Design of an RC Aircraft

Seth Kitchen
Missouri University of Science and Technology

Isaac Foster
Missouri University of Science and Technology

UNIVERSITY OF MISSOURI SCIENCE AND TECHNOLOGY

AE2780: INTRODUCTION TO AEROSPACE DESIGN

Spirit in the Sky Report 2

Authors:
Bria BROILS
Isaac FOSTER
Seth KITCHEN
Zach MAZZONI

Professor:
Dr. Warner MEEKS

Department of Mechanical and Aerospace Engineering
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1 Introduction

1.1 Executive Summary

A remote controlled aircraft was designed, constructed, and tested for Aerospace Engineering 2780: Introduction to Aerospace Design on the campus of Missouri University of Science and Technology. The aircraft goals were to takeoff within 35 feet, be able to clear a 6 foot obstacle, and maintain flight for several laps around a gymnasium. Calculations were made for base aircraft dimensions and predictions for takeoff distance and distance to clear a 6 foot obstacle were made using a Runge-Kutta fourth order computer program. The designed aircraft was simulated in RealFlight simulation software to further improve dimensions such as elevator and rudder area and polyhedral angle. The aircraft was built and finally tested on May 1, 2017.

1.1.1 Step 1: Construct Fuselage With Landing Gear and Motor

This step consisted of assembling pre-made pieces into a basic fuselage. A 28" piece of balsa wood was used to form the basic fuselage. A motor, landing gear assembly, and tail skid were added along with velcro for attaching the electronics. The motor mount was used as the datum for calculation moments and flight parameters.

1.1.2 Step 2: Determine Available Thrust as a Function of Forward Speed

In this step, 3 different propeller sizes were tested to determine their thrust characteristics. These propellers were the 8"x6", the 9"x6", and the 9"x3.8". Two methods of testing were used. The first method was experimental, involving the use of a wind tunnel to record experimental data. The second method was analytical, using equations to predict the thrust characteristics. The 9"x6" propeller was selected for this aircraft.

1.1.3 Step 3: Develop Conceptual Wing and Tail Designs

During this step, the overall design goals for the aircraft were decided. A wing plan form area of 4 square ft and an aspect ratio of 6.25 were chosen resulting in a stall speed of 12.95 ft/s using a lift coefficient of 1.2. Using this information, the AG-35 airfoil was chosen.

1.1.4 Step 4: Estimate Aircraft Performance and Iterate Wing and Tail Designs as Needed

Take off performance for the aircraft was estimated in MATLAB using the RK4 method. Excel spreadsheets were used to calculate additional performance characteristics. The maximum angle of climb was found to be 20.5 degrees. Minimum take off distance was predicted to be 35.7 ft. No major design changes were made from steps 2 and 3.

1.1.5 Step 5: Prepare Initial Weight and Balance Report, Examine Aircraft Stability, Select Wing Location

Using basic data from a previous semester, the longitudinal center of gravity for the aircraft was determined. With those distances, the method of moments was used to find the center of gravity. Then, using an excel spreadsheet, the static margin was calculated for every reasonable wing leading edge distance. This was done to determine the predicted wing location that met the requirements for the aircraft.

1.1.6 Step 6: Develop and Test a Flight Simulation Model, Further Iterate Design as Needed and Prepare Aircraft Drawings

Using RealFlight Software, different types of wing and tail surfaces were tested to determine aircraft performance. Different wing loadings were tested to determine flight performance properties such as maximum aircraft velocity in straight level flight. The result of this data is found in Table 6.4. This data was used to determine the length of the wing center section and polyhedral angle. The polyhedral angle was chosen to be 25 degrees. Further tests were done, varying the length of center flat section, length and angle of polyhedral sections, areas of horizontal and vertical surfaces, and percent of elevator and rudder surfaces.

1.1.7 Step 7: Construct Aircraft Wing and Empennage and Complete Fuselage Construction

After the designs of the wing and tail surfaces had been finalized, AutoCAD was used to create template drawings for the wing and tail surfaces. These drawings were used as templates for construction. Following construction, MonoKote was used to cover the surface, using a sealing iron and heat gun to adhere the MonoKote to the constructed surfaces.

1.1.8 Step 8: Conduct Wind Tunnel Drag Testing of the Motor-Fuselage Combination, Update Estimated Aircraft Performance Characteristics

Once the fuselage was fully constructed, it was placed into a wind tunnel to determine its drag. This process was similar to step 2. The final location of the wing was also determined using the method of moments with the completed aircraft. The location selected was 11.5 inches from the datum to the leading edge without a payload, and 10.5 inches with a payload.

1.1.9 Step 9: Conduct Final Flight Testing

Once the aircraft was fully constructed and all lab tests completed, the aircraft was ready for flight testing. A performance prediction sheet was filled out and used to evaluate the aircraft. Following a pre-flight checklist, the aircraft was assembled and prepared for flight. During testing, the take off distance, take off distance with 4oz payload, distance to clear a 6ft obstacle, and max velocity in straight and level flight were tested and compared against the predicted values.

1.1.10 Step 10: Critique of Finished Aircraft

Flight testing demonstrated that the aircraft met and exceeded the design goals. The aircraft was stable yet maneuverable, able to navigate the testing facility with ease. The plane was able to take off in 21.4ft, considerably faster than predicted, and was able to clear the 6 ft obstacle at the predicted distance of 42ft. With the 4oz payload, the aircraft took off in 31.2 ft, again considerably faster than predicted. The max velocity in straight and level flight was shown to be 30.3 ft/sec. This was slower than predicted, but could be due to the very short testing distance. Overall, the aircraft met and exceeded expectations. It is formally recommended that Redienhcs Corporation market this aircraft to those seeking an airplane that is stable, maneuverable, and easy to control.

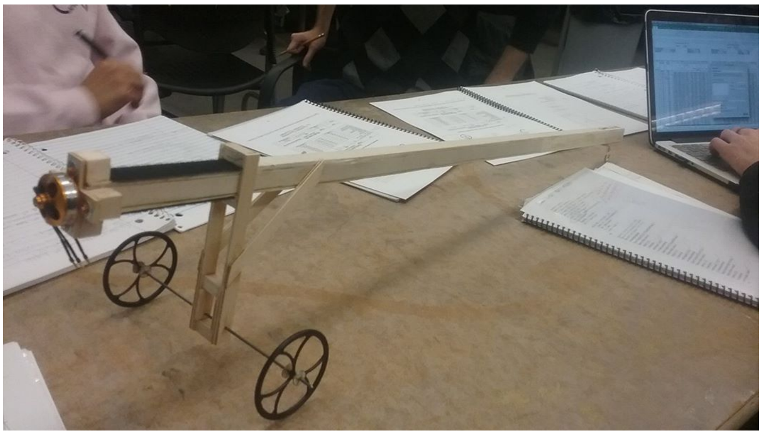
2 Step 1: Construct Fuselage, Attach Landing Gear and Motor

2.1 Summary:

This step was comprised of attaching the landing wheels, structural trusses, and motor to the fuselage. The landing struts were cut out of a stencil using a bandsaw, glued together, and attached to the fuselage. To attach the wheels, a 2-part epoxy was mixed and applied between the hubs and wheels after ensuring the wheels were equidistant from the fuselage. As the epoxy dried, the wheels were spun to ensure they did not fuse with the hubs. Oil was added during this process to ensure the wheels would freely rotate.

Other tasks accomplished included attaching a BP A2204 Brushless Outrunner Motor (83 W max) to the front of the fuselage and gluing on velcro strips where the battery will be mounted. Finally, a landing skid was glued to the end of the fuselage. The initial fuselage construction is shown in Figure 1.

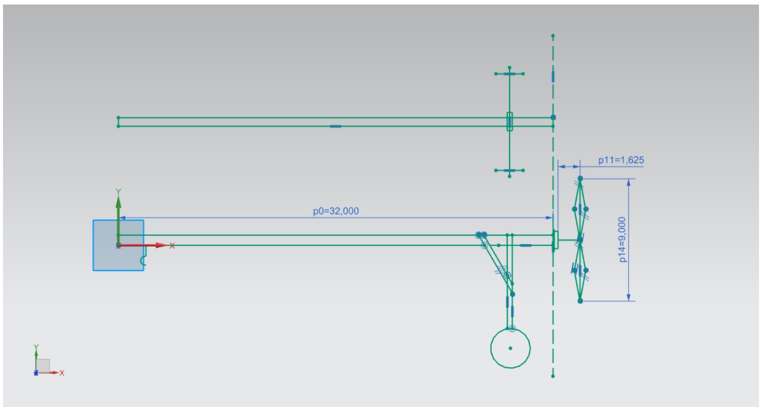
Figure 1: Initial Fuselage



2.2 Computerized Drawing

The computer drawing of the fuselage and landing gear is shown in Figure 2. The dotted line is the datum line where measurements (i.e. moments) were taken from.

Figure 2: Computerized Drawing of Fuselage and Landing Gear



2.3 Problems:

An error was made while assembling the landing gear assembly. The piece sliding onto the the fuselage was constructed too wide which caused a gap between the landing gear assembly and fuselage. It was decided that this would not cause a problem structurally to the aircraft, nor would it affect its aerodynamics enough to be relevant during flight.

3 Step 2: Determine Available Thrust as a Function of Forward Speed

3.1 Setup

To setup the experiment, an 8x6 propeller, BP A2204 brush-less Outrunner motor, battery pack, and speed controller were put inside the test section of an 18” X 18” subsonic wind tunnel. In addition, a pitot-tube was placed inside the test section to measure the pressure of the wind tunnel.

Four people were needed to perform the experiment. One person controlled airflow of the wind tunnel, one person recorded the thrust and velocity into Excel, one person measured the propeller thrust, and one person measured wind tunnel velocity.

3.2 Procedure

First, the person controlling the propeller let the propeller run for one minute to normalize the electrical properties. Second, the person controlling the wind tunnel increased the wind tunnel velocity in increments of 5 feet/second up to 50 feet/second. Third, the person measuring wind tunnel velocity called out velocities, informing the person controlling the wind tunnel whether to increase or decrease the speed. Finally, the person recording data entered the thrust and velocity said by the other two people.

