MEM 425: Aircraft Performance and Design
Final Report
Fall 2020-2021 (202015)

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Abstract

The purpose of the project is to deliver a wing that can be mounted onto a customer's airplane. The wing should be able to provide the maximum range of velocity for the airplane to hold cruise flights at sea-level. The group recommends a cruise velocity of 3.62ft/s, and a static margin between 0.05 and 0.15. The maximum payload that can be carried is 0.1156 lb, this was calculated through the flight testing. The flight testing ran at three different angles and three different velocities. During the flight trial, it was determined that the one that optimized angle of attack, led to an increase of maximum range of velocities. Overall, the team's model airplane was successful for the actual flight test and met the performance expectations that the team had, with a few adjustments to make the glider more efficient and stable.

Member Participation

Team: A

Name	**% Deserved**	**Signature**
Nelli Khalatyan	100%	*Nelli Khalatyan*
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Introduction

Cruise flight is one of the simplest forms of flight analysis. In cruise flight, lift and weight are equal in magnitude but opposite in direction and the same is true for thrust and drag. The net moment at the aerodynamic center is also zero. The design process for a successful aircraft flight is iterative in nature. In this project, designing for maximum range is the ultimate objective. The important parameters for this project include wing area, total weight, stability, aspect ratio, the selected airfoils. These parameters must be designed for using aerodynamic modeling and then be tested using real flight. Iterative testing takes place to refine the wing and locations of center of gravity and neutral point. The lift and drag forces must also be analyzed as well as the stability of the glider.

Initially, the first part of the process of airplane design requires description of the wing parameters such as the size and shape of the wing. Once that process has been complete, construction of the glider is completed and testing begins. The glider is then further refined from test data in form of iterative testing and refinement. In theory, the final glider should be better at performing its intended task than the first design derived only from aerodynamics. Once a sufficiently stable structure is created and tested, data is collected to determine the neutral point and center of gravity as well as the coefficient of lift and other parameters. Stability conditions such as vertical tail stability can also be adjusted after the initial theoretical calculations.

An important distinction in this project is the difference between the given theoretical values of the propeller engine plane and the experimental glider. It is a given that the glider that is fabricated and tested has no sustained thrust as it is not running on an engine like the propeller plane. The propeller plane has given values of thrust that will help during the fabrication and test phases of the project.

These assumptions, along with the other givens such as the tail aspect ratio, span and moment arm, will help for the initial design of the glider and is a good starting point for the iterative design.

Design Process

For this project, there were a lot of constraints with only a few variables given. Most of the variables had to be assumed and then solved for. The crucial design parameter was to design for maximum range of velocities over which the airplane can hold cruise flights at sea-level. Given the limited resources of materials and tools that would be able to be used to build this aircraft, the team decided that the most important design factor was the wing size. The team started looking at airfoils initially to decide the shape of the chamber. With the foam material used for the construction of the wings, there was a maximum opportunity to build a chord length of 6 inches. Eventually the team went with a chord length of 5.026 to increase the aspect ratio of the wing. The team decided to use a smaller Reynold's number because the efficiency of the wing increases with a smaller Reynold's number. For that purpose the team chose Reynold's value as 50,000. Using these values and the equation below a theoretical cruise velocity can be calculated. It is important to note that this number would only be exact without any design errors or other error sources.

$$Re = 546 * velocity * chord \qquad \text{Equation 1}$$

$$Velocity = \frac{50{,}000}{546 * (5.0625in)} = 3.62\ ft/s$$

The theoretical cruise velocity is a reasonable assumption for flight and is used in later calculations. The following step would be to choose an airfoil that would be easy to fabricate given the constraints. After careful research, comparison and deliberation, the NACA6412 was chosen as the ideal airfoil. Assuming the angle of attack was equal to zero, the coefficient of lift was obtained from the provided data. The angle of attack was assumed to be 0 to limit the effects of angle of attack (α) on the report. For cruise flight, it is assumed that weight is equal to lift. Using that equation and the given values of density (ρ), weight at 20 grams, and the velocity as was yielded from Equation 1. Although the assumption of 20 grams for weight was taken, the actual weight will not be equivalent to 20 grams but instead it is still unknown due to the unknown weight of the wing. The final weight is approximated to be within a range of 5-8 grams. Using Equation 1 above, the values listed in Table 1, these are the design parameters that will be used to construct the wing for the model. Although, this wing will be built by hand and not with the help of a machine, the team expects a margin of error for the wing area to be $\pm\ 2.5\ inches$ and the wing span to be $\pm$ 1 inches.

Table 1: various mathematically determined characteristics of each airfoil with the chosen airfoil highlighted in yellow

Airfoil	**Coefficient of Lift**	**Wing Area (S)**	**Wingspan (b)**	**Max Camber Location**	**Max Camber**
NACA 6409	0.2639	171.8547292	33.94661318	2.00475	0.30375
NACA 4412	0.1804	251.3994625	49.65915309	2.025	0.2025
NACA 4415	0.0526	862.2141262	170.3139015	2.035125	0.2025
NACA 6412	0.27112	167.2781906	33.04260555	2.00475	0.30375

Estimated Aerodynamic Characteristics

After the selection of the airfoil, databases can be found and used concerning the coefficient of lift and coefficient of drag against the angle of attack. The online database “Airfoil Tools” [1] is such an example. The coefficient of lift versus angle of attack and the coefficient of drag versus angle of attack graphs for the NACA 6412 airfoil are shown below in Figure 1. These values are the ones which will be compared to the experimental.

