MEM 425:
Final Report

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Name	% Deserved	Signature
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Design Process and Decision

The design process for this project started with selecting an airfoil and a wing planform. An elliptical wing planform was selected for its outstanding aerodynamic properties given that it has the lowest possible induced drag according to thin airfoil theory. Choosing an elliptical planform also provides a constant oswald efficiency factor of 1. Additionally, the AG03 airfoil was chosen for its close resemblance to a flat plate, which was the original intended wing material. Though elliptical planform wings are notorious for their poor manufacturability, the aerodynamic efficiency was reason enough to overcome the manufacturing difficulties.

Once the wing planform and airfoil were chosen, an aspect ratio and wing span were selected after listening to professor Harding's lecture. Knowing that a long thin wing with a relatively large aspect ratio would be ideal, the aspect ratio and wing span were chosen to be 6 inches and 24 inches, respectively. The reason a long and thin wing with respect to the body of the aircraft is favorable, is because it will produce less drag for the generated amount of lift; making it highly efficient.

After the aspect ratio and oswald efficiency factor were known, the K value was then determined to be 0.0531. Plugging the wing span and aspect ratio into the wing area and chord length equations resulted in a wing area of 95.9 square inches, a mean chord length of 4 inches, and a root chord length of 5.0929 inches. Since we initially planned on using balsa wood for the fuselage and foam plates for the wing, a weight of 20 grams was assumed. This assumed weight did not account for the additional weight of the battery. Because the weight of the battery was equal to 10 grams, the total estimated weight for the glider was determined to be 30 grams.

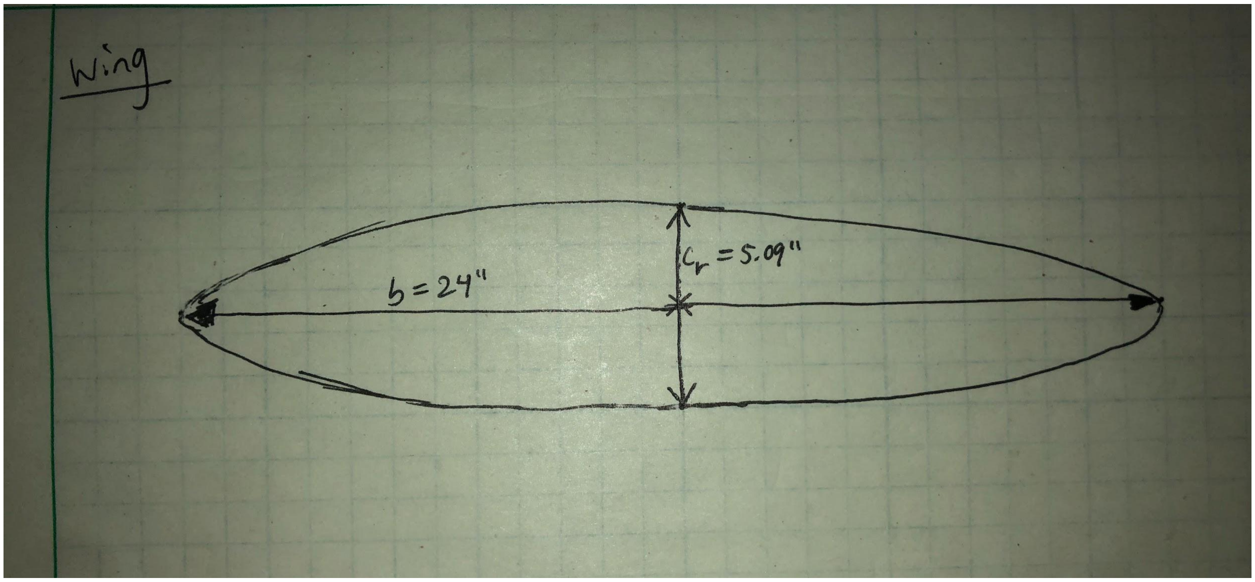


Figure 1: Elliptical wing dimensions

Table 1: Glider dimensions

Glider Dimension	**Value**
Airfoil	AG03
Wing Shape	Elliptical
Aspect Ratio (AR)	6
Wing Span (b)	24 inches
Oswald Efficiency Factor (e)	1
K	.0531
Wing Area (S)	95.9 inches^2
Mean Chord Length (c)	4 inches
Root Chord Length (c)	5.0929 inches
Assumed Weight (W) *no battery*	20 grams
Assumed Weight (W) *with battery*	30 grams

Estimated Aerodynamic Characteristics of the Wing

For the purpose of calculating the estimated aerodynamic properties of the wing, data was obtained specific to the AG03 airfoil providing a $(\frac{C_L}{C_D})_{MAX}$ ratio of 32.7 for an angle of attack of 4.75 degrees with a reynolds number of 50,000. Using the provided lift drag ratio, a glide angle was able to be obtained and was equal to 1.75 degrees. The calculated glide angle was then used in the lift and drag equations specific to glide flight, and resulted in a lift of 0.2942 newtons and a drag equal to 0.009 newtons. With lift and drag known, the coefficients for lift and drag were then solved for and determined to be 0.1402 and 0.0043, respectively.

In order to determine the lift curve slope of 0.0295 inverse degrees, the lift coefficient was divided by the corresponding angle of attack. Next, the values for parasitic drag, coefficients of lift and drag, as well as lift and drag were all made into functions of alpha. Due to designing for a maximum range flight, the minimum drag relationships for a propeller aircraft was used, as well as the drag polar equation to solve for the parasitic drag coefficient. With the previous mentioned parameters now all expressed as functions of alpha, three plots were constructed; lift coefficient vs. alpha, drag coefficient vs. alpha, and an overlay of both lift and drag vs. alpha all found in the appendix.

Verification of Estimations From Flight Tests

The final construction materials for the glider were changed from the original plans, however not significantly enough to make an impact on the original theoretical values. The final fuselage was constructed out of a rectangular basswood shaft, the wing was constructed out of foam board 1/8th of an inch thick, and they were connected using super glue. Additionally, when it came time to attach the battery, the battery was connected using duct tape as it would allow movement in the case that it was positioned incorrectly for ideal flight. The fuselage for the glider was constructed to be 8.5 inches with a weight of 9 grams. For the purpose of manufacturing accuracy, one solid wing was manufactured as opposed to two separate wings which had a weight of 22 grams. Although the battery proposed in the Soft Soar data weighed 10 grams, the battery used for the glider's payload had a minor weight increase at 11 grams. The total weight of the glider with no payload was 31 grams and with the payload, 42 grams. The single solid wing was glued to the top side of the fuselage, while the battery was attached to the bottom of the fuselage using duct tape. The angle of incidence of the wing was assumed to be 0 degrees because of the wing being mounted flat against the fuselage. All glider dimensions can be found below in Table 2, with a schematic of the aircraft shown below Table 2 in Figure 1

