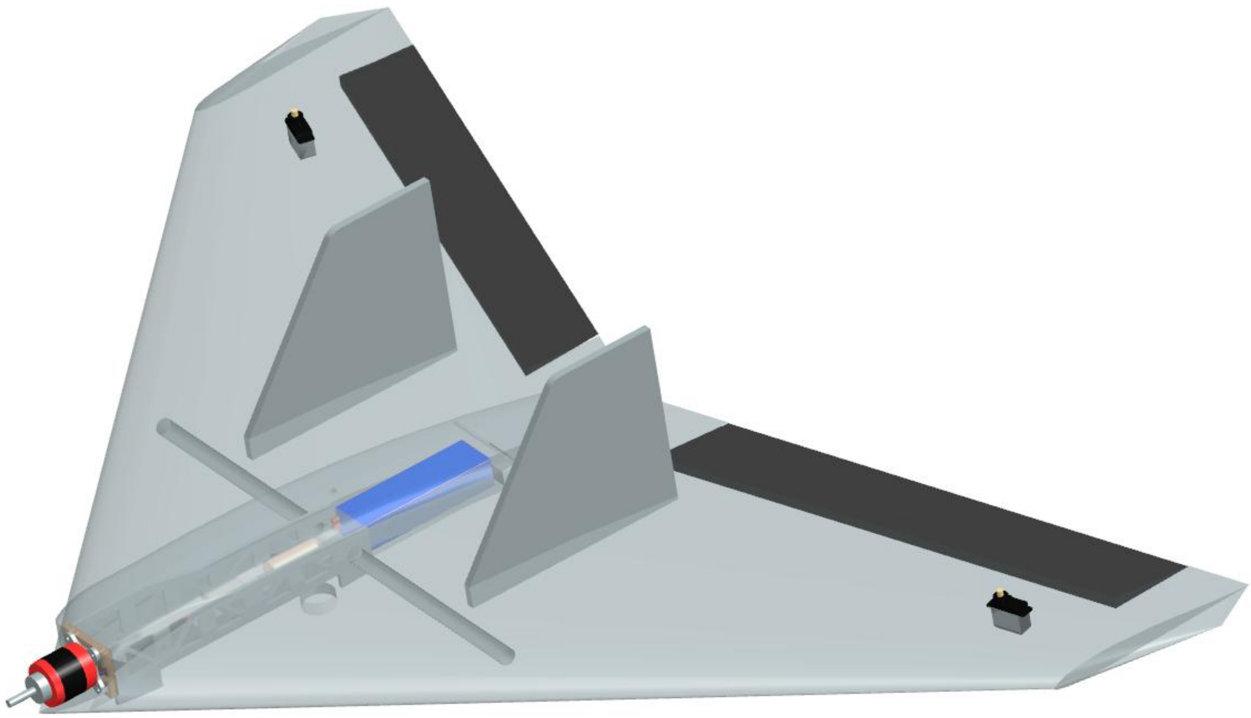
AE 484 Homework 5

by

Justin Abel, Luke Brown, Tyler Gralewski, Patrick Swiatek, Jaden Tran



Submitted on Friday, March 14th, 2025.

All part and assembly drawings are found at the end of this report.

I. Wing Geometry

The team chose to create a low aspect ratio delta wing for a fast, maneuverable aircraft. The taper ratio is close to the manufacturer requirement of 0.3 or greater. The wingspan is short of the max wingspan of 70 in. This reduces the weight of the aircraft dramatically for the size being built. The wing loading is decreased and the thrust to weight much higher. These changes allow the aircraft to fly quickly. The general characteristics of the wing are listed in Table 1.

Table 1: Characteristics of the Wing Planform.

Parameter	**Unit**	**Value**
Airfoil	-	MH64
Aerodynamic Center	in	14.21
Wing Area	sq in	539.88
MAC	in	14.66
LE Sweep	degrees	45
Taper Ratio	-	0.325
Root Chord	in	20.37
Tip Chord	in	6.621
Twist	degrees	0
Aspect Ratio	-	2.96

The wing has an MH64 airfoil designed for RC delta wings. This airfoil provides ample lift for the mission and the aircraft is statically stable in pitch. No twist was added as the team is not concerned with the lift distribution of the wing nor the effects of stall as the aircraft operates below its stall angle of attack.

II. Design of the Aircraft

The overall design of the aircraft was to minimize fuselage wetted area to keep the aircraft as close to a wing shape as possible. The fuselage top was created to follow the airfoil shape. All the electronic components were kept in the centerline of the wing with a narrow fuselage body to reduce fuselage cross-section. In terms of removable panels, the wings are the only component of the design that can be detached. To secure the wings to the design, the team is utilizing two spars that are attached to the fuselage structure. While the vertical stabilizers do originally need to be slotted into the airframe structure, after using epoxy the stabilizers will become a permanent feature of the model. In order to hold the aircraft for hand launching, a cut-out out of the wing will be utilized to have a hand hold. In terms of moving the battery to adjust CG, the model unfortunately does not provide much wiggle room in that regard. Due to the fuselage shape, the battery must remain around its current position where only minimal adjustments can be done. This is suitable for the aircraft as it is sufficiently stable in the longitudinal axis.

III. Force and Moment Balance of the Aircraft

The center of gravity was calculated using a part-by-part buildup of the aircraft centroids in NX. Purchased parts were weighed to find their masses. The manufactured components' masses were found by using the volumes in NX and known material densities from online research or the project description. These accumulated values provided a CG position at 9.7 in from the leading edge of the root of the wing. The mass moments of inertia were calculated using the CG position in NX. The mass values in NX are arbitrary and were scaled by the real masses to find the total moment of inertia for the aircraft. The center of gravities and mass moments of inertia are listed in Table 2. The CG positions are measured from the leading edge of the root of the wing.

Table 2. Mass Properties of the Aircraft.

Part	**Mass (lbm)**	**CGx (in)**	**CGy (in)**	**CGz (in)**	**Ixx (lbm sq in)**	**Iyy (lbm sq in)**	**Izz (lbm sq in)**
Total Aircraft	2.270	9.735	0.006	0.438	78.480	93.677	170.468
Wing	0.333	13.061	0.000	0.166	25.405	9.993	35.297
Elevon	0.155	22.009	0.000	0.074	20.970	20.534	41.468
Winglet	0.025	23.184	0.000	0.049	10.191	3.960	14.145

Part	Mass (lbm)	CGx (in)	CGy (in)	CGz (in)	Ixx (lbm sq in)	Iyy (lbm sq in)	Izz (lbm sq in)
Vertical Stabilizer	0.057	15.184	0.000	2.741	1.979	1.861	2.900
Motor Bulkhead	0.016	1.875	0.000	0.778	0.011	1.247	1.247
Fuselage Top	0.038	10.081	0.000	1.240	0.071	0.856	0.853
Fuselage Side	0.533	5.986	0.000	0.520	0.497	15.560	15.767
Servo	0.054	21.082	0.014	0.284	14.776	5.995	20.761
Spar	0.204	10.185	0.000	0.374	4.359	0.039	4.392
Motor	0.278	0.926	0.000	0.479	0.104	26.048	26.008
Receiver	0.019	6.209	0.000	0.372	0.002	0.299	0.300
Spinner	0.050	0.000	0.000	0.000	0.000	0.000	0.000
Battery	0.404	12.921	0.029	0.450	0.094	2.870	2.904
ESC	0.090	3.742	0.000	0.157	0.013	4.305	4.308
Torsion Pins	0.002	16.000	0.000	0.167	0.007	0.070	0.077
Payload	0.011	9.083	0.102	0.400	0.001	0.038	0.039

IV. Load Paths of the Aircraft

At the center of the model, the fuselage will be the major load-bearing component of the design. The fuselage will be created as a rigid structure in order to be capable of supporting the airframe structure. In terms of forces on the fuselage, the fuselage carries the battery, receiver, ESC, spinner, and motor. Weight from these components will produce load along the fuselage shape. Since the battery weight will have the highest concentrated load, extra reinforcement may be necessary. Additionally, thrust force from the motor will create additional tension force that the fuselage structure will support. Reinforcing the motor mount and ensuring a stiff fuselage box will help maintain structural integrity in these locations. Bending moments due to lift will also be supported by the fuselage. Transferred to the fuselage by the spars, the fuselage-wing attachment point will be a key area of high stress concentration.

The wing structure will be built with a foam core and balsa wood around it. This is done to reduce weight but to also provide stiffness where needed. The foam in the center of the wing mainly aids in maintaining shape and reducing overall weight, while the balsa wood around it provides strength to handle aerodynamic loads and impacts. While there will be a further discussion of spars shortly, the spars help transfer various loads and bending forces to the wing where these stresses are then distributed throughout the wing structure. With the shape of the wing being supported by the stiffness of the balsa wood, the wing structure will be capable of withstanding torsional loads and yawing moments due to control surface deflections and aerodynamic forces. In addition to this, the airframe will also support loading from lift and wing weight along with the shear stresses that occur throughout the wing. Force due to the thrust generated by the motor will also be transferred through the fuselage and spars eventually into the foam wings.

In terms of providing load paths, the two spars will be integrated into the fuselage. For various bending loads such as from torsion and lift, the spars will assist in transferring these stresses to the rigid fuselage structure. Additionally, as previously mentioned, the spars will help transfer other aerodynamic forces and stresses into the wing structure which will then be evenly distributed throughout. The front spar will be the one taking a majority of the bending loads from lift as well as providing bending resistance against torsional loads. The rear spar on the other hand, which is positioned below the vertical stabilizers, is utilized to distribute stresses from yawing, twisting, and bending moments on the vertical stabilizers.

For other components of the design, the vertical stabilizers will endure aerodynamic forces that produce twisting and bending moments. The rear spar helps in transferring these forces into the wing structure. The elevons will introduce stresses at hinge points which may need to be reinforced to prevent wear.

To summarize the major regions of where stress concentrations may be the largest concern, the fuselage-wing attachment point (the wing root location) will be a key location to reinforce. More specifically, the spar attachment points will have a majority of stress concentration on them. In this region, shear stress will need to be monitored to ensure the structural integrity of the design.

Pictured below are various sketches illustrating loading and stresses on the design. In the sketches, T represents thrust, L represents lift, D represents drag, M represents a bending moment, and S represents shear stress. Additionally, an image showcase a load path diagram can be found below illustrating how the force from the motor transfers through the motor plate through the fuselage truss into the spar where it then goes into the foam wing.

