ASEN 2002
Aerodynamics of a Cambered Airfoil
Experimental Lab
Group 1 - December 13, 2017

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The main premise of this experiment was to relate the lift and drag to the different angles of attack of a cambered airfoil. This was done experimentally using a Clark Y-14 airfoil model with 19 surface pressure ports in the ITLL wind tunnel. The differential pressure between the ports was found and used to find the Coefficient of Pressure. These were then used to find the axial and perpendicular coefficients, which are easily converted into the Coefficient of Drag and the Coefficient of Lift. These coefficients were calculated for each of the 32 angles that the section of Aerodynamics measured and graphs were generated to match and compare to the graphs in the NACA report 628 from 1938. The goal was to find the maximum lift that this airfoil can produce, if it has an angle of attack where the drag and coefficient of pressure are minimum. All of this data was run for higher airspeeds to see how the graphs change with higher Reynolds and Mach numbers. These data and the analysis are presented here.

Nomenclature

α = Angle of Attack [deg]
a = Axial Force [N]
C_a = Coefficient of the Axial Force
C_D = Coefficient of Pressure Drag
c_D = Coefficient of Sectional Pressure Drag
C_L = Coefficient of Lift
c_L = Coefficient of Sectional Lift
C_n = Coefficient of the Normal Force
C_P = Coefficient of Pressure
D = Drag [N]
L = Lift [N]
n = Normal Force [N]

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I. Introduction

Airfoils, the fundamental cross-sectional geometry in an aircrafts wing, are critical to the production of drag and lift on aerodynamic vehicles. Because the distribution of pressure and shear-stress on the surfaces of a vehicle make up all of the aerodynamic forces acting on that vehicle, quantifying and integrating the pressure distribution around the geometry of an airfoil leads to an understanding how lift and drag are created on aerodynamic vehicles. For the purposes of this experiment, a cambered Clark Y-14 airfoil, instrumented with 19 flush mounted pressure taps around its surface, was mounted in a wind tunnel such that pressure distribution on the surface could be measured for varying angles of attack and airspeed. In this lab, the aerodynamic performance of the airfoil is calculated by determining how pressure distribution varies with time and angle of attack, by determining if there are indications of flow separation of the surface, and by finding the largest change in pressure between an upper and lower surface. This performance is then analyzed to gain an understanding of how angle of attack affects lift and drag, and how the results compare to those produced by NACA in 1938.

II. Experimental Setup and Measurement Techniques

For this experiment, the Clark Y-14 airfoil was mounted vertically in the wind tunnel on a rotating plate such that the angle of attack could be adjusted between trials. Group 1 started the experiment with an angle of attack of -8 and an airspeed velocity of 9 meters per second. The velocity was then increased to 17 meters per second and finally 34 meters per second at the same angle of attack. The group then brought the velocity back to zero, opened up the test section, adjusted the angle of attack to zero, and resealed the test section. This same process was repeated for angles of attack zero, eight, and 16°. Special care was taken to ensure that the velocity was raised gradually for the protection of the wind tunnel. This same procedure was repeated by the other lab groups in order to account for all angles of attack between -15 and 16°, at the same three testing velocities. All data was recorded on the LabVIEW VI on the wind tunnel computer and saved according to the given file naming convention.

III. Post-processing and Calculation of Force Coefficient

One major source of uncertainty in the calculation of the lift and pressure drag was the fact that only 16 flush mounted pressure taps were used to collect data. The wing was outfitted with 19 such pressure taps, but only 16 were used for the purposes of the experiment. Uncertainty could have been reduced if more pressure taps were utilized. The method used to compute the trailing edge pressure was to extrapolate pressure measurements from the top and bottom of the wing section to the trailing edge, and average these extrapolations. While this method can provide a reasonable estimation of the pressure at the trailing edge, it is not as accurate as it would be if pressure at that point could be directly measured. The differential pressures were then numerically integrated around the airfoil surface to find the net axial and normal forces acting on the wing. From these forces the force coefficients and henceforth the lift and drag coefficients were derived.