Table 2: Calculated glider dimensions

Glider Dimension	**Value**
Actual Glider Weight (W) *no battery*	31 grams
Actual Glider Weight (W) *with battery*	42 grams
CG measured from nose *no battery*	4.69 inches
CG measured from nose *with battery*	4.8 inches
Battery center location from nose	2.375 inches
Wing center location from nose	4.875 inches
Neutral Point from nose	1 inch

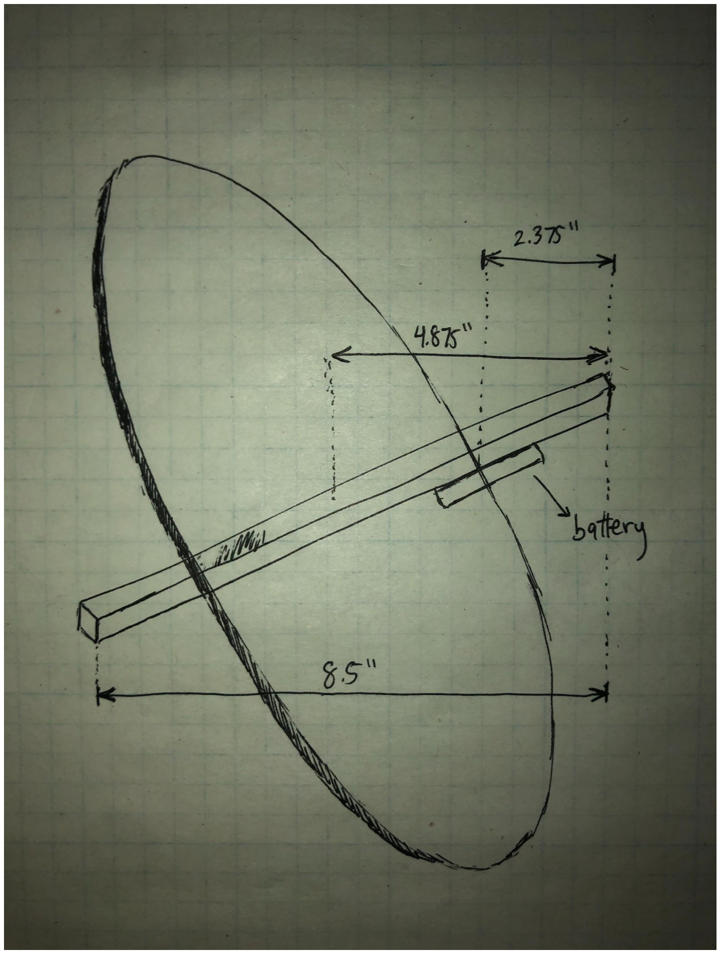


Figure 1: Schematic drawing of wing and fuselage

There were a total of over twenty test flights for the constructed glider however, a majority of those tests were inconclusive due to extremely restricted flight locations. Because of the aggressive and constant winds ever-present in Philadelphia combined with the absence of access to Drexel facilities, the only reasonable test location available to the group was inside the hallways of University Crossings. This location was not ideal as the location had tight parameters for flight, i.e. an 8 foot ceiling and a hallway width of less than 6 feet. Though the boundaries were tight, this location provided a haven from violent winds. The flight data from three different flights were recorded for use in the experimental calculations and resulted in an average of 38 feet traveled from a constant launch height of 6 feet. Using the average distance traveled paired with the average total flight time, an average velocity was able to be calculated as 11 feet per second.

Additionally, with such a large number of test flights in cramped conditions eventually led to the foamboard starting to take noticeable damage from the crash landings. Most of the damage was occurring to the wing tips with minor damage along the leading edge of the wing itself. The weakened foam board was causing the glider to veer off to the right for every flight, interfering with accurate data collection. The repeated collisions caused the wing tips to become more pliable than originally intended. Once the damage started impacting the flight performance of the glider, the wing tips were repaired and reinforced with duct tape; which solved the main performance issue. The video provided shows the glider in flight for one of the trials and it can be observed that the glider travels along a fairly straight path down the hallway, until it veers right slightly colliding with the wall and therefore shortening the ultimate horizontal distance traveled. Although the ultimate horizontal distance traveled was shortened, the measured distance was 34.6 feet.

Table 3: Experimental flight data

	Distance (ft.)	Flight Time (s)	Velocity (ft/s)	Glide Angle (deg)
Trial 1	40.26	3.53	11.93	9.25
Trial 2	39.14	3.27	11.53	9.035
Trial 3	34.6	2.95	9.54	8.625

Performance Analysis

Under the conditions of COVID-19, the constructed wing was not able to be flown on the Soft Soar RC aircraft. Instead the performance was analyzed utilizing matlab and data provided by the instructor.

The analysis was based on a flight involving the constructed wing at cruise, where weight was equal to lift and drag was equal to thrust. Expressions were found to calculate the coefficients of drag as function of velocity. Eventually these expressions were used to calculate drag as a function of velocity, and compared to the maximum thrust available graph shown below.

