Aerodynamic Force and Moment measurements for Full Model Airplane

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This experiment measures the aerodynamic forces and moments acting on a DC-6B aircraft in SDSU's subsonic wind tunnel to evaluate the students ability so separate performance characteristics. The objectives include determining lift, drag, pitching, and yaw moment coefficients at various angles of attacks and sideslip angles. The key outcomes are to measure the major characteristics of an airplane and using the external strain gauge balance, and to obtain better understandings of airplane performance based on the theory learned in the class and this experiment.

Nomenclature

q	=	Dynamic Pressure
rho	=	Air Density
V	=	Air Velocity
Re	=	Reynolds Number
alpha	=	Angle of Attack
beta	=	Sideslip Angle
S	=	Wind reference Area
c	=	Mean aerodynamic chord
b	=	Wingspan
AR	=	Aspect Ratio
F	=	Forces in x,y,z
M	=	Moments in x,y,z
CL	=	Lift Coefficient
CD	=	Drag Coefficient
CM	=	Pitching Moment Coefficient
CN	=	Yawing Moment Coefficient
e	=	Oswald Efficiency Factor
K	=	Induced Drag Constant
R	=	Gas Constant
mu	=	Dynamic Viscosity

I. Introduction

THIS lab involves aerodynamic testing of an aircraft model to find insights on stability and control within the wind tunnel. For this lab, we will be testing the forces and moments applied to a DC-6B aircraft model in the SDSU subsonic wind tunnel. This will lead to calculations of lift, drag, pitching, and yawing moments across a set of angles of attacks and sideslip angles. These results graphed will allow for real-world understanding of the necessity of this kind of testing, and knowledge of the applied theory.

This lab bridges theoretical aerodynamics with the empirical validation found through the data collected in the lab. This will reinforce thin airfoil theory as was taught in our class. The result will be compared to predictions and will allow for us to compare the results and identify shortcomings with the lab, the calculations, and the experiment as a whole.

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I. Theory

The importance of the theory section is to inform the readers of every equation used in the experiment to be as clear and transparent as possible in the efficacy of the lab. To start this lab off, the calculation of dynamic pressure in order to get the air velocity in the wind tunnel. The significance of this ensures that results are scalable to real flight conditions. This equation comes from Bernoulli's equation for incompressible flow.

$$V = \sqrt{\frac{2q}{\rho}} \qquad \text{equation 1}$$

This is a fundamental fluid mechanics equation. Next we have Reynolds number, which characterizes the flow regime between laminar and turbulent flows based on both the inertial and viscous forces in the fluid. In this formula, Rho is the air density from the ideal gas law, V is the flow velocity, c is the mean aerodynamic chord, and mu is the dynamic viscosity derived using Sutherland's Formula.

$$Re = \frac{\rho V c}{\mu} \qquad \text{equation 2}$$

C is given in this lab as 3.466 inches, velocity is derived from equation 1, density and viscosity are calculated using the ambient temperature and pressures collected. The results section of this lab will contain these results. After Reynold's we describe the Force and Moment corrections described in the lab handout. This is calculated by subtracting and isolating the structural interference. By doing this we remove tare effects and strut aerodynamics such as the model weight.

$$F_{aero} = [F_{model\ on,wind\ on} - F_{model\ on,wind\ off}] - [F_{model\ off,wind\ on} - F_{model\ off,wind\ off}] \qquad \text{equation 3}$$

This equation yields pure aerodynamic forces acting on the model, and can be calculated using all of the separate tests collected in the experiment portion of this lab. It should be noted that moments will work the same way, since they were calculated at the same time as the forces. The next equations are the aerodynamic coefficients. The first is for lift.

$$C_L = \frac{L}{qS} \qquad \text{equation 4}$$

Where L is the lift (or force in the z direction), S is the reference area, and q is the dynamic pressure. This comes from the thin airfoil theory equation, and it determines lift generation efficiency by normalizing the dynamic pressure and wing area. The second is the drag coefficient:

$$C_D = \frac{D}{qS} \qquad \text{equation 5}$$

Where D is the drag (or force in the x direction). This quantifies the drag at any given lift coefficient and comes from the drag decomposition. Thirdly is the pitching Moment coefficient.