IV. Airfoil Static Pressure Coefficient Distribution

The pressure coefficient versus normalized chord-wise position for several angles of attack is plotted in Fig. 1. See how for large negative angles of attack, the C_p for the bottom of the foil is "above" that of the top surface of the foil. This means the airfoil is creating negative lift at these angles of attack. As the angle hits -5°, the bottom surface C_p values start to go back "below" the top surface as lift stops acting downward on the foil. At 0°, the C_p curve looks similar to the traditional example of the pressure coefficient curve. As the angle creeps into 5 and 10°, the C_p values get larger in magnitude and the bottom surface starts to contribute more lift.

The angles of attack measured by group 1 were -8, 0, 8, and 16°. The pressure distributions at these angles are plotted in Fig. 2. See that the pressure distributions are consistent with the trends set by the rest of the class like in Fig. 1. At larger negative angles of attack, the lower surface distribution is above the upper surface, indicating a force downward on the wing. For 0 angle of attack, the profile looks like the standard C_p distribution, and for the larger angles of attack, the C_p distributions for both surfaces tend

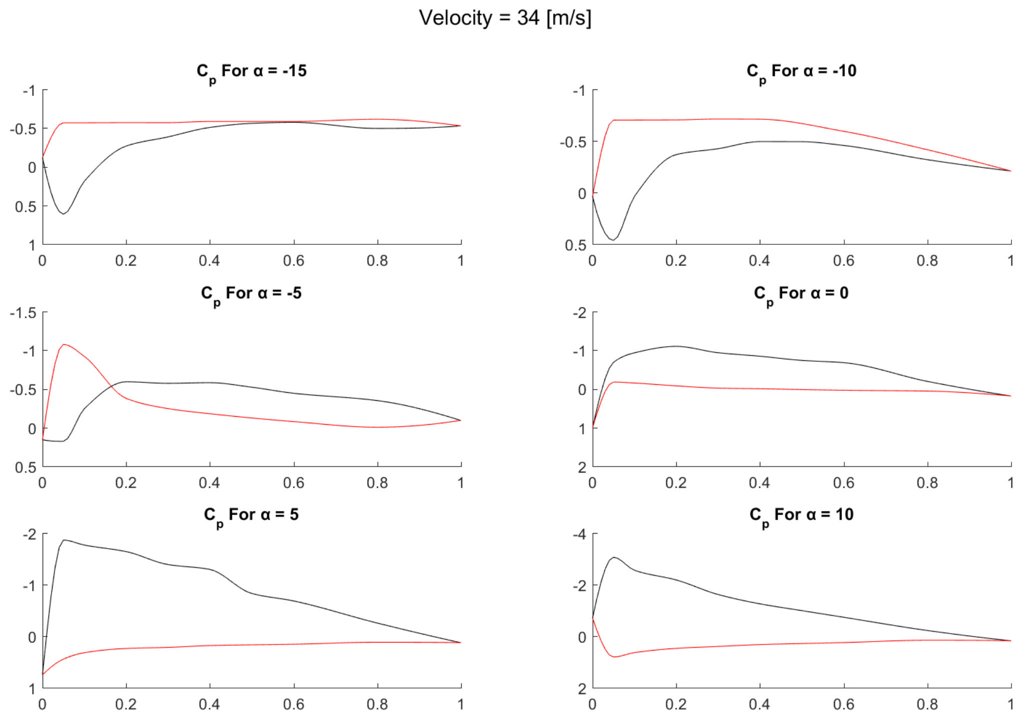


Figure 1: Coefficient of Pressure *These plots show the top and bottom pressure coefficients for the Clark Y-14 (normalized to chord-wise position.) Note that the red curve is the bottom surface and the black curve is the top surface.*

toward larger negative values as the bottom of the airfoil faces more of the free stream air.

The velocity of the air also seems to smooth out the C_p distributions as shown in Fig. 3. For $\alpha = -5°$, the increased velocity moves the intersection of the top and bottom surface distributions closer to the leading edge. At 9 m/s, the intersection happens at about .35 of the chord, but as velocity increases, this intersection moves to less than .2 of the chord. For 0 angle of attack, increasing velocity causes the magnitudes of the coefficients to increase slightly, generating more lift. The same is true for the 5° angle of attack. The reason the graphs seem to smooth out at high speed is because the random error in measurements becomes less significant as the magnitudes increase. With the higher velocity comes more pressure and thus more force and the wing will have higher C_p values to create more lift.