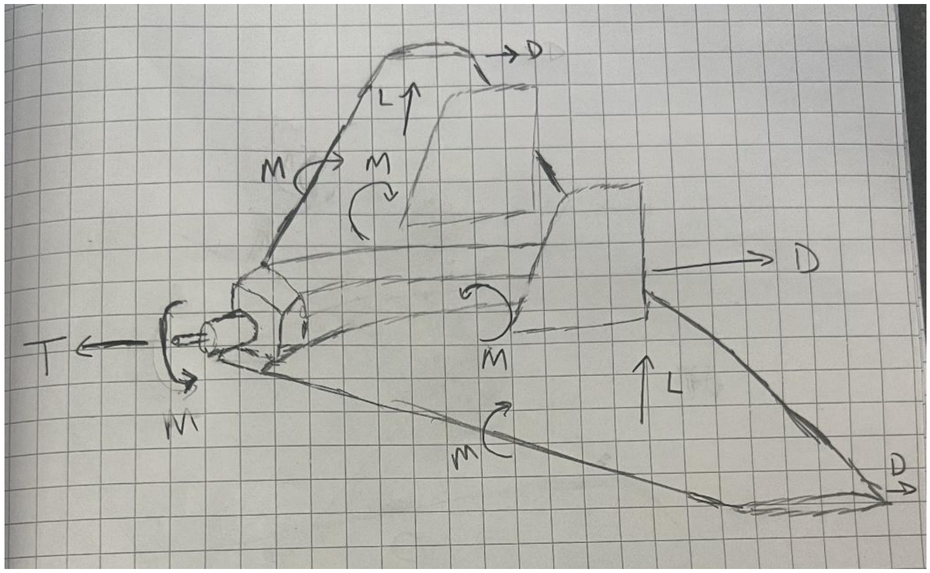


Figure 1: View of Loads and Moments

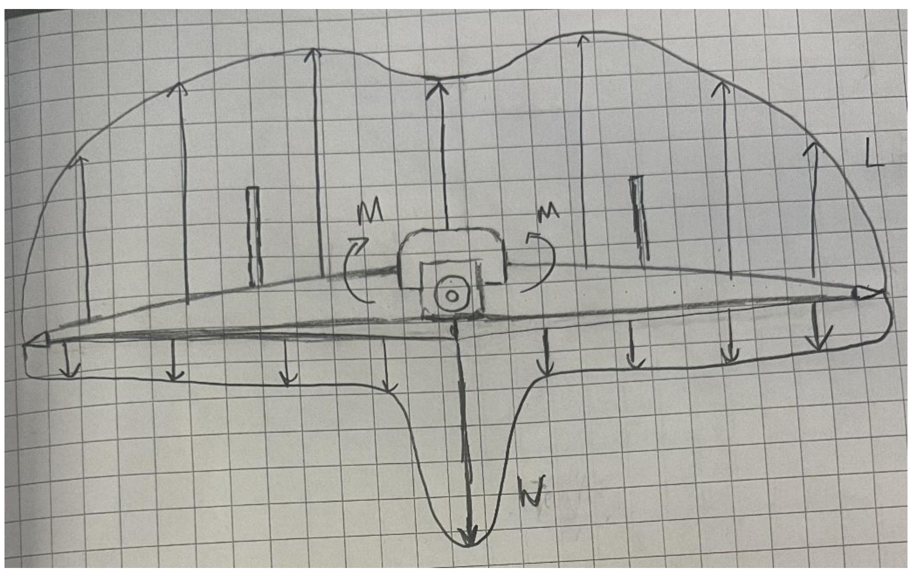


Figure 2: Front View of Lift and Weight Loading with Lift Bending Moment

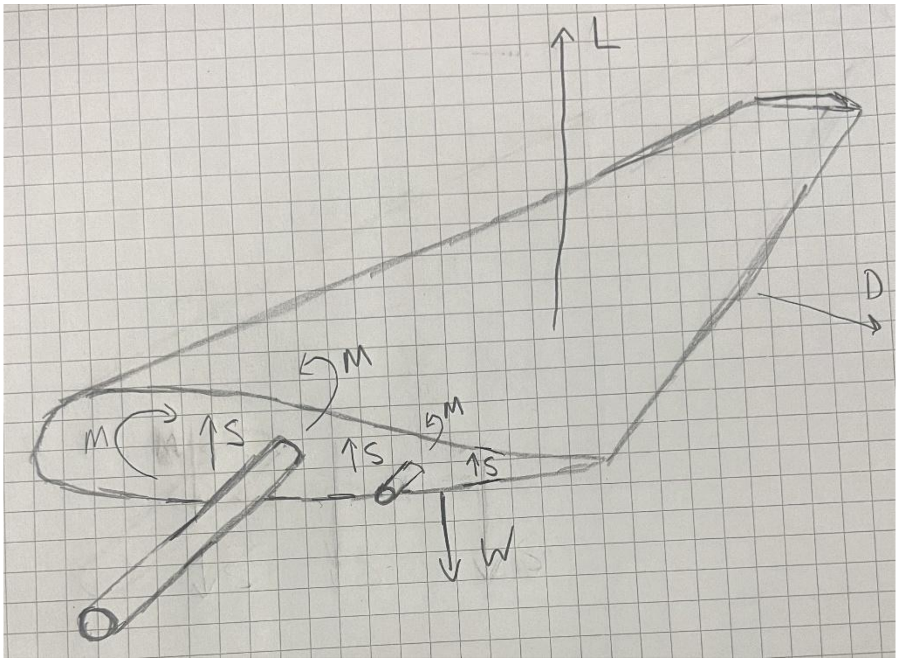


Figure 3: Cross-Sectional View of Wing and Spar Shear and Moment Stresses

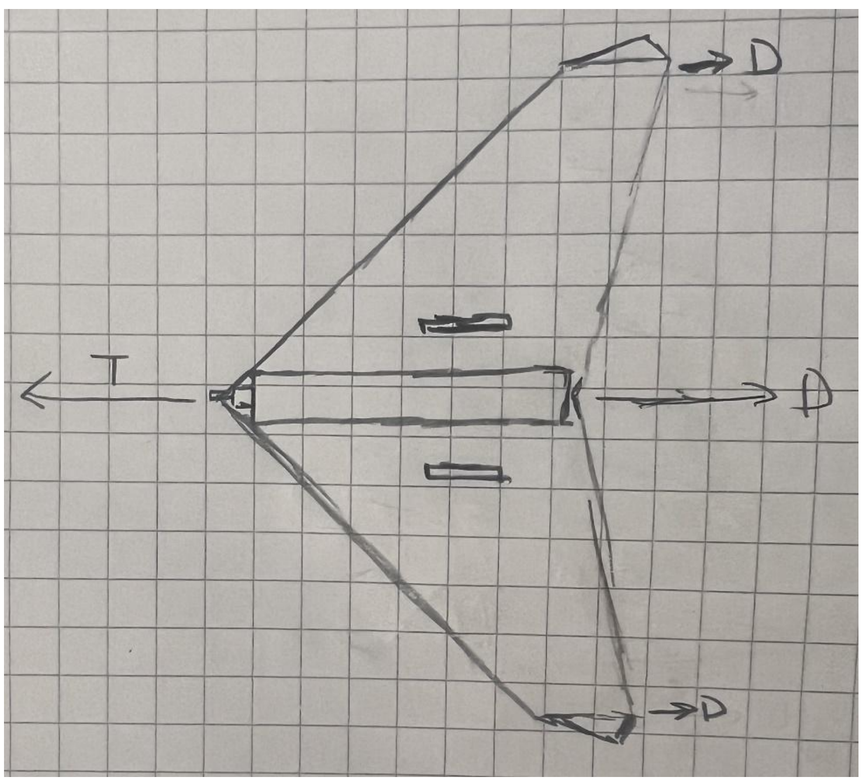


Figure 4: Top View of Model

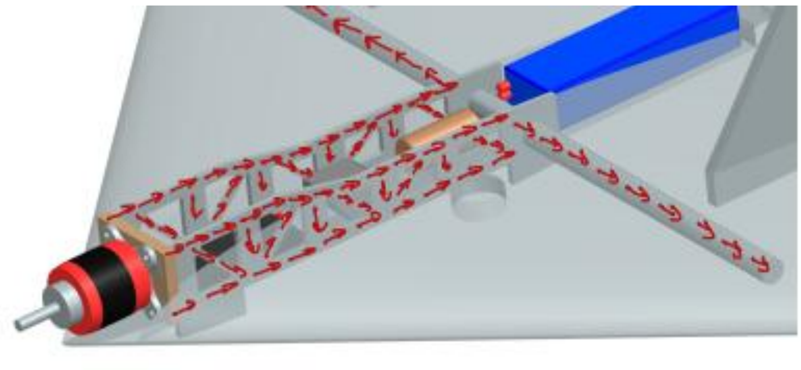


Figure 5: Motor Load Path Diagram

V. Sizing of the Vertical Stabilizers

Table 3: Planform Characteristic of the Vertical Stabilizer.

Vertical Stabilizer	
Root Chord (in)	7
Tip Chord (in)	3.05
Taper Ratio	0.44
Width (in)	0.125
Height(in)	5 (5.7 if including extension into fuselage)
Volume (in^3)	15.32
Planform Area (in^2)	30.03
Mean Aerodynamic Chord (in)	5.28
Mean Chord Length (in)	6.01

To determine the planform area of the vertical stabilizer, the following equation was utilized:

$$S_{VT} = \frac{V_{VT}\, b_w\, S_w}{L_{VT}} \quad (1)$$

Where V_{VT} represents the vertical tail volume ratio, L_{VT} represents the vertical tail moment arm length (¼ wing chord to ¼ vertical tail chord), S_w represents the wing planform area, and b_w represents the wingspan. Since two vertical stabilizers were incorporated into the design, calculations were split up into two halves. From the lecture slides, V_{VT} is equal to around 0.04 - 0.05 based on AE 498 2012 data [1]. 0.04 was selected for the design. Based on the team's initial drawings and through the use of an online measuring tool, $L_{VT}$is equal to 7.19 in. As previously mentioned, to account for the two vertical stabilizers, the half-span (20 in) and half-wing planform area (269.94 in^2) values are utilized. Plugging in numbers and solving, S_{VT} is equal to 30.03 in^2.

To verify this result, the lecture slides were referred to once again. From Chevalier, $\frac{S_{VT}}{S_W}$ should equal between 0.06 and 0.1 [1]. Calculating $\frac{S_{VT}}{S_w}$ for our design using 30.03 in^2 for S_{VT} and 269.94 in^2 for S_w, we find a ratio of 0.11. While slightly outside the range, the model's ratio is only 0.01 outside of the desired range. Through this analysis, the team settled on a vertical stabilizer planform area of 30.03 in^2.

To now determine the dimensions of the vertical stabilizers some values were selected to constrain the stabilizers. Based upon initial design drawings, a root chord of 7 in was utilized. The team also looked for a taper ratio between 0.3 to 0.6 for the vertical stabilizer. Attempting to find a taper ratio somewhere in the middle of this range, the team utilized a tip chord of 3.3 in. This resulted in a taper ratio of 0.47 for the vertical stabilizer.

The directional stability from the vertical stabilizer was calculated using the method in Nelson [2, pg. 76] and estimates for the lift-curve slope from McCormick [3, pg. 526]. The value for C_{n_β} is 0.057. This positive, indicating directional stability.

VI. Stability Analysis of the Aircraft

The team's goal was to achieve a static margin (SM) between 20-30% MAC. This range is desirable because as the team is designing a hobbyist aircraft with multiple first-time builders/designers. A higher static margin range limits overall maneuverability but provides a cushion for the pilot and for error propagation in design assumptions. For experienced designers and builders, a static margin range between 10-20% MAC would be acceptable as the pedigree of design and manufacturing of this type of aircraft would exist. For a design that is new to the team, a higher static margin range provides a safety factor in design. In order to do so, the CG had to be placed in front of the Neutral point (NP), which from the wing sizing homework (homework 4), the NP was around 14 inches from the leading edge (LE) of the wing. To find the SM of a flying wing, an equation utilizing NP, CG, and MAC can be used, as seen below as Eq. (2). As mentioned in Section I, the MAC is around 14.66. Therefore, using equation Eq. (2) to solve for CG knowing NP and the desired range of %MAC, the CG should be placed between 9.602 to 11.068" from the LE of the wing.

$$SM = \frac{NP - CG}{MAC} * 100\% \quad (2)$$

To evaluate the NP of the wing within XFLR5, the geometry from Section I and weight outlined in section III was utilized. Specifically, the wing was assigned its respective mass and the mass of the rest of the components were assigned as a point mass at a location iterated until the wings CG from Section III was obtained. Then a fixed speed analysis was ran at 52 ft/s (approximately 35 mph), with an AOA range of -5 degrees to 18 degrees. The resulting Neutral point was at 14.208 in from the LE. To find the NP as a percentage of MAC, the NP was simply divided by the MAC and converted to a percentage. The longitudinal SM was then calculated using Eq. (2) and can be seen below in 4. Then the provided Flying Wing software was utilized to find the wings CG which could later be used in Eq. (2) to find its SM in %MAC once again.

To generate an accurate model within the Flying Wing calculator, the wings geometry was first input from Section I. Then the mass and the wing cores CG from Section III was input and to account for the rest of the missing weight from the various components, a part mass was added. This was added as the remaining aircraft weight, around 1.937 lbs, as the shape of the Fuselage Cap along with the Caps CG position as listed in Section III. A screenshot of the inputs and the resulting outputs from the online calculator can be seen below in Fig. 6. Using the resulting CG output from the calculator along with the previous MAC and the NP from XFLR5 within Eq. (2), the SM can be calculated and can be seen below in 4.