$$C_M = \frac{M}{qSc} \qquad \text{equation 6}$$

For this, c is the reference length which is used to balance the moment across the equation. It evaluates the longitudinal stability, with M being the moment along the y axis. Negative instances of this coefficient are often proportional to stability. Finally we have Yawing Moment Coefficients:

$$C_N = \frac{N}{qSb} \qquad \text{equation 7}$$

B here is the width for the airfoil, and N comes from the moment in the z axis from our data. The purpose is to assess directional stability, and a positive slope implies a stable aircraft. Moving onto an alternate method for coefficients, there is the Drag Polar and Oswald Efficiency.

$$C_D = C_{D,0} + KC_L^2 \qquad \text{equation 8}$$

$$K = \frac{1}{\pi e AR} \qquad \text{equation 9}$$

$$AR = \frac{b^2}{S} \qquad \text{equation 10}$$

$$e = 1.78(1 - 0.045AR^{0.68}) - 0.64 \qquad \text{equation 11}$$

For this derivation, K utilized the aspect ratio of the aircraft, which is defined above in this telescoping set of equations. $C_{D,0}$ comes from the data, and the value of the drag coefficient at the moment the angle of attack is equal to zero. This combined with the empirical Oswald efficiency factor allows us to predict drag at varying lift conditions, and links the wing geometry in order to reduce drag in the system. This also is used to split drag into its component parts. Oswald's efficiency factor quantifies how aerodynamically clean a wing is. This is done by accounting for non ideal lift and additional drag from the wingtip and fuselage interference, and allows for design optimization for our system.

Sample Calculations:
Using the following raw data for angle of attack and sideslip angle equal to zero, for each of the table instances and their testing runs. These are used for the calculations shown below using the equations explained earlier in the theory:

Fx **(lb)**	***Fy*** **(lb)**	***Fz*** **(lb)**	***Mx*** **(lb·in)**	***My*** **(lb·in)**	***Mz*** **(lb·in)**
0.02748	-0.02599	0.1092	-0.9071	-0.7408	0.005568
3.197	0.01982	8.496	-0.5368	-17.54	0.9833
3.337	0.1271	7.016	-1.644	-24.04	-1.073
2.586	0.3248	1.039	-1.670	-26.95	0.2558
0.009616	0.002915	0.02409	-0.3341	-0.3104	0.06663

Table 1: Raw data for sample calculations

Constant	Value	Units
q	0.1805	psi
S^2	93.81	in²
c	3.466	in
b	27.066	in
R	1716	$\frac{ft \cdot lb}{slugs \cdot R}$
T	534.8	°R
P	14.74	psi

Table 2: Constants for sample calculations

$$V = \sqrt{\frac{2q}{\rho}} = \sqrt{\frac{2 \cdot .1805}{.002308}} = 149.9 \frac{ft}{s}$$

$$\mu = 2.629e-7 \frac{T}{T+198.72} = 2.629e-7 \frac{534.8}{534.8+198.72} = 3.831e-7 \left(\frac{slugs}{ft \cdot s}\right)$$

$$Re = \frac{\rho V c}{\mu} = \frac{.002308 \cdot 149.9 \cdot \frac{3.466}{12}}{3.831e-7} = 3.252e5$$

$$F_{aero} = (F_{1,1,1} - F_{1,1,0}) - (F_{0,0,1} - F_{0,0,0}) =$$

$$F_z = (8.496 - .1092) - (1.039 - .02409) = 7.372\ lb$$

$$F_x = .6006\ lb, \qquad M_y = 9.40\ lb \cdot in, \qquad M_z = .7886\ lb \cdot in$$

$$C_L = \frac{F_z}{qS} = .4348$$

$$C_D = \frac{F_x}{qS} = .03542$$

$$C_M = \frac{M_y}{qSc} = .1703$$

$$C_N = \frac{M_z}{qSc} = .001721$$

II. Setup and Procedure

Having a clear setup, apparatus descriptions, and procedure is important for any lab report to be able to display only the needed information to allow someone from outside the class to be able to understand and replicate the experiment using only the information provided. This section will split into two sections, first the Equipment and Apparatus (in paragraph format) and secondly the setup and procedure (in bullet format). With this, any reader should be able to accurately replicate the data we have collected for this lab.