At low angles of attack where the flow has not separated, the coefficient of pressure distributions are all very similar. Though small features change from plot to plot, the distributions do not change overall in a significant way. The most noticeable change for each velocity is the shape of the C_p distribution over the lower surface of the wing. It becomes flatter. See Fig. 4 for a comparison of the distributions at small angles of attack at a free stream velocity of 9 m/s.

From all the figures, generally the greatest difference between the C_p on the top surface of the airfoil and bottom of the airfoil occurs around 10% of the chord. This then would be the location where the greatest lift force magnitude would occur on the wing. Even when the wing is being pushed downward (negative lift), the greatest difference between the top and bottom surface occurs around this location. Since this is fairly close to the front of the airfoil, one can expect a relatively large pitching moment on the airfoil and would likely have a noticeable effect on the design of an aircraft using this airfoil.

As for finding flow separation on the airfoil, one can look for a sudden decrease in the pressure coefficient values on the top surface of the airfoil for positive angles of attack. This can be seen when looking at the graphs of the pressure coefficient vs. the chord-wise position, that the spike in negative pressure coefficient with positive angles of attack will have flow separation at the correlating chord-wise position. This occurs around 0 or 1° for free stream 9 m/s, 5° at 17 m/s, and 12° at 34 m/s. Note that the lift coefficient plot in

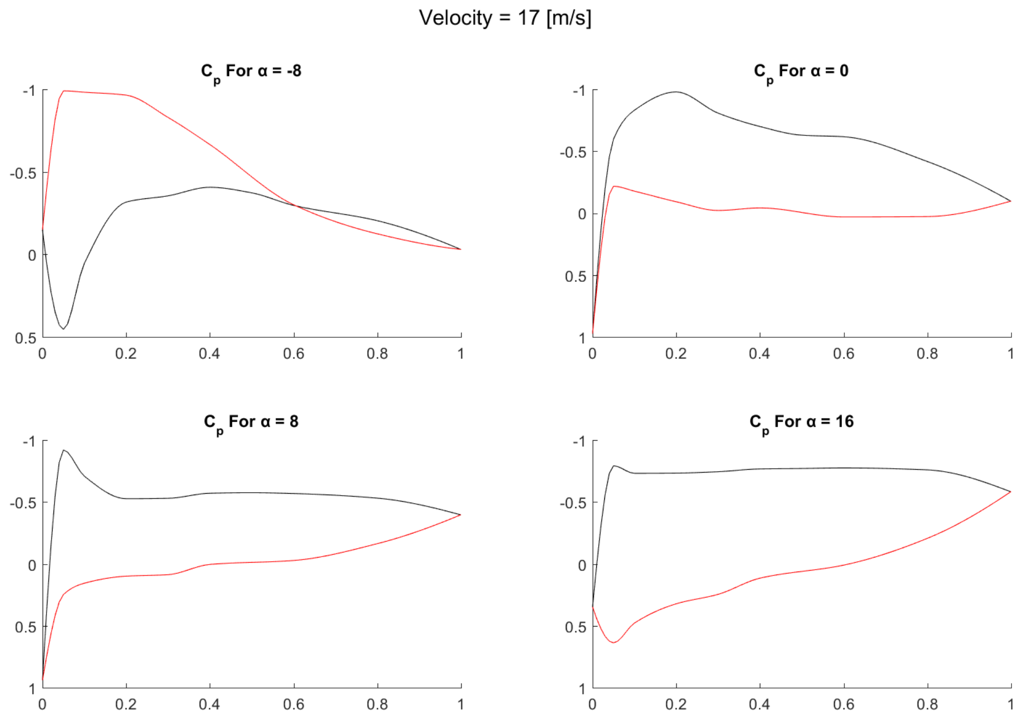


Figure 2: Group 1 Angles of Attack *These are the coefficients of pressure distribution for the angles of attack measured by Group 1. The red curve is the bottom surface of the airfoil while the black is C_p values for the top surface.*

Fig. 5a corroborate these observations.