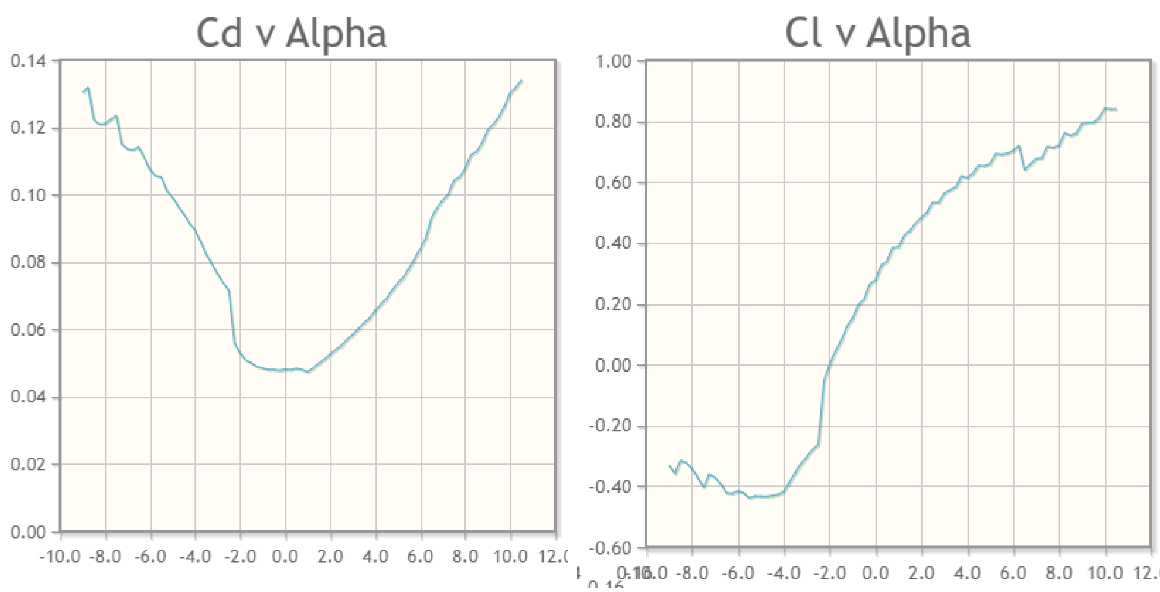

Figure 1: Coefficient of lift and drag versus the angle of attack for NACA 6412 airfoil

The coefficient of moment versus the angle of attack of the NACA 6412 was also found and can be used for comparison purposes. This graph is shown below in Figure 2.

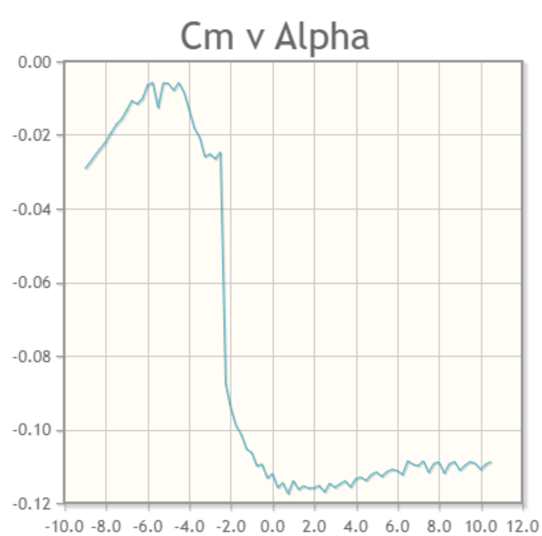

Figure 2: Coefficient of moment versus the angle of attack for NACA 6412 airfoil

Flight Test Confirmation

Once the appropriate theoretical values of the glider were found, the glider was then fabricated. The material of the wing used in the glider design was styrofoam. This material was used due to the fact that it is relatively light and strong. The lightweight nature of the material is good for producing good lift. Wood dowels were used for the fuselage of the glider due to consistent weight distribution and low surface area which allows for smaller values of drag. The wing is rectangular and the airfoil is the NACA 6412, a close approximation to the flat plate airfoil.

Once the materials were chosen, the plane was then built. During the process of building and testing the glider, adjustments had to be made accordingly, which resulted in two separate builds of the plane.

The first iteration was built with the dimensions shown in Table 2, shown below. It had a horizontal stabilizer added in the tail section for added stability. Additionally, a piece of duct tape was used at the nose to minimize the moment arm at the nose. The two half-wings were attached to the front of the fuselage center with duct tape. The first design of the iterative process of the glider was then tested. An image of the first glider is shown below in Figure 3.

Table 2: Glider parameters of first design

Parameter		**Value (in)**
Wing	Chord	3
	Thickness	0.375
	Span	25.75
	Mounting Location: leading edge of wing measured from nose of aircraft	4
Fuselage	Length	23.875
Tail	Span: Top	2.25
	Span: Bottom	2.25
	Cord	0.875
	Thickness	0.875

Figure 3: First plane fabrication with predetermined parameters

Once the plane was tested, numerous issues in the flight of the plane became apparent. The gliding capabilities of the airplane were subpar at best and the plane also had issues maintaining stability with its erratic flipping, most likely due to a non zero pitching moment. Additionally, there was a weak point on the fuselage of the aircraft where the 12” wooden dowels were connected to extend to 24”. Therefore, a new plane was fabricated in an attempt to mitigate these issues.

