Preliminary Design Review

PLEASE NEGLECT AIR RESISTANCE

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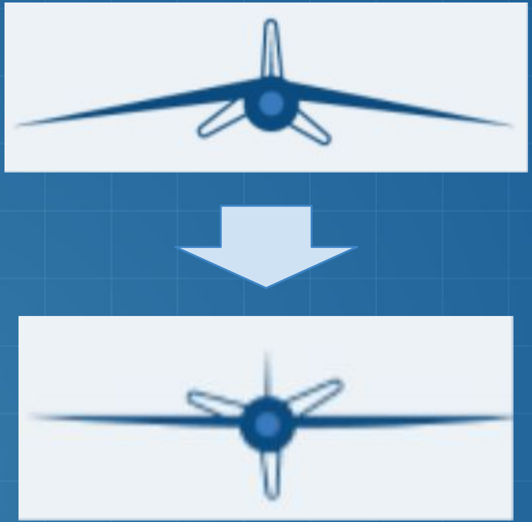
CHANGES FROM CDR

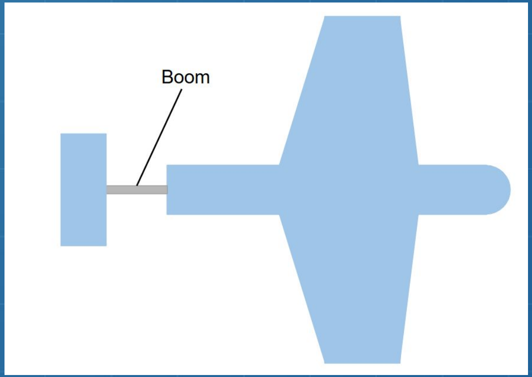
Changed from Anhedral to Straight-Wing

- \- Less Unique
- \+ Easier to Manufacture
- \+ More Lift



Added Single Boom Tail

- \- Possibly harder to manufacture
- \+ Less Weight
- \+ More Unique



WING DESIGN: MANUFACTURING PLAN

- The wings will be manufactured by connecting ribs with spars.
- The rib-spar structure will then be wrapped with UltraCote film.
- Additionally, balsa sheeting will be added to the leading and trailing edges for the UltraCote to attach to.
    - This plan will minimize weight, and provide ease of manufacturability as the ribs will be laser-cut.
- The ribs will be cut from ⅛” thick plywood, and the main structural spars will be aluminum.