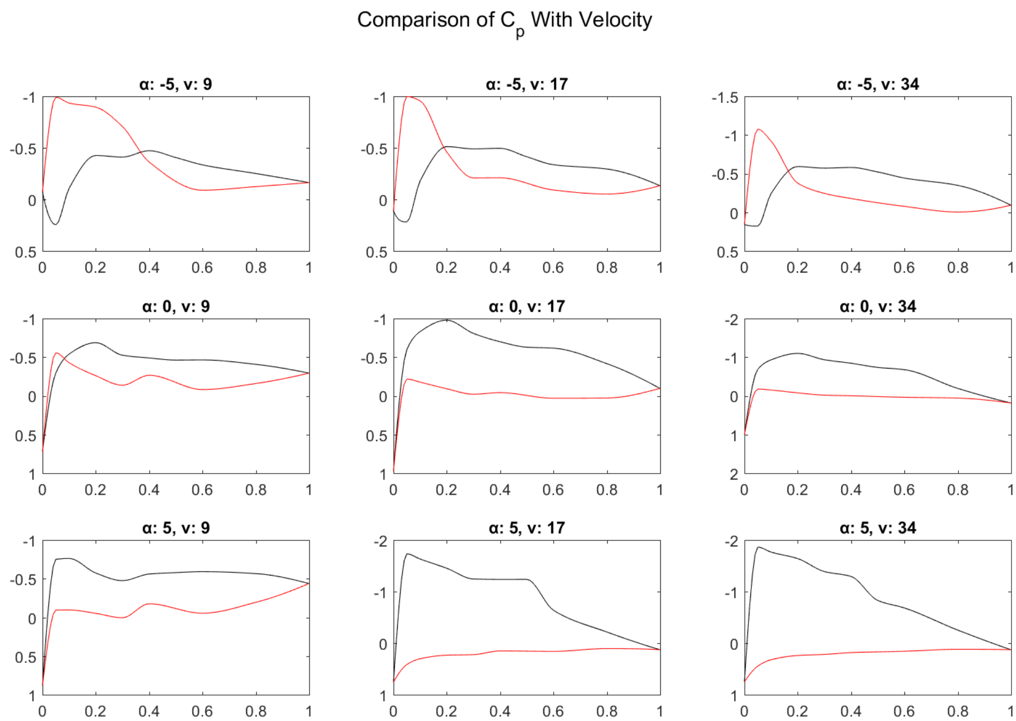


Figure 3: Velocity Effects on C_p *These plots show how velocity affects the pressure coefficient distributions for three angles of attack.*

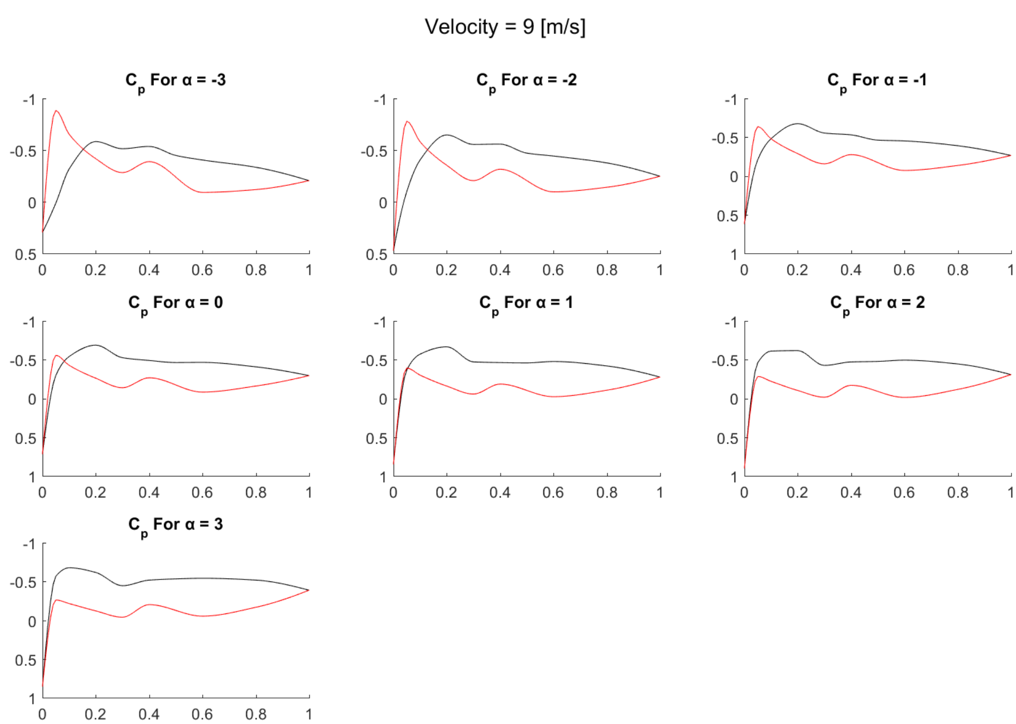
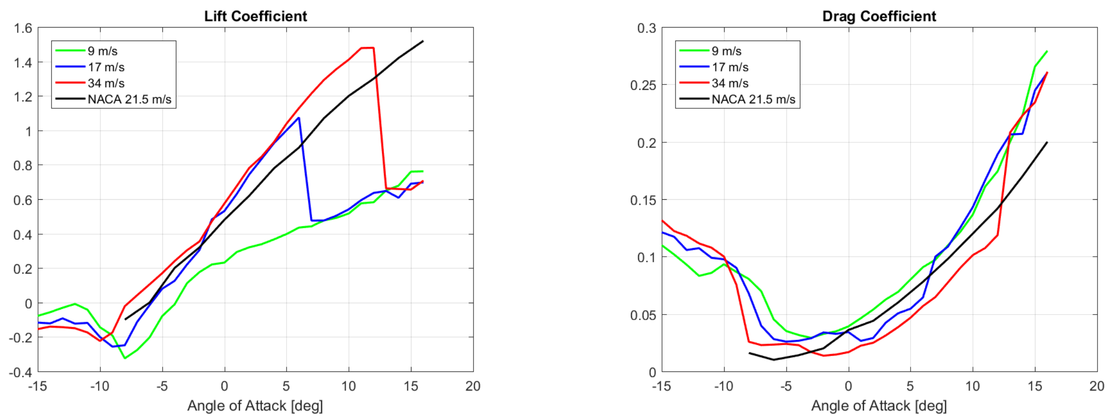