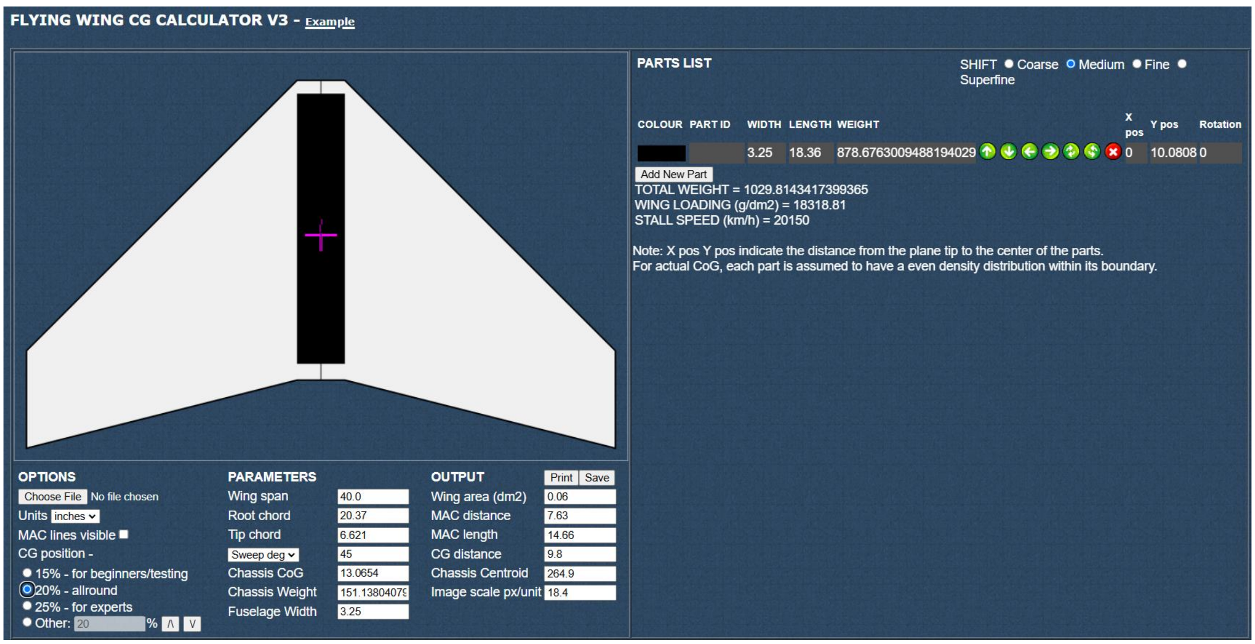


Figure 6: Screenshot of inputs and outputs from the Flying Wing Calculator

Table 4: Longitudinal Static Stability of the Aircraft.

	XFLR5	Flying Wing Calculator
NP [in]	14.208	14.208
CG [in]	9.7347	9.8000
MAC [in]	14.660	14.660
NP [%MAC]	96.917	96.917
SM [%MAC]	30.514	30.068

From the above SM in %MAC results, both software produced inputs into Eq. (2) that produced very similar SM in %MAC. The CG from the Flying Wing calculator was within 0.07in of the CG produced from Section III which was input into XFLR5. This along with the fact that the same NP value and MAC length was used to calculate the SM in %MAC for both software explains how the two produced very similar results.

Using the same simulation that was used to obtain the NP of the wing in XFLR5, plots of C_D, C_L, C_L/C_D, and C_m vs Alpha were obtained and can be seen below as Figs. 7-10.

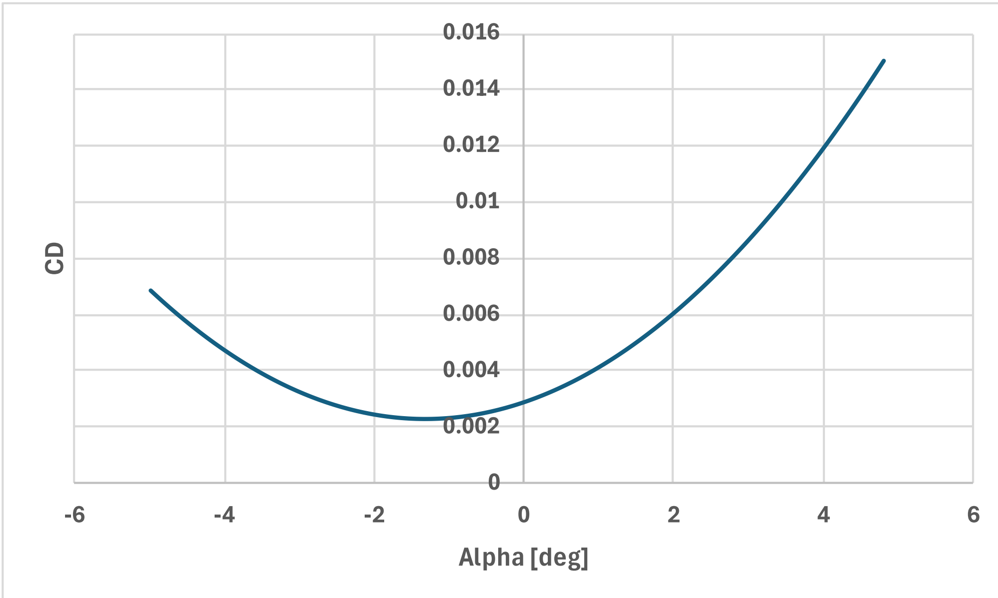


Figure 7: Drag coefficient at different angles of attack

The plot of the Drag coefficient versus angle of attack shown in Fig. 7 above shows a minimum drag at about -1 degrees AoA. But based on the maximum L/D from Fig. 9 and its corresponding AoA of about 1 degree, the drag can be estimated at about 40 counts. This drag value is not the minimum but it is only roughly 20 counts higher, which to obtain a large L/D the team feels it is a good trade. Considering the large power to weight ratio explained below in Section VII, the relative low drag estimate at a 1 degree AoA should not negatively impact the performance of the teams design. Therefore, the drag versus AoA plot shows how the overall drag from a clean wing configuration remains relatively low at small angles of attack, which the aircraft will likely be flying at during cruise.

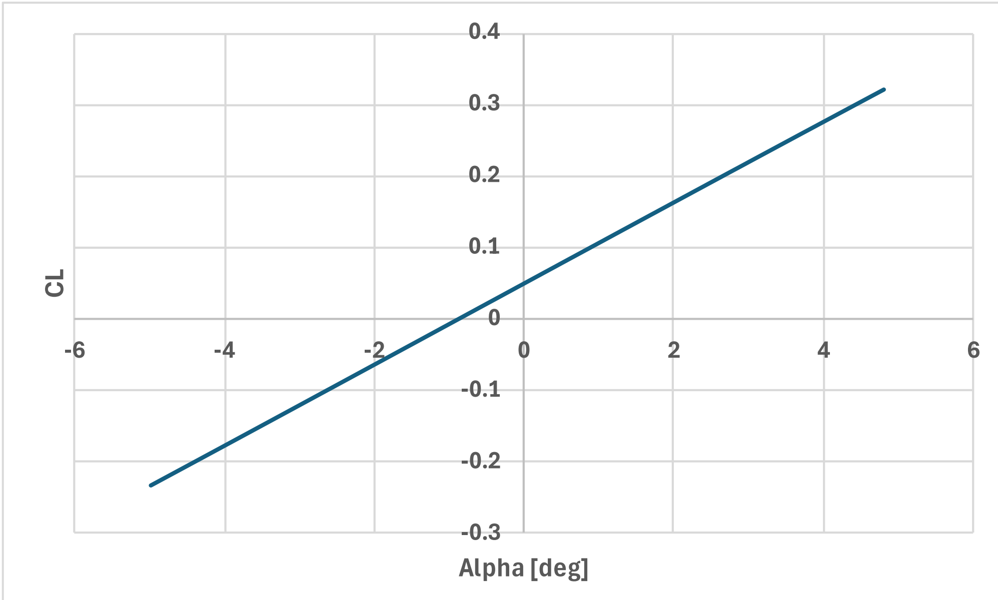


Figure 8: Lift coefficient at different angles of attack

The lift versus angle of attack curve has a lower lift-curve slope than the maximum of 2π. From thin air foil theory, the slope is about 2. Since the aircraft has a high power to weight ratio, the aircraft doesn't need a very optimal lift curve as the thrust can handle less aerodynamic performance. The aircraft is very light so the required CL is low for cruise also making the lift curve acceptable.

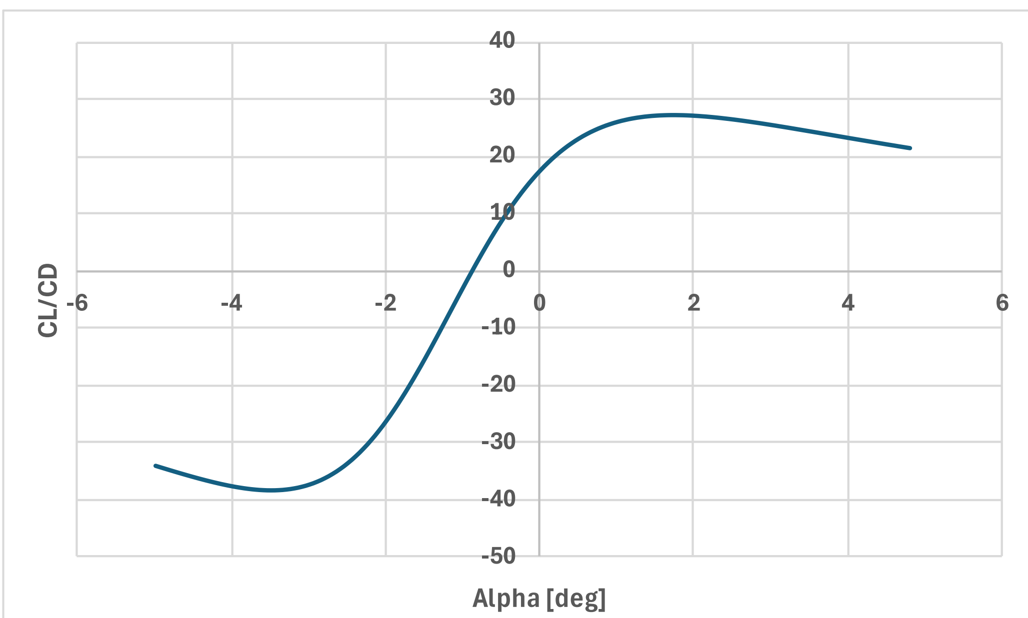


Figure 9: Lift to Drag ratio at different angles of attack.

The aircraft has a high lift-to-drag ratio of 16 at zero angle of attack and 25 around a cruising angle of attack of 2°. These values are in the range for a high aspect ratio glider. These values are inflated as the XFLR5 model only accounts for the wing and wingtips. The vertical stabilizers and fuselage will create a large addition to the parasitic drag of the aircraft. The servo/control arms will create protuberance drag that will add to the parasitic drag. The elevator deflection required from Fig. 4_5 will create trim drag on the aircraft as well.

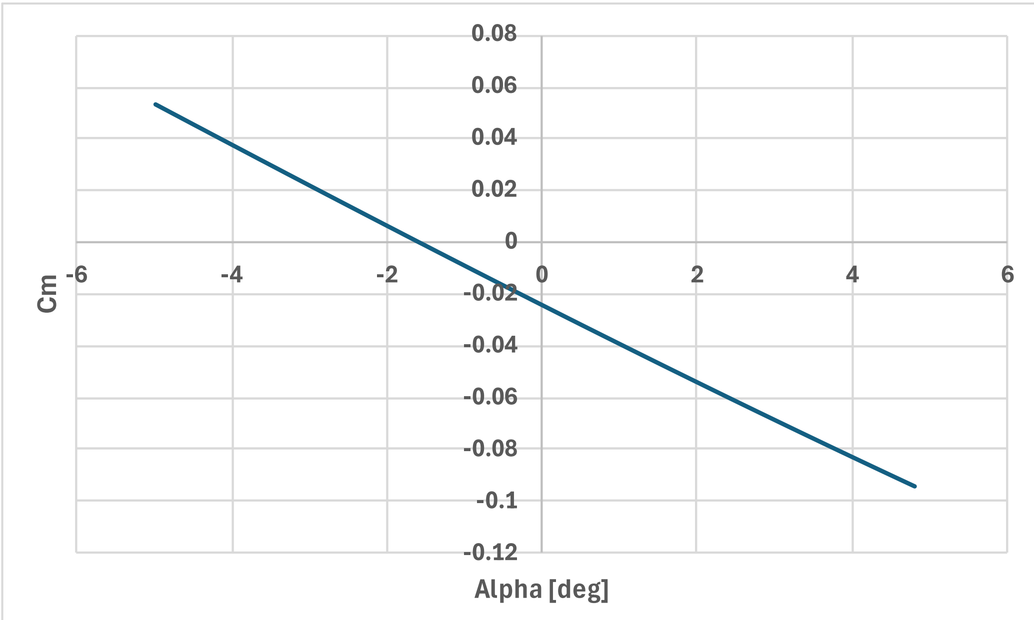


Figure 10: Moment coefficient at different angles of attack

The pitching moment versus angle of attack for the aircraft has a negative slope meaning the aircraft is statically stable in the longitudinal axis. The aircraft is untrimmable with no control surface deflection as the pitching moment is negative at zero angle of attack. This can be fix with elevon deflections during flight. Since the elevons consist of ~75% of span, there should be enough $C_{m\delta e}$ to correct the pitching moment.