The SDSU subsonic wind tunnel is a closed return type with test section dimensions of 45” x 32” x 67” and a speed range from 0 to 180 mph. There is an expected turbulence factor of 1.27. The wind tunnel uses a 150 HP variable pitch 4 blade propellor, and is equipped with Particle Image Velocimetry (PIV) and Laser Doppler Velocimetry (LDV) with three degree translation systems for flow field analysis. There is also a 6 component load cell strain gauge balance system. The balance system has the following load limits: Lift = 150 lb, Drag = 50 lb, Side force = 100 lb, Pitch = 1000 inch-pounds, roll = 1000 inch-pounds, and Yaw = 1000 inch-pounds. This is calibrated for <1% interaction error under combined loads.

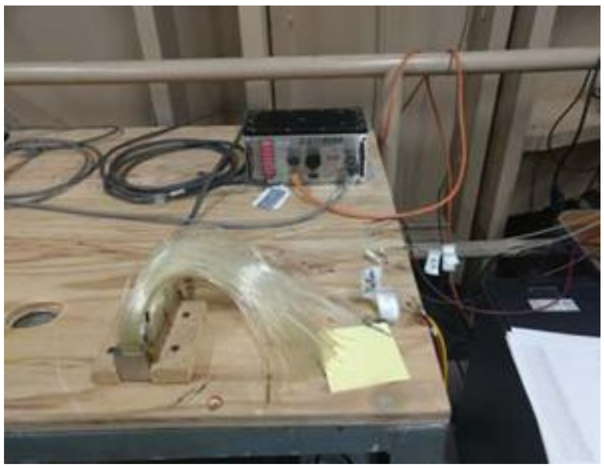

Figure 1: Pressure Transducer for wind tunnel

The Data Acquisition System uses LabView software to calculate 1000 runs at 5000 Hz sampling rate, and measures the forces and moments within the chamber. The recording of ambient temperature and pressure is done through electronic thermometers and barometers respectively.

Figure 2: DSM4000 data intepreter

The airplane model used in this experiment is a DC-6B with a reference area of 93.81 square inches, a reference length of 3.466 inches, and a reference wing span of 27.066 inches. This lab will be conducted at either q=5 inH2O or q=0 inH2O, depending on the necessary requirements at that time. A water manometer is used to measure this dynamic pressure maintained throughout the testing.

Procedure:

Wind Tunnel Preparation

- Inspect test section for debris or obstructions
- Ensure access doors are closed and latched
- Verify all equipment is secured properly inside test section

Cooling System Activation

- Turn on cooling System to stabilize the internal temperature
- Allow the system to run for at least 10 minutes to ensure minimal temperature drift throughout testing

Dynamic Pressure Calibration

- Connect the pitot static probe used for testing to the pressure scanner
- Set the wind tunnel speed to q = 0 inH2O, and initiate a baseline pressure reading
    - This will be used later to zero data
- When required to set speed to q = 5 inH2O, gradually increase speed while monitoring manometer
    - Allow time for the manometer reading to stabilize before collecting any data

Model Installment

- Secure DC-6B model into the strut assembly, and fasten securely
- Attach strut to the 6 component load cell strain gauge for proper load measurements
- Use inclinometer to set the angle of attack to zero
    - Adjust rear strut screw to fine tune the angle if necessary
- Confirm sideslip angle is zero and the fuselage is centered
- Zero the balance to account for model weight, and record the initial forces (Fx, Fy, Fz, Mx, My, Mz)

Execution:

Run	Model	Tail	Wind	Purpose	Angles Tested	Key steps
1	On	On	Off	Measure weight and structural interference	α: -6 to 15 (β = 0) β = 0, 5, 10 (α = 0)	Zero balance Adjust between α & β
2	On	On	On	Measure full aerodynamic loads	α: -6 to 15 (β = 0) β = 0, 5, 10 (α = 0)	Set q = 5 inH2O Monitor q stability
3	On	Off	On	Isolate tail effects	α: -6 to 15 (β = 0) β = 0, 5, 10 (α = 0)	Remove tail Isolate effects
4	Off	Off	On	Measure strut drag	β = 0, 5, 10 (α = 0)	Remove model Record strut forces
5	Off	Off	Off	Measure strut weight	β = 0, 5, 10 (α = 0)	All off for tare forces