A new glider design was designed from scratch. The fuselage for this design was built with the wooden dowels glued in such a way that the support was even on all the sides which eliminated the weak point. Then the wings were designed with a larger span and chord length, but a smaller thickness. Additionally, a new tail was designed, with a smaller thickness relative to the first iteration, in order to keep the horizontal stabilizer, but eliminate the extra weight. Next, hot glue was attached to the nose of the plane to act as a counterweight to minimize the moment arm. These parameters are shown below in Table 3.

Table 3: Glider parameters of second design

Parameter		**Value (in)**
Wing	Chord	5.0625
	Thickness	0.125
	Span	26
	Mounting Location: leading edge of wing measured from	5.5

	nose of aircraft	
Fuselage	Length	24
Tail	Span: Top	2
	Span: Bottom	3.75
	Cord	0.875
	Thickness	0.25

This glider did a better job of gliding than the previous design; however, this glider was still unstable and doing flips so this design was modified, but a new plane was not built from scratch. The second plane design is shown below in Figure 4.

Figure 4: Second Plane Design

The second design continued to be modified until the aircraft was able to glide with stability. The wings were sanded down and straightened to fit the flat plate design more effectively. Additionally, the horizontal tail stabilizer was increased in area and a vertical tail stabilizer was added for additional stability. Once this design was updated, the aircraft successfully glided with reasonable stability. Figure 5 below shows the final glider design with the added vertical stabilizer. Table 4 shows the final modifications to this design. Additionally, the final weights of this plane are shown in Table 5 below.

Figure 5: Final glider design

Table 4 : Final modifications of the design all of which were done on the tail.

Tail Parameter	**Value (in)**
Span: Top	2.875
Span: Bottom	2
Cord	1.125
Thickness	0.5
Additional horizontal stabilizer	6.125
Additional vertical stabilizer	5.5

Table 5: Final Glider Design Weight

Parameter (Weight)	**Value (g)**
Wing	14.5
Fuselage	23
Tail	9
Nose	5
Total	**66**

Data analysis

Upon fabrication of the glider as was described in Figure 3, the glider was launched and the flightpath was determined using the trial videos. This flightpath is shown below in Figure 6.

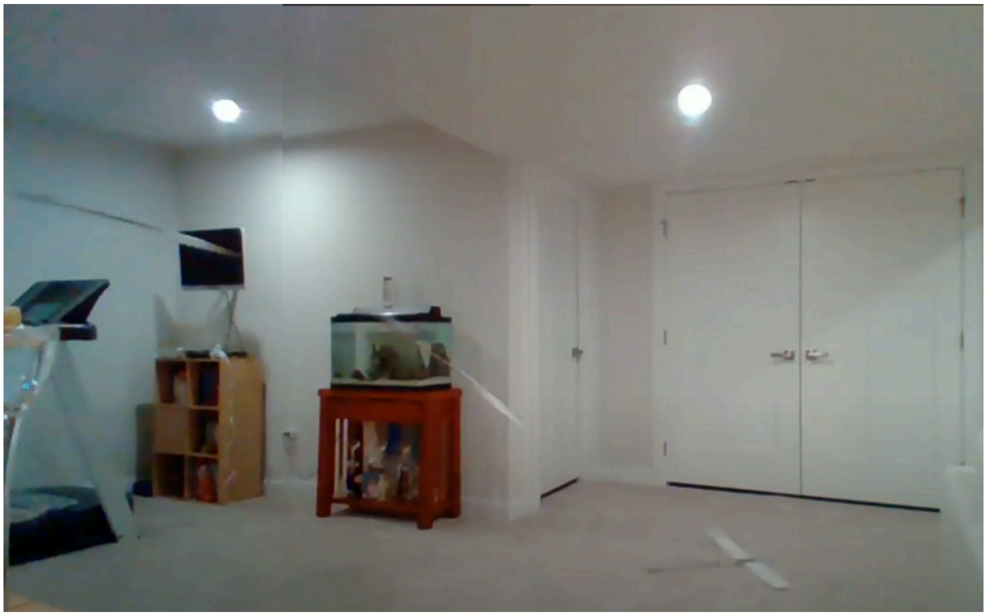

Figure 6: Test flight of glider final design

Using the selected airfoil (NACA 6412) and the data collected from the flight, the aerodynamic characteristics can be compared to test the integrity of the glider. The NACA 6412 airfoil has a given coefficient of lift of 0.2776 at an angle of attack of 0 degrees. Since the wing of the glider is mounted on the same axis as the fuselage (i.e the angle of the wing relative to the fuselage is 0 degrees), the glide angle can be taken as the angle of attack for comparative purposes.

The glider was tested with different glide angles and six different initial heights. The coefficient of lift for these different angles can be calculated from the coefficient of lift equation when combined with the lift equation for a glide angle. These equations are shown below in Equations 2 and 3.

$$C_L = \frac{L}{qS} = \frac{L}{\frac{1}{2}\rho v^2 S} \quad \text{and} \quad L = W cos\alpha \qquad \text{Equation 2}$$

$$C_L = \frac{2W cos\alpha}{\rho v^2 S} \qquad \text{Equation 3}$$

Since all the variables to solve for the coefficient of lift are known from the experimental data, determining the coefficient of lift is simply a matter of plugging in these values. The values for these substitutions are shown below in Table 6.