VII. Performance of the Aircraft

Table 5: Performance Characteristics of the Aircraft.

Total weight of the aircraft	2.27 lbs.
Power to Weight Ratio	198.3 watts/lb.
Wing Loading	9.69 oz/ft^2

In order to determine the total weight of the aircraft two methods were used. Given parts, such as the motor and battery, were weighed directly using a digital scale, and then converted from grams to pound mass. For fabricated parts, the total volume for each part provided by NX was recorded. Then, the pound mass for each component was determined by multiplying the estimated density of the material of the part by its volume. From combining the measured or estimated weight for each component, a total weight of about 2.27 pounds was obtained for our aircraft. The power to weight and wing loading are listed in Table 5. The power to weight ratio falls within the range of ~50 to 200 watts/lb. suggested in Professor Elliott's sides, thus making it a reasonable value for our aircraft. This is on the very high end; this meets the team's goal creating a fast delta-wing aircraft. The wing loading falls very close to what is generally considered a "soaring glider", which has a wing loading of 9.7 oz/ft$^2 \pm 4$. This is the range that we were aiming to achieve when designing, making it a reasonable calculated value.

Sample Calculations

1. Center of Gravity (Moment Balance Method)

$$CG = \frac{\sum (m_i \cdot x_i)}{\sum m_i}$$

- CG = center of gravity [in]
- m_i = mass of each component [lbm]
- x_i = location of that component's centroid along a chosen reference axis [in]

$$CG = \frac{4 \cdot 13.5 + 3 \cdot 11.2}{4 + 3} = 12.51\ in$$

2. Taper Ratio

$$\lambda = \frac{C_{tip}}{C_{root}}$$

- λ = taper ratio
- C_{root} = root chord [in]
- C_{tip} = tip chord [in]

$$\lambda = \frac{3}{7} = 0.429$$

3. Mean Aerodynamic Chord Calculation

$$MAC = \frac{2}{3} \cdot C_{root} \frac{(1 + \lambda + \lambda^2)}{(1 + \lambda)}$$

- MAC = mean aerodynamic chord [in]
- C_{root} = root chord [in]
- λ = taper ratio [in]

$$MAC = \frac{2}{3} \cdot 7 \frac{(1 + 0.4 + 0.4^2)}{(1 + 0.4)} = 5.2\ in$$

4. Static Margin

$$SM = \frac{X_{NP} - X_{CG}}{MAC} \times 100\%$$

- SM = Static Margin (% of MAC)
- X_{NP} = Location of the neutral point [in]
- X_{CG} = Location of the center of gravity [in]
- MAC = Mean Aerodynamic Chord [in]

$$SM = \frac{13 - 9.5}{14} \times 100\% = 25\%$$

5. Vertical Tail (Stabilizer) Planform Area Calculation

$$S_{VT} = \frac{V_{VT}\, b_w\, S_w}{L_{VT}}$$

- S_{VT} = vertical tail planform area [in^2]
- V_{VT} = vertical tail volume ratio
- b_w = wing span [in]
- S_w = wing planform area [in^2]
- L_{VT} = vertical tail length [in]

$$S_{VT} = \frac{0.04 \cdot 20 \cdot 250}{10} = 20\ in^2$$

6. Lift-Curve Slope of the Vertical

$$C_{L\alpha v} = 1.6 \frac{2\pi}{1 + \frac{2\pi}{AR_{VT}}}$$

- $C_{L\alpha v}$ = lift-curve slope of the vertical [1/rad]
- AR_{VT} = aspect ratio of the vertical [1/rad]

$$C_{L\alpha v} = 1.6 \frac{2\pi}{1 + \frac{2\pi}{2.96}} = 1.53\ [\frac{1}{rad}]$$

7. Combined Sidewash and Tail Efficiency

$$\eta_v \left(1 + \frac{d\sigma}{d\beta}\right) = 0.724 + 3.06 \frac{S_{VT}/S_W}{1 + \cos \Lambda_{c/4w}} + 0.4 \frac{z_w}{d} + 0.009 AR_W$$

- $\eta_v \left(1 + \frac{d\sigma}{d\beta}\right)$ = combined tail efficiency and sidewash
- S_{VT} = vertical tail planform area [in^2]
- S_W = wing planform area [in^2]

- $\Lambda_{c/4w}$ = quarter-chord sweep of wing [degrees]
- z_w = distance, parallel to the z-axis, from quarter chord of the root of the wing to fuselage centerline [in]
- d = maximum fuselage depth [in]
- AR_W = aspect ratio of the wing

$$\eta_v\left(1+\frac{d\sigma}{d\beta}\right) = 0.724 + 3.06\frac{(0.11)}{1+\cos(39.6)} + 0.4(0) + 0.009(2.96) = \ 0.941$$

8. Change in Yawing Moment due to Sideslip

$$C_{n_\beta} = V_{VT}\eta_v C_{L\alpha v}(1+\frac{d\sigma}{d\beta})$$

- C_{n_β} = change in yawing moment due to sideslip [1/rad]
- V_{VT} = vertical tail volume ratio
- η_v = tail efficiency factor
- $(1+\frac{d\sigma}{d\beta})$ = sidewash parameter
- $C_{L\alpha v}$ = lift-curve slope of the vertical [1/rad]

$$C_{n_\beta} = (0.04)(1.53)(0.916) = 0.056\ [\frac{1}{rad}]$$

<u>References</u>

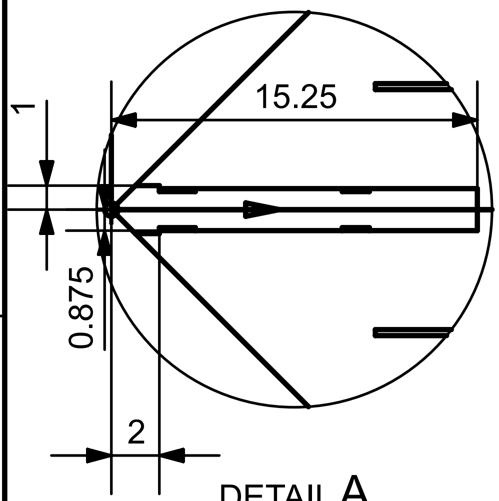
[1] Elliott, Greg. *AE 484 Lecture 06: UAV Design, Fabrication, and Performance Analysis*, 2025. University of Illinois Urbana-Champaign.

[2] Nelson, R., *Flight Stability and Automatic Control* ¸ 1997, McGraw-Hill.

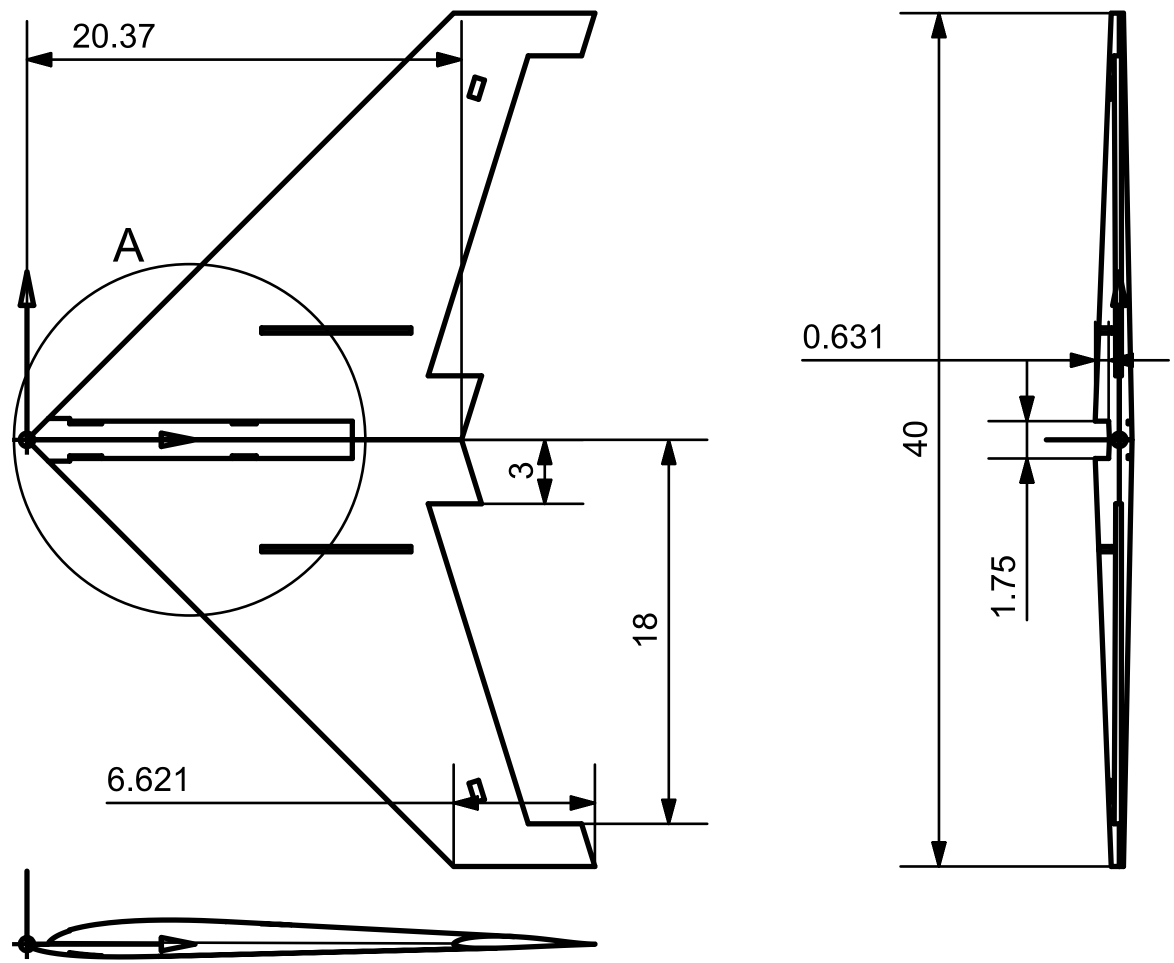
[3] McCormick, B. W., *Aerodynamics, Aeronautics, and Flight Mechanics*, John Wiley and Sons, 1979.