Table 3: Execution steps for each run, with purpose and key steps

From table 1, we should be able to gleam what is to be done for each step. The angles will each have data recorded for them, and their values averaged for the final raw data. The major purpose of each of these runs is to observe the effects of each aspect of the forces at play in the wind tunnel. This will allow for isolation each aspect and will later be used to calculate the adjusted forces and moments at play in the wind tunnel.

III. Results

The following section will go over each of the aspects of this lab we were expected to calculate, and explain the results and their implications for this lab. To begin, we will go over the following table for the Reynolds number:

Run #	1	2	3	4	5
Reynolds (x10^4)	2.2385	2.2626	2.2774	2.2391	2.2379

Table 4: Reynold's number for each run

As far as can be concluded, the Reynolds numbers for this experiment are reasonable and accurate with an extremely small error of <.5%, and numbers that are to be expected with the lab requirements, since the testing speed is small and the temperatures and pressures remain consistent throughout the lab if you consider the raw data.

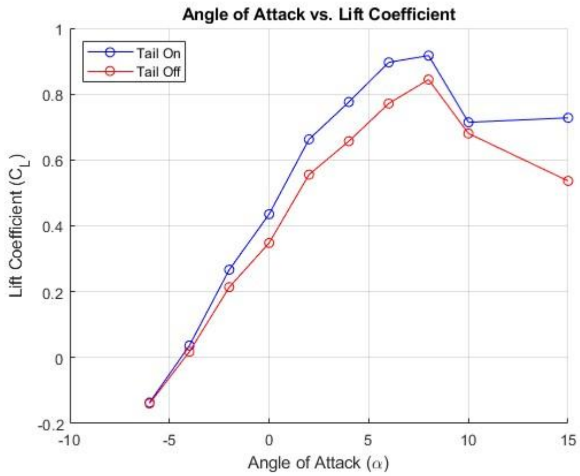

Figure 3: Angle of Attack vs. Lift Coefficient for Tail off and on

There are many takeaways to be had from the data here. This graph should show how lift generation changes in accordance to the angle of attack for each experiment. The linear region between -6 and 8 degrees is the attached flow over the airfoil, and the stall point can be seen for angles > 8 degrees. This is a short angle for stall, which is normally attributed to 15+ degrees, but is also not an unreasonable angle for this to occur at. Between the tail on and tail off graphs, the lift becomes slightly shorter when closing in on the stall, which does make conceptual sense, as the lack of tail down force would reduce the overall lift. The data scatter has a wobble from the raw data which would be used better to flatten the curve. And the experiment could also be improved upon if we were given different q values at which to test and to compare with these results.

The lift slope $\frac{dC_L}{d\alpha}$ should be ~0.1-0.12 theoretically to match thin airfoil theory, unfortunately and strangely we do not achieve this number. From the code, the lift slope is calculated to be ~0.08. This is not terribly far from the bottom number of the expected lift slope, but it is clear that there is error in this system that is causing this to be inaccurate. This will be expanded on later in the conclusion. In addition, the maximum list coefficient is calculated to be $C_{L,max} = 0.9169$, which relates conceptually as an accurate figure. This value is identified at the critical angle of attack, $\alpha_{max} = 8.0$, which is the value at which we see the maximum coefficient of lift in the system.