Figure 4: C_P Distribution at Small Angles of Attack *These plots show how the pressure distribution for small angles of attack does not change much while the flow is mostly still attached to the body.*

V. Lift and Pressure Drag Coefficients

For all the test data gathered in section 014, the coefficients of lift and drag as a function of angle of attack are plotted in Fig. 5. Also included in the graphs are NACA test results from Technical Report 628 [4].



(a) Coefficient of Lift *This plot shows how the coefficient of lift changes with angle of attack at a few different velocities.*

(b) Coefficient of Drag *This plot shows how the coefficient of drag changes with angle of attack at a few different velocities.*

Figure 5: Coefficients of Lift and Drag

From Fig. 5a it can be seen that the coefficient of lift generally increases with increasing angle of attack. At angles of attack near 0, the lift coefficients for every velocity increase almost linearly. At higher angles of attack the lift coefficient decreases suddenly. This is the point of trailing edge stall where the sudden loss of lift occurs due to flow separation. As evident in Fig. 5a, the stall occurs later for higher velocities. For each measured velocity, the lift coefficient continues to rise after the drop-off, however. This is not quite the same linear slope, and the lift coefficients appear to resume another linear regime once flow has separated. The regime at extremely low angles of attack is a loss of lift since the angle of attack is too steep. This is leading-edge stall, and all velocities react similar to this effect.

The coefficient of drag also increases with angle of attack above large negative angles of attack (around -7°). The drag coefficient increases in a non-linear fashion, however. The nonlinear nature of the drag coefficient curve indicates exponentially increasing drag as angle of attack is increased. At the angles where leading edge stall occurred in Fig. 5a, the drag coefficient is larger. The flow separation causes not only drops in lift, but there is high pressure drag at these angles of attack.

Taking all of this into consideration, the maximum coefficient of lift this Clark Y airfoil produced is 1.5. This occurs at an angle of attack of 11 to 12° and a free stream velocity of 34 m/s. This coefficient is highly dependent on tunnel velocity. As shown in Fig. 5a, the higher free stream velocities stay attached to the airfoil at higher angles of attack.

At zero angle of attack, the Clark Y-14 airfoil creates a positive lift coefficient no matter the measured velocity. The reason the airfoil creates lift with no angle of attack is due to the camber of the airfoil. At 0 angle of attack symmetric airfoils generate no lift, but a cambered airfoil produces an asymmetrical pressure distribution; the air moves faster over the top of the wing and with the higher local velocity comes lower pressure. As a result, the total pressure over the top of the wing is less than the pressure acting on the bottom surface and thus lift is generated. Zero lift is generated when the angle of attack is less than zero (which counteracts the camber). This angle ranges from around -9° to about -4°, depending on the free stream velocity.