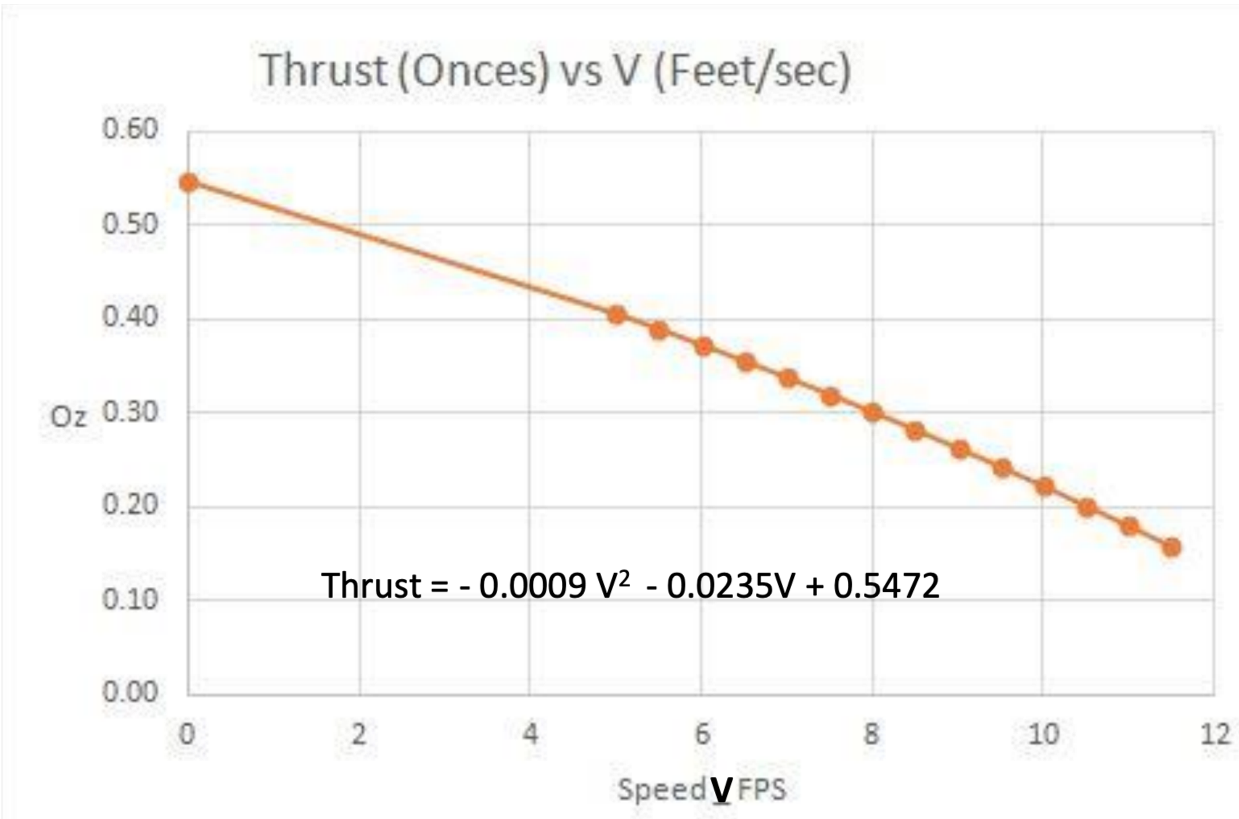


Figure 2: Maximum thrust available vs. Velocity.

The expression provided by the soft soar data was plotted in comparison to the drag expressions derived using the experimental data shown in the plot below.

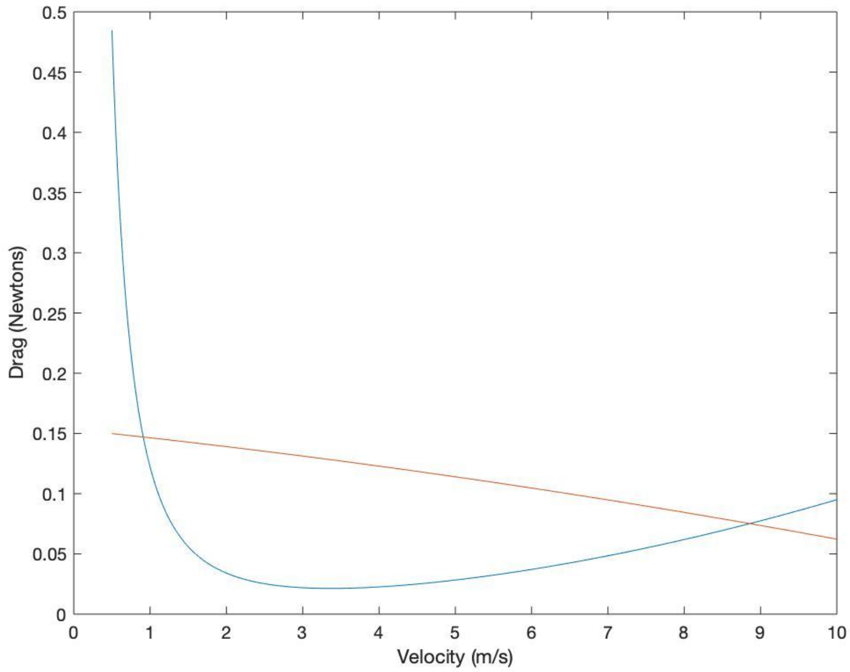


Figure 3: Resulting Drag of the wing compared to maximum thrust available, in newtons and meters per second

Cruise can be obtained when the thrust available is greater than the drag. The graph above indicates that cruise can be obtained between 1 and 9 m/s or 3.28 and 29.5 ft/s.

In order to calculate the maximum take off weight, additional weight was analytically added to the model, until the resulting drag forces exceed the maximum thrust available. When an additional 120 grams were added to the weight, the drag was no longer within the boundaries of the available thrust. The maximum weight was 160 grams total, including the weight of the body, battery, and wings.

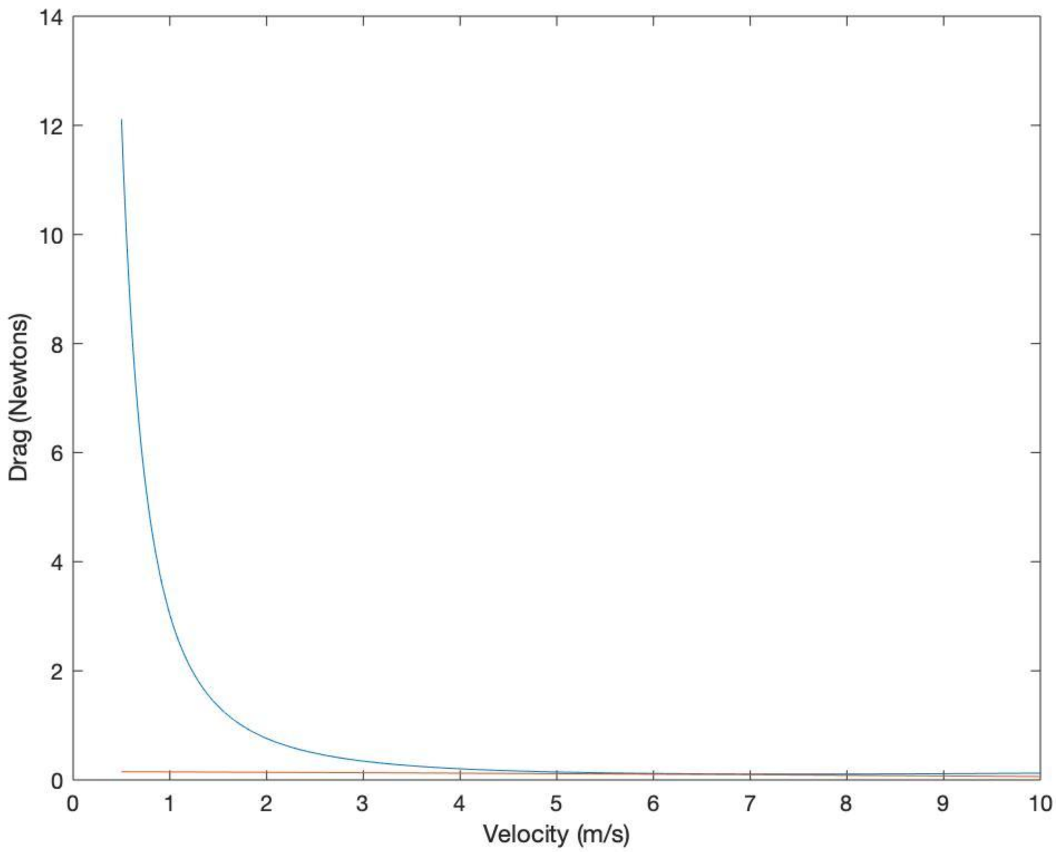


Figure 4: Adjusted drag and thrust graph (figure 3) but with an additional 120 g of weight

Takeoff and landing distances were determined using the two equations below, respectively. C_{Lmax} was determined to be 31.049 using the glide test data, and μ was set equal to 0.8 as the coefficient of friction for rubber wheels.

$$Sg = \frac{1.21}{g\rho C_{Lmax}}\left(\frac{W}{S}\right)\left(\frac{W}{T}\right)$$ Equation 1

$$S_l = \frac{1.3225W^2}{D+\mu_{br}W}$$ Equation 2

The takeoff range was found to be 3.5 inches and the landing distance was 11.97 inches.