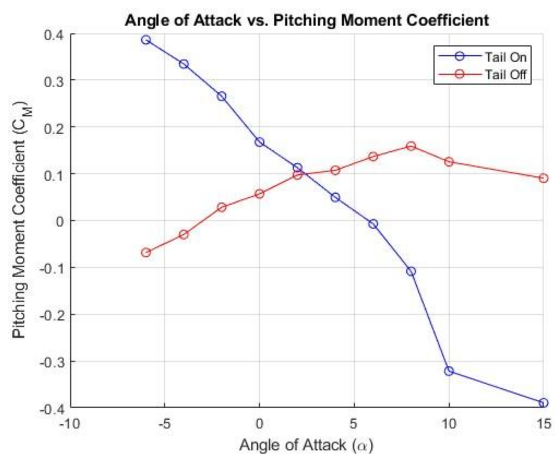

Figure 4: Angle of Attack vs. Pitching Moment Coefficient for Tail off and on

In this figure we are meant to measure the longitudinal stability of the aircraft. In this case, a negative slope should suggest a stable aircraft. This would mean that nose pitches down as the angle of attack increase. From the graph we can see that the tail on shows a negative slope, however the tail off does not, which means that without the tail this aircraft is unstable. There is a zero moment angle where we can find the trim point. The zero lift angles are as follows: Tail on = -4.4225, Tail off = -4.2266. These are expected zero lift angles and can be used to trim the aircraft to make it the most generally stable it can be.

There are moment wobbles in the data that could distort the slopes and strut interference that can mess with drag, despite our efforts to zero them. The pitch moment slope is calculated to be $\frac{dC_m}{d\alpha}$=-0.0360, which is between the expected range and is a considerably accurate number from these calculations.

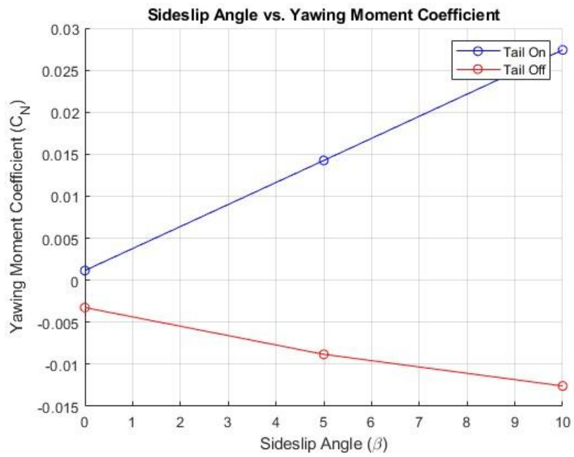

Figure 5: Sideslip Angle vs. Yawing Moment Coefficient for Tail off and on

The yawing angle vs. the sideslip angle evaluates the directional stability, otherwise known as the “weathercock” effect. In this case, a positive slope will suggest a stable aircraft since the nose yaws into the wind. From the graph, the tail on becomes more positive as the sideslip increases, which is opposite for the tail off. This is to be expected and suggests that the tail off is unstable. This error can also come from wobble noise and a possible beta misalignment which would lead to large error later in the data.

The yaw moment slope is calculated to be $\frac{dC_n}{d\alpha}$ = 0.0026. This is not a ridiculous number to calculate, however it is lower than could be expected. This seems likely if not a little unexpected, but error could explain much of this.

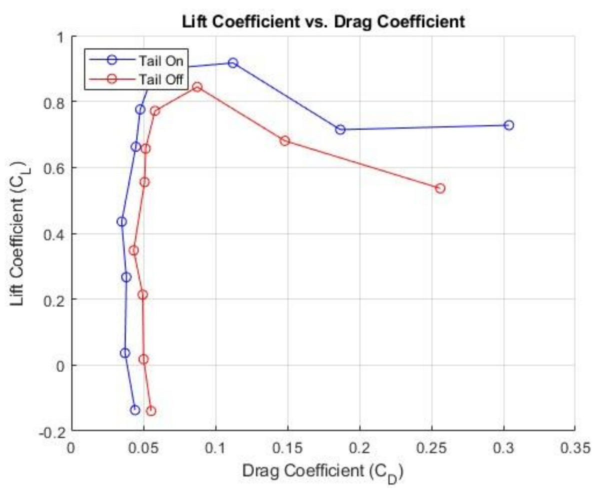

Figure 6: Lift Coefficient vs. Drag Coefficient for Tail off and on

For the final figure, we attempt to quantify the efficiency of the aircraft by relating Cl to Cd. The peak of the curve is also the optimal cruise relation. The curve for both tail on and off is parabolic, which confirmed that there is an induced drag at high lift coefficients, which is another way to observe stall in the aircraft. The tail causes the lift to be shorter, which could also suggest that there is less lift thanks to not having the tail, but there is also less drag. This would mean that at high angles of attack there is a parasite drag from the tail.