Though close to the NACA data, the data in Fig. 5a lie slightly above the NACA data. One possible reason for this slight discrepancy is blockage effects. Since the airfoil blocks a significant amount the test section in this wind tunnel, the airfoil almost acts as a nozzle in the tunnel and affects the static pressure at a point on the wing. One crude way to start correcting this effect is to scale down the coefficient of lift by the open area of the test section: $\frac{A_{test}-A_{blocked}}{A_{test}}$. When the coefficients of lift are adjusted in this way, the plot in Fig. 6 is obtained. Though this does not fully solve the disparity in the data from the NACA results, one can see this plot is closer to the expected results.

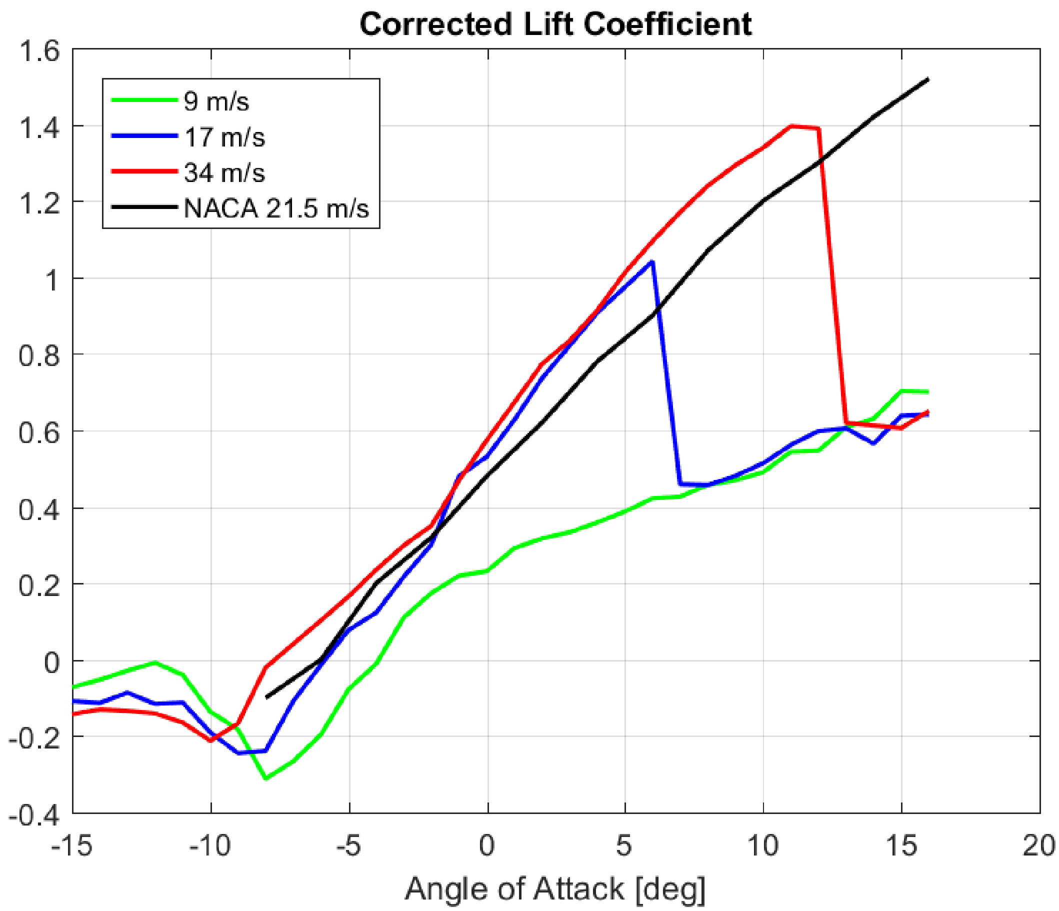


Figure 6: Corrected Lift Coefficient *These are the lift coefficients corrected for blockage in the test section caused by the airfoil as it gets higher angles of attack.*

VI. Conclusions

When compared to the 1938 NACA data, the findings in this lab are largely similar, but do have some key differences. For the lift coefficient, the NACA data for a velocity of 21.4 m/s appears to show no stall, even at angles of attack upwards of 15. In contrast, the lab data only shows no stall for the low velocity of 9 m/s. For 17 and 34 m/s, our data shows stall at angles of attack of about 6 and 12, respectively. For the drag coefficient, the lab data has a tighter fit to the NACA data, but does not include considerations of the effect of viscosity, which contributes significantly to drag force. It follows the values slightly more closely than the lift coefficient, and follows the trend of the data throughout all tested angles of attack much more closely. Based on this comparison, this lab does a reasonable job of quantifying lift forces about a Clark Y-14 airfoil, but not as good of a job at determining drag, due to the lack of analysis of viscous forces.