The center of gravity was determined by taking the total moment around the fuselage tip and then dividing by the total force. Without the battery, the center of gravity was found to be 4.69 inches. Note that the front most edge of the wing was placed 2.38 inches from the leading edge of the fuselage. The battery was placed 2.38 inches away from the fuselage tip and resulted in a center of gravity at 4.08 inches. These placements were determined during the flight test. The wing and battery were moved until relatively stable flight was achieved.

The stability margin of this orientation was determined using the equation below.

$$SM = \frac{x_{np} - x_{cg}}{c} \qquad \text{Equation 3}$$

The “x” values were measured from the leading edge of the wing. The neutral point was equal to a quarter of the chord length, and the location of the center of gravity was equal to 1.71 inches. The stability margin was found to be -.1778, which is considered weakly unstable.

Appendix

MEM 425 Project

```
clear;clc;
```

List Known

```
   W=(20+10)*10^-3*9.81 %Weight including wieght of wing (Assumed to be 10 g), Newtons
W = 0.2943
   AR=6; %Assumed Aspect Ratio
   b=24*.0254 %Assumed Wingspan, m
b = 0.6096
   rho=1.225; %kg/m^3, density at sea level
   St=14*(.0254)^2 %Tail Area, m^2
St = 0.0090
   bt=6 * 0.254 %Tail Span, m
bt = 1.5240
   ARt=2.6; %Tail Aspect Ratio
   syms Vexp; T=(-0.0009*Vexp^2-0.0235*Vexp+0.5472); %Thrust in ounces, velocity in ft/sec
   mu=1.5111E-5; %Kinematic Viscocity at 25 degC, m^2/s
```

Part 1

```
   S=b^2/AR %Wing Area, m^2
S = 0.0619
   c=b/AR %mean Chord Length,m
c = 0.1016
   c_in=c*39.3701 %mean chord length, in
c_in = 4.0000
   cr=4*S/(pi*b) %root Chord Length, m
cr = 0.1294
   cr_in=cr*39.37 %root chord length, in
```

```
cr_in = 5.0929
```

Part 2

```
CoefficientRatio=32.7; %Max Cl/Cd
alpha=4.75; %angle of attack
Re=50000;
%These values are from our Airfoil Data, The Airfoil selected is AG03


GlideAngle=atand(1/CoefficientRatio)
```

```
GlideAngle = 1.7516
```

```
syms V;
V=double(solve(Re==V*c/mu,V)) %Velocity, m/s
```

```
V = 7.4365
```

```
V_in=V*3.28 %Velocity, inches
```

```
V_in = 24.3918
```

```
L=W*cosd(GlideAngle) %lift, newtons
```

```
L = 0.2942
```

```
D=W*sind(GlideAngle) %drag, newtons
```

```
D = 0.0090
```

```
Cd=2*D/(rho*V^2*S) % Coefficient of drag
```

```
Cd = 0.0043
```

```
Cl=2*L/(rho*V^2*S) %coefficient of lift
```

```
Cl = 0.1402
```

```
a=Cl/alpha %lift curve slope, 1/deg
```

```
a = 0.0295
```

```
clear Cl alpha ; syms Cl alpha Cdo;
```

```
e=1; %assumed for elliptical wings
K=1/(pi*AR*e)
```

K = 0.0531

```
Cl=a*alpha %Cl as a function of alpha
```

Cl =

$$\frac{8508434434132263\,\alpha}{2882303761517117440}$$

```
Cdo=vpa(solve(Cl==sqrt(Cdo/K),Cdo),4)
```

Cdo = $4.623\text{e-}5\,\alpha^2$

```
Cd=Cdo+K*Cl^2 %Cd as a function of alpha
```

Cd = $0.00009245889284459352097655937541276\,2\alpha^2$

```
L=(Cl*rho*V^2*S)/2 %lift as a function of alpha
```

L =

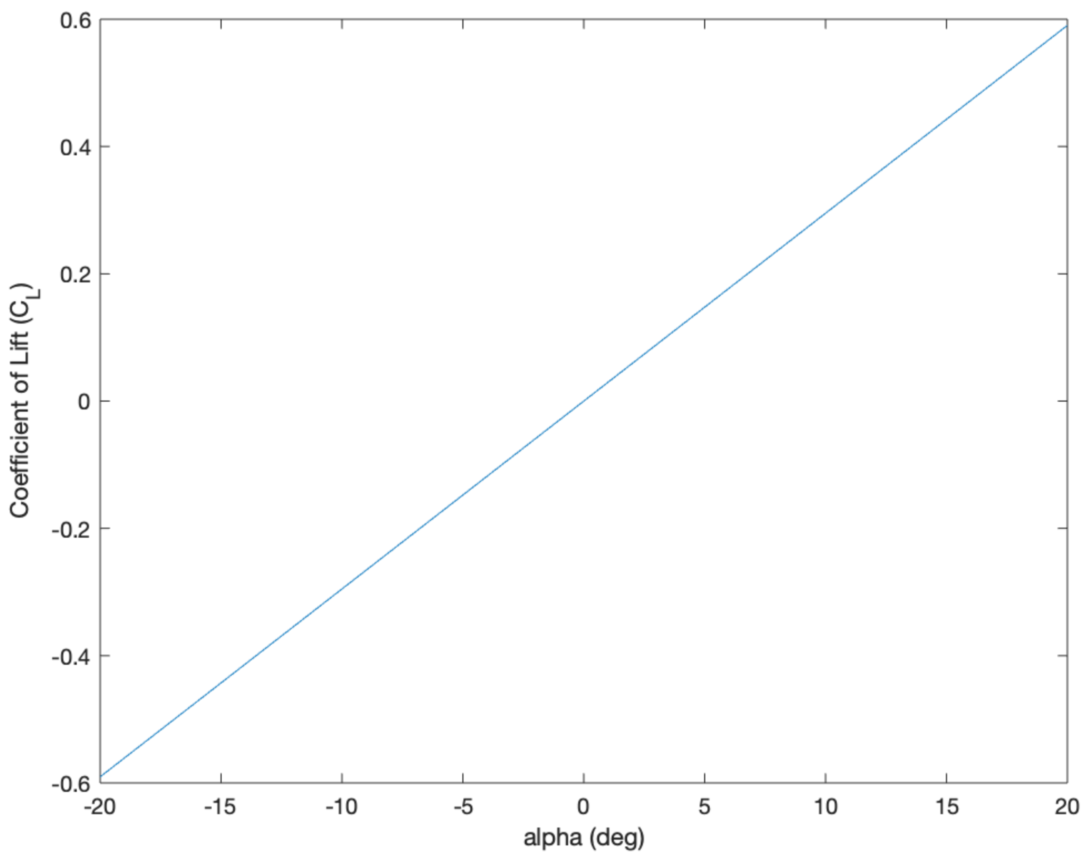
$$\frac{28962960757661037142537723829952515898895043575791\,\alpha}{467680523945888933825179146469210566289898413752320}$$

```
D=(Cd*rho*V^2*S)/2 %Drag as a function of alpha
```