3.3 Method 1

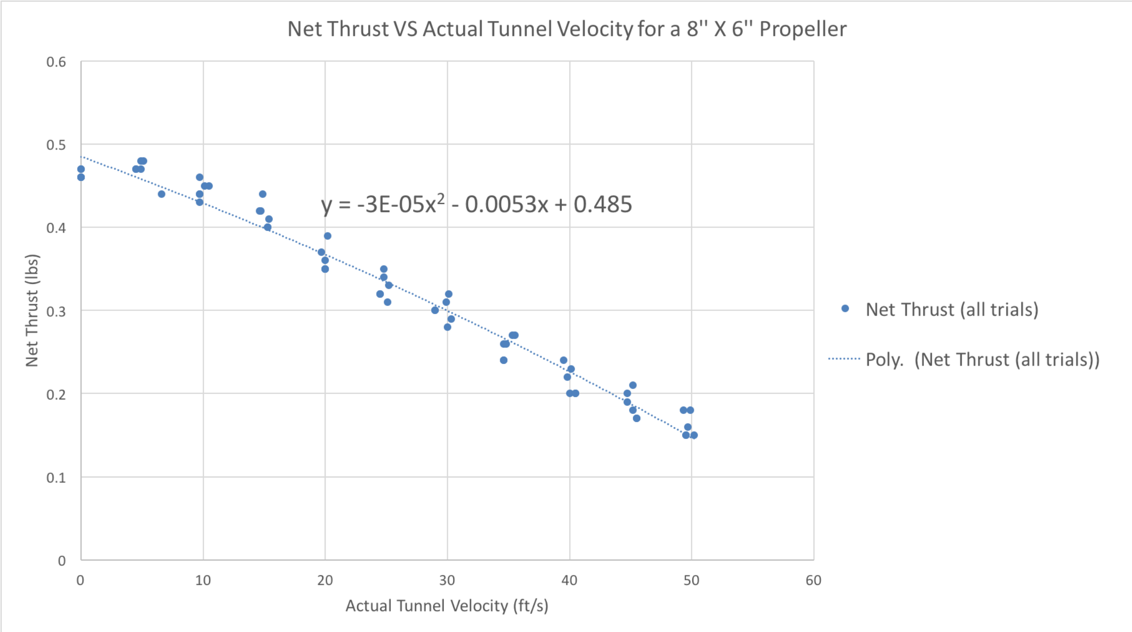
The first method of determining available thrust was to analyze the experimental data in excel. Based on wind tunnel velocity and measured propeller thrust, the net thrust (lb), thrust regression equation(lb), motor power(ft $\cdot$ lb/sec, Watts), battery power supplied (Watts), and system efficiency were calculated. The raw data collected from the wind tunnel was plotted as data points and a regression polynomial curve was added.

The equation came out to be

$$\text{Thrust Available} = (-3 \cdot 10^{-5}) \cdot (V^2) - (0.0053 \cdot V) + 0.485 \tag{1}$$

This was then used to determine the power available. The net thrust versus tunnel velocity for the propeller tested (8”x6”) is shown in Figure 3.

Figure 3: Net Thrust Vs. Tunnel Velocity for 8”x6”



3.4 Method 2

The available engine thrust is given by:

$$1 = \frac{T}{2T_0}\left(\left(\frac{V}{W_0}\right) + \sqrt{\left(\frac{V}{W_0}\right)^2 + \frac{4T}{T_0}}\right) \tag{2}$$

where W_0 is the induced velocity at the propeller face which is determined by,

$$W_0 = \sqrt{\frac{T_0}{2\rho A}} \tag{3}$$

Solving for W_0

$$W_0 = \sqrt{\frac{.46}{2 \cdot 2.0381 \cdot 10^{-3} \cdot \pi \cdot .333^2}} = 18\frac{\text{ft}}{\text{s}} \tag{4}$$

The use of Wolfram Alpha to plot the equation where y is thrust and x is velocity is shown in Figure 4.

Figure 4: Thrust Vs. Velocity Method 2

```
ContourPlot[{1 == 1.08696 y (x / 18 + Sqrt[x ^ 2 / 324 + 8.69565 y])}, {x, 0, 60}, {y, 0, 0.6}]
```

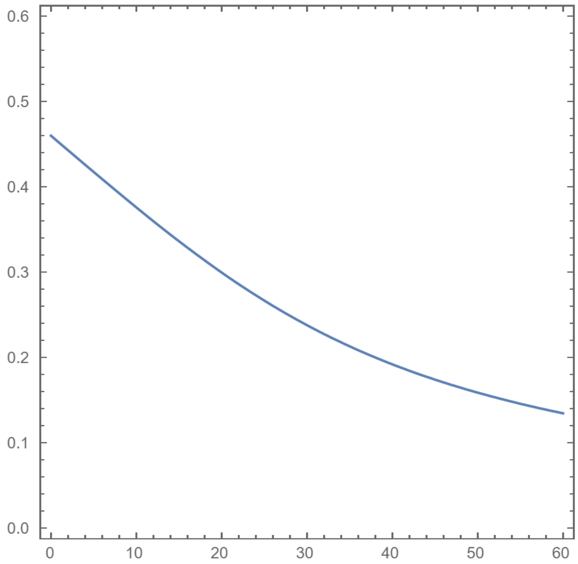


Figure 5: Thrust Vs. Velocity Method 2

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ContourPlot[{(25 y (x / 18 + Sqrt[x ^ 2 / 324 + (200 y) / 23])) / 23 == 1, y == -3 * 10 ^ {-5} x ^ 2 - .0053 x + .485}, {x, 0, 50}, {y, 0, .7}]
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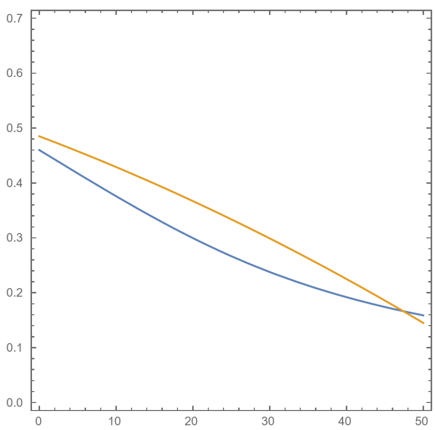
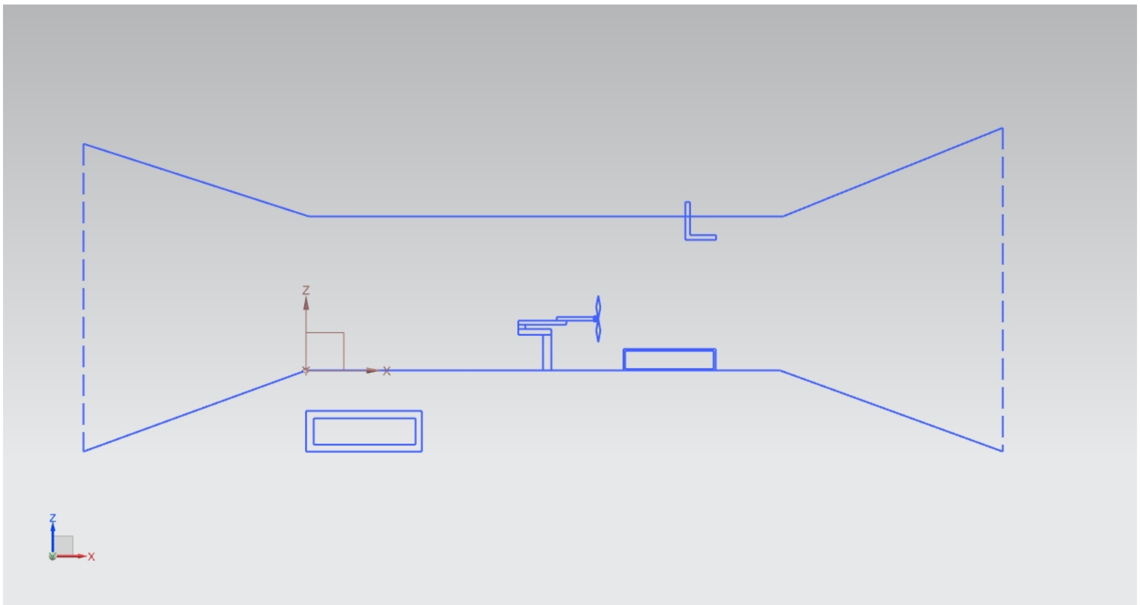


Method 1 and Method 2 yield similar results, however the dip in the curve of the graph created by Method 2 is a little steeper. The discrepancies in the graphs are due to the second method being theoretical. When the theoretical equation was created, assumptions were made which make the data less accurate. There is human error in Method 1 because data cannot be measured perfectly. There is a possibility that critical values such as air pressure fluctuated slightly during the experiment.

3.6 Computer Drawing

The experiments were carried out in a subsonic wind tunnel on the campus of Missouri S&T. The setup is shown in Figure 6. Note the propeller in the middle of the tunnel, the pitot tube sticking out of the top, the load cell readout in the bottom left, and the multi-function meter in the bottom right.

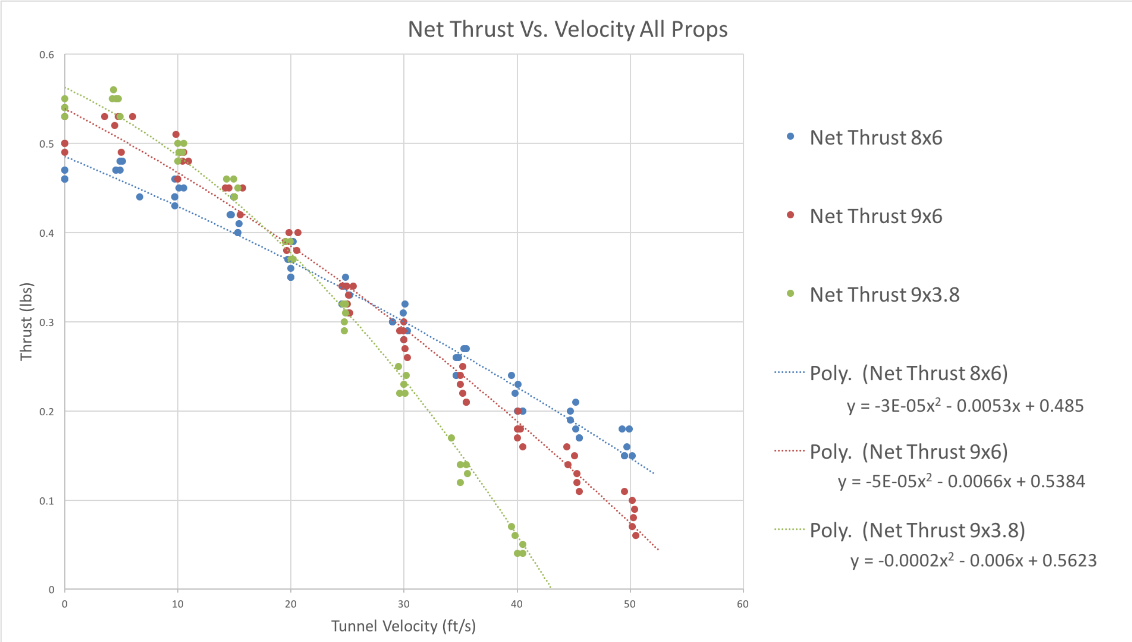
Figure 6: Test Setup



3.7 Propeller Selection:

Three different propellers were tested: 8"x6", 9"x6", 9"x3.8". By compiling the data from method 1 experimentation of the propellers, a graph comparing power available was created and is shown in Figure 7. Based off of its performance at low velocities and increased power and thrust compared to the other propellers,the 9"x6" propeller was selected.