References

[1]Anderson, John David. Introduction to Flight. 8th ed., McGraw Hill Education, 2016.

[2]Taylor, John Robert. An Introduction to Error Analysis: the Study of Uncertainties in Physical Measurements. 2nd ed., University Science Books, 1997.

[3]Farnsworth, John. Aerodynamics of a Cambered Airfoil. 15 Nov. 2017, learn.colorado.edu/d2l/le/content/215270/viewContent/3291372/View.

[4]Pinkerton, Robert M., Greenberg, Harry. Aerodynamic Characteristics of a Large Number of Airfoils Tested in the Variable Density Wind Tunnel. Report No. 628., National Advisory Committee for Aeronautics, 1938.

Appendix A: MATLAB Code

```
%ASEN 2002 Lab 2
%Group 1
%Data Analysis

%Fresh Start
clear;clc;close all;

%Some Constants
R = 287; %[J/(kg*K)]
velocities=[9 17 34];
attacks = -15:16;
c = sqrt(.14665^2 + 3.5^2);

%Gather all data
nameID = {'01','03','05','07','09','11'};%,'13','15'};
losCps=zeros(32,16,3); %angle down rows, port across columns, velocity in 3D


for i=1:length(nameID)
    name=['AirfoilPressure_S014_G' nameID{i} '.csv'];
    data=xlsread(name);
    ports=data(:,7:end-6);
    data = data(any(ports,2),:); %Delete rows of zeros
    p_atm=data(:,1);
    t_atm=data(:,2);
    airspeed=data(:,4);
    pitotDynamicPres = data(:,5);
    ports=data(:,7:end-6);%Get good data for ports again
    angle=data(:,end-5);

    for j=1:3 %Each team tested three velocities
        inds=not(abs(sign(sign(velocities(j)-3 - airspeed) + sign(velocities(j)+3 - airspeed))));
        %Further subdivide by angle
        for k=1:32 %Just check each angle
            a = attacks(k);
            inds2 = (angle(inds) == a); %indices in each velocity where there is a specific attack angle
            if sum(inds2)>0 %this angle was tested
                for l = 1:16 %16 port measurements
                    pDp = pitotDynamicPres(inds);
                    q = mean(pDp(inds2));

                    portaloons = ports(inds,l);
                    deltaP = mean(portaloons(inds2));

                    Cp = deltaP/q;
                    losCps(k,l,j) = Cp;
                end
            end
        end
    end
end
```

```

%Chordwise positions
pos_xc = [0 .175 .35 .7 1.05 1.4 1.75 2.1 2.8 2.8 2.1 1.4 1.05 0.7 0.35 0.175];
samples = [6 16 26]; %Indices of "attacks" for the angles we want to look at
n=length(samples);

%Plot Cp at a sampling of attack angles
for i=1:3 %A plot for each velocity
    figure
    for j=1:n
        subplot(1,n,j)
        plot(pos_xc(1:9)/c,losCps(samples(j),1:9,i),'k')
        hold on
        plot(pos_xc(10:16)/c,losCps(samples(j),10:16,i),'r')
        set(gca,'Ydir','reverse')
        subname = sprintf('C_p For \x3B1 = %1.f',attacks(samples(j)));
        title(subname);
    end
    name = sprintf('Velocity = %1.f [m/s]',velocities(i));
    suptitle(name);
end


%%%%%%%%%%%%%%%%%%%%%%
% LIFT AND DRAG COEFFICIENTS
%%%%%%%%%%%%%%%%%%%%%%
close all;
clear all;
clc
%Read in data
C_l = zeros(3,32);
C_d = zeros(3,32);
aoa = zeros(1,32);
index = 1;
for i = 1:8
    num = (i-1)*2 + 1;
    groups = ['01','03','05','07','09','11','13','15'];