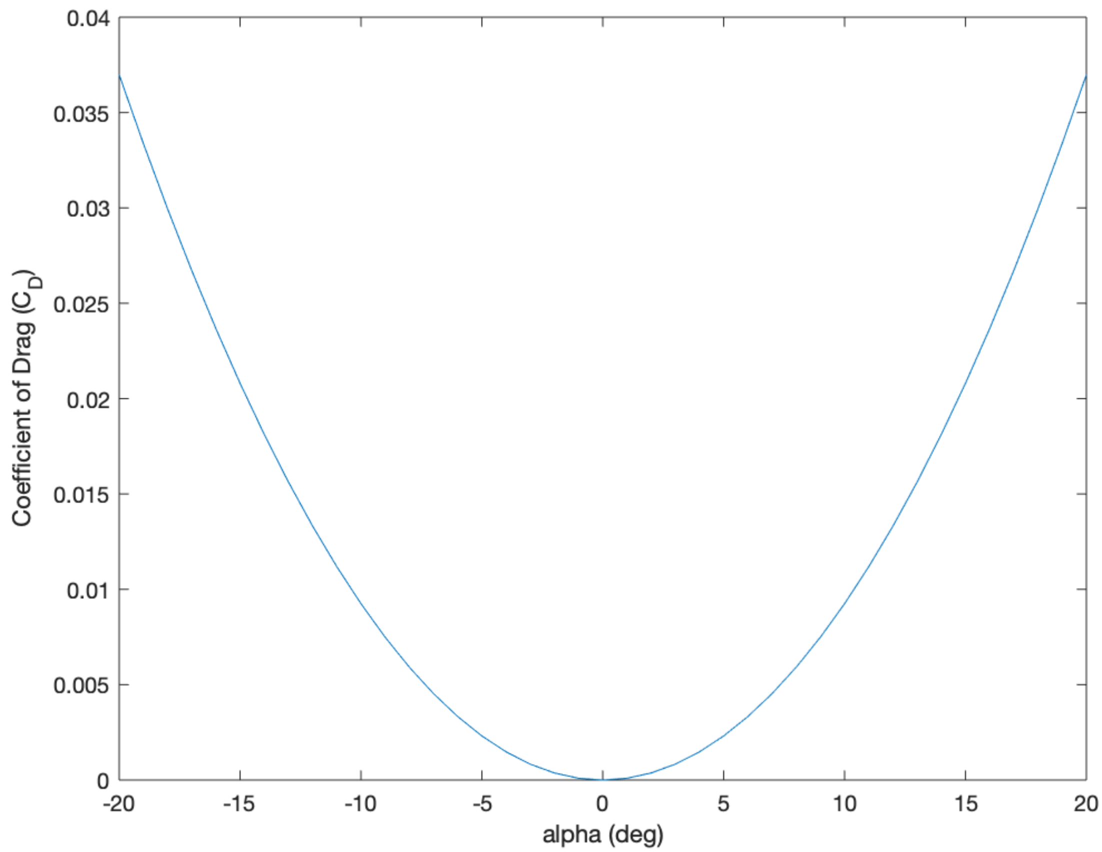
D = $0.00019396905586350261086224832756993\,\alpha^2$

```
%Assumed: Cruise flight

Cd=matlabFunction(Cd);
Cl=matlabFunction(Cl);
L=matlabFunction(L);
D=matlabFunction(D);
```

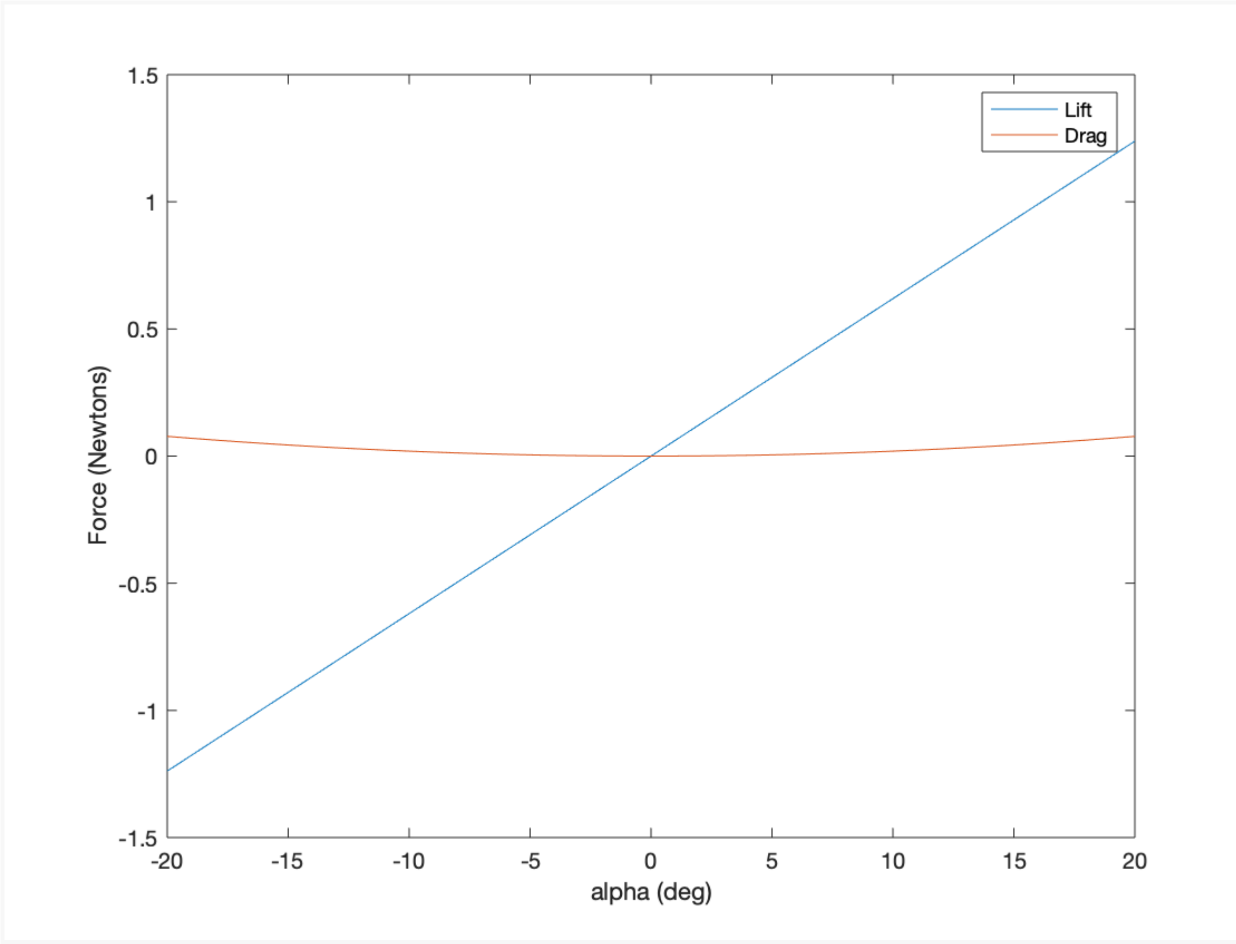
```
alpha=[-20:1:20];
plot(alpha,Cl(alpha))
xlabel('alpha (deg)')
ylabel('Coefficient of Lift (C_L)')
```




```
figure
plot(alpha,Cd(alpha))
xlabel('alpha (deg)')
ylabel('Coefficient of Drag (C_D)')
```




```
figure
plot(alpha,L(alpha))
hold on
plot(alpha,D(alpha))
xlabel('alpha (deg)')
ylabel('Force (Newtons)')
legend('Lift','Drag')
```