Figure 7: Thrust of all Propellers



4 Step 3: Develop Conceptual Wing and Tail Design

4.1 Preliminary Design Efforts

The primary goal in selecting wing and tail designs was to maximize stability and low speed performance. To accomplish this, the aspect ratio was maximized along with wing surface area. Testing several different configurations, a wingspan of 60 inches and a wing surface area of 4 square feet were selected, allowing for a wing aspect ratio of 6.25.

For designing the tail surfaces, a percentage of wing surface area was chosen for the horizontal and vertical stabilizers. For the total area of horizontal surfaces, a percentage of 30 percent was chosen. For the total area of vertical surfaces, a percentage of 11 percent was chosen.

For the wing, the AG-35 airfoil was selected for its high maximum coefficient of lift and small drag bucket. In addition, it was selected for its relatively flat bottom which would make construction easier.

4.2 Weights

To estimated weights of the wing, horizontal and vertical surfaces as follows:

Wing:

$$W_{wing} = 1.40\frac{\text{oz}}{\text{ft}^2} \cdot 4\ \text{ft}^2 = 5.6\ \text{oz}\ (.35\ \text{lb}) \tag{5}$$

$$W_{horizontal} = 1.25\frac{\text{oz}}{\text{ft}^2} \cdot 1.2\ \text{ft}^2 = 1.5\ \text{oz}\ (.094\ \text{lb}) \tag{6}$$

$$W_{vertical} = 1.25\frac{\text{oz}}{\text{ft}^2} \cdot .454\ \text{ft}^2 = .5675\ \text{oz}\ (.0.036\ \text{lb}) \tag{7}$$

Estimated weights of all aircraft components are as follows:

Figure 8: Weights of all aircraft components

Component	Weight (oz.)	Weight (lb)
Propeller (9x6)	.32	.02
Li-Po Battery	1.64	.1025
Motor	.435	.0270
Radio Receiver	.725	.0453
Servos	.575	.035
Fuselage	2.805	.1753
Wing	5.6	.35
Horizontal Surface	1.5	.094
Vertical Surface	.5675	.035
TOTAL EST. WEIGHT	14.1675	.885

4.3 Infinite Wing

Data for the infinite wing is shown in the table below.

Figure 9: Infinite Wing Dimensions

Max Coefficient of Lift	.11667
Coefficient of Drag at Zero Lift	.013
Angle of Attack at Zero Lift	$-2\,^{\circ}$
Reynold's Number	128339.015

4.4 Finite Wing

The area for the finite wing was maximized at 4 square feet. By increasing the wing surface area, the capability for the wing was increased. In addition, by maximizing the wingspan, the aspect ratio was increased reducing the induced drag.

Figure 10: Finite Wing Dimensions

Wing Surface Area	4 (ft^2)
Wingspan	5 (ft)
Wing Chord	.8 (ft)
Wing Loading	.221 (lb per ft^2)
Lift Curve Slope	.0783
Oswald Efficiency Factor	.69

4.5 Horizontal Surfaces:

For the total area of horizontal surfaces, a percentage of 30 percent was chosen.

Figure 11: Horizontal Surface Dimensions

Horizontal Tail Surface Area	1.2 (ft^2)
Horizontal Tail Span	1.416 (ft)
Horizontal Tail Chord	.85 (ft)
Horizontal Tail Aspect Ratio	1.67

4.6 Vertical Surfaces:

For the total area of vertical surfaces, a percentage of 11 percent was chosen.

Figure 12: Vertical Surface Dimensions

Vertical Tail Surface Area	.454 (ft^2)
Vertical Tail Span	.583 (ft)
Vertical Tail Chord	.775 (ft)
Vertical Tail Aspect Ratio	.75

4.7 Airfoil Graphs

The AG-25 airfoil was selected for its high coefficient of lift curve and low drag. The shape is listed in Figure 13 and its lift curve and drag bucket are listed in Figure 14. The chord length for the implemented AG-25 airfoil is 9.6 inches.

Figure 13: AG-35 Airfoil

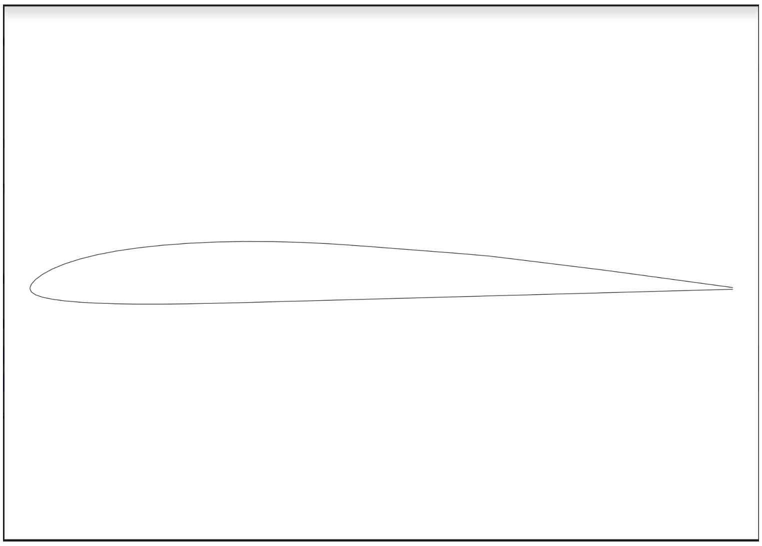
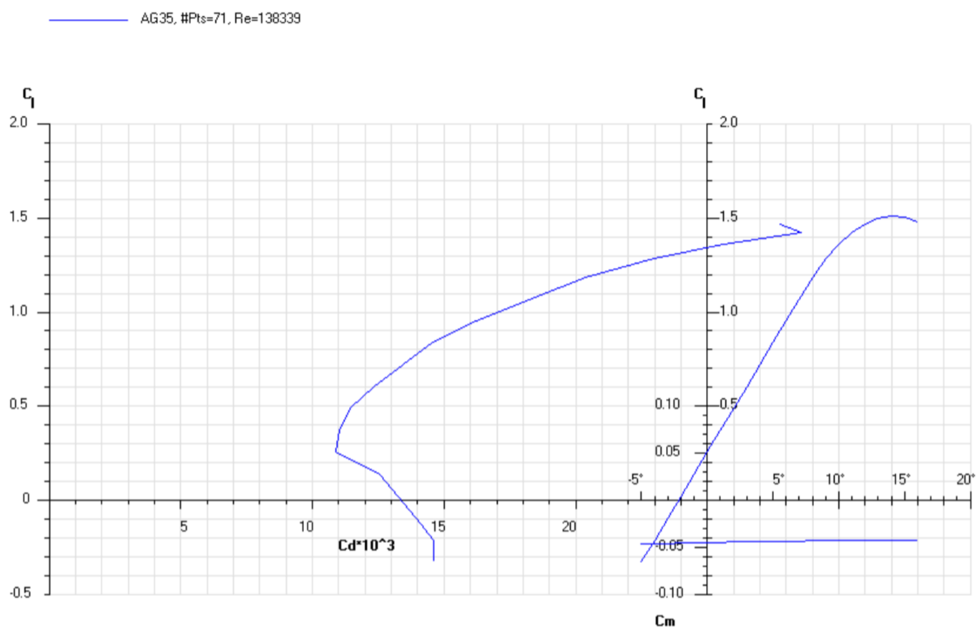


Figure 14: Eppler Plot for AG-35



5 Step 4: Estimate Aircraft Performance and Iterate Conceptual Wing and Tail Design

5.1 Procedure

To fill out the spreadsheet for performance calculation, the lift slope of the wing airfoil and the lift slope of the finite wing were calculated by hand. The lift slope of the finite wing and the horizontal surface were determined by the aspect ratio. From those calculations, the Oswald efficiency factor for the wing could be solved.

The design goal was an aircraft that possesses the ability to fly at lower speeds and maintain control. Using a lower stall velocity, along with a larger wing plan form area allows the aircraft to accomplish this goal. However, by increasing the area of the wings and control surfaces, the weight of the aircraft also increases. This can have a detrimental effect on lift. To overcome this increase in weight, a propeller that could output a high static thrust and larger thrusts at slower speeds was needed. By looking at the data, the 9" X 6" propeller offered the thrust profile that met this need.

No major design changes were made from steps 2 and 3. A comprehensive sheet of the performance data for the 9x6 can be found in appendices.

5.2 Performance Data

Thrust, power, velocity, and distance data were analyzed to estimate performance. Figure 15 contains data calculated from the Runge-Kutta code and Excel performance spreadsheets. The Thrust Available and Thrust Required graphs for the 9" X 6" are listed on the same graph in Figure 16. The Power Available and Power Required graphs for the 9" X 6" Propeller are listed on the same graph in Figure 17. The Hodograph for the 9" X 6" is listed in Figure 18.

Figure 15: Runge-Kutta Calculated Performance Data

Max Velocity in Level Flight (ft/s)	36.5
Power Required off Stall Velocity (ft*lbs/s)	0.16409
Power Available off Stall Velocity (ft*lbs/s)	5.23155
Aircraft Rate of Climb for Max Climb Angle (ft/s)	5.5111
Aircraft Velocity for Max Climb Angle (ft/s)	14.7683
Minimum Takeoff Distance of the Aircraft (ft)	35.79
Takeoff Distance with 4oz Payload (ft)	75.61
Distance to Clear 6 foot Obstacle (ft)	51.87

Figure 16: Thrust vs Velocity for the 9” X 6”