Table 6: Trial # versus coefficient of lift of experimental data

Trial	**CL**
1	1.26998
2	0.63638
3	1.32689
4	0.29215
5	0.58155
6	0.70213

The coefficient of drag of the experimental data can also be determined. The equation for the coefficient of drag as it relates to the coefficient of lift is shown below in Equation 4.

$$C_D = C_L \tan\alpha \qquad \text{Equation 4}$$

Using the glide angles which were determined from the sample flights of the glider along with the already calculated coefficient of lift values, the coefficient of drag values were also determined. These results are summarized below in Table 7.

Table 7: Trial # versus coefficient of drag of experimental data

Trial	**CD**
1	0.38099
2	0.15910
3	0.32363
4	0.08243
5	0.28968
6	0.12789

The coefficient of lift and drag coefficients can be compared to the NACA 6412 values for the corresponding angles of attack. The percent error for the sample flights can be calculated. The calculation for trial 6 was conducted since the angle of attack of that flight was within the values from the provided tables of the NACA 6412. The results are shown below in Table 8.

Table 8: Percent error of coefficient of lift and drag values

Trial	Theoretical CL	Actual CL	Percent Error	Theoretical CD	Actual CD	Percent Error
6	0.842	0.702	-16.424	0.132	0.128	-3.337

Error Analysis

The percent error values for the coefficient of lift and drag for this trial were 16.424% and 3.337%, respectively. Based on the circumstances and possible sources of error, the difference between the theoretical values with the experimental was not truly significant. That being said, there are still potential sources of error that were introduced into the flight tests that could have yielded these differences.

The flight time for the glider was short on all tests and the time was difficult to approximate. Due to this human element, it is possible that the times recorded were off by as much as half a second. This discrepancy in the data could change the coefficient of lift and drag due to the velocity element in their equations. Other possible sources of error also stem from the fabrication of the aircraft wing itself and the environment. Since it was difficult to make the dimensions of the wing as precise and as efficiently as possible in a machine shop with all the tools available at hand. Additionally, this plane could only be made and tested by one person due to the quarantine and all teammates were scattered throughout different states. Small differences between the measured area of the wing and the air density could also affect the values of coefficient of lift and drag. Sudden wind gusts can also contribute to the error.

Performance Analysis of Soft SoarTM

The thrust versus velocity curve of the Soft SoarTM model is given in the problem statement. The thrust velocity curve describes the battery thrust as a function of flight velocity. The curve is shown in graphical form below in Figure 7.

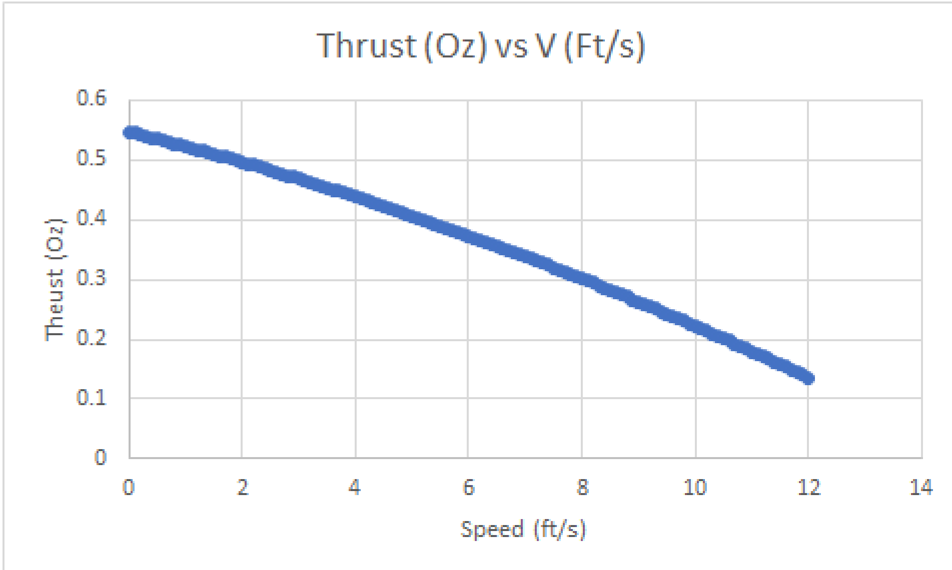

Figure 7: Propeller thrust versus velocity

All the step-by-step work for all the below calculations and plots are shown below in the appendices. The drag versus velocity curve can be generated from the parasite drag coefficient determined from the experimental data. The drag-polar equation can be used to generate the coefficient of drag at varying velocities. The drag-polar equation is shown below in Equation 5. The value for K is determined from Equation 6 where e is 0.7 for a rectangular wing.

$$C_D = C_{D,0} + KC_L^2 \qquad \text{Equation 5}$$

$$K = 1 / \pi eAR \qquad \text{Equation 6}$$

Once the coefficients of drag values are solved for the varying velocities, the same can be done to solve for the drag values with the drag equation. This equation is shown below in Equation 7.

$$D = 0.5C_D\,\rho V^2 S \qquad \text{Equation 7}$$

The drag versus velocity curve was computed and the results are shown below in Figure 8. The minimum drag found was 0.0301 lbs at 10.6 ft/s.