Part 3

```
Range=38; %Vertical Range, ft
Height=6; %Takeoff Height, ft
t=3.5; %time of flight, sec
Vv=Height/t %Vertical Velocity, ft/s
```

```
Vv = 1.7143
```

```
Vh=Range/t %Horizontal Velocity, ft/s
```

```
Vh = 10.8571
```

```
V=sqrt(Vv^2+Vh^2) %Net Velocity, ft/s
```

```
V = 10.9916
```

```
V=V*.3048 %Net Velocity, m/s
```

```
V = 3.3503
```

```
expGlideAngle=atand(Height/Range) %deg
```

```
expGlideAngle = 8.9726
```

```
expGlideRatio=1/tand(expGlideAngle) %Cl/Cd
```

```
expGlideRatio = 6.3333
```

```
L=W*cosd(expGlideAngle) %Lift, Newtons
```

```
L = 0.2907
```

```
D=W*sind(expGlideAngle) %Drag, Newtons
```

```
D = 0.0459
```

```
Cl=2*L/(rho*S*V^2) %Coefficient of Lift
```

```
Cl = 0.6827
```

```
clear Cdo; syms Cdo;
Cdo=Cl^2*K
```

```
Cdo = 0.0247
```

```
Cd=2*Cdo
```

```
Cd = 0.0495
```

Part 4

```
clear V; syms V
L=W; D=T; %Cruise Flight, Lift=Weight, Drag= Thrust
q=vpa(1/2*rho*V^2,4)
```

q = $0.6125\, V^2$

```
Cl=vpa(L/(q*S),4)
```

Cl =

$$\frac{7.758}{V^2}$$

```
Cd=vpa(Cdo+(K*Cl^2),4)
```

Cd =

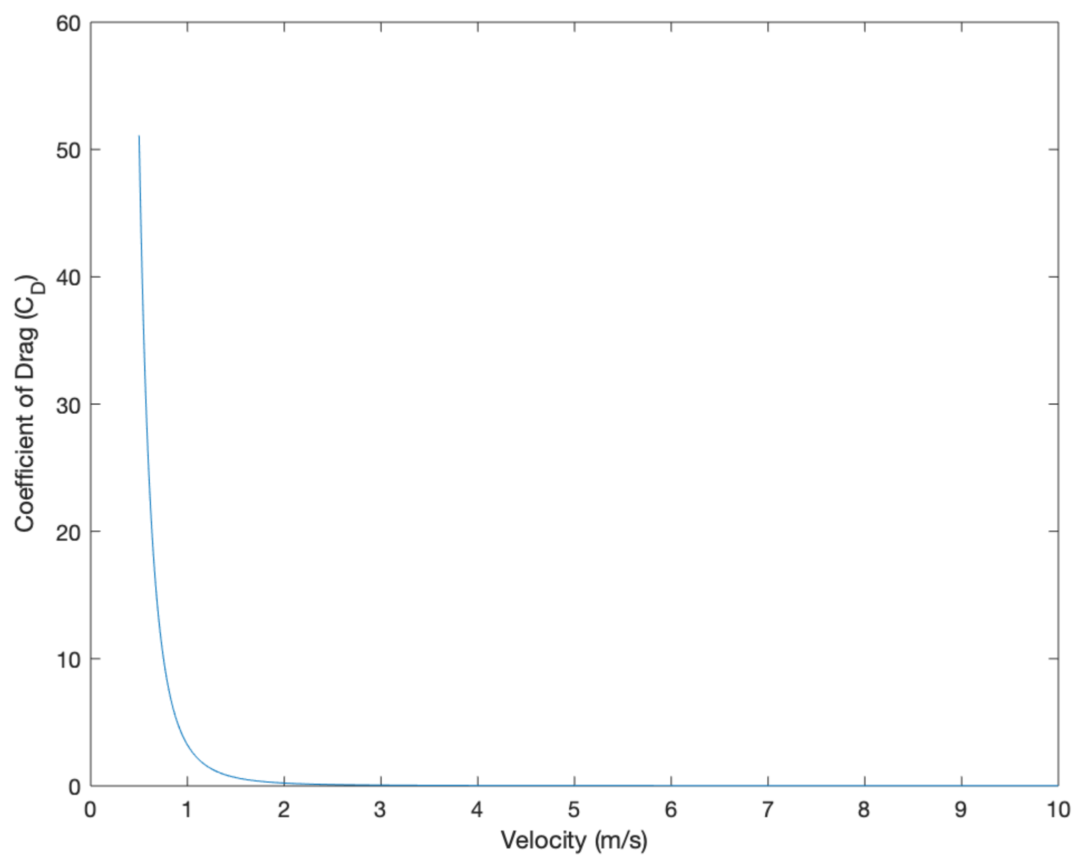
$$\frac{3.193}{V^4} + 0.02473$$

```
D=matlabFunction(q*S*Cd);
GlideRatio=vpa(Cl/Cd,4)
```