Table of Contributions:

Tyler Gralewski	- CAD for Fuselage Side plate, helped adjust wing cutout cad - CAD for Fuselage Top Piece - Section VI, III
Justin Abel	- Assembly and Drawings - CAD of Wing, Winglets, Elevons, Integration of Components - Cnbeta calculation - Section I, III, VI
Patrick Swiatek	- CAD of Vertical Stabilizer - Sketches and Load Path Diagram - Section II, IV, V
Jaden Tran	- CAD of spar - CAD of torsion pin
Luke Brown	- CAD for all non-fabricated components (motor, battery, etc.) - CAD for the motor mount/bulkhead - Individual part drawings - Section VII



DETAIL A
SCALE 1:8



NOTE: Detailed dimensions for complex geometries will be provided electronically

ALL DIMENSIONS IN INCHES

SIEMENS

THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

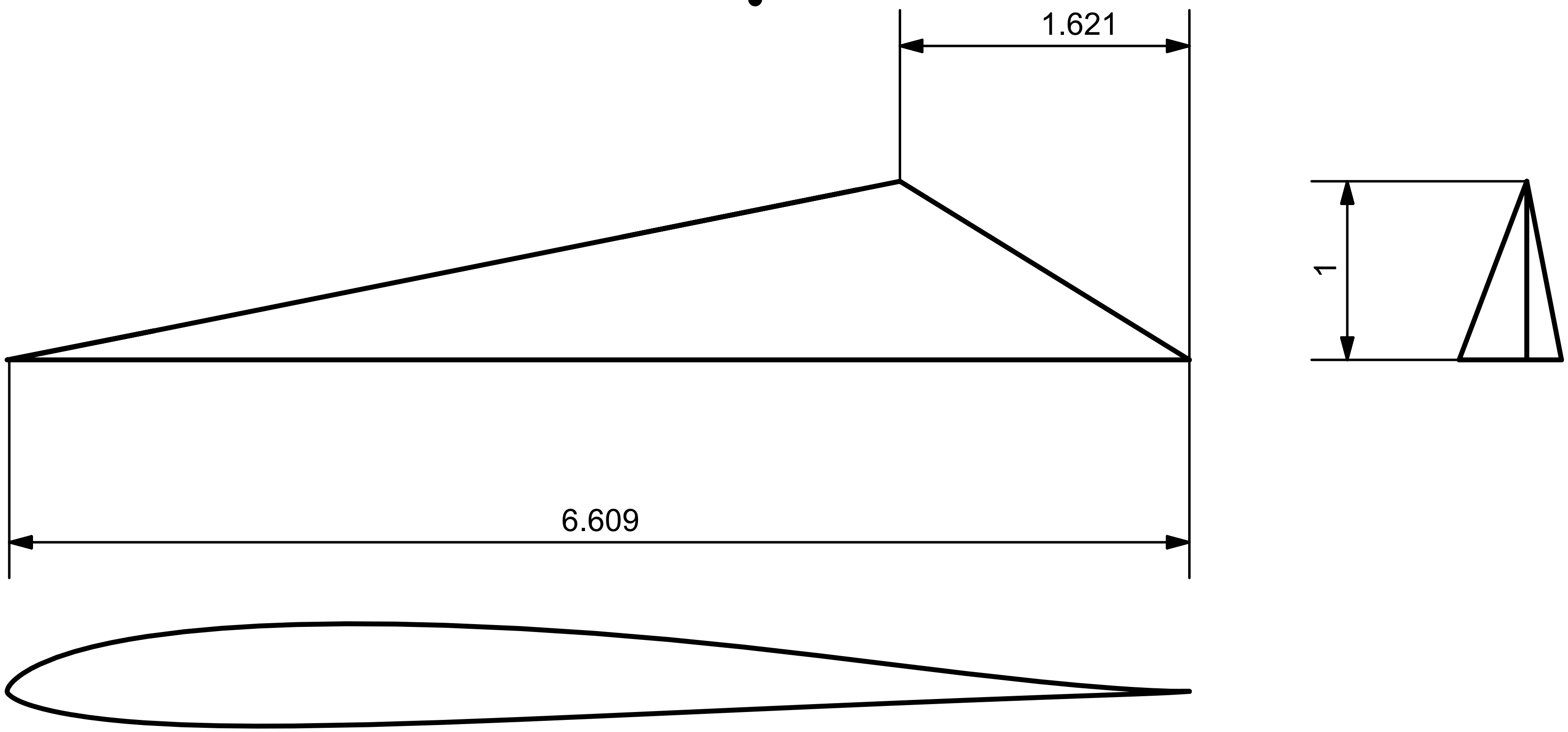
TEAM	1
DRAWN BY	
CHECKED BY	
QTY	1
MATERIAL	FOAM/BALSA

TITLE

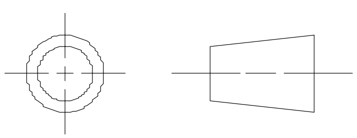
Wings

SIZE	DRG NO.	SHEET REV
A		A

SCALE 1:9		SHEET 1 OF 1

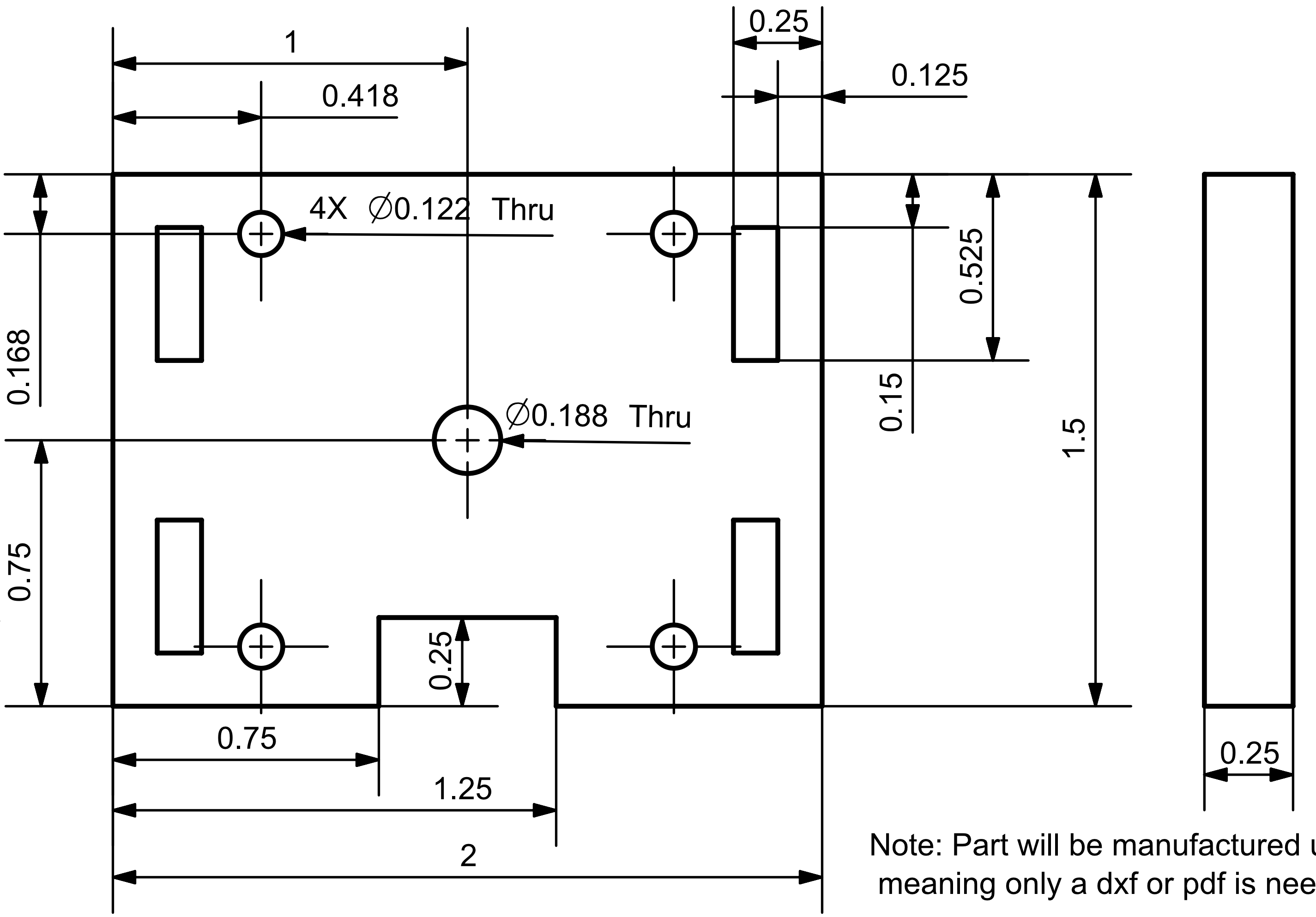


NOTE: left winglet is mirrored

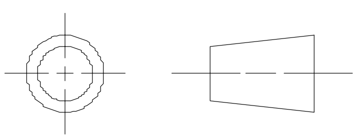


ALL DIMENSIONS IN INCHES

SIEMENS		THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE
TEAM	1	TITLE
DRAWN BY		
CHECKED BY		Winglet (right)
QTY	2	SIZE A / DRG NO. / SHEET REV A
MATERIAL	BALSA	
		SCALE 1:1 / SHEET 1 OF 1



Note: Part will be manufactured using a laser cutter, meaning only a dxf or pdf is needed to manufacture



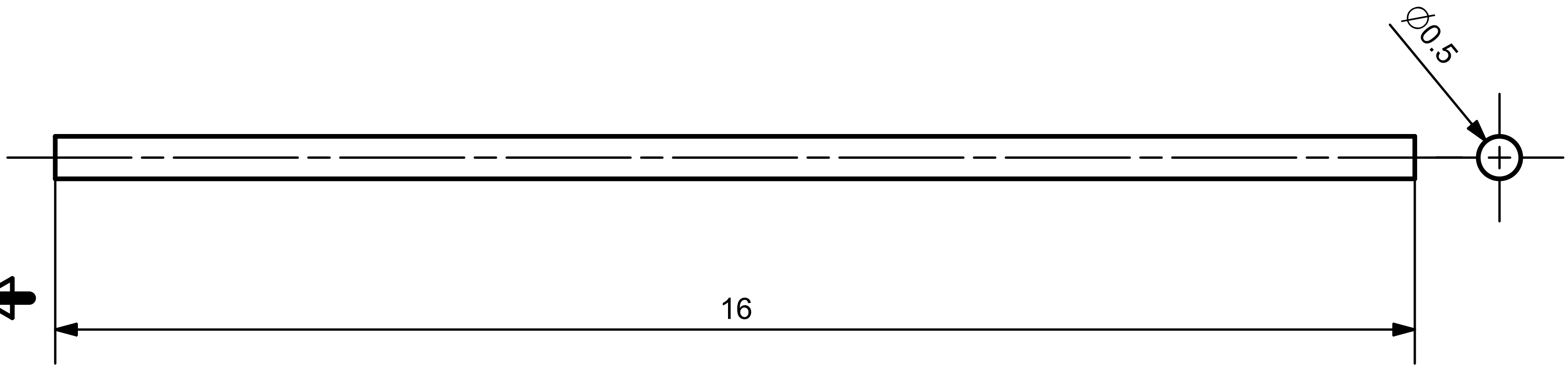
ALL DIMENSIONS IN INCHES

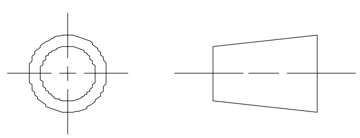
SIEMENS	THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE
Team	1
DRAWN BY	
CHECKED BY	
QTY	1
Material	Plywood