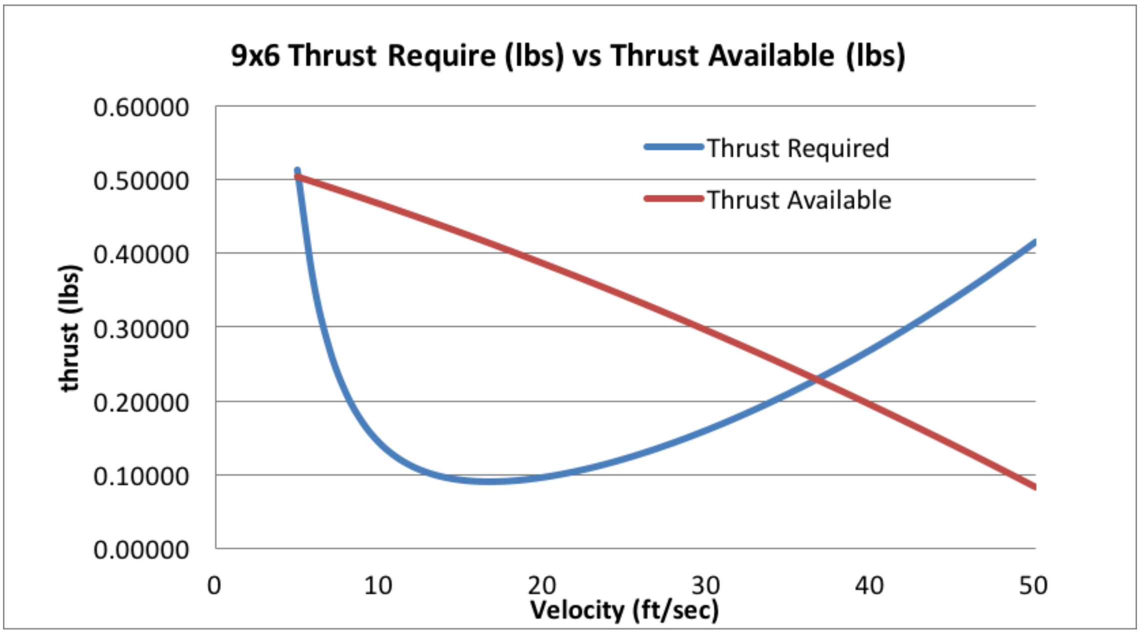


Figure 17: Power vs Velocity for the 9” X 6” Propeller

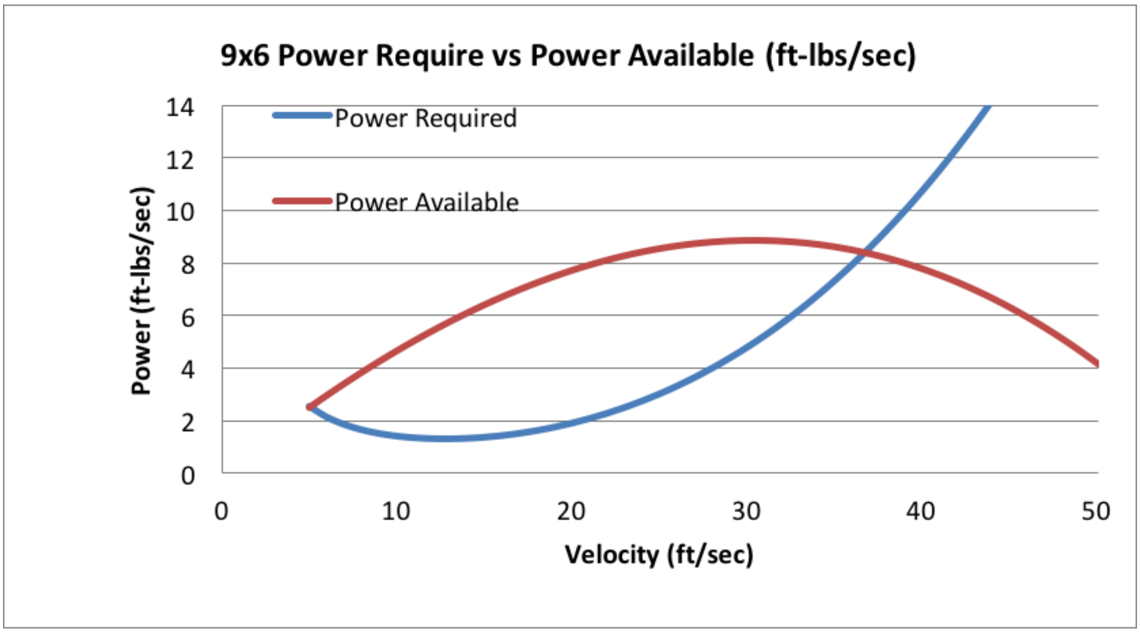
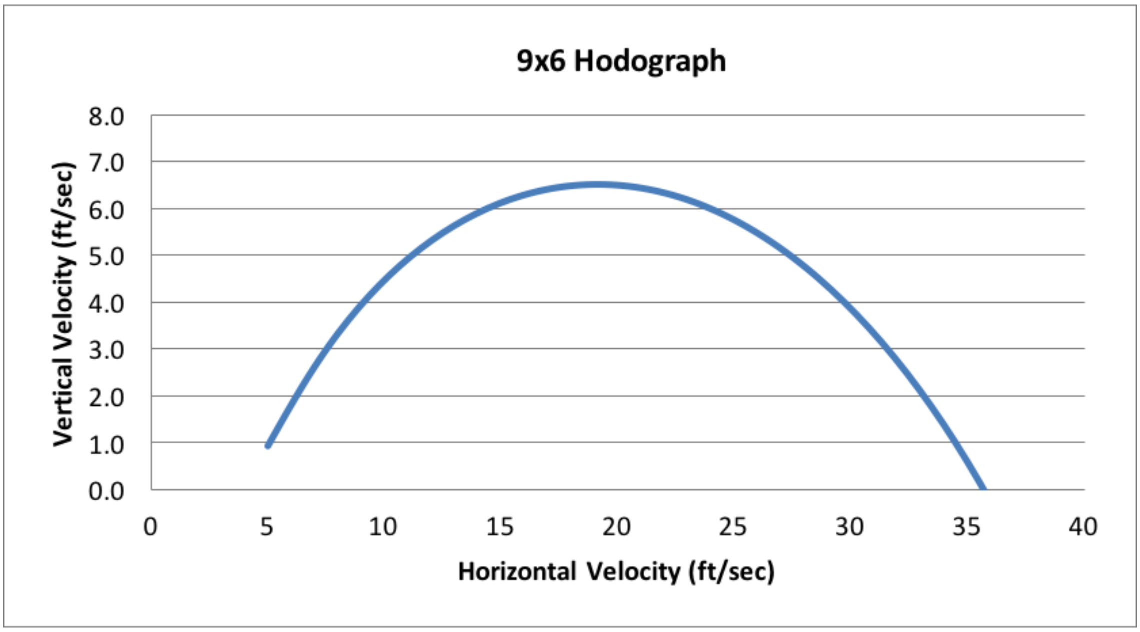


Figure 18: Hodograph Diagram



6 Step 5: Prepare Initial Weight and Balance Report, Examine Stability of Wing-Fuselage Combination and Select Wing Location

6.1 Procedure

The purpose of this section is to decide where to place the wing on the fuselage that makes the air craft stable. To obtain the best stable location of the wing, the longitudinal distance of the aircraft's center of gravity and the neutral point from the datum line was determined. The neutral point is where the center of gravity is to make the stability neutral. The position of the neutral point relative to the center of gravity is important because if the neutral point is in front of the center of gravity the plane will be unstable. The static margin is the distance between the center of gravity and the neutral point, resulting in the aircraft having a large positive static margin.

A previous semesters aircraft was used to get basic measurements to determine the aircraft longitudinal center of gravity. With those distances, the method of moments was used to find the center of gravity.

$$X_{cg} = \frac{\Sigma_{fixed}\text{moments} + \Sigma_{movable}\text{moments}}{\Sigma_{fixed}\text{weights} + \Sigma_{movable}\text{weights}} \tag{8}$$

With the use of the weight and balance sheet, the static margin was calculated for every reasonable wing leading edge distance to see if the different locations met the requirements for the aircraft. The optimal location of the wing is when the wing's quarter chord matches with the wing's center of gravity.

Figure 19: Weight and Balance

Weight and Balance	*Values*
Distance from datum to wing leading edge (in)	11
Distance from datum to wing center of gravity (in)	14.84
Moment of wing about the datum (in-oz)	83.10
Total Moment about datum (in-oz)	190.61
Location of aircraft cg about datum (in)	13.45
Wing 1/4 chord from datum (in)	13.40
Location of Neutral Point (in)	14.55
Static Margin (in)	1.09
Static Margin as percent of wing chord	11%

Figure 20: Center of Gravity for a 4oz Payload Using a Moment Table

CG Location from Datum (in)	SM (in)	SM as percent of wing chord
9.214	−2.347	−0.244
9.478	−2.131	−0.222
9.742	−1.915	−0.200
10.036	−1.700	−0.177
10.270	−1.434	−0.155
10.535	−1.258	−0.132
10.799	−1.052	−0.110
11.063	−0.837	−0.087
11.327	−0.621	−0.065
11.591	−0.426	−0.042
11.866	−0.189	−0.020
12.120	0.026	0.003
12.364	0.242	0.025
12.645	0.458	0.048
12.913	0.674	0.070
13.177	0.899	0.093
13.441	1.105	0.225
13.705	1.321	0.138
13.969	1.537	0.160
14.234	1.752	0.182
14.498	1.968	0.205
14.762	2.184	0.227
15.005	2.400	0.250

7 Step 6: Develop Flight Simulation Model, Make Design Changes as Needed and Prepare Aircraft Drawings

7.1 Procedure

The plane was simulated in RealFlight to tune pilot specific dimensions. For instance, a slight change in polyhedral angle may make the plane feel smoother on turns to the pilot. With the use of RealFlight Software, different types of wing and tail surfaces can be estimated and tested to see which design will work best for the desired performance.Before simulating, the data sheet in Appendix D in the lab booklet needed to be filled out. The recorded data is found in Table 7.1.1-7.1.6. Using the recorded data in RealFlight basic aircraft wing loading was tested. The wing loading was increased to determine the performance flight properties, such as maximum aircraft velocity in straight level flight, take off distance and velocity, and distance from takeoff to clear six foot obstacle. The result of this data is found in Table 7.2.1. To test for better performance, the aspect ratio, length of center flat section, length and angle of polyhedral sections, areas of horizontal and vertical surfaces, and percent of elevator and rudder surfaces could be adjusted.

Table 7.1.1: Vertical Tail Dimensions

Vertical Tail	*Values*
Chord of the vertical tail at the root (in)	9.33
Chord of the vertical tail at the tip (in)	9.33
Leading edge sweep (degrees)	0
Estimated weight of the vertical tail (oz)	0.55
Wing length (in)	7

Table 7.1.2: Rudder Dimensions

Rudder	*Values*
Length (in)	7
Percent of chord at root (percent)	50
Percent of chord at tip (percent)	50

Table 7.1.3: Horizontal Tail Dimensions

Horizontal Tail	$Values$
Chord at the root (in)	10.20
Chord at the tip (in)	10.20
Leading edge sweep (degrees)	0
Weight - half the span of the horizontal tail (oz)	0.753
Wing length - half the span of the horizontal tail (in)	8.50

Table: 7.1.4: Elevator Dimensions

Elevator	$Values$
Length (in)	17
Percent of chord at root (percent)	30
Percent of chord at tip (percent)	30

Table 7.1.5: Main Wing Dimensions

Main Wing	$Values$
Airfoil name at root)	$AG35$
Airfoil name at tip	$AG35$
Chord at the root (in)	9.60
Chord at tip (in)	9.60
Dihedral (degrees)	15
Incidence at root (degrees)	0
Weight - half the span of the wing (oz)	2.8
Wing length - half the span of the wing (in)	30

Table 7.1.6: Fuselage Dimensions

Fuselage	$Values$
Weight - with landing gear, tail skid, motor, and propeller (oz)	3.56
Propeller diameter (in)	9
Propeller pitch (in)	6

Table 7.2.1: Data From Flight Simulation

Weight 1/2 wing (oz)	Wing Loading (oz/ft^2)	Takeoff Distance (feet)	Takeoff Velocity (mph)	Takeoff Velocity (ft/sec)	Distance to clear a 6 foot obstacle	Maximum Velocity Straight & Level Flight (mph)	Maximum Velocity Straight & Level Flight (ft/sec)
2.8	2.799	20	15	22	60	25	36.67
		23	13	19.1	56		
		24	13	19.1	69		
		31	15	22	68		
		28	14	20.5	66		
	AVG	25.2	14	20.54	63.8	25	36.67
4.3	3.391	41	16	23.47	87	25	36.67
		43	16	23.47	105		
		47	16	23.47	100		
		41	16	23.47	92		
		40	15	22	94		
	AVG	42.4	15.8	23.17	95.6	25	36.67
5.8	3.983	48	15	22	95	25	36.67
		63	17	24.93	111		
		51	15	22	109		
		63	17	24.93	119		
		59	16	23.47	105		
	AVG	56.8	16	23.47	107.8	25	36.67
7.3	4.575	69	17	24.93	105	24	35.2
		84	17	24.93	137		
		69	16	23.47	119		
		78	16	23.47	121		
		66	15	22	121		
	AVG	73.2	16.2	23.76	120.6	24	35.2
8.8	5.167	91	16	23.47	139	24	35.2
		91	16	23.47	144		
		93	16	23.47	134		
	AVG	91.7	16	23.47	139	24	35.2

Below are the comments made by the designated pilot about the varying performance parameters.