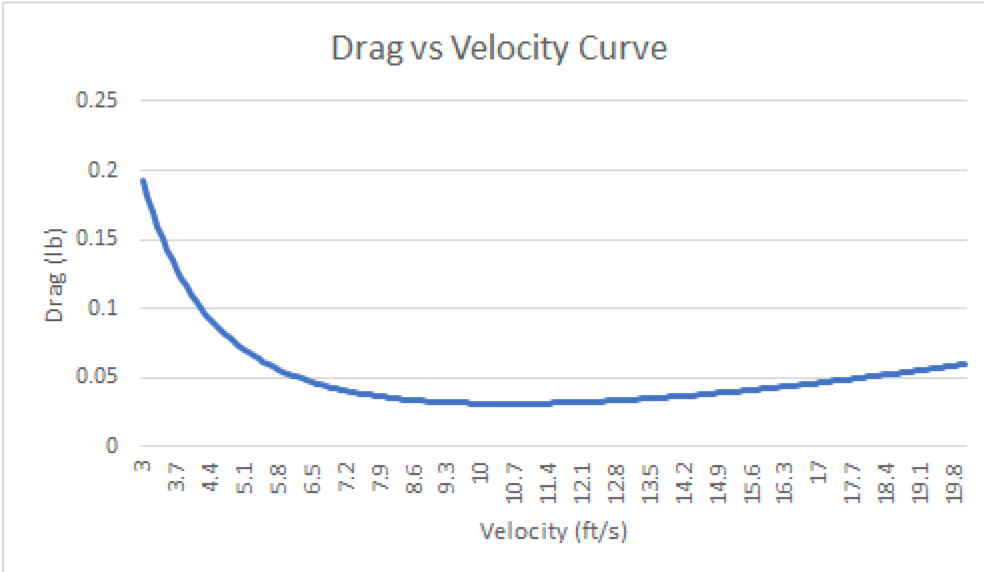

Figure 8: Drag vs velocity curve of SoftSoarTM model

The drag profile of the propeller plane was compared to the thrust capabilities of the battery to find the range of cruise flights. This region encompasses all the velocities where the thrust is greater than the drag. The graph that describes this is shown below in Figure 9.

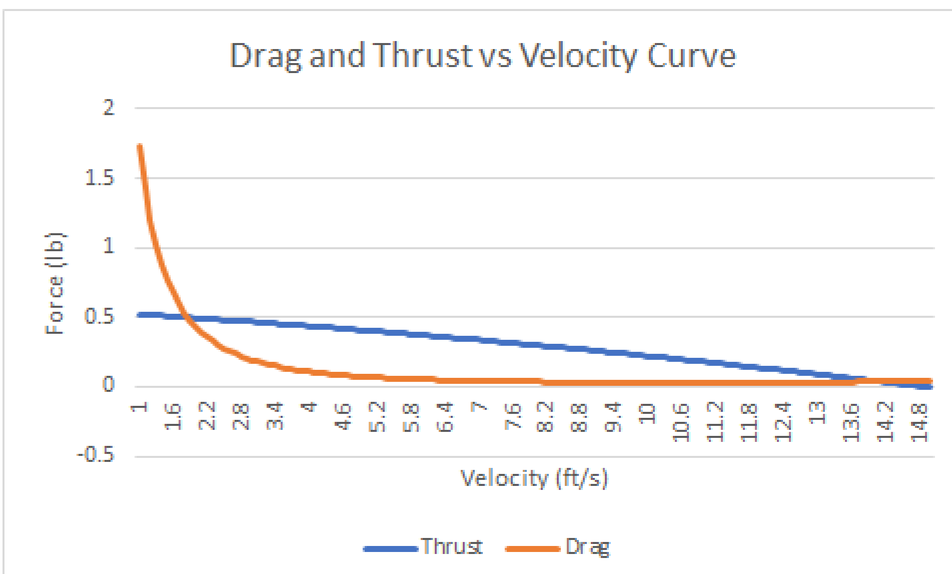

Figure 9 : Drag and thrust vs velocity curve of SoftSoarTM model

Based on the graph above, the range cruise flights is from 1.9 ft/s to 14.2 ft/s. All other velocity values have a drag value that exceeds the thrust and the plane cannot fly in cruise.

The maximum payload of the aircraft can also be determined by finding the maximum take off weight. The weight of the aircraft is then subtracted by the maximum take off weight and the result is the maximum payload. Equation 8 shows how to calculate the maximum take off weight.

$$W = \frac{1}{2} v_{LO}{}^{2} \rho S C_L$$

Equation 8

The take off velocity used in the above equation is the maximum velocity the plane can fly in cruise conditions. In this case, the maximum velocity is 14.2 ft/s. As a result, the maximum take off weight is 0.2611 lb. Equation 9 shows how to calculate the maximum payload as a function of the take off weight and the weight of the aircraft.

$$W_{payload} = W_{\max tax-off} - W_{aircraft}$$

Equation 9

Given that the weight of the aircraft is 0.1455 lb, the maximum payload can be calculated as 0.1156 lb or 52.4 grams.

The takeoff distance can also be calculated. This value is important because it can be used to determine if an aircraft can take off at a runway. The equation for calculating the takeoff distance is shown below in Equation 10.

$$S_g = \frac{1.21W^2}{\rho g S C_L T}$$

Equation 10

Where W is aircraft weight, ρ is air density, g is gravity, S is wing area C_L is coefficient of lift and T is thrust. After substituting all the values into the equation, the takeoff distance was determined to be 5.15 ft.

The landing distance can also be calculated. This value can be used to determine if an aircraft can land at a certain airport. The landing distance is a function of take off velocity, weight, thrust, drag, the friction factor, lift and gravity. The relationship between these terms is shown below in Equation 11.

$$S_g = \frac{v_{LO}{}^{2}(\frac{W}{g})}{2(T - (D + \mu(W - L)))}$$

Equation 11

Substituting the values for this equation, which are all known, yields a result of 6.60 ft.