The Maximums in this relationship are as follows: $\frac{C_L}{C_D}$ for tail on = 3.0173, for tail off = 3.2956. With this we can see that there is far more drag dominance from the lack of a tail, once again showing the importance of the tail for a properly flown aircraft. In addition to this, we have calculated the Oswald Efficiency Factor using equation 11 in the theory section. This would suggest that from the curve fit, a value of 0.8 would align with theory for straight wings. This is proven, as the calculation shows a value of e = 0.8160. This is an appropriate value for this experiment.

IV. Conclusion

In conclusion the lab had almost entirely reliable data. From each of the tables we are able to see what we generally would expect from the data to show us, especially with Tail On data points. For tail off, we do see slightly differing results from theoretical, but in all cases they merely suggest that there is less stability in the aircraft without the tail which we should expect. The zero lift angles, Cl/Cd maximums, and maximum lift coefficients discussed in the results also are accurate and acceptable for what we know theoretically should be the result. The only issue I can see in this lab would be a surprisingly low lift slope, which I must conclude comes from some data error, which we can see is prevalent throughout the lab. Thanks to the calibration and the zeroing of the data through equation 3, we are able to properly relate the data to the coefficients and discuss them properly and with full understanding of the fundamentals. In the appendix I have a proof included for this equation.

Design Recommendations

As for design recommendations, as mentioned previously, it may be handy to provide different dynamic pressures for the raw data, which we can use to identify separate trends in the data. Besides that, it might be handy to minimize the angles we test at, and focus more on some of the other extremes. If we had been able to test at lower angles of attack we could see downward stall, which tends to be important in extreme flight cases. Besides changes like that, there are likely many wobbles in the data that play a roll in all of the calculations and error. This can be ruled out with stronger equipment and potentially a filter to the data to take out noise.

Appendix

The following is a proof for equation 3:

$$F_{aero} = \left[F_{model\ on,wind\ on} - F_{model\ on,wind\ off}\right] - \left[F_{model\ off,wind\ on} - F_{model\ off,wind\ off}\right]$$

This equation is meant to remove two sources of interference, gravitational tares, and support strut forces. Firstly we should define the forces in the wind tunnel and balance the combined effects:

$$Model + strut + wind\ (F_{measured}) = F_{model} + F_{strut} + F_{weight} + F_{noise}$$

We can get to the model forces by cancelling out the other terms. Model on and wind on forces account for model, strut, and weight, Model on wind off accounts for weight, model off wind on accounts for both aero forces on and weight of strut, and lastly model off wind off gives only the weight strut

$$F_{on,on} = F_{model} + F_{strut} + F_{weight}$$
$$F_{on,off} = F_{weight}$$
$$F_{off,on} = F_{aero\ strut} + F_{weight\ strut}$$
$$F_{off\ off} = F_{weight\ strut}$$

By combining these terms into the derived equation, we can create the following equation:

$$F_{aero} = F_{model} + F_{strut} + F_{weight} - F_{weight} - F_{aero\ strut} + F_{weight\ strut} + F_{weight\ strut}$$

By canceling out weight strut, weight, and aero strut, we are left with the final equation:

$$F_{aero} = F_{model}$$

And thus we are able to separate the data into only observing the aerodynamic forces on the model. This same application is used for moments, and all the same without the tail. This is a powerful method to being able to analyze the effects of an object in a wind tunnel without the equipment causing much error.