TITLE

Motor Bulkhead

SIZE	DRG NO.	SHEET REV
A		A
SCALE 2:1		SHEET 1 OF 1





ALL DIMENSIONS IN INCHES

SIEMENS

THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

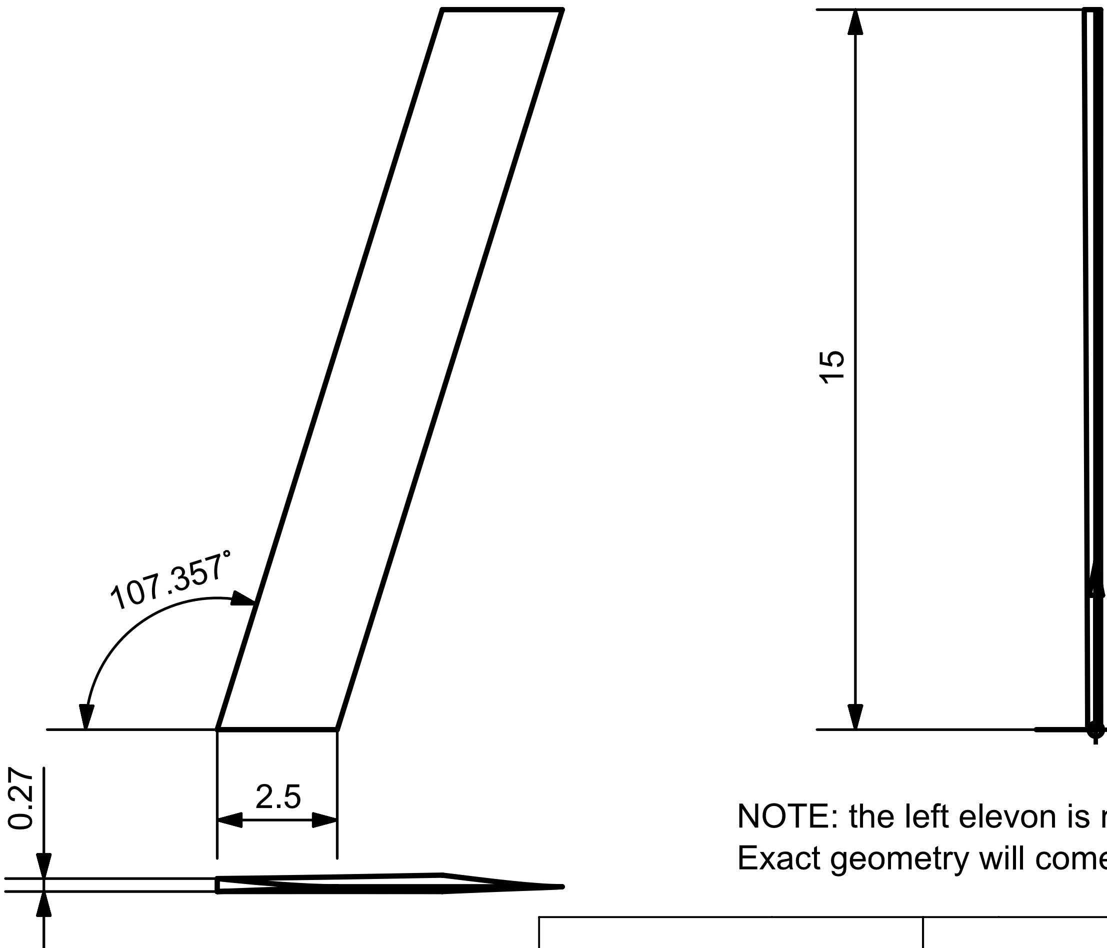
TEAM	1
DRAWN BY	
CHECKED BY	
QTY	1
MATERIAL	CARBON FIBER ROD

TITLE

Spar

SIZE	DRG NO.	SHEET REV
A		A

SCALE 1:2		SHEET 1 OF 1



NOTE: the left elevon is mirrored with the same dimensions.
Exact geometry will come from the airfoil

ALL DIMENSIONS IN INCHES

SIEMENS

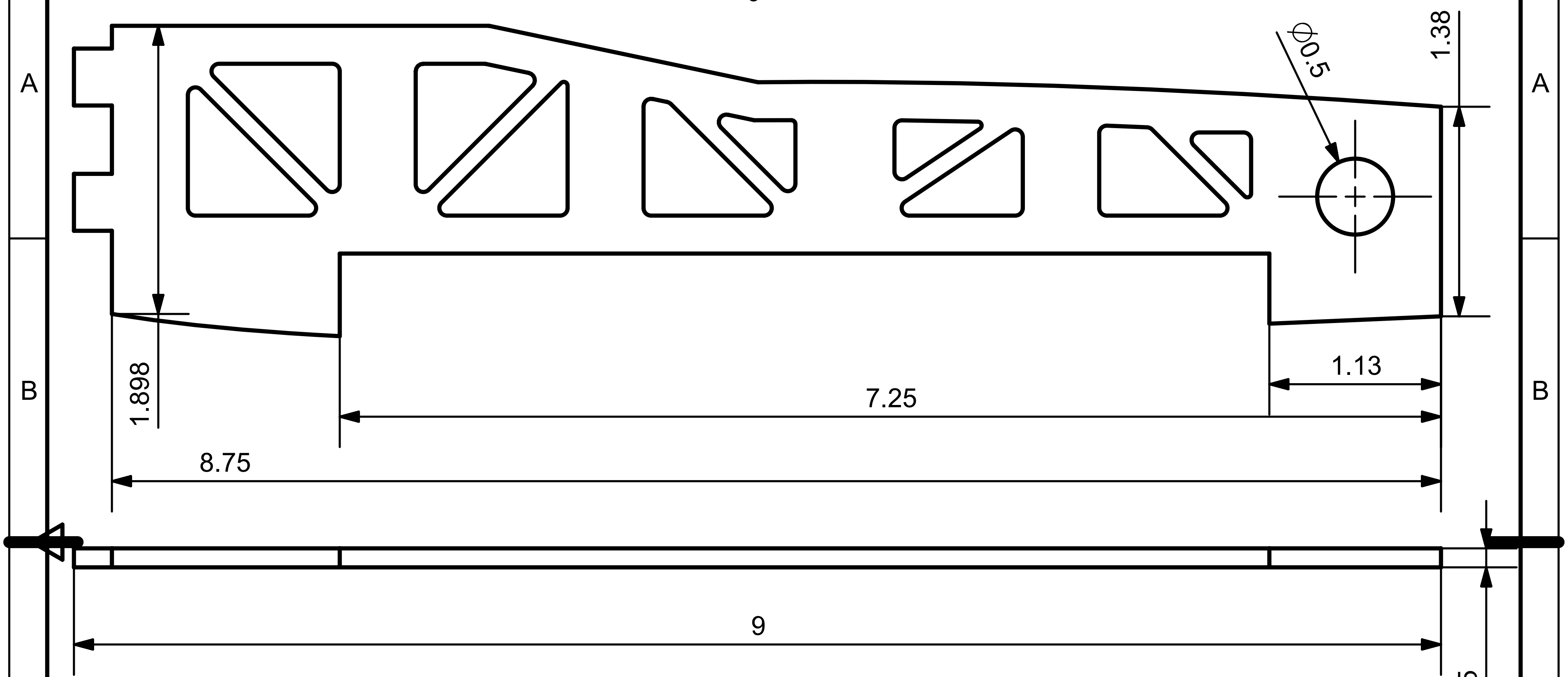
THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

TEAM	1
DRAWN BY	
CHECKED BY	
QTY	2
MATERIAL	FOAM/BALSA

TITLE
Elevon (Right)

SIZE	DRG NO.	SHEET REV
A		A

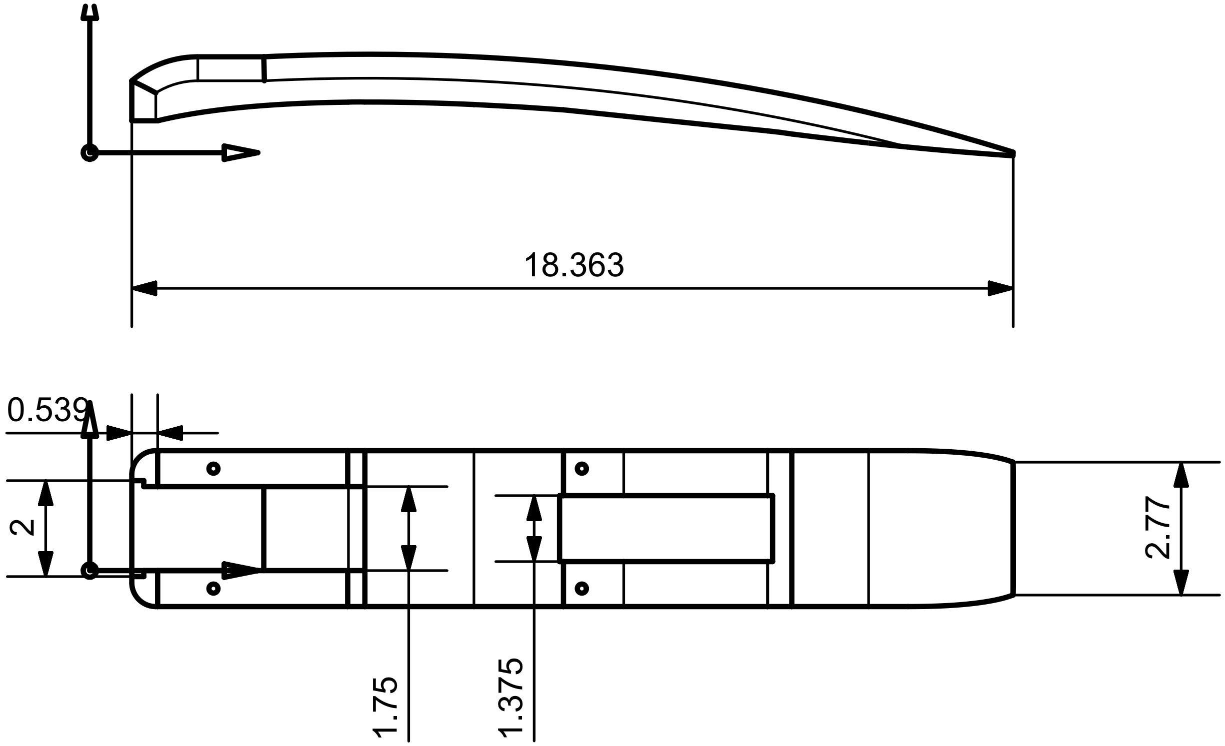
SCALE 1:4 — SHEET 1 OF 1



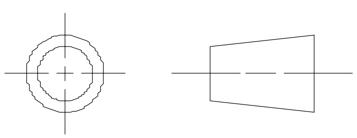
Note: Full dimensioning not given due to presence of complex geometries.
Part will be manufactured using a laser cutter, meaning only a dxf or pdf file is needed to manufacture

ALL DIMENSIONS IN INCHES

SIEMENS		THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE
TEAM	1	TITLE
DRAWN BY		
CHECKED BY		Fuselage Side
QTY	2	SIZE A / DRG NO. / SHEET REV A
MATERIAL	Plywood	SCALE 1:1 / SHEET 1 OF 1



Note: Part will be manufactured using CNC foam cutting.
Complex geometries will be given electronically