    filename = ['AirfoilPressure_S014_G' groups(num:num+1) '.csv'];
    datain = xlsread(filename);
    % Replacing zero rows
    if i == 3
        datain = [datain(1:219,:); mean(datain(1:219,:),1); datain(221:end,:)];
    elseif i ==8
        datain = [datain(1:79,:); mean(datain(1:79,:),1); datain(81:end,:)];
    end
    %cutting up data for each aoa
    for j = 0:3
        data2 = datain(j*60 + 1:j*60 + 60,:);
        %cutting up data for each velocity 9,17,34 m/s
        for k = 0:2
            data = data2(k*20 + 1:k*20 + 20,:);
            [C_l(k+1,index),C_d(k+1,index),aoa(index)] = getCo(data);
        end
```

```
            index = index + 1;
        end
    end
%% Plotting
% Sorting data by AOA
plot_data = [aoa./0.0174533; C_l; C_d];
plot_data = sortrows(plot_data',1)';
%NACA data
naca_aoa = -8:2:16;
length(naca_aoa)
naca_lift = [-0.1,0,0.2,0.32,0.48,0.62,0.78,0.9,1.07,1.2,1.3,1.42,1.52];
naca_drag = [0.08,0.05,0.07,0.1,0.18,0.22,0.3,0.39,0.49,0.6,0.71,0.85,1]*2/10;
naca_v = 21.5; %m/s
% Actually plotting it
    %Lift
    subplot(1,2,1)
    l_plot = plot(plot_data(1,:),plot_data(2,:),'g',plot_data(1,:),plot_data(3,:),'b',plot_data(1,:),plot_data(4,:),'r',naca_aoa,naca_lift,'k');
    grid on;
    title('Lift coefficient');
    xlabel('AOA [deg]');
    legend('9 m/s','17 m/s','34 m/s','NACA 21.5 m/s');
    %Drag
    subplot(1,2,2)
    d_plot = plot(plot_data(1,:),plot_data(5,:),'g',plot_data(1,:),plot_data(6,:),'b',plot_data(1,:),plot_data(7,:),'r',naca_aoa,naca_drag,'k');
    grid on;
    title('Drag coefficient');
    xlabel('AOA [deg]');
    legend('9 m/s','17 m/s','34 m/s','NACA 21.5 m/s');

    set(l_plot,'linewidth',1.5);
    set(d_plot,'linewidth',1.5);
    suptitle('Lift and Drag coefficients vs AOA')
%% Forces
function [C_l,C_d,aoa] = getCo(data)
    %Coordinates of ports in meters, this includes trailing edge
    coords = 0.0254*[0,0.14665; 0.175,0.33075; 0.35,0.4018; 0.7,0.476; 1.05,0.49; 1.4,0.4774; 1.75,0.4403; 2.1,0.38325; 2.8,0.21875; 3.5,0; 2.8,0; 2.1,0; 1.4,0; 1.05,0; 0.7,0.0014; 0.35,0.0175; 0.175,0.03885];
    c = 0.0254*3.5; %chord in meters
    normal = zeros(1,16);
    axial = zeros(1,16);
    P_dyn = mean(data(:,5)); %Freestream dynamic pressure
    diffP = mean(data(:,7:22),1); %Differential pressure averages
    aoa = 0.0174533*data(1,23); %Angle of attack in radians
    P_t = mean([diffP(9) diffP(10)]); %Trailing edge pressure
    diffP = [diffP(1:9) P_t diffP(10:16)]'; %inserting trailing edge pressure into diffP
    for j = 1:16 %integrating pressures to get forces
        dx = coords(j+1,1) - coords(j,1);
        dy = coords(j+1,2) - coords(j,2);
        normal(j) = -0.5*(diffP(j) + diffP(j+1))*dx;
        axial(j) = 0.5*(diffP(j) + diffP(j+1))*dy;
    end
```

```
    F_n = sum(normal); %normal net force
    F_a = sum(axial); %axial net force
    %Normal and axial force coefficients
    C_n = F_n/P_dyn/c;
    C_a = F_a/P_dyn/c;
    %Drag and lift coefficients
    C_l = C_n*cos(aoa) - C_a*sin(aoa);
    C_d = C_n*sin(aoa) + C_a*cos(aoa);
end
```