The neutral point can also be calculated through a three step process. The neutral point location, when determined relative to the center of gravity of the aircraft, gives a good prediction about the stability. The stability margin is defined as the ratio between the distance between the center of gravity and the neutral point to the chord length of the wing. The equation that summarizes this relationship is shown below in Equation 12.

$$\text{S.M.} = \frac{x_{\text{np}} - x_{\text{cg}}}{c}$$

Equation 12

A good range for the stability margin is from 0.05 to 0.15; numbers outside this region predict unstable behavior for the aircraft. The center of gravity of the aircraft is a given quantity from the problem statement. The neutral point can be calculated using the aspect ratios of the wing and horizontal tail along with the horizontal tail volume coefficient. The horizontal tail volume coefficient can be calculated from Equation 13 below.

$$\text{V}_{\text{h}} \equiv \frac{S_{\text{h}}\,\ell_{\text{h}}}{S\,c}$$

Equation 13

Where S_h and l_h are the horizontal tail area and moment arm respectively. Once the horizontal tail volume coefficient has been calculated, the neutral point location can be found using Equation 14.

$$\frac{x_{\text{np}}}{c} \simeq \frac{1}{4} + \frac{1 + 2/AR}{1 + 2/AR_{\text{h}}}\left(1 - \frac{4}{AR + 2}\right)\text{V}_{\text{h}}$$

Equation 14

Where AR_h is the aspect ratio of the horizontal tail, a given value from the problem statement. Upon calculating and substituting these values into the relevant formulas, the neutral point was determined to be 3 inches from the nose. This yielded a result of a static margin of -0.5039, a value that predicts unstable behavior.

Conclusion

The DBF (Design, Build, and Fly) project allowed the team to use both technical as well as practical knowledge to design, build and fly a glider. Iterative testing was a key focal point of the project that allowed for a final product with aerodynamic properties similar to that of a theoretical airfoil. These experimental results were then used to predict the behavior of a battery powered propeller plane. The maximum payload, takeoff distance, landing distance, stability margin, neutral point, center of gravity and drag curves were all calculated for the propeller plane. The main focus of the project, the maximum payload was found to be 52.4 grams.

The fabricated wing, when compared to the airfoil, behaved in a similar manner when comparing the coefficients of lift and drag. Errors in the aerodynamic analysis can be traced to a number of factors but most important is the imprecise timing mechanism which changed the coefficient values drastically. Despite this, the percent error remained anywhere from 3.3% to 16.4%, a small percentage given the quality of the glider itself.

One area of potential improvement would be to improve the design to change the static margin to a positive value. The group noticed many instances in the testing where the glider did not glide properly but also flipped around. This is consistent with the static margin value prediction of the behavior of the aircraft.

References

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[2] bclarkrestonbclarkreston 3, aeroaliasaeroalias 95.2k44 gold badges260260 silver badges407407 bronze badges, and KoyovisKoyovis 47k99 gold badges125125 silver badges224224 bronze badges, “What is the meaning of the percentage used to describe the thinness of this glider's airfoil?,” *Aviation Stack Exchange*, 01-Aug-1966. [Online]. Available: https://aviation.stackexchange.com/questions/39690/what-is-the-meaning-of-the-percentage-used-to-describe-the-thinness-of-this-glid. [Accessed: 01-Dec-2020].

[3] *Performance*. [Online]. Available: http://www.dept.aoe.vt.edu/~lutze/AOE3104/range&endurance.pdf.

[4] “Summary of Low-Speed Airfoil Data,” *Illinois.edu*. [Online]. Available: https://m-selig.ae.illinois.edu/uiuc_lsat/Low-Speed-Airfoil-Data-V5.pdf.

Appendix A: Calculations to find the SM

$$W = \frac{1}{2} v_{LO}{}^2 \rho S C_L$$

W = 0.5(14.2)^2(0.002377)(0.9140625)(1.19204) = 0.2611

$$W_{payload} = W_{\max\, tax-off} - W_{aircraft}$$

$W_{\text{max tax-off}}$ = 0.2611 - 0.1455 = 0.1156

$$S_g = \frac{1.21W^2}{\rho g S C_L T}$$

(1.21(0.145505)^2)/((32.2)(0.002377)(0.9140625)(0.792531)(0.0896)) = 5.15

$$S_g = \frac{v_{LO}{}^2 (\frac{W}{g})}{2(T - (D + \mu(W - L)))}$$

S_g= (13)(0.145505/32.2)/(2(0.0896-(0.020537) = 6.60 ft

$$\text{S.M.} = \frac{x_{\text{np}} - x_{\text{cg}}}{c}$$

SM = ((3 - 5.55)/5.0625) = -0.503

$$\text{V}_\text{h} \equiv \frac{S_\text{h}\, \ell_\text{h}}{S\, c}$$

V_H = (14*5.25)/(131.625)(5.0625) = 0.1103

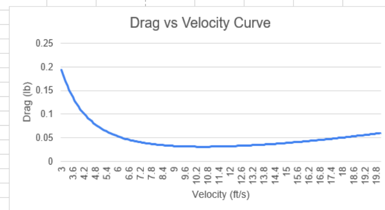
$$\frac{x_{\text{np}}}{c} \simeq \frac{1}{4} + \frac{1 + 2/AR}{1 + 2/AR_\text{h}} \left(1 - \frac{4}{AR + 2}\right) \text{V}_\text{h}$$