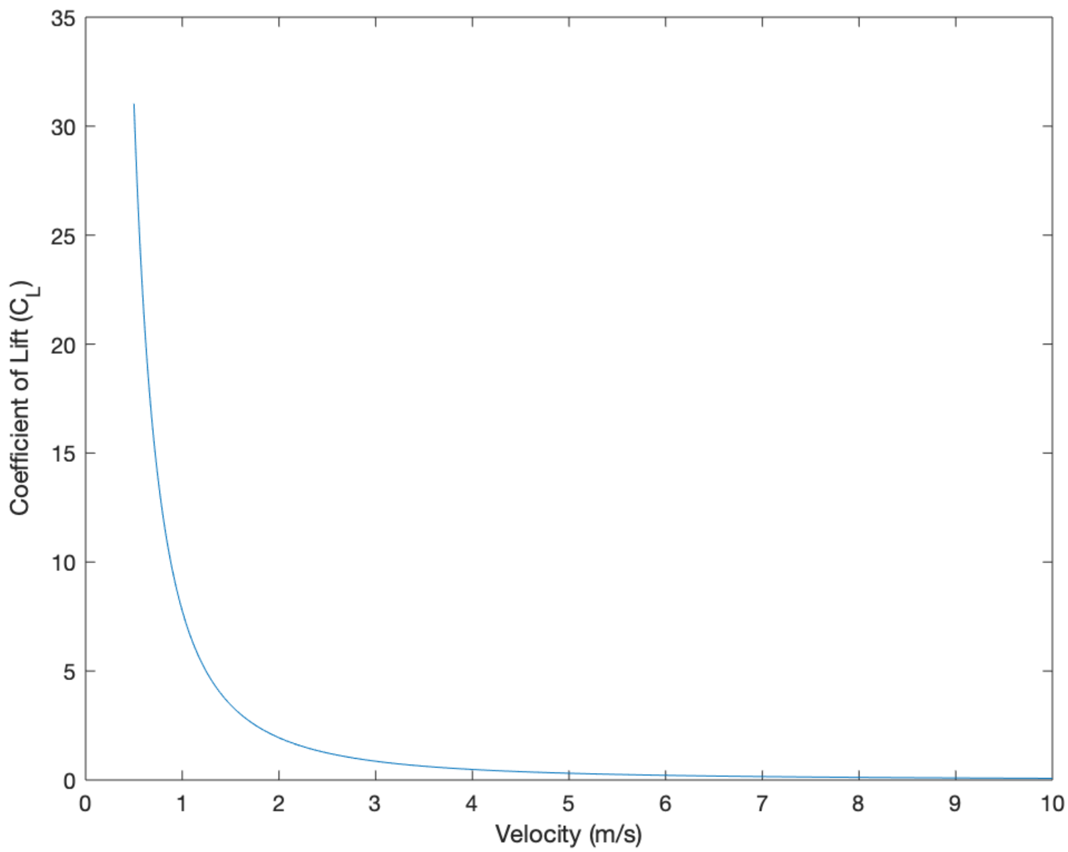
GlideRatio =

$$\frac{7.758}{V^2\,\left(\frac{3.193}{V^4} + 0.02473\right)}$$

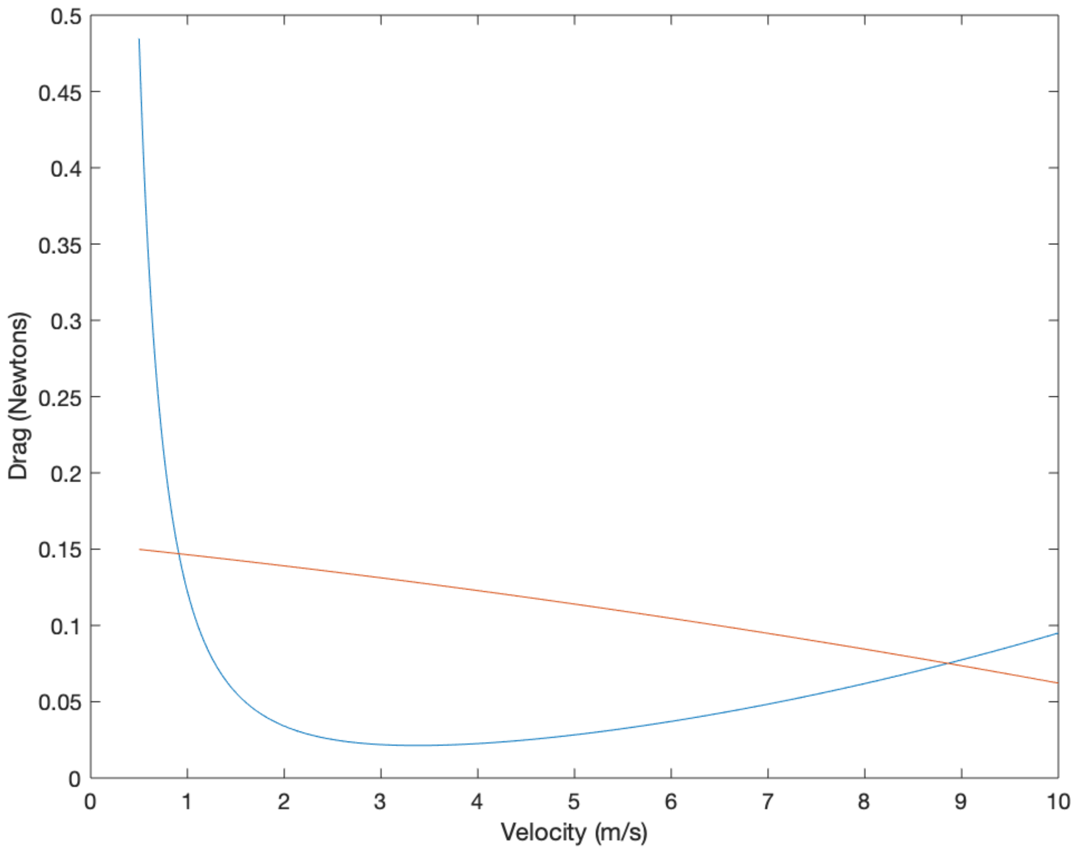
```
Cl=matlabFunction(Cl);Cd=matlabFunction(Cd);

V=[.5:.01:10];

figure
plot(V,Cd(V))
xlabel('Velocity (m/s)')
ylabel('Coefficient of Drag (C_D)')
```




```
figure

plot(V,Cl(V))

xlabel('Velocity (m/s)')

ylabel('Coefficient of Lift (C_L)')
```




```
figure
T=(-0.0009*Vexp^2-0.0235*Vexp+0.5472)*.28;
T=matlabFunction(T);
plot(V,D(V))
hold on
plot(V,T(V))
xlabel('Velocity (m/s)')
ylabel('Drag (Newtons)')
```




```
figure
clear V; syms V;

L=W+120E-3*9.81; D=T; %Cruise Flight, Lift=Weight, Drag= Thrust
q=vpa(1/2*rho*V^2,4)
```

