a3 = dimlessX_0006;
        dimlessZ_0006a3 = dimlessZ_0006;
        Cp_0006a3 = Cp_0006;
    end

end

% Sort the data
all_0006 = sort(round(all_0006, 5));

%% Math for NACA 0012

% For the number data files
for itr = 1:length(dataFiles_0012)
    % Build path to the file
    filePath = fullfile(dataPath_0012, dataFiles_0012(itr).name);
    dataFile = importdata(filePath);

    % Parse the filename to get the angle of attack
    subs = split(dataFiles_0012(itr).name, '_'); lastSub = subs{end};
    alphadeg = erase(lastSub, '.txt');

    % Determine if alpha is negative and convert to int
    if contains(lastSub, 'm')
        alphadeg = erase(alphadeg, 'm');
        alphadeg = -1*str2double(alphadeg);
    else
        alphadeg = str2double(alphadeg);
    end

    % Convert to rad
    alpha = alphadeg*pi/180;

    % Extract x and z columns and ipc column
    dimlessX_0012 = flipud(dataFile.data(:, 1));
    dimlessZ_0012 = flipud(dataFile.data(:, 3));
    Cp_0012 = flipud(dataFile.data(:, 4));

    % Find force coefficients for x and z
    cfx = trapz(dimlessZ_0012, Cp_0012); cfz = trapz(dimlessX_0012, -Cp_0012);

    % Lift and drag coefficient calculations
    cl = cfz*cos(alpha)-cfx*sin(alpha);
    cd = cfz*sin(alpha)+cfx*cos(alpha);

    all_0012(itr,1) = alphadeg;
    all_0012(itr,2) = cl; all_0012(itr,3) = cd;

    if (alphadeg == 3)
```

```
        dimlessX_0012a3 = dimlessX_0012;
        dimlessZ_0012a3 = dimlessZ_0012;
        Cp_0012a3 = Cp_0012;
    end

end

% Sort the data
all_0012 = sort(round(all_0012, 5));

%% Math for NACA 0018

% For the number data files
for itr = 1:length(dataFiles_0018)
    % Build path to the file
    filePath = fullfile(dataPath_0018, dataFiles_0018(itr).name);
    dataFile = importdata(filePath);

    % Parse the filename to get the angle of attack
    subs = split(dataFiles_0018(itr).name, '_'); lastSub = subs{end};
    alphadeg = erase(lastSub, '.txt');

    % Determine if alpha is negative and convert to int
    if contains(lastSub, 'n')
        alphadeg = erase(alphadeg, 'n');
        alphadeg = -1*str2double(alphadeg);
    else
        alphadeg = str2double(alphadeg);
    end

    % Convert to rad
    alpha = alphadeg*pi/180;

    % Extract x and z columns and ipc column
    dimlessX_0018 = flipud(dataFile.data(:, 1));
    dimlessZ_0018 = flipud(dataFile.data(:, 3));
    Cp_0018 = flipud(dataFile.data(:, 4));

    % Find force coefficients for x and z
    cfx = trapz(dimlessZ_0018, Cp_0018); cfz = trapz(dimlessX_0018, -Cp_0018);

    % Lift and drag coefficient calculations
    cl = cfz*cos(alpha)-cfx*sin(alpha);
    cd = cfz*sin(alpha)+cfx*cos(alpha);

    all_0018(itr,1) = alphadeg;
    all_0018(itr,2) = cl; all_0018(itr,3) = cd;

    if (alphadeg == 3)
        dimlessX_0018a3 = dimlessX_0018;
        dimlessZ_0018a3 = dimlessZ_0018;
        Cp_0018a3 = Cp_0018;
    end

end
```

```
% Sort the data
all_0018 = sort(round(all_0018, 5));

%% Plots

% Plot the x-z cross section and camber line
tiledlayout(figure(621), 1, 3); clf;
xz0006 = nexttile(1); hold on;
xlabel('x (Chord length)'); ylabel('z (Thickness)');
title('Airfoil 0006 Cross-Section');
grid on; axis equal; xlim([0,1]); ylim([-0.3,0.3]);
plot(dimlessX_0006, dimlessZ_0006, 'k-');
hold off;

xz0012 = nexttile(2); hold on;
xlabel('x (Chord length)'); ylabel('z (Thickness)');
title('Airfoil 0012 Cross-Section');
grid on; axis equal; xlim([0,1]); ylim([-0.3,0.3]);
plot(dimlessX_0012, dimlessZ_0012, 'k-');
hold off;

xz0018 = nexttile(3); hold on;
xlabel('x (Chord length)'); ylabel('z (Thickness)');
title('Airfoil 0018 Cross-Section');
grid on; axis equal; xlim([0,1]); ylim([-0.3,0.3]);
plot(dimlessX_0018, dimlessZ_0018, 'k-');
hold off;

tiledlayout(figure(622), 'flow'); clf;
Cp0006 = nexttile(); hold on;
xlabel('x (Chord length)'); ylabel('Pressure Coefficient');
title('Airfoil 0006 Pressure Distribution, alpha = 3');
axis ij; grid on; axis([0 1 -2 1]); n = length(dimlessX_0006);
plot(dimlessX_0006a3(n/2:n), Cp_0006a3(n/2:n), 'b', dimlessX_0006a3(1:n/2+1), Cp_0006a3(1:n/2+1), 'b--');
legend('Upper Surface', 'Lower Surface');
hold off;