X_{np} = 0.25 + (((1)+(2/5.135))/((1+2/2.6))(1-4/5.135 +2)(0.1103)) = ~3

Appendix B: Calculations to plot Drag and Thrust v. Velocity Curves

F5 =2*$B$1/($B$2*E5^2*$B$3)

	A	B	E	F	J	K	L	M	O
1	W (lb)	0.145505							
2	p	0.002377							
3	A	0.9140625	V	Cl	T	Cd02	Cd2	D2	
4	k	0.08854	0	#DIV/0!	0.5472	0.12789	#DIV/0!	#DIV/0!	
5			0.1	13393.8	0.54484	0.12789	15883463.1	172.552	
6			0.2	3348.44	0.54246	0.12789	992716.566	43.138	
7			0.3	1488.2	0.54007	0.12789	196092.264	19.1725	
8			0.4	837.111	0.53766	0.12789	62044.9053	10.7845	min D
9			0.5	535.751	0.53523	0.12789	25413.6687	6.90212	0.0309678
10			0.6	372.049	0.53278	0.12789	12255.8864	4.79316	
11	V	10.6	0.7	273.342	0.53031	0.12789	6615.48108	3.52154	
12	Cl	1.1920408	0.8	209.278	0.52782	0.12789	3877.92648	2.69622	
13			0.9	165.355	0.52532	0.12789	2421.01846	2.13039	
14			1	133.938	0.5228	0.12789	1588.47419	1.72566	
15			1.1	110.692	0.52026	0.12789	1084.98979	1.42622	
16			1.2	93.0123	0.5177	0.12789	766.112796	1.19848	
17			1.3	79.2531	0.51513	0.12789	556.252081	1.02125	
18			1.4	68.3356	0.51254	0.12789	413.587464	0.88064	
19			1.5	59.5279	0.50993	0.12789	313.875307	0.76721	
20			1.6	52.3194	0.5073	0.12789	242.490302	0.67439	
21			1.7	46.3452	0.50465	0.12789	190.301176	0.59747	
22			1.8	41.3388	0.50198	0.12789	151.43355	0.53302	
23			1.9	37.1019	0.4993	0.12789	122.007426	0.47849	
24			2	33.4844	0.4966	0.12789	99.3995338	0.43194	
25			2.1	30.3714	0.49388	0.12789	81.798917	0.39189	
26			2.2	27.6731	0.49114	0.12789	67.9317584	0.35719	
27			2.3	25.319	0.48839	0.12789	56.8867745	0.32692	
28			2.4	23.2531	0.48562	0.12789	48.0019466	0.30037	
29			2.5	21.43	0.48283	0.12789	40.7895553	0.27695	
30			2.6	19.8133	0.48002	0.12789	34.8856519	0.25619	
31			2.7	18.3728	0.47719	0.12789	30.0154279	0.23771	
32			2.8	17.0839	0.47434	0.12789	25.9691134	0.22118	
33			2.9	15.926	0.47148	0.12789	22.5849658	0.20634	
34			3	14.882	0.4686	0.12789	19.7371036	0.19297	
35			3.1	13.9373	0.4657	0.12789	17.3267009	0.18089	



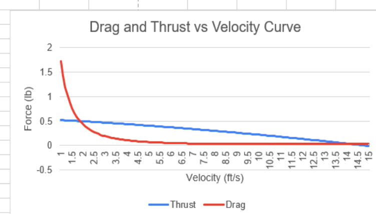
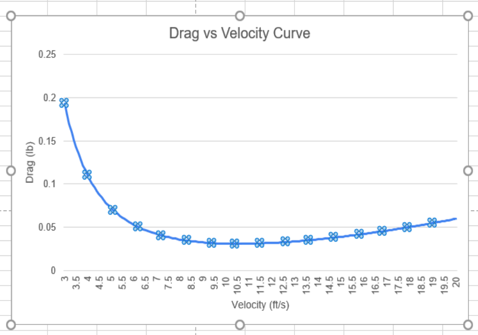


Chart 1 =SERIES(,'Drag and Thrust Curves'!$E$34:$E$204,'Drag and Thrust Curves'!$M$34:$M$204,1)