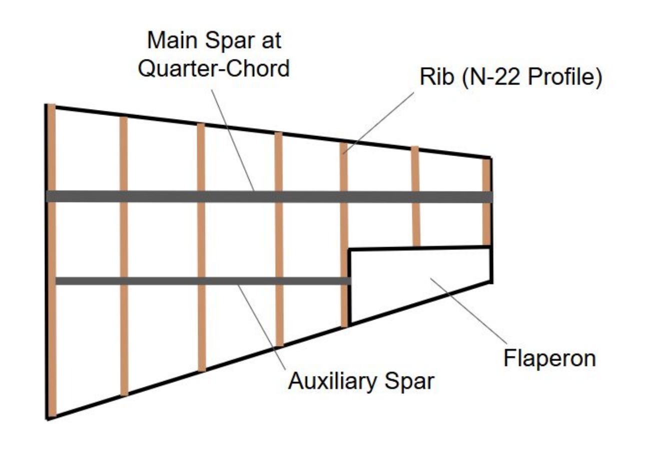


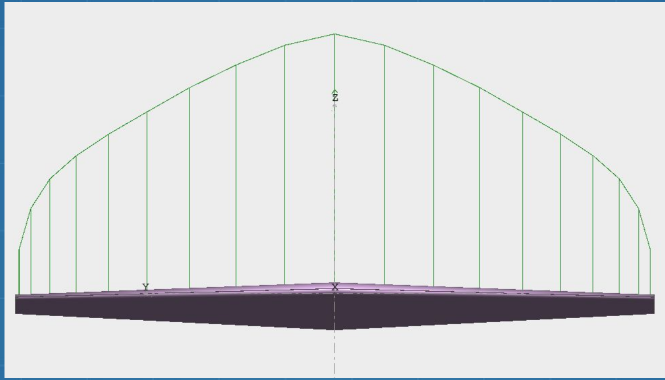
*Image not to scale

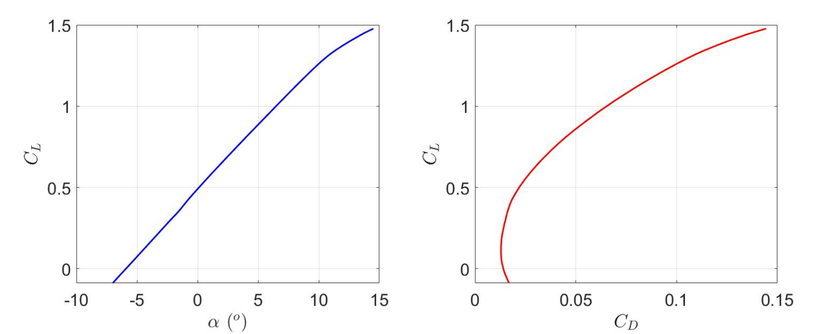
WING DESIGN: AERODYNAMICS SUMMARY

Geometric Parameters

Parameter	Value
Airfoil	N-22
Area, S	816.66 in^2
Span, b	70 in.
Aspect Ratio, AR	6.0
Root Chord, c_r	16.66 in.
Tip Chord, c_t	6.66 in.
Taper Ratio, λ	0.4
Oswald's Efficiency, e	0.8691
MAC, $\bar{c}$	12.38 in.
LE Sweep, Λ_{LE}	4.336°
$\bar{c}/4$ Sweep, $\Lambda_{\bar{c}/4}$	0°
$\bar{c}/4$ Thickness, $(t/\bar{c})_{\bar{c}/4}$	12.215%
Max Thickness, $(t/\bar{c})_{max}$	12.4%

XFLR5 Analysis





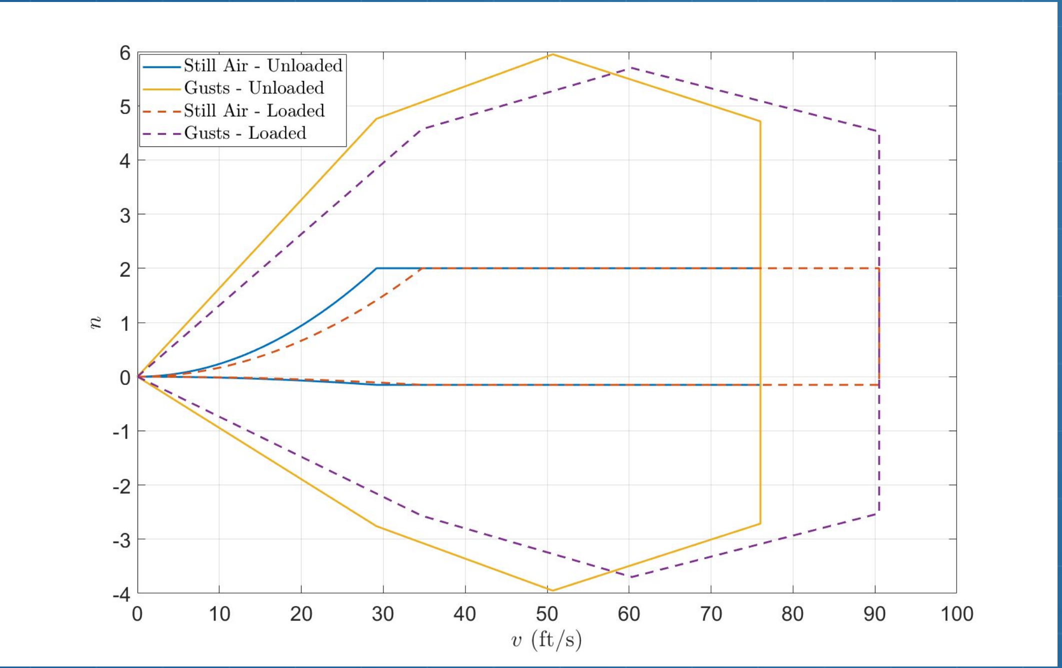
Initial Aerodynamic Parameters

Parameter	Value
C_{L0}	0.49
$(C_{L,\alpha})$	4.584 /rad
C_{Lmax}	1.48
C_{D0w}	0.043
Stall Speed, v_s	30.1 ft/s
Cruise Speed, v_c	50.7 ft/s
Dive Speed, v_d	76.05 ft/s

*Initial speeds were calculated solely based on the wing. Full plane speeds are stated later.