Cp0018 = nexttile(); hold on;
xlabel('x (Chord length)'); ylabel('Pressure Coefficient');
title('Airfoil 0018 Pressure Distribution, alpha = 3');
axis ij; grid on; axis([0 1 -2 1]); n = length(dimlessX_0018);
plot(dimlessX_0018a3(n/2:n), Cp_0018a3(n/2:n), 'b', dimlessX_0018a3(1:n/2+1), Cp_0018a3(1:n/2+1), 'b--');
legend('Upper Surface', 'Lower Surface');
hold off;


% Plot Cl data
figure(623); clf; hold on;
scatter(all_0006(:,1), all_0006(:,2), 'b*');
scatter(all_0012(:,1), all_0012(:,2), 'k*');
scatter(all_0018(:,1), all_0018(:,2), 'r*');
ClCv_0006 = polyfit(all_0006(1:(end-3),1), all_0006(1:(end-3),2), 1);
ClCv_0012 = polyfit(all_0012(1:(end-3),1), all_0012(1:(end-3),2), 1);
ClCv_0018 = polyfit(all_0018(1:(end-3),1), all_0018(1:(end-3),2), 1);
ClCv_thinAirfoil = [0.11,0];
fplot(poly2sym(ClCv_thinAirfoil), 'k-');
```

```
fplot(poly2sym(ClCv_0006), 'b--');
fplot(poly2sym(ClCv_0012), 'k--');
fplot(poly2sym(ClCv_0018), 'r--');
xlabel('Angle of Attack (deg)'); ylabel('Lift Coefficient (dimless)');
title('Lift Coefficient vs. Angle of Attack');
legend('NACA 0006, m = 0.1046', 'NACA 0012, m = 0.0972', 'NACA 0018, m = 0.0962', 'Thin Airfoil Theory, m =
0.11', Location='northwest');
grid on; hold off;
zeroLiftAoA_0006 = roots(ClCv_0006);
zeroLiftAoA_0012 = roots(ClCv_0012);
zeroLiftAoA_0018 = roots(ClCv_0018);

% Plot Cd data
figure(624); clf; hold on;
xlabel('Angle of Attack (deg)'); ylabel('Drag Coefficient (dimless)');
title('Drag Coefficient vs. Angle of Attack');
xlim([-6,15]); ylim([0,0.35]);
plot(all_0006(:,1), all_0006(:,3), 'b.-');
plot(all_0012(:,1), all_0012(:,3), 'k.-');
plot(all_0018(:,1), all_0018(:,3), 'r.-');
CdCv_0006 = polyfit(all_0006(:,1), all_0006(:,3), 3);
CdCv_0012 = polyfit(all_0012(:,1), all_0012(:,3), 3);
CdCv_0018 = polyfit(all_0018(:,1), all_0018(:,3), 3);
legend('NACA 0006', 'NACA 0012', 'NACA 0018', Location='northwest');
% fplot(poly2sym(CdCv_0006), 'b--');
% fplot(poly2sym(CdCv_0012), 'k--');
% fplot(poly2sym(CdCv_0018), 'r--');
grid on; hold off;


% Plot the x-z cross section and camber line
tiledlayout(figure(6210), 1, 3); clf;
xz0006 = nexttile(1); hold on;
xlabel('x (Chord length)'); ylabel('z (Thickness)');
title('Airfoil 0006 Cross-Section');
grid on; axis equal; xlim([0,1]); ylim([-0.3,0.3]);
plot(dimlessX_0006, dimlessZ_0006, 'k-');
hold off;

xz0018 = nexttile(3); hold on;
xlabel('x (Chord length)'); ylabel('z (Thickness)');
title('Airfoil 0018 Cross-Section');
grid on; axis equal; xlim([0,1]); ylim([-0.3,0.3]);
plot(dimlessX_0018, dimlessZ_0018, 'k-');
hold off;

figure(6220); clf;
Cp0012 = nexttile(); hold on;
xlabel('x (Chord length)'); ylabel('Pressure Coefficient');
title('Airfoil 0012 Pressure Distribution, alpha = 3');
axis ij; grid on; axis([0 1 -2 1]); n = length(dimlessX_0006);
plot(dimlessX_0012a3(n/2:n), Cp_0012a3(n/2:n), 'b', dimlessX_0012a3(1:n/2+1), Cp_0012a3(1:n/2+1), 'b--');
legend('Upper Surface', 'Lower Surface');
hold off;

close all;
```