Changing the Rudder:

- 45 percent of the vertical surface- The plane handles well, flies smooth when the flying straight. When the plane makes hard turns it does not react well.
- 50 percent of the vertical surface- When turning the plane tries to over corrects itself to much.
- 55 percent of the vertical surface- During the turns the plane starts to slip, and jerks back and forth.

Changing the Elevator:

- 20 percent of the horizontal surface- The plane was fairly responsive and did not over correct itself too much.
- 25 percent of the horizontal surface- The plane is more responsive and did not over correct itself.
- 30 percent of the horizontal surface- The place can handle more aggressive turns. Feels more stable in level flight, but slips in turns.

Changing the center of gravity wing location:

§ Original location is 6 inches from leading edge.

- 5 inches from leading edge- The plane moves to much making it easy to flip.
- 6 inches from leading edge- The plane climbs fast at level flight.
- 7 inches from leading edge- When in level flight the plane does not dive to much. Without touching the controls the plane increases altitude.
- 8 inches from leading edge- When in straight level flight the plane dives.
- 9 inches from leading edge- The plane dives more than most locations. The plane resists in turns.

Change wing dihedral angle with decided wing center of gravity:

§ The wing center of gravity is fixed at 6 inches

- -10 degrees- The plane is almost impossible to control in turns. The plane starts diving the as soon as the plane rolls.

- -5 degrees- The plane still dives hard in turns, but is better than -10 degrees.
- 0 degrees- The plane does not roll, just yaws.
- 5 degrees- This angel is better than the previous ones, but the plane is still difficult to control.
- 10 degrees- There is less slip when rolling.

Change length of center flat section and polyhedral section:

§ The Polyhedral is fixed at 15 degrees

- 3 inches flat section/ 27 inches polyhedral section- The plane flew jittery and over responsive trying to over correct itself.
- 6 inches flat section/ 24 inches polyhedral section- The plane was less over corrective, but was prone to nose dive.
- 9 inches flat section/ 21 inches polyhedral section- When the plane increased in altitude the stability decreased. There was difficulty trying to pitch the nose up. The plane did well rolling.
- 12 inches flat section/ 18 inches polyhedral section- The plane rolled with ease.

Change angle of polyhedral with decided length of flat center section and polyhedral section:

§ The length of the flat section is 12 inches and the polyhedral length is 18 inches.

- 20 degrees- The plane starts to dive in turns.
- 25 degrees- The plane was more responsive, and did not try to over correct itself as much.

From adjusting the different aspects of the aircraft surfaces, a conclusion was made for the final aircraft design.

- Rudder percentage: 45 percent
- Elevator percentage: 25 percent
- Location of wing center of gravity: 6 inches
- Length of flat section of the wing: 12 inches
- Length of polyhedral section of the wing: 18 inches
- Angle of polyhedral section of the wing: 25 degrees

8 Step 7: Construct Aircraft Wing & Empennage and Complete the Fuselage

The construction of the wing was separated into three steps. The first step consisted of preparing templates to be used in the construction of the wing. A CAD drawing of the wing sections was created to serve as an assembly blueprint for step 2. The wing center drawing is in Figure 24. Another CAD drawing of the AG-35 airfoil was created and glued onto thin strips of balsa wood and then cut using a band saw. These cut sections served as the ribs for the wing.

The second step involved constructing and assembling the wing frame. The wing frame was assembled in three parts: the two polyhedral sections and the center section. The method of construction was the same for all three. Using the assembly blueprint from step 1, the ribs were aligned with the trailing edge and glued together using two aluminium blocks to hold the ribs steady. Then, stiffeners were added to the ribs to ensure they did not separate from the trailing edge. After this, the leading edge was glued onto the ribs. After the polyhedral sections and the center section were constructed, they were glued together with thick plywood cut to the selected polyhedral angle. Shims were added to ensure a good connection between the sections. Finally, the wing was sanded down.

The third and final part involved covering the wing and heat shrinking the covering. The covering was carefully applied and glued onto the wing using a heating iron. After the covering was applied, a heat gun was used to shrink the fabric. Finally, the wing was checked for warping and was corrected by manually by bending the wing and using the heat gun to shrink the fabric.

After construction, the wing was balanced by adding metal washers to the end of the wing. This completed the construction of the wing.

Other computer animated drawings of the rudder, vertical stabilizer and horizontal surface are added below in figure 21, 22, and 23.