ALL DIMENSIONS IN INCHES

SIEMENS	THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

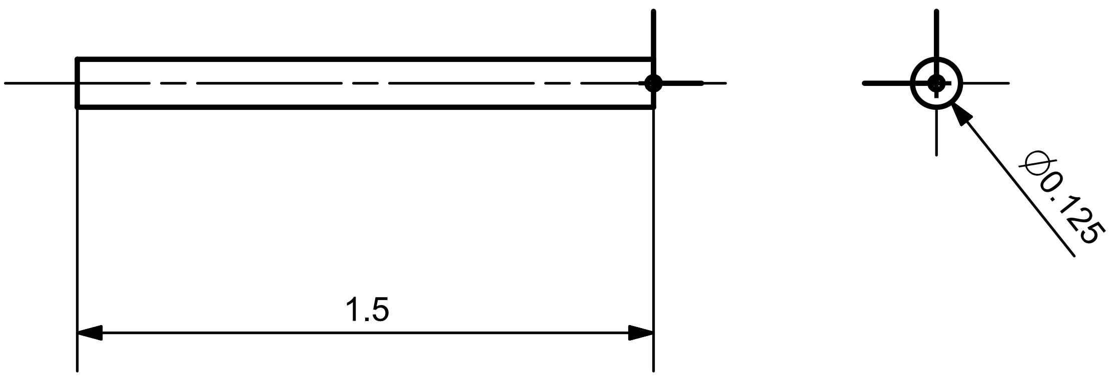
TEAM	1
DRAWN BY	
CHECKED BY	
QTY	1
MATERIAL	FOAM

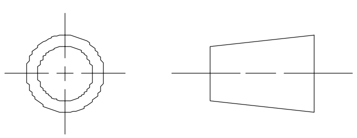
TITLE

Fuselage Cover

SIZE	DRG NO.	SHEET REV
A		A

SCALE 1:4		SHEET 1 OF 1





ALL DIMENSIONS IN INCHES

SIEMENS

THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

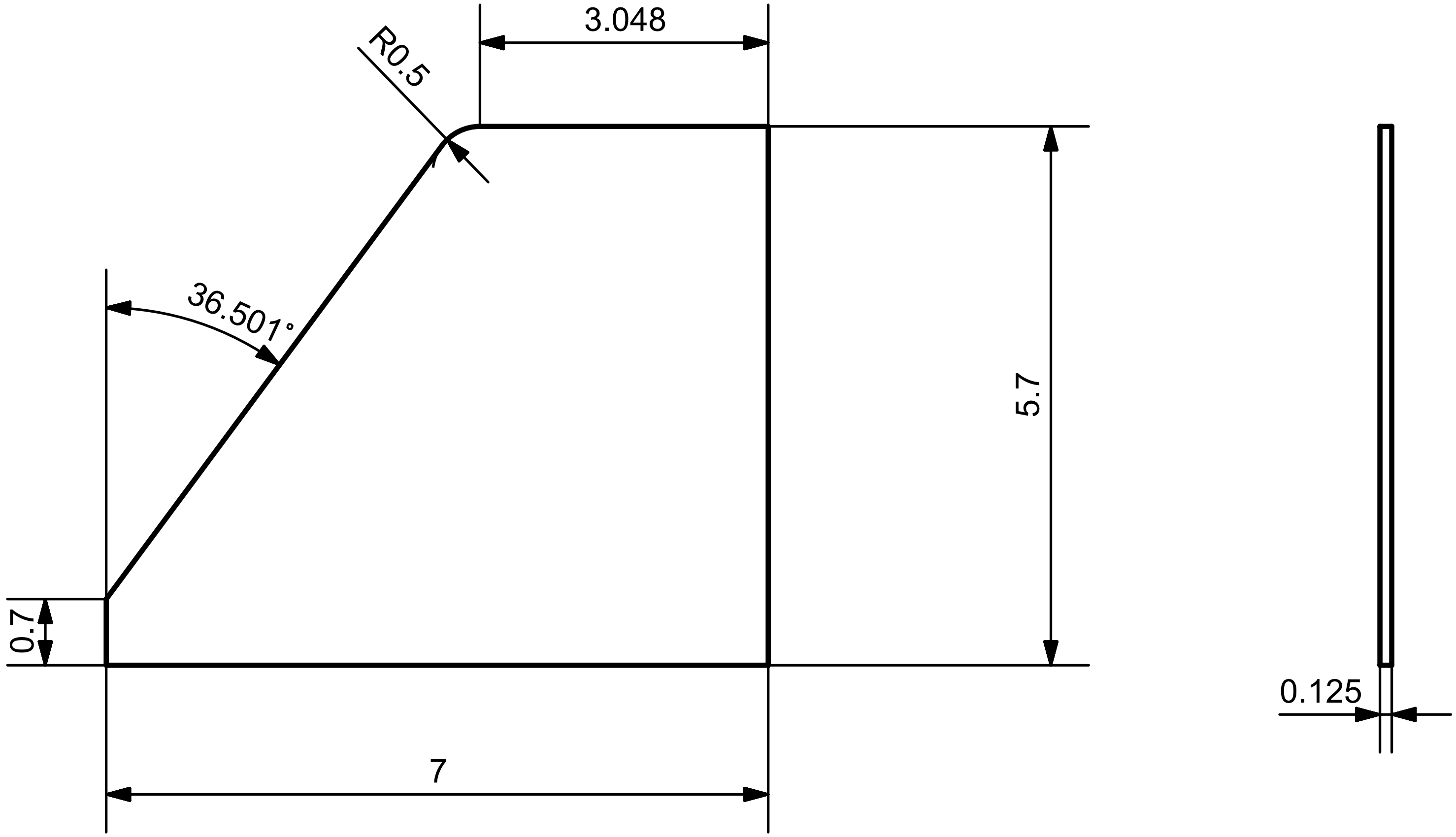
TEAM	1
DRAWN BY	
CHECKED BY	
QTY	1
MATERIAL	CARBON FIBER

TITLE

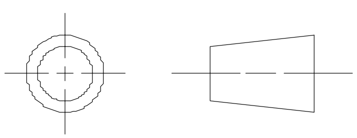
Torsion Pin

SIZE	DRG NO.	SHEET REV
A		A

SCALE 2:1		SHEET 1 OF 1



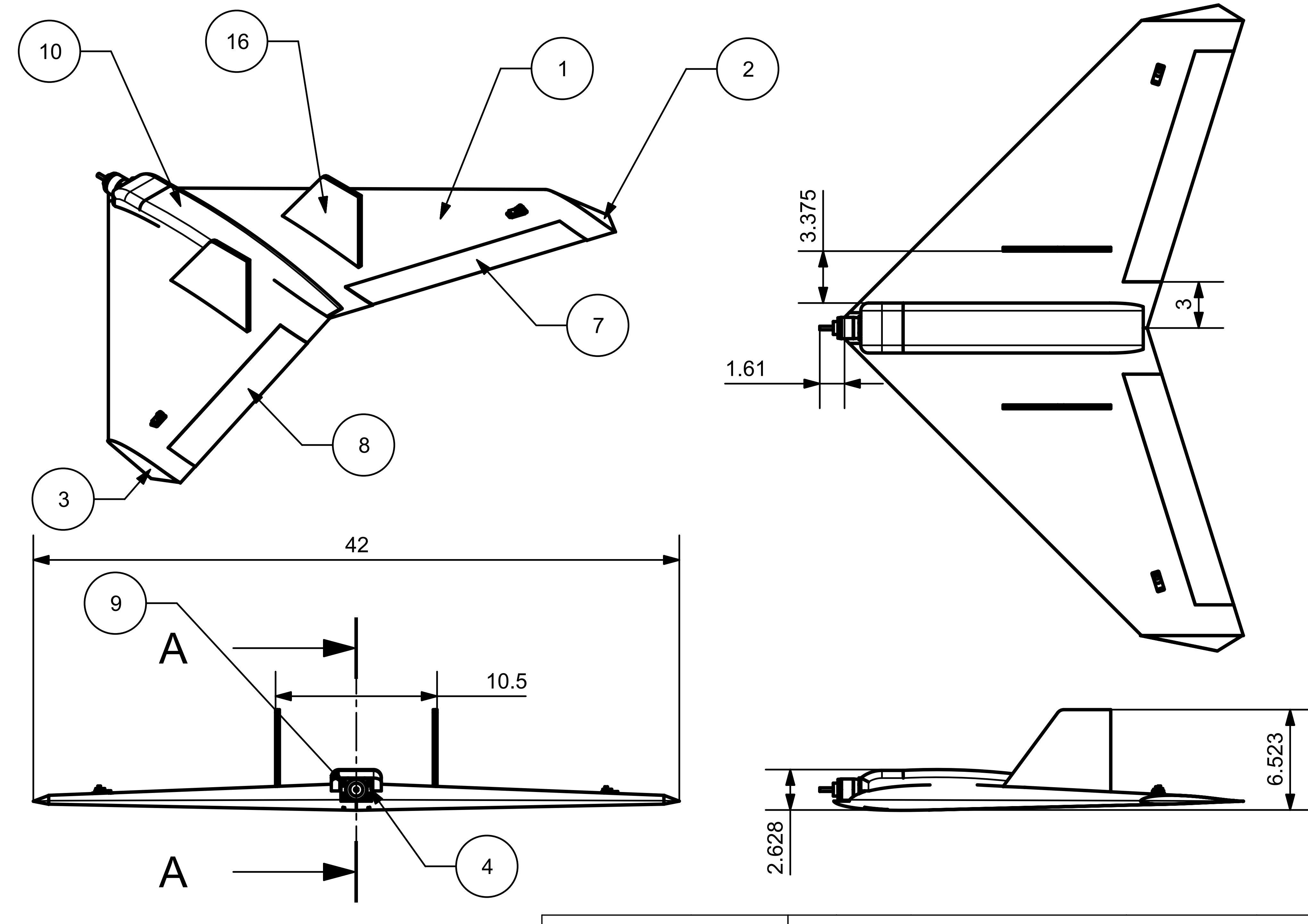
NOTE: The vertical stabilzer will be manufactured with a laser cutter.
Two fins will be created by placing 1/8in fins together

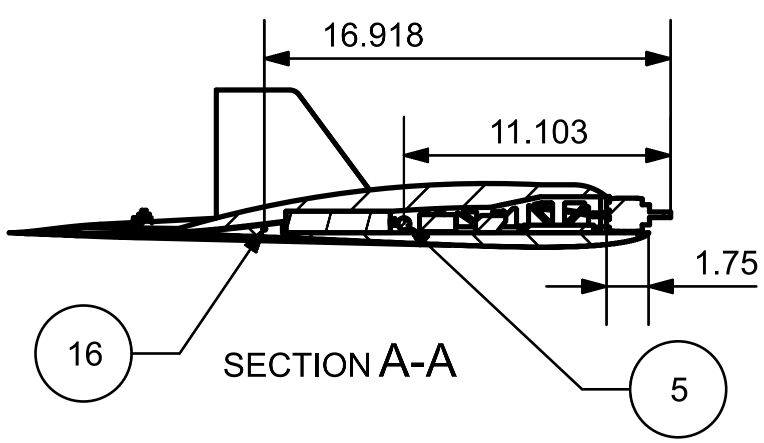


ALL DIMENSIONS IN INCHES

SIEMENS		THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE	
TEAM	1	TITLE Vertical Stabilizer	
DRAWN BY			
CHECKED BY			
QTY	4	SIZE A / DRG NO.	SHEET REV A
MATERIAL	BALSA WOOD		
		SCALE: 1:2	SHEET 1 OF 1

Item No.	Part Name	Qty
1	Wing	1
2	Right Winglet	1
3	Left Winglet	1
4	Motor Mount	1
5	Spar	1
6	Motor	1
7	Right Elevon	1
8	Left Elevon	1
9	Fuselage Side Plate	2
10	Fuselage Top	1
11	ESC	1
12	Receiver	1
13	Battery	1
14	Payload	1
15	Torsion Pin	2
16	Vertical Stabilizer	4
17	Servo	2