CODE

```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%
% AE302 Lab 5 - Wyatt Welch
%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
clc, clear all

% Load Data
load("dat111.mat"); dat111 = dat111(2:end,:);
load("dat110.mat");
load("dat101.mat"); dat101 = dat101(2:end,:);
load("dat001.mat");
```

```
load("dat000.mat");

% Corrections
fill001 = dat001(1,:);
dat001 = [repmat(fill001, 10, 1); dat001(2:3,:)];
fill000 = dat000(1,:);
dat000 = [repmat(fill000, 10, 1); dat000(2:3,:)];

% Reordered Data
q = 5 * .0360912; %Psi
S = 93.81;
c = 3.466;
b = 27.066;
R = 1716; %ftlb/slugR
angA = dat111(:,1);
angB = dat000(10:12,2);

T111 = 75.1; P111 = 30.06;
T110 = 79; P110 = 29.97;
T101 = 72.5; P101 = 30.08;
T001 = 78.8; P001 = 29.96;
T000 = 79.1; P000 = 29.97;
T_array = [T111 T110 T101 T001 T000] + 459.67; %R
P_array = [P111 P110 P101 P001 P000] .* .49115; %Psi

% Reynolds Numbers
rho = (P_array * 144) ./ (R * T_array);
V = sqrt((2 * (5 * 5.204)) ./ rho);
mu = 2.629e-7 * (T_array.^1.5) ./ (T_array + 198.72);
Re = (rho .* V * (c / 12)) ./ mu;
fprintf("Reynolds numbers [111 110 101 001 000]:\n")
fprintf("%4.4e ",Re)
fprintf("\n\n")

% Set-up Data
Fxd111 = dat111(:,4);
Fxd110 = dat110(:,4);
Fxd101 = dat101(:,4);
Fxd001 = dat001(:,4);
Fxd000 = dat000(:,4);

Fyd111 = dat111(:,5);
Fyd110 = dat110(:,5);
Fyd101 = dat101(:,5);
Fyd001 = dat001(:,5);
Fyd000 = dat000(:,5);

Fzd111 = dat111(:,6);
Fzd110 = dat110(:,6);
Fzd101 = dat101(:,6);
Fzd001 = dat001(:,6);
Fzd000 = dat000(:,6);

Mxd111 = dat111(:,7);
```

```
Mxd110 = dat110(:,7);
Mxd101 = dat101(:,7);
Mxd001 = dat001(:,7);
Mxd000 = dat000(:,7);

Myd111 = dat111(:,8);
Myd110 = dat110(:,8);
Myd101 = dat101(:,8);
Myd001 = dat001(:,8);
Myd000 = dat000(:,8);

Mzd111 = dat111(:,9);
Mzd110 = dat110(:,9);
Mzd101 = dat101(:,9);
Mzd001 = dat001(:,9);
Mzd000 = dat000(:,9);

% Force/Moment Corrections
FxT1 = (Fxd111 - Fxd110) - (Fxd001 - Fxd000);
FyT1 = (Fyd111 - Fyd110) - (Fyd001 - Fyd000);
FzT1 = (Fzd111 - Fzd110) - (Fzd001 - Fzd000);

MxT1 = (Mxd111 - Mxd110) - (Mxd001 - Mxd000);
MyT1 = (Myd111 - Myd110) - (Myd001 - Myd000);
MzT1 = (Mzd111 - Mzd110) - (Mzd001 - Mzd000);


FxT0 = (Fxd101 - Fxd110) - (Fxd001 - Fxd000);
FyT0 = (Fyd101 - Fyd110) - (Fyd001 - Fyd000);
FzT0 = (Fzd101 - Fzd110) - (Fzd001 - Fzd000);

MxT0 = (Mxd101 - Mxd110) - (Mxd001 - Mxd000);
MyT0 = (Myd101 - Myd110) - (Myd001 - Myd000);
MzT0 = (Mzd101 - Mzd110) - (Mzd001 - Mzd000);


% Coefficient Calculations
Cl1 = FzT1 / (q * S);
Cl0 = FzT0 / (q * S);

Cd1 = FxT1 / (q * S);
Cd0 = FxT0 / (q * S);

Cm1 = MyT1 / (q * S * c);
Cm0 = MyT0 / (q * S * c);

Cn1 = MzT1 / (q * S * b);
Cn0 = MzT0 / (q * S * b);

% Cd Components
AR = b^2 / S;
e = 1.78 * (1-.045 * AR ^ .68) - .64;