Figure 21: Rudder

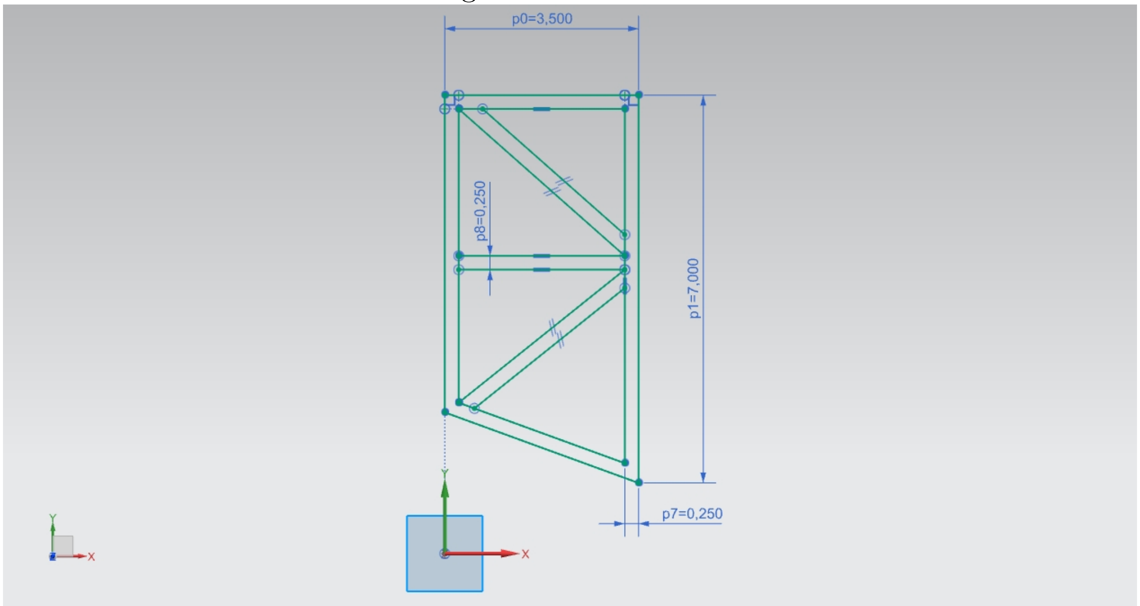


Figure 22: Vertical Stabilizer

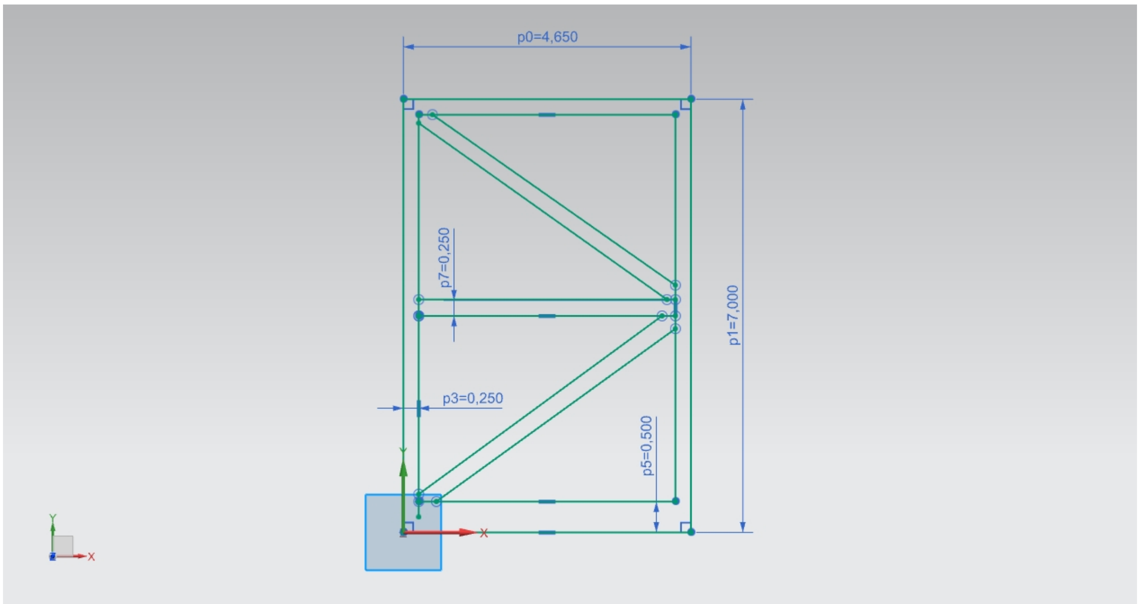


Figure 23: Horizontal Stabilizer

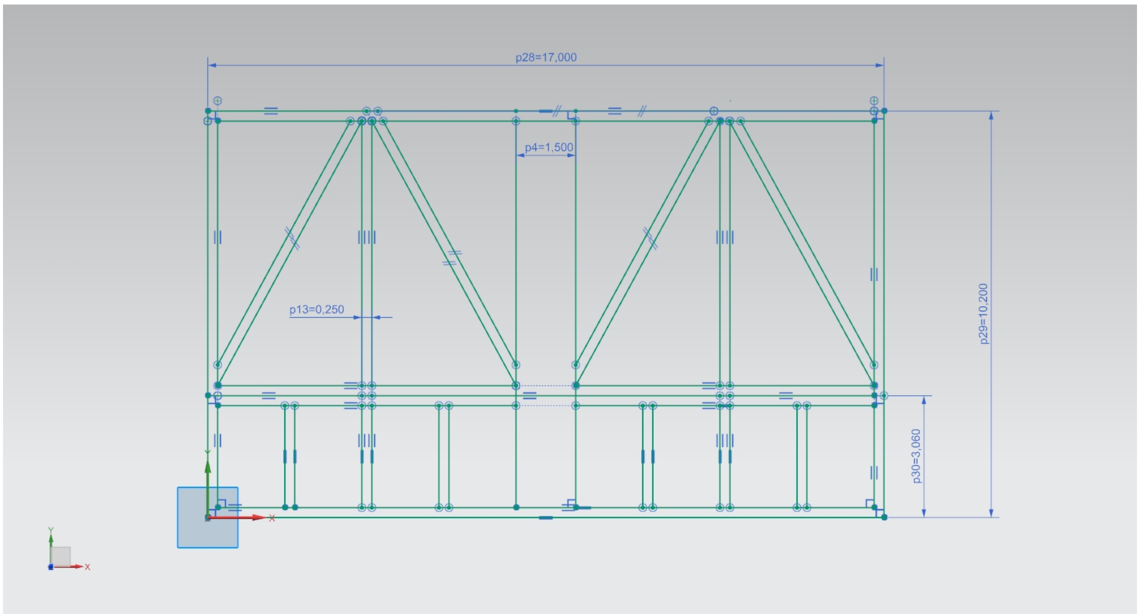
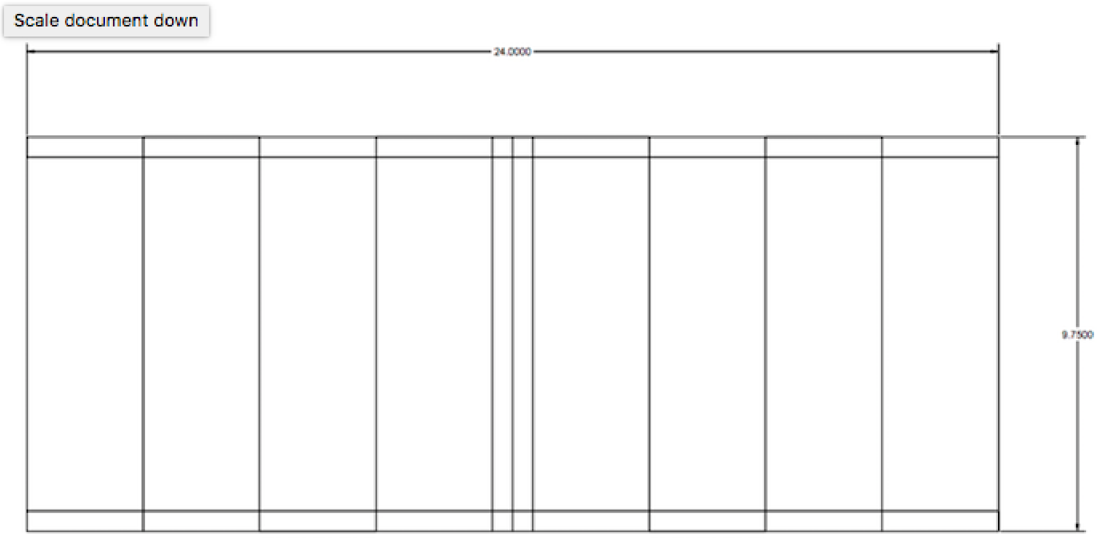


Figure 24: Wing Center



9 Step 8: Conduct Wind Tunnel Drag Testing of Motor-Fuselage Assembly and Recalculate Aircraft Performance

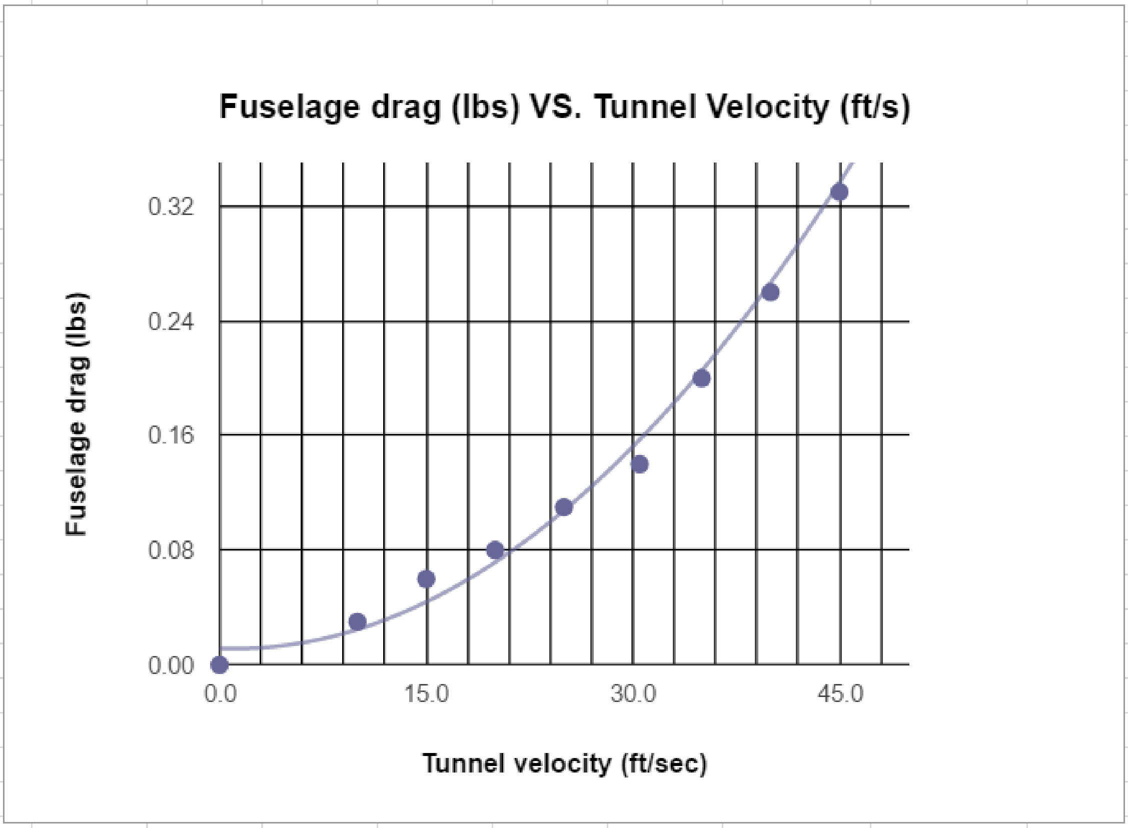
To setup the experiment, the constructed fuselage was put inside the test section of an 18” X 18” subsonic wind tunnel. In addition, a pitot-tube was placed inside the test section to measure the pressure of the wind tunnel. A drawing of the test section in Figure 26.

Three people were needed to perform the experiment. One person controlled airflow of the wind tunnel, one person recorded data from a load cell into Excel, and one person measured wind tunnel velocity. In Figure 26 the load cell readout is at the bottom and the Flow Kinetics Multi-Function Meter is at the top next to the pitot-probe tube.

First, the person controlling the wind tunnel increased the wind tunnel velocity in increments of 5 feet/second up to 50 feet/second. Second, the person measuring wind tunnel velocity called out velocities, informing the person controlling the wind tunnel whether to increase or decrease the speed. Finally, the person recording data entered the drag data into an excel spreadsheet.

After collecting all the data the results were plotted in Excel and can be seen below in Figure 25.

Figure 25: Fuselage Tail Drag



Equation (9) represents the regression curve for the total fuselage and tail drag as a function of velocity.

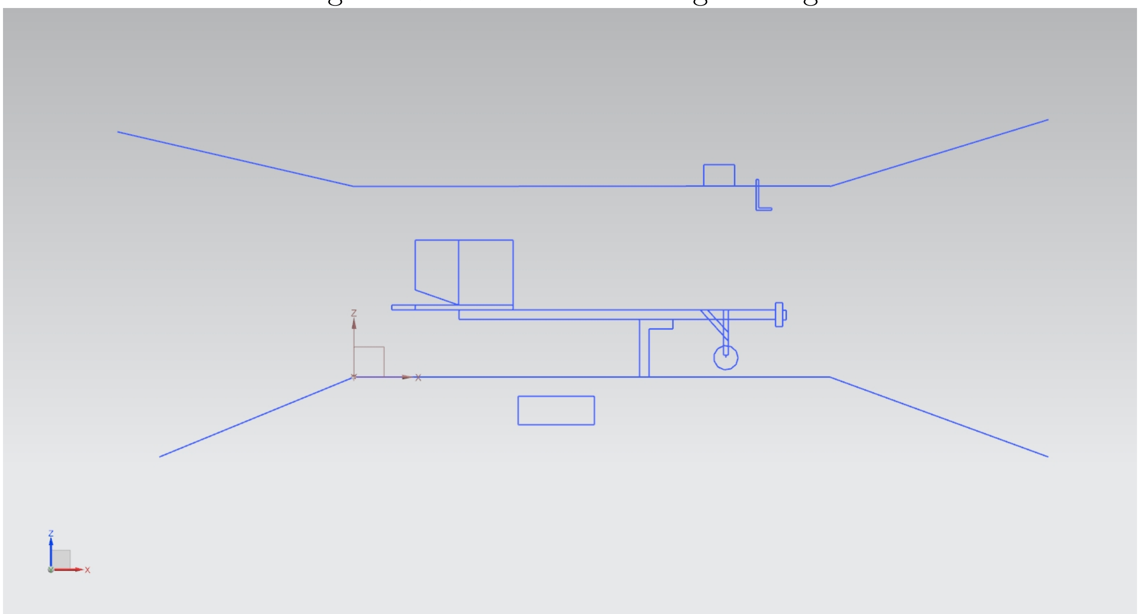
$$\mathrm{Drag}_{F/T} = 0.00017V^2 - 0.00038V + 0.01154 \tag{9}$$

The data for the experiment was collected and entered into table 8.1 on the next page. The last column in the chart labeled Drag of Fuselage Regression equation (lbs) is the calculated thrust using equation (9) above. It is easy to compare the predicted thrust based on the regression equation with the actual measured tunnel thrust and net thrust and see that the values are very close. This confirms the validity of the equation and promotes confidence in its use to recalculate some performance characteristics.