WING DESIGN: LOADING AND GUSTS

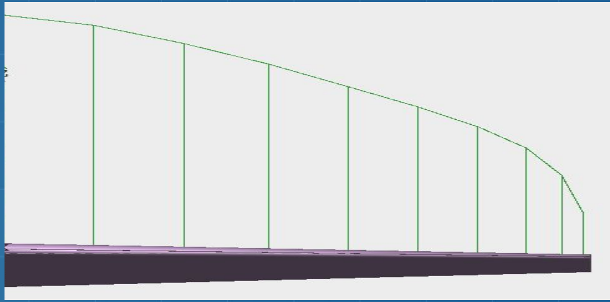
- Load factors were calculated for loaded and unloaded flight with and without gusts in MATLAB

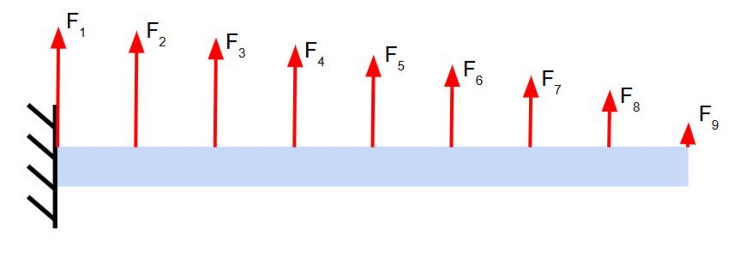


Max Loads	
Loading Condition	**Value**
Still Air - Unloaded	2.1
Gusts - Unloaded	5.95
Still Air - Loaded	2.1
Gusts - Loaded	5.70

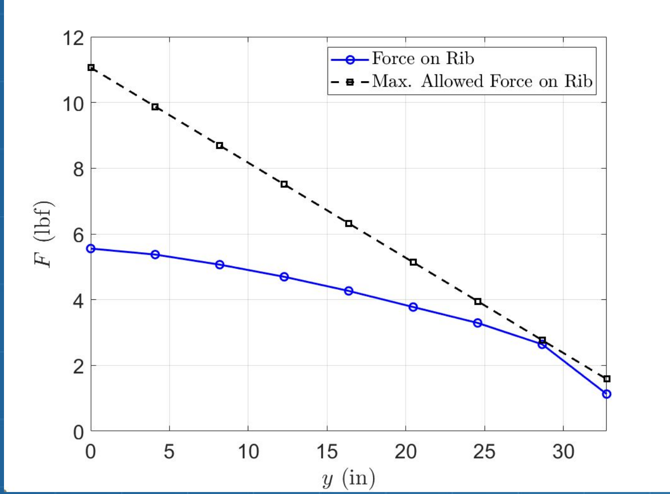
WING DESIGN: RIB STRESS ANALYSIS

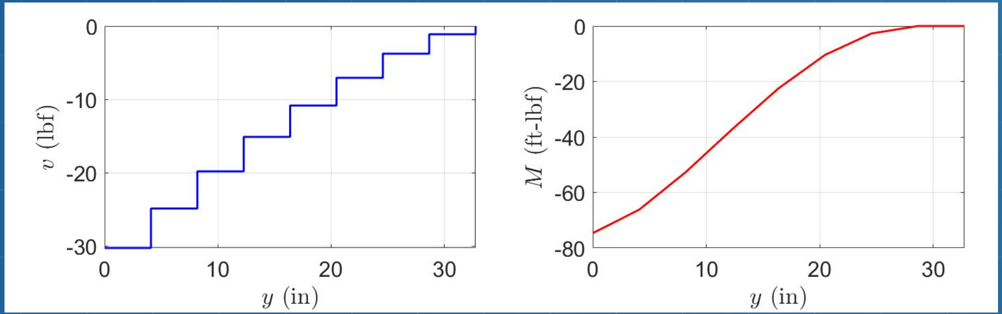
- Total force was calculated via: $\sum F = \frac{n_{max} W}{2}$
- Yield stress of plywood in rolling shear: 250 psi
- Assuming a *FoS* of 2 and that all the weight will be on the main spare, the force on each beam is given as: $F = t \cdot b \cdot \tau_y$
  - Where *t* is the rib thickness (⅛”), and *b* is the minimum distance between the main spar and top of rib
- Assuming the weight distribution follows the XFLR5 lift distribution, an FEA analysis was run to find the weights on each rib.
- Those numbers were compared to the allowed forces, and the number of ribs was adjusted so that the max force would not be exceeded.
- **<u>Result:</u> 9 ribs for each wing**