fprintf("The Oswald Efficiency Factor, e = %4.4f\n\n",e)
K = 1/(pi() * e * AR);
```

```
alpha_dat = angA(1:end-2);
Cl_dat = Cl1(1:end-2);
Cl_dat0 = Cl0(1:end-2);
cross = find(diff(sign(Cl_dat)));
cross0 = find(diff(sign(Cl_dat0)));
alpha0 = interp1(Cl_dat(cross:cross+1), ...
               alpha_dat(cross:cross+1), ...
               0, 'linear');
alpha01 = interp1(Cl_dat0(cross0:cross0+1), ...
               alpha_dat(cross0:cross0+1), ...
               0, 'linear');
fprintf("Zero lift angles \nTail On: %4.4f\nTail Off:
%4.4f\n\n",alpha0,alpha01)
[~,idx] = min(abs(angA-alpha0));
C_d0 = Cd1(idx);
Cd = C_d0 + K * Cd1.^2;

ClCdMax1 = max(max(Cl1(1:end-2) / Cd1(1:end-2)));
ClCdMax0 = max(max(Cl0(1:end-2) / Cd0(1:end-2)));
fprintf("Cl/Cd maximums\nTail on: %4.4f \nTail off:
%4.4f\n\n",ClCdMax1,ClCdMax0)

% dCl/da, dCm/da, dCn/db
lin_range = (alpha_dat >= -6) & (alpha_dat <=8);
pCl = polyfit(alpha_dat(lin_range), Cl_dat(lin_range),1);
Clslope = pCl(1);

Cm_dat = Cm1(1:end-2);
pCm = polyfit(alpha_dat(lin_range), Cm_dat(lin_range),1);
Cmslope = pCm(1);

beta_dat = angB;
lin_range = (beta_dat >= 0) & (beta_dat <=10);
Cn_dat = Cn1(10:12,:);
pCn = polyfit(beta_dat(lin_range), Cn_dat(lin_range),1);
Cnslope = pCn(1);

fprintf("Lift Slope dCl/da = %4.4f \nPitch Moment Slope dCm/da = %4.4f " +
...
    "\nYaw Moment Slope dCn/db = %4.4f\n\n",Clslope,Cmslope, Cnslope)


% Stall
[Cl_max, idx_max] = max(Cl_dat);
alpha_stall = alpha_dat(idx_max);
fprintf("Maximum lift coefficient, C_lmax = %4.4f" + ...
    "\nCritical angle of attack, \alpha_max =
%4.4f\n\n",Cl_max,alpha_stall)


% Plots
figure(1)
hold on, grid on
```

```
plot(angA(1:end-2), Cl1(1:end-2), '-ob')
plot(angA(1:end-2), Cl0(1:end-2), '-or')

legend("Tail On", "Tail Off", 'Location','northwest')
xlabel('Angle of Attack (\alpha)')
ylabel('Lift Coefficient (C_L)')
title('Angle of Attack vs. Lift Coefficient')


figure(2)
hold on, grid on
plot(angA(1:end-2), Cm1(1:end-2), '-ob')
plot(angA(1:end-2), Cm0(1:end-2), '-or')

legend("Tail On", "Tail Off", 'Location','northeast')
xlabel('Angle of Attack (\alpha)')
ylabel('Pitching Moment Coefficient (C_M)')
title('Angle of Attack vs. Pitching Moment Coefficient')


figure(3)
hold on, grid on
plot(angB, Cn1(10:12,:), '-ob')
plot(angB, Cn0(10:12,:), '-or')

legend("Tail On", "Tail Off", 'Location','northeast')
xlabel('Sideslip Angle (\beta)')
ylabel('Yawing Moment Coefficient (C_N)')
title('Sideslip Angle vs. Yawing Moment Coefficient')


figure(4)
hold on, grid on
plot(Cd1(1:end-2), Cl1(1:end-2), '-ob')
plot(Cd0(1:end-2), Cl0(1:end-2), '-or')

legend("Tail On", "Tail Off", 'Location','northwest')
ylabel('Lift Coefficient (C_L)')
xlabel('Drag Coefficient (C_D)')
title('Lift Coefficient vs. Drag Coefficient')
```