q = $0.6125\ V^2$

```
Cl=vpa(L/(q*S),4)
```

Cl =

$$\frac{38.79}{V^2}$$

```
Cd=vpa(Cdo+(K*Cl^2),4)
```

```
Cd =
```

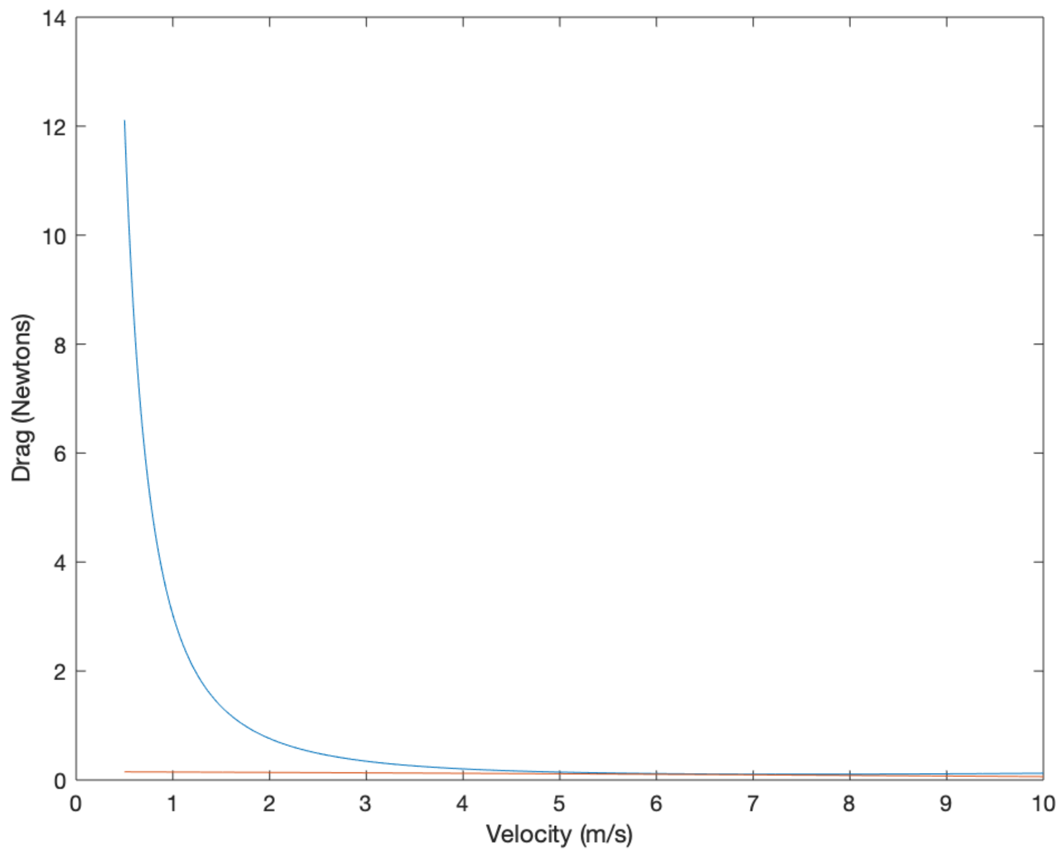
$$\frac{79.82}{V^4} + 0.02473$$

```
D=matlabFunction(q*S*Cd);
GlideRatio=vpa(Cl/Cd,4)
```

```
GlideRatio =
```

$$\frac{38.79}{V^2\,\left(\frac{79.82}{V^4} + 0.02473\right)}$$

```
T=(-0.0009*Vexp^2-0.0235*Vexp+0.5472)*.28;
T=matlabFunction(T);
V=[.5:.01:10];
plot(V,D(V))
hold on
plot(V,T(V))
xlabel('Velocity (m/s)')
ylabel('Drag (Newtons)')
```




```
Clmax=31.049; V=sqrt(Vv^2+Vh^2);

L=31.049*(1/2)*rho*V^2*S
L = 142.3043
addWeight=(L-W)/9.81
addWeight = 14.4760

Sg=1.21/(9.81*rho*Clmax)*(W/S)*(1/(T(V)/W))
Sg = 0.0899
Sg_in=Sg*39.07
```

```
Sg_in = 3.5123

   mu_br=.8;

   Sl=(1.3225*W^2)/(D(V)+mu_br*(W))

Sl = 0.3064

   Sl_in=Sl*39.07

Sl_in = 11.9712



   batterycenter=(1+1/16)+(2+10/16)/2 %inch

batterycenter = 2.3750

   batterycenter=batterycenter*.0254 %meters

batterycenter = 0.0603

   WingCenter=(5/2)+(2+3/8) %inch

WingCenter = 4.8750

   WingCenter=WingCenter*.0254 %meters

WingCenter = 0.1238

   ShaftCenter=8.5/2 %inches

ShaftCenter = 4.2500

   ShaftCenter=ShaftCenter*.0254 %meters

ShaftCenter = 0.1079

   batteryweight=11E-3*9.81 %N

batteryweight = 0.1079

   WingWeight=22E-3*9.81 %N

WingWeight = 0.2158

   shaftWeight=9E-3*9.81 %N

shaftWeight = 0.0883

Mtip=(batteryweight*batterycenter)+(WingWeight*WingCenter)+(shaftWeight*ShaftCent
er)

Mtip = 0.0428

   Ftot=WingWeight+batteryweight+shaftWeight
```

```
Ftot = 0.4120
   cg=Mtip/Ftot %meters
cg = 0.1038
   cgin=cg*39.37
cgin = 4.0863
   Mtip_nobattery=(WingWeight*WingCenter)+(shaftWeight*ShaftCenter)
Mtip_nobattery = 0.0363
   Ftot_nobattery=WingWeight+shaftWeight
Ftot_nobattery = 0.3041
   cg_nobattery=Mtip_nobattery/Ftot_nobattery
cg_nobattery = 0.1192
   cg_nobatteryin=cg_nobattery*39.37
cg_nobatteryin = 4.6935
   xcgin=cgin-(2+3/8)
xcgin = 1.7113
   xcg=xcgin/39.37
xcg = 0.0435
   xnp=c/4
xnp = 0.0254
   SM=(xnp-xcg)/c
SM = -0.1778

   q=vpa(1/2*rho*V^2,4)
q = 74.0

Mcg=((WingWeight*(cg-WingCenter))+(batteryweight*(cg-batterycenter))*(+shaftWeigh
t*(cg-ShaftCenter)))
Mcg = -0.0043
   Cm=Mcg/(q*S*c)
Cm = -0.0092883983507904671163711207356283
```