WING DESIGN: SPAR STRESS ANALYSIS

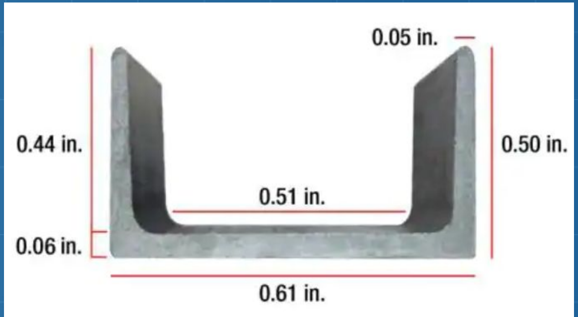




- The rib analysis allowed for the creation of shear-moment diagrams for the spars.
- Results:
  - v_{max} = -30.2 lbf
  - M_{max} = -74.7 ft-lbf
  - Both maximums occur at the wing root

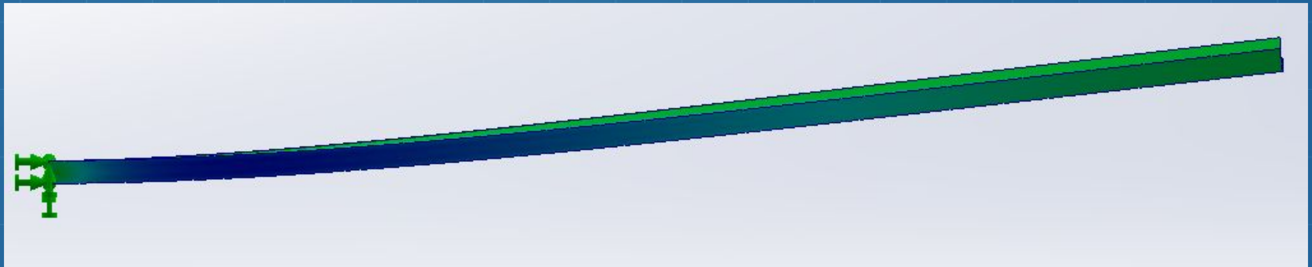
WING DESIGN: SPAR STRESS ANALYSIS

- An aluminum spar was chosen with cross-sectional dimensions as shown.
- Bending and shear stress were calculated via: $\tau = \frac{v}{A}$ $\sigma = \frac{My}{I}$
  - $\tau_{max} = 374.7$ psi, $\sigma_{max} = 2978$ psi
- Yield Shear Stress of Aluminum: 30000 psi
- Yield Tensile Stress of Aluminum: 40000 psi
- Therefore, predicted stresses will be much lower than their maximum allowable limits.