SECTION A-A

ALL DIMENSIONS IN INCHES

SIEMENS

THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

FIRST ISSUED	3/13/25
DRAWN BY	JA
CHECKED BY	LB
APPROVED BY	TG

TITLE

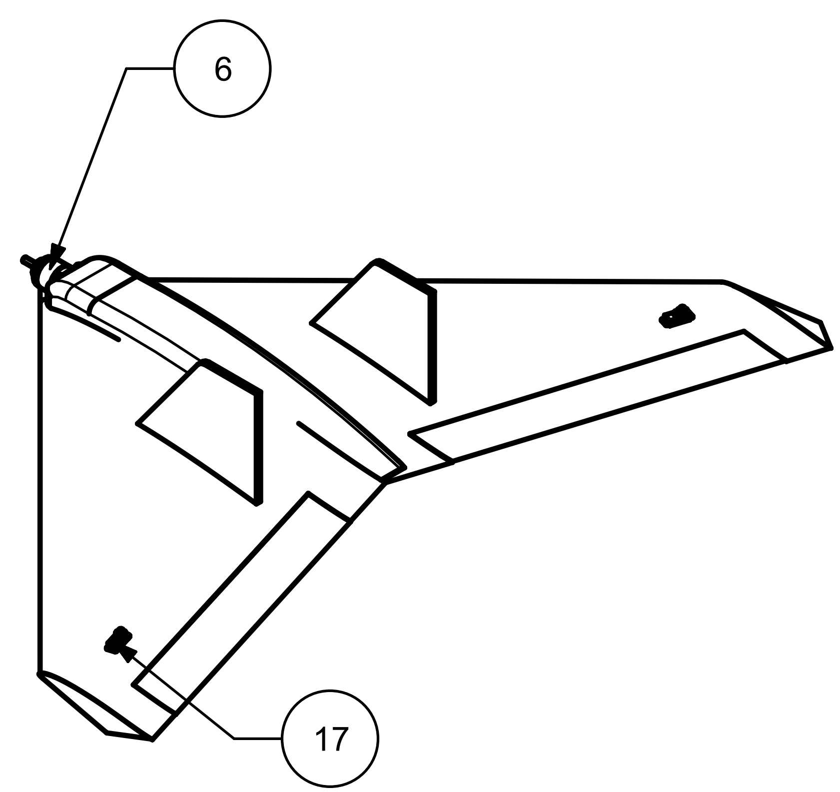
Layout of Manufactured Components

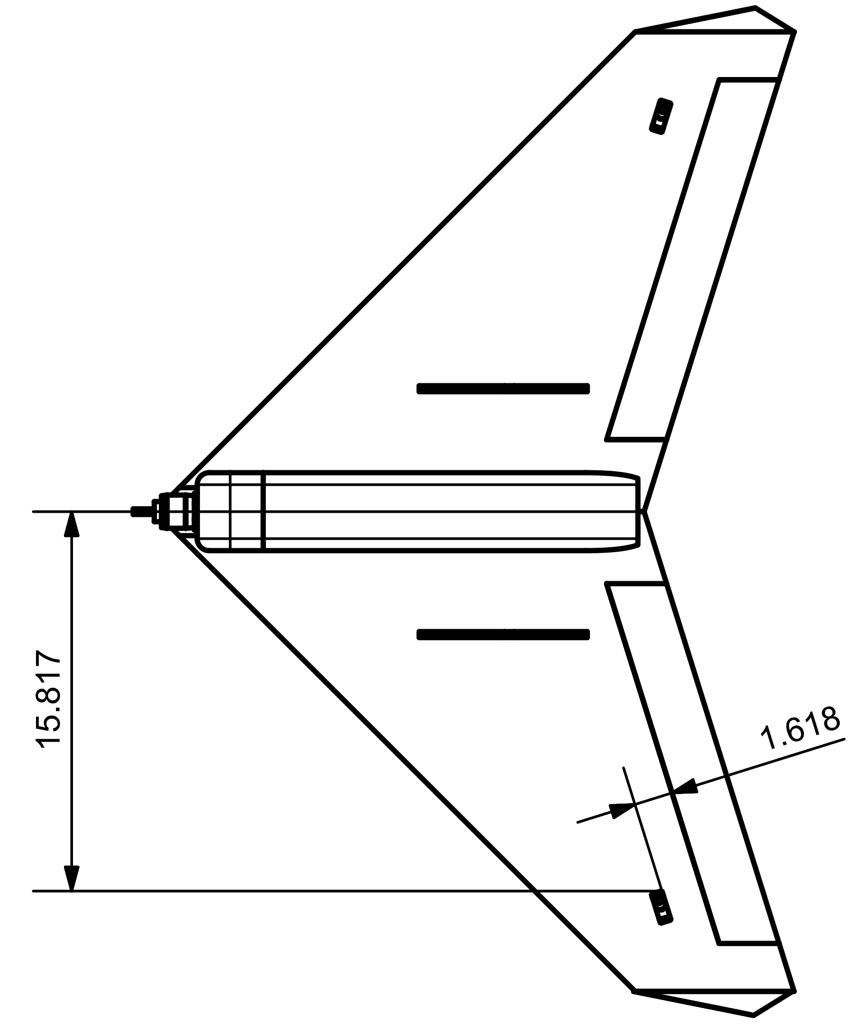
SIZE	DRG NO.	SHEET REV
B	AE484_overall	A

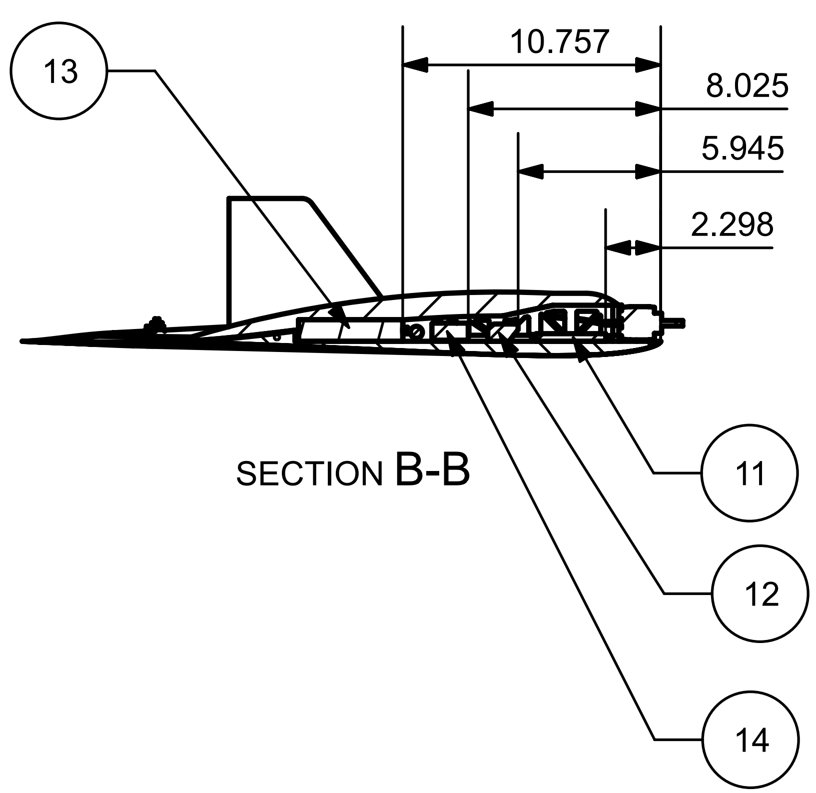
SCALE 1:8

SHEET 1 OF 3

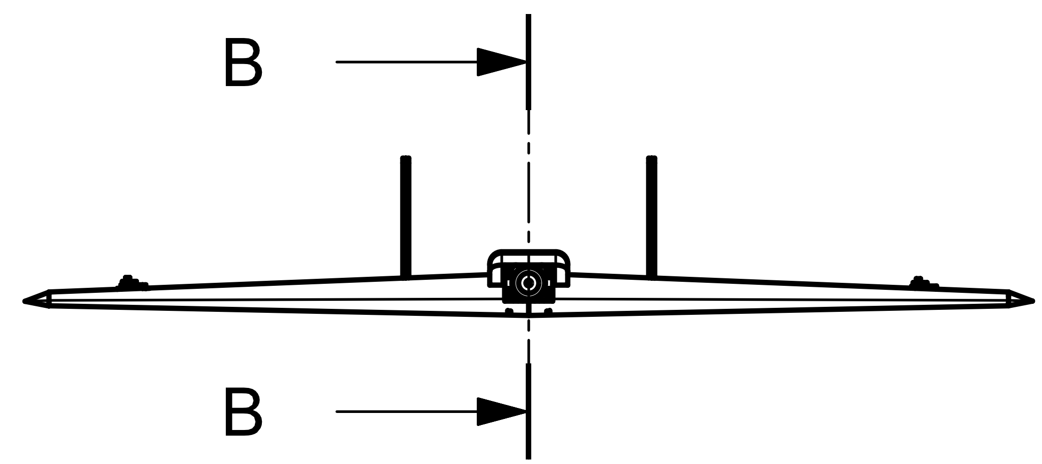
Item No.	Part Name	Qty
1	Wing	1
2	Right Winglet	1
3	Left Winglet	1
4	Motor Mount	1
5	Spar	1
6	Motor	1
7	Right Elevon	1
8	Left Elevon	1
9	Fuselage Side Plate	2
10	Fuselage Top	1
11	ESC	1
12	Receiver	1
13	Battery	1
14	Payload	1
15	Torsion Pin	2
16	Vertical Stabilizer	4
17	Servo	2

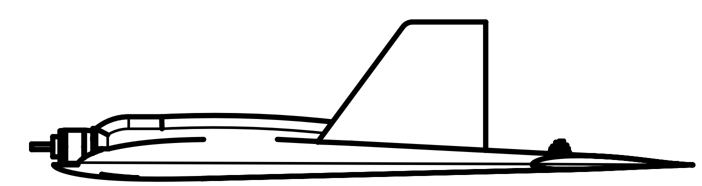






SECTION B-B

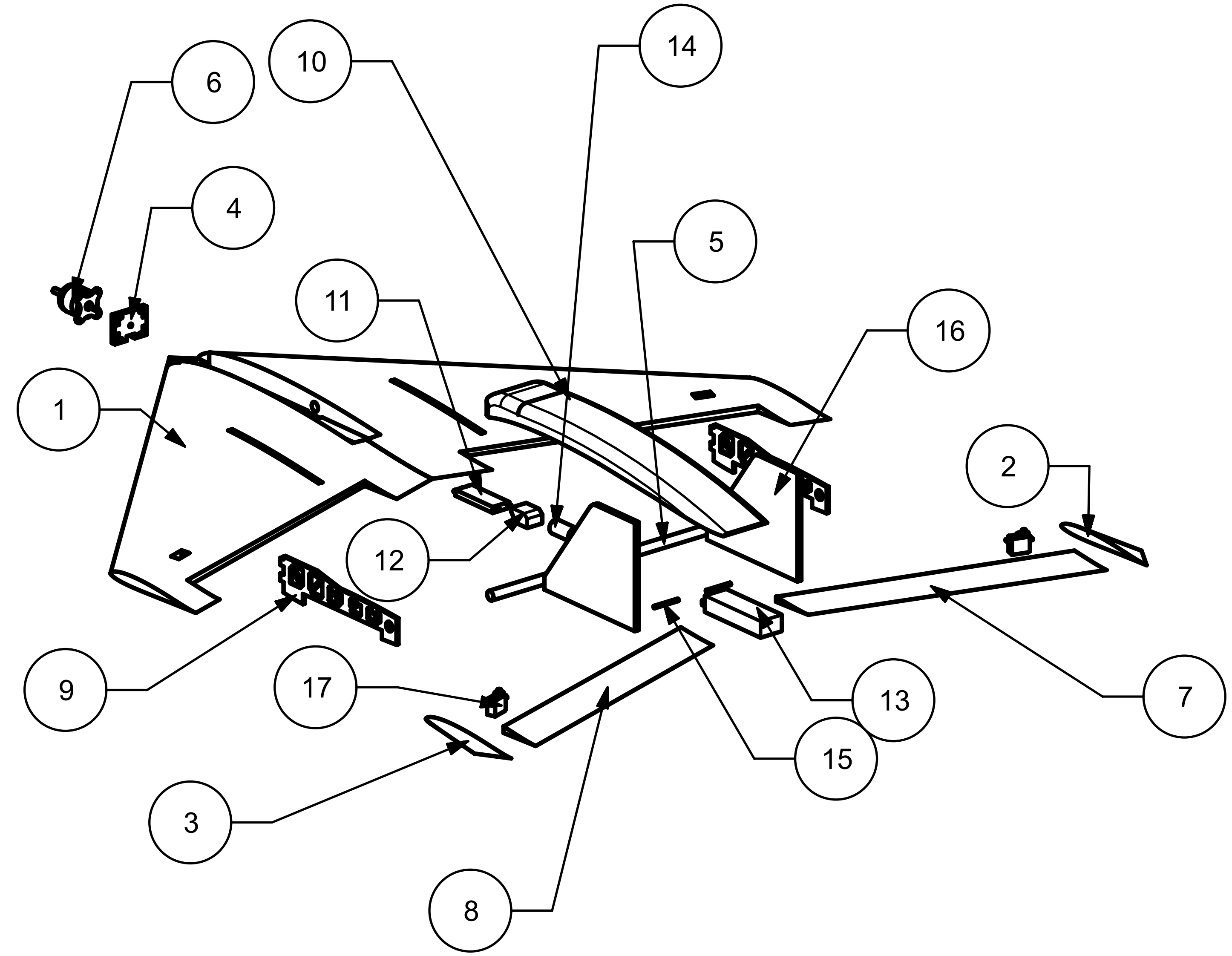




ALL DIMENSIONS IN INCHES

SIEMENS	THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE		
FIRST ISSUED	3/13/25	TITLE	Layout of Purchased Parts
DRAWN BY	JA		
CHECKED BY	LB		
APPROVED BY	TG	SIZE: B	DRG NO.: AE484_overall — SHEET REV: A
		SCALE 1:8	SHEET 2 OF 3

Item No.	Part Name	Qty
1	Wing	1
2	Right Winglet	1
3	Left Winglet	1
4	Motor Mount	1
5	Spar	1
6	Motor	1
7	Right Elevon	1
8	Left Elevon	1
9	Fuselage Side Plate	2
10	Fuselage Top	1
11	ESC	1
12	Receiver	1
13	Battery	1
14	Payload	1
15	Torsion Pin	2
16	Vertical Stabilizer	4
17	Servo	2



ALL DIMENSIONS IN INCHES

SIEMENS

THIS DRAWING HAS BEEN PRODUCED USING AN EXAMPLE TEMPLATE PROVIDED BY SIEMENS PLM SOFTWARE

FIRST ISSUED	3/14/25
DRAWN BY	JA
CHECKED BY	JT
APPROVED BY	LB

TITLE

Exploded View of Aircraft

SIZE	DRG NO.	SHEET REV
B	AE484_overall	A

SCALE 1:8

SHEET 3 OF 3