Table 8.1: Fuselage and Tail Drag

	Fuselage Drag Worksheet				
Target Velocity	Actual Velocity	Meter Reading	Meter Tare	Net Drag	Drag of Fuselage Regression Equation
ft/sec	ft/sec	lbs	lbs	lbs	lbs
0	0.0	0.00	0.00	0.00	0.01
10	10.0	-0.03	0.00	0.03	0.02
15	15.0	-0.06	0.00	0.06	0.04
20	20.0	-0.09	-0.01	0.08	0.07
25	25.0	-0.12	-0.01	0.11	0.11
30	30.5	-0.15	-0.01	0.14	0.16
35	35.0	-0.22	-0.02	0.20	0.21
40	40.0	-0.28	-0.02	0.26	0.27
45	45.0	-0.35	-0.02	0.33	0.34
50	50.0	-0.45	-0.02	0.43	0.42

Figure 26: Wind Tunnel Drag Testing



10 Step 9: Conduct Final Flight Testing and Analyze Flight Test

Before the aircraft was tested, the final location of the leading edge with respect to the datum, location of the neutral point with respect to the datum, static margin (as percent of wing chord), and weight of the aircraft were measured with and without a 4 oz. payload. The calculated results are listed in Figure 27. The best placement of the wing differed from the spreadsheet to the 2-finger test, and in the end, the best 2-finger test placement of the wing is what the group went with.

Figure 27: Final Aircraft Dimensions

	Distance from datum to wing leading edge	Actual Location of Neutral Point	Static Margin as % of wing Chord	Weight
Nominal	11.5	15.09	24	15.23
With 4 oz Payload	10.5	14.15	21	19.23

Flight testing was conducted inside the Missouri S&T Gale-Bullman Athletics Building in the Upper Gym. The aircraft was flown by Jim, the lab instructor for the AE2780 course. The primary differences between the predicted and actual were the distance to take off. The aircraft consistently achieved flight well before the predicted distance. This could be due to letting the motor to achieve full power before release or more likely, having an angle of attack during ground roll higher than was used for calculating the estimated distances.

In addition, the plane was slightly more maneuverable than expected. This could be due to the experience of the pilot. The large size of the control surfaces could have also influenced this in ways not accounted for in initial estimates of maneuverability. The pilot commented favorably on the performance and stability of the aircraft, as well as the ease in which it landed. Overall, the pilot was very pleased with the aircraft with no suggestions as to improvements being offered. The pilot mentioned, "It's beautiful."

The takeoff distance for the base aircraft is described in Table 9.1, the takeoff distance for the aircraft with a 4 ounce payload is given in Table 9.2, the distance to clear a 6 foot obstacle is given in Table 9.3, and the maximum velocity in straight and level flight is given in Table 9.4.

Table 9.1: Takeoff Distance for Base Aircraft

Test Number	Distance (ft)
1	22.5
2	20
3	22
4	21
AVG	21.375

Table 9.2: Takeoff Distance for Base Aircraft+4 Ounce Weight

Test Number	Distance (ft)
1	33
2	30.5
3	30
AVG	31.1667

Table 9.3: Takeoff Distance for Base Aircraft+4 Ounce Weight

Test Number	Distance (ft)
1	42
AVG	42

Table 9.4: Distance to Clear a 6' obstacle

Time A-B (sec)	Velocity A-B (ft/s)	Time B-A (sec)	Velocity B-A (ft/s)
1.54	32.4675	1.85	27.027
1.60	31.25	1.64	30.4878
2.01	24.8756	1.84	26.8817
1.07	46.729	1.83	27.3224
1.73	28.9017	1.61	31.0559
1.66	30.1205	1.61	31.0559
		1.96	25.5012
AVG	30.2928		

Figure 28: Final Fuselage Side View

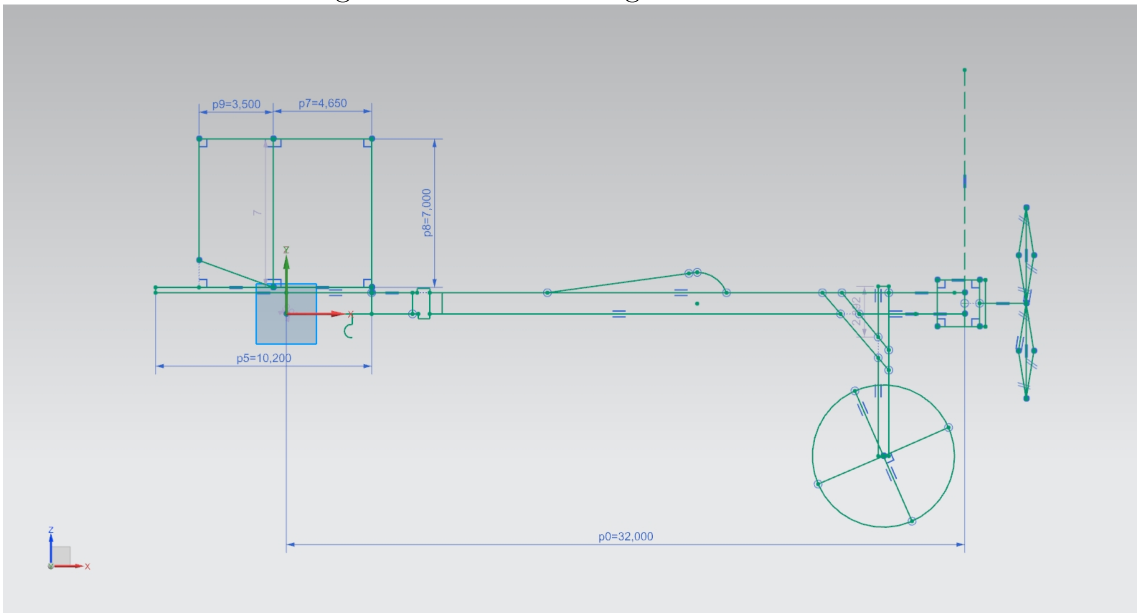
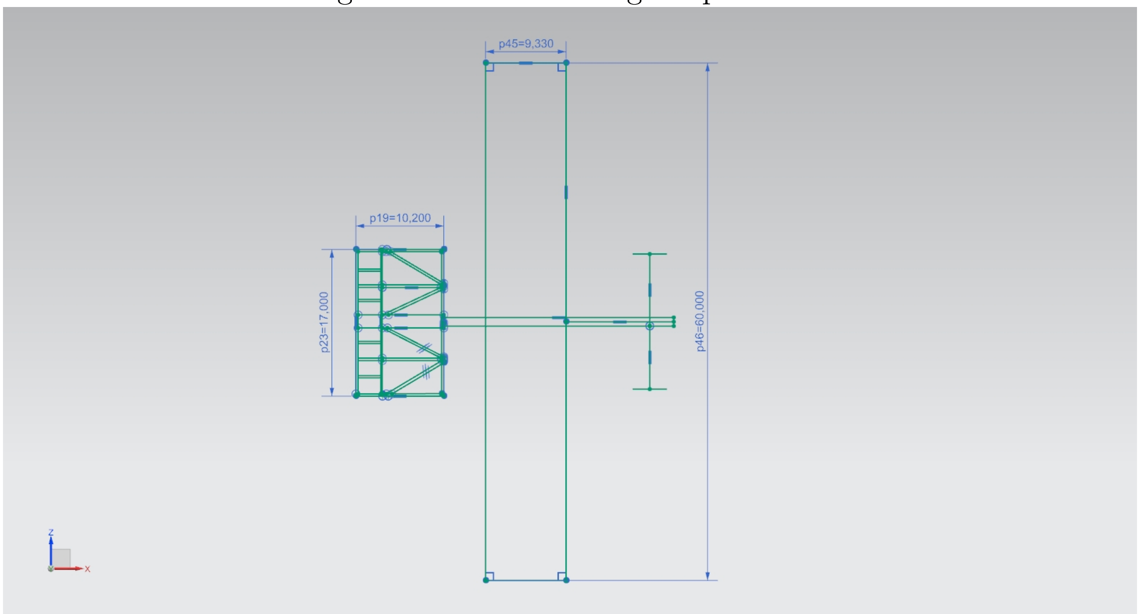


Figure 29: Final Fuselage Top View



A Equations

$$V_{stall} = \sqrt{\frac{2 \cdot w}{p \cdot s \cdot C_{L_{max}}}} \tag{10}$$

$$C_{L_{max}} = \frac{2 \cdot w}{p \cdot s \cdot v^2} \tag{11}$$

$$\text{Reynolds Number} = \frac{p \cdot V_{cruise} \cdot s}{\mu} \tag{12}$$

$$\mu = 3.7373 \cdot 10^{-7} \frac{\text{slug}}{\text{ft} \cdot \text{s}}$$

$$K_{AR<4} = 1 + (1.87 - .00233 A_{LE}) \cdot \frac{AR}{100} \tag{13}$$

$$K_{AR>4} = 1 + ((9.2 - 2.3 \cdot A_{LE}) - (.22 - .153 \cdot A_{LE}) \cdot AR) \cdot \frac{1}{100} \tag{14}$$

$$B = \sqrt{1 - M^2} \tag{15}$$

$$C_{L_\alpha} = \frac{AR}{K} \cdot \left(\frac{2 \cdot \pi}{2 + \sqrt{\frac{A \cdot R^2 \cdot B}{K^2} \cdot (1 + \frac{tan^2(A_{c/2})}{B^2} + 4)}} \right) \tag{16}$$

$$a = \frac{a_0}{1 + (\frac{57.3 \cdot a_0}{\pi \cdot e \cdot AR})} \tag{17}$$

$$C_{D_w} = C_{D_0} + \frac{C_L^2}{\pi \cdot e \cdot AR} \tag{18}$$

$$D_W = \frac{W}{\frac{L}{D}} \tag{19}$$

$$D_{F/T} = 0.00016 \cdot V^2 - 0.0001 \cdot V \tag{20}$$

$$R/C = \frac{\text{excess power}}{\text{weight}} \tag{21}$$

$$h_n = h_{ac_{wb}} + (\frac{l_t \cdot S_t}{c \cdot S}) \cdot \frac{a_t}{a} \cdot (1 - \frac{\partial \epsilon}{\partial \alpha}) \tag{22}$$

$$SM = (h_n - h) \tag{23}$$

A Performance Data for 9”x6”

Velocity (ft/s)	Aircraft Coefficient of lift	Wing Coefficient of Drag	Wing L/D	Wing Drag (lb)	Fuselage Drag (lb)
5	7.773	4.418	1.759	0.510	0.004
6	5.398	2.131	2.533	0.354	0.005
7	3.966	1.151	3.445	0.260	0.007
8	3.036	0.675	4.496	0.200	0.009
9	2.399	0.422	5.684	0.158	0.012
10	1.943	0.277	7.005	0.128	0.015
11	1.606	0.190	8.457	0.106	0.018
12	1.349	0.134	10.035	0.089	0.022
13	1.150	0.098	11.733	0.076	0.026
14	0.991	0.073	13.543	0.066	0.030
15	0.864	0.056	15.456	0.058	0.035
16	0.759	0.043	17.461	0.051	0.039
17	0.672	0.034	19.545	0.046	0.045
18	0.538	0.028	21.694	0.041	0.050
19	0.486	0.023	23.890	0.038	0.056
20	0.441	0.019	26.114	0.034	0.062
21	0.441	0.016	28.347	0.032	0.068
22	0.401	0.013	30.569	0.029	0.075
23	0.367	0.011	32.755	0.027	0.082
24	0.337	0.010	34.886	0.026	0.090
25	0.311	0.008	36.939	0.024	0.098
26	0.287	0.007	38.894	0.023	0.106
27	0.267	0.007	40.731	0.022	0.114
28	0.248	0.006	42.433	0.021	0.123
29	0.231	0.005	43.986	0.020	0.132
30	0.216	0.005	45.318	0.020	0.141
31	0.202	0.004	46.601	0.019	0.151
32	0.190	0.004	47.648	0.019	0.161
33	0.178	0.004	48.519	0.018	0.171
34	0.168	0.003	49.213	0.018	0.182
35	0.178	0.003	49.733	0.018	0.193
36	0.178	0.003	50.087	0.018	0.204
37	0.178	0.003	50.282	0.018	0.215
38	0.178	0.003	50.327	0.018	0.227
39	0.178	0.003	50.234	0.018	0.239