	E	F	J	K	L	M
34	3	14.882	0.4686	0.12789	19.7371036	0.19297
35	3.1	13.9373	0.4657	0.12789	17.3267009	0.18089
36	3.2	13.0799	0.46278	0.12789	15.2755407	0.16993
37	3.3	12.2991	0.45985	0.12789	13.5212467	0.15996
38	3.4	11.5863	0.4569	0.12789	12.0137204	0.15087
39	3.5	10.9337	0.45393	0.12789	10.7124551	0.14256
40	3.6	10.3347	0.45094	0.12789	9.58449378	0.13494
41	3.7	9.78362	0.44793	0.12789	8.60286254	0.12794
42	3.8	9.27546	0.4449	0.12789	7.745361	0.1215
43	3.9	8.8059	0.44186	0.12789	6.99362075	0.11556
44	4	8.37111	0.4388	0.12789	6.33236774	0.11007
45	4.1	7.96774	0.43572	0.12789	5.74884061	0.10498
46	4.2	7.59284	0.43262	0.12789	5.23232919	0.10027
47	4.3	7.24379	0.42951	0.12789	4.77380622	0.09589
48	4.4	6.91827	0.42638	0.12789	4.36563178	0.09182
49	4.5	6.61421	0.42323	0.12789	4.00131491	0.08802
50	4.6	6.32976	0.42006	0.12789	3.67532028	0.08449
51	4.7	6.06327	0.41687	0.12789	3.38291077	0.08118
52	4.8	5.81327	0.41366	0.12789	3.12001854	0.07809
53	4.9	5.57841	0.41044	0.12789	2.88313914	0.0752
54	5	5.35751	0.4072	0.12789	2.66924408	0.07249
55	5.1	5.14947	0.40394	0.12789	2.47570834	0.06995
56	5.2	4.95332	0.40066	0.12789	2.30025012	0.06757
57	5.3	4.76816	0.39737	0.12789	2.14088046	0.06533
58	5.4	4.5932	0.39406	0.12789	1.99586112	0.06323
59	5.5	4.42769	0.39073	0.12789	1.86366903	0.06124
60	5.6	4.27097	0.38738	0.12789	1.74296646	0.05938
61	5.7	4.12243	0.38401	0.12789	1.63257563	0.05762
62	5.8	3.9815	0.38062	0.12789	1.53145724	0.05597
63	5.9	3.84768	0.37722	0.12789	1.43869216	0.05441
64	6	3.72049	0.3738	0.12789	1.35346585	0.05293
65	6.1	3.59951	0.37036	0.12789	1.27505491	0.05154
66	6.2	3.48433	0.3669	0.12789	1.20281568	0.05023
67	6.3	3.3746	0.36343	0.12789	1.13617428	0.04899
68	6.4	3.26996	0.35994	0.12789	1.07461817	0.04782

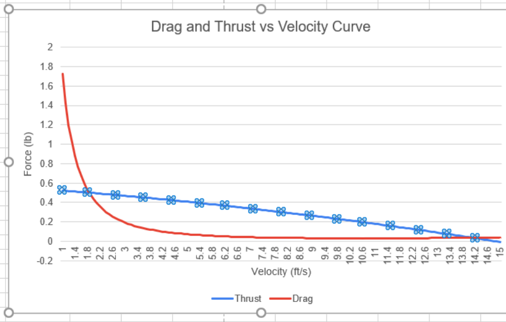


Drag and Thrust Curves | Flight Tests | Pre-Flight + Post-Flight Calcul | Aircraft Measurements

Chart 2

=SERIES("Thrust",'Drag and Thrust Curves'!$E$14:$E$154,'Drag and Thrust Curves'!$J$14:$J$154,1)

	E	F	J	K	L	M
16	1.2	93.0123	0.5177	0.12789	766.112796	1.19848
17	1.3	79.2531	0.51513	0.12789	556.252081	1.02125
18	1.4	68.3356	0.51254	0.12789	413.587464	0.88064
19	1.5	59.5279	0.50993	0.12789	313.875307	0.76721
20	1.6	52.3194	0.5073	0.12789	242.490302	0.67439
21	1.7	46.3452	0.50465	0.12789	190.301176	0.59747
22	1.8	41.3388	0.50198	0.12789	151.43355	0.53302
23	1.9	37.1019	0.4993	0.12789	122.007426	0.47849
24	2	33.4844	0.4966	0.12789	99.3995338	0.43194
25	2.1	30.3714	0.49388	0.12789	81.798917	0.39189
26	2.2	27.6731	0.49114	0.12789	67.9317584	0.35719
27	2.3	25.319	0.48839	0.12789	56.8867745	0.32692
28	2.4	23.2531	0.48562	0.12789	48.0019466	0.30037
29	2.5	21.43	0.48283	0.12789	40.7895553	0.27695
30	2.6	19.8133	0.48002	0.12789	34.8856519	0.25619
31	2.7	18.3728	0.47719	0.12789	30.0154279	0.23771
32	2.8	17.0839	0.47434	0.12789	25.9691134	0.22118
33	2.9	15.926	0.47148	0.12789	22.5849658	0.20634
34	3	14.882	0.4686	0.12789	19.7371036	0.19297
35	3.1	13.9373	0.4657	0.12789	17.3267009	0.18089
36	3.2	13.0799	0.46278	0.12789	15.2755407	0.16993
37	3.3	12.2991	0.45985	0.12789	13.5212467	0.15996
38	3.4	11.5863	0.4569	0.12789	12.0137204	0.15087
39	3.5	10.9337	0.45393	0.12789	10.7124551	0.14256
40	3.6	10.3347	0.45094	0.12789	9.58449378	0.13494
41	3.7	9.78362	0.44793	0.12789	8.60286254	0.12794
42	3.8	9.27546	0.4449	0.12789	7.745361	0.1215
43	3.9	8.8059	0.44186	0.12789	6.99362075	0.11556
44	4	8.37111	0.4388	0.12789	6.33236774	0.11007
45	4.1	7.96774	0.43572	0.12789	5.74884061	0.10498
46	4.2	7.59284	0.43262	0.12789	5.23232919	0.10027
47	4.3	7.24379	0.42951	0.12789	4.77380622	0.09589
48	4.4	6.91827	0.42638	0.12789	4.36563178	0.09182
49	4.5	6.61421	0.42323	0.12789	4.00131491	0.08802
50	4.6	6.32976	0.42006	0.12789	3.67532028	0.08449



Drag and Thrust Curves | Flight Tests | Pre-Flight + Post-Flight Calcul | Aircraft Measurements