40	0.178	0.002	50.014	0.018	0.252
41	0.178	0.002	49.678	0.018	0.265
42	0.178	0.002	49.241	0.018	0.278
43	0.178	0.002	48.712	0.018	0.292
44	0.178	0.002	48.105	0.019	0.305
45	0.178	0.002	47.430	0.019	0.320
46	0.178	0.002	46.648	0.019	0.334
47	0.178	0.002	45.918	0.020	0.349
48	0.178	0.002	45.101	0.020	0.364
49	0.178	0.002	44.253	0.020	0.379
50	0.178	0.002	43.382	0.021	0.395

Thrust Required (lb)	Thrust Available (lb)	Power Required (ft-lb/s)	Power Available (ft-lb/s)	Rate of Climb (ft/s)	Horizontal Velocity (ft/s)
0.513	0.504	0.504	2.561	−0.051	5.000
0.359	0.497	2.156	2.982	0.921	5.929
0.267	0.490	1.872	3.428	1.734	6.782
0.209	0.482	1.672	3.859	2.439	7.619
0.170	0.475	1.529	4.275	3.061	8.464
0.143	0.467	1.430	4.674	3.616	9.323
0.124	0.460	1.368	5.5057	4.113	10.202
0.111	0.452	1.335	5.424	4.559	11.100
0.102	0.444	1.329	5.774	4.956	12.018
0.096	0.436	1.347	6.107	5.307	12.955
0.093	0.428	1.388	6.422	5.612	13.911
0.091	0.420	1.452	6.720	5.873	14.833
0.090	0.412	1.537	7.000	6.090	15.872
0.091	0.403	1.645	7.261	6.261	16.876
0.093	0.395	1.775	7.504	6.387	17.894
0.096	0.386	1.927	7.728	6.467	18.926
0.100	0.378	2.102	7.933	6.500	19.969
0.105	0.369	2.301	8.118	6.485	21.022
0.110	0.360	2.524	8.283	6.421	22.085
0.115	0.351	2.771	8.429	6.307	23.156
0.122	0.342	3.045	8.554	6.142	24.234
0.129	0.333	3.344	8.658	5.924	25.316
0.136	0.324	3.671	8.741	5.652	26.402
0.144	0.314	4.026	8.803	5.326	27.489
0.152	0.305	4.410	8.844	4.943	28.576

0.161	0.295	4.823	8.862	4.503	29.660
0.170	0.286	5.267	8.858	4.003	30.740
0.179	0.276	5.743	8.832	3.444	31.814
0.189	0.266	6.251	8.783	2.823	32.879
0.200	0.256	6.793	8.711	2.138	33.933
0.211	0.246	7.369	8.615	1.390	34.972
0.222	0.236	7.980	8.496	0.575	35.995
0.233	0.226	8.628	8.353	−0.306	36.999
0.245	0.215	9.312	8.185	−1.257	37.979
0.257	0.205	10.035	7.993	−2.277	38.933
0.270	0.194	10.797	7.776	−3.368	39.858
0.283	0.184	11.600	7.534	−4.533	40.749
0.296	0.173	12.443	7.266	−5.771	41.602
0.310	0.162	13.328	6.972	−7.085	42.412
0.324	0.151	14.256	6.653	−8.477	43.176
0.338	0.140	15.229	6.307	−9.946	43.887
0.353	0.129	16.246	5.934	−11.496	44.540
0.368	0.118	17.309	5.534	−13.127	45.130
0.384	0.106	18.419	5.107	−14.840	45.648
0.400	0.095	19.577	4.653	−16.638	246.089
0.416	0.083	20.784	4.170	−18.522	46.443

A MATLAB Code For Takeoff Distance

```
%chord length of the wing
cw=0.8; %(ft)

%wingspan
b=5; %(ft)

%wing planform area
s=b*cw; %(ft^2)

%Aspect ratio
AR=(b^2)/(s);

%Lift slope of the finite wing (1/degrees)
a_0=0.0784;
```

```
%Max coefficent of lift
cl_max=1.2;

%span efficiency factor
e=0.69;

%height of the wings above the ground
h=0.5; %(ft)

%free stream density
rho=0.0023; %(slugs/ft^3)

%Obstacle height for clearance (ft)
ob=6;

%gravitaional constant
g=32.2; %(ft/s^2)

%nominal weight of the aircraft
W=14.1675/16; %(lbs)

%Wing parasite drag coefficent
cd_0=0.012;

%angle where cl is 0
alpha_0=-2; %(degrees)

%ground roll angle
alpha_g=1.5; %(degrees)

%coefficent of lift (actual)
cl=a_0*(alpha_g-alpha_0);

%coeffiecnt of rolling friction
mu_r=0.02;

%ground roll parameter
phi=(16*h/b)^2/(1+(16*h/b)^2);

%coefficent for fusalage drag equation
d0=-0.0001;
d1=0.00016;
```

```
%coefficent of drag of the wing
Cd_w= cd_0+ phi*((cl^2)/(3.14*e*AR));

%Total drag equation as a function of velocity
D=@(v) (0.5*rho*(v^2)*s*Cd_w)+ d0*(v)+d1*(v^2);

%coefficents for thrust equation
c0=0.485;
c1=-0.0053;
c2=-0.00003;

%Thrust available as a function of velocity
T=@(v) c0+c1*(v)+c2*(v^2); %(lbf)

%Lift as a function of velocity
L=@(v) 0.5*rho*(v^2)*s*cl; %(lbf)

%Pt.1 of program. Estimating distance, velocity, and time to achieve
    flight

%Initial position, velocity, and time
x0=0;
v0=0;
t0=0;
Da=0;
Ta_0=0.46; %static thrust = max thrust
i=1;
x(1)=x0;
v(1)=v0;
t(1)=t0;
Da(1)=Da;
Ta(1)=Ta_0;
max_iterations=5000;
dt=0.005; %differential time change

%while loop for the rk4
while (L(v(i))<W && i<max_iterations)

    k1=(g/W)*(T(v(i))-D(v(i))-mu_r*(W-L(v(i))));
    k2=(g/W)*(T(v(i)+0.5*k1*dt)-D(v(i)+0.5*k1*dt)-mu_r*(W-L(v(i)+0.5*k1*dt)));
    k3=(g/W)*(T(v(i)+0.5*k2*dt)-D(v(i)+0.5*k2*dt)-mu_r*(W-L(v(i)+0.5*k2*dt)));
    k4=(g/W)*(T(v(i)+k3*dt)-D(v(i)+k3*dt)-mu_r*(W-L(v(i)+k3*dt)));
    t(i+1)= t(i)+dt;
```

```
    v(i+1)= v(i)+(1/6)*(k1+2*k2+2*k3+k4)*(dt);
    x(i+1)= x(i)+(v(i)*dt)+(0.5*k1*(dt^2));
    Da(i+1)=D(v(i));
    Ta(i+1)=T(v(i));
    i=(i+1);
end

%Deliverables for pt.1
figure(1);
plot(t,x)
title('Position vs Time')
xlabel('Time (s)')
ylabel('Position (ft)')

figure(2);
plot(t,v)
title('Velocity vs Time')
xlabel('Time (s)')
ylabel('Velocity (ft/s)')

figure(3);
plot(x,Da,x,Ta)
title('Drag and Thrust vs Position')
xlabel('Position (ft)')
ylabel('Force (lbf)')
legend('Drag','Thrust')


%Pt 2 of program. using aircraft performance to estimate thrust, power,
    and
%maximum rate of climb angle

%cl after ground roll
cl_var =@(v) (2*W)/(rho*v^2*s);

%cd after ground roll
cd_var= @(v) cd_0 + ((cl_var(v))^2)/(3.14*e*AR);

%Thrust Required for unaccelerated flight
TR =@(v) 0.5*rho*(v^2)*s*(cd_var(v));

%Power required for unaccelerated flight
PR =@(v) TR(v)*v;
```

```
%Power available for unaccelerated flight
P =@(v) T(v)*v;

%Stall velocity
v_stall=12.6653;

%Rate of climb (ft/s)
RoC = @(v) (P(v)-PR(v))/W;


%conditions for loop
i=1;
v(1)=v_stall;
Thrust_avail(1)=0.41306; %plug in stall velocity into thrust available
    eq above
Thrust_req(1)=0.012956; %Plug in stall velocity to the drag equation
Power_avail(1)=5.23155; %Thrust_avail*v_stall
Power_req(1)=0.16409; %Thrust_req*v_stall
Rate_of_climb(1)=5.7229; %Plug values into RoC eq above
dv=0.001;

while(TR(v(i))<T(v(i)))
    Thrust_avail(i+1)=T(v(i));
    Thrust_req(i+1)=TR(v(i));
    Thrust_diff1= T(v(i))-TR(v(i));
    Thrust_diff2=Thrust_avail(i)-Thrust_req(i);
    Power_avail(i+1)=P(v(i));
    Power_req(i+1)=PR(v(i));
    Rate_of_climb(i+1)=RoC(v(i));
    v(i+1)=v(i)+dv;
    i=i+1;

    if(Thrust_diff2<Thrust_diff1)
        velocity_theta_max=v(i-1); %horizontal velocity at max climb
            angle
        RoC_theta_max=RoC(v(i-1)); %Rate of climb at max climb angle
    end


end
```

```
theta_max=atan(RoC_theta_max/velocity_theta_max); %max climb angle
    (radians)
x_air=(ob)/tan(theta_max); %Horizontal distance covered by the aircraft
    while in climb
x_total=x+x_air; %Total Horizontal distance to clear 6ft obstacle

%Deliverables for pt.2 (graphs)
figure(4);
plot(v,Thrust_avail,v,Thrust_req)
title('Thrust available and required vs Velocity')
xlabel('Velocity (ft/s)')
ylabel('Thrust (lbf)')
legend('available','required')

figure(5);
plot(v,Power_avail,v,Power_req)
title('Power available and required vs Velocity')
xlabel('Velocity (ft/s)')
ylabel('Power (lb*ft/s')
legend('available','required')

figure(6);
plot(v,Rate_of_climb)
title('Hodograph Diagram')
xlabel('Horizontal velocity (ft/s)')
ylabel('Rate of climb (ft/s)')
```