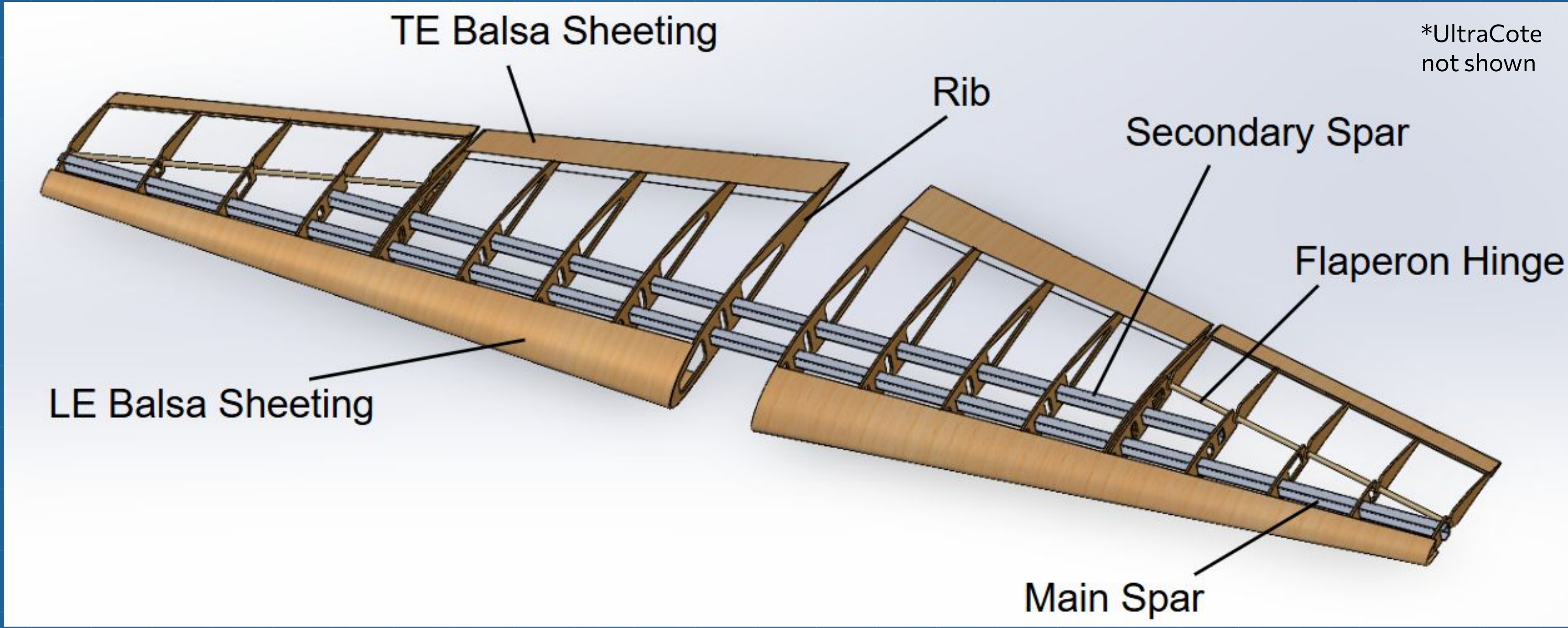


WING DESIGN: SPAR STRESS ANALYSIS

- A SolidWorks FEA was also conducted, approximating the wing as the main spar.
- The applied load magnitude was determined by the maximum load factor.
- The applied load distribution was determined by the lift distribution from XFLR5.
- <u>Results:</u>
    - τ_{max} = 240.9 psi
    - σ_{max} = 1568 psi
    - FoS = 5.908



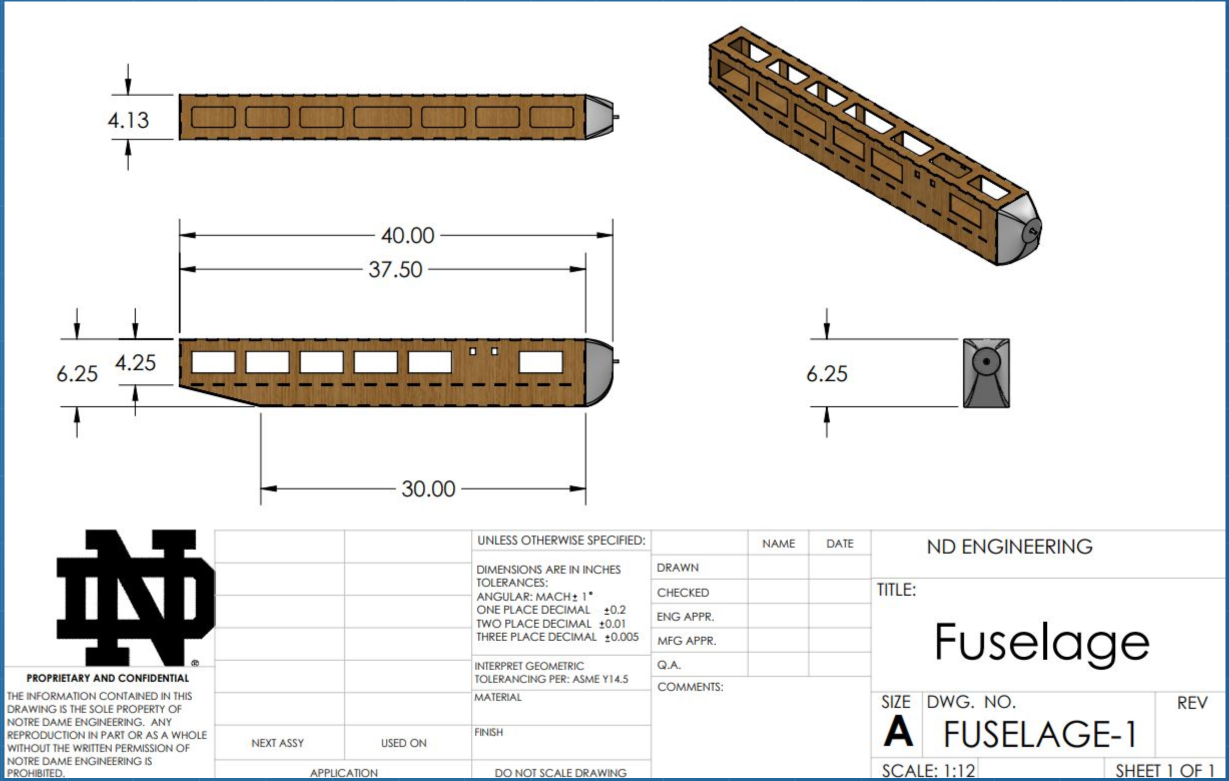
WING DESIGN: FINAL



- Wings will be built around the fuselage
- Components for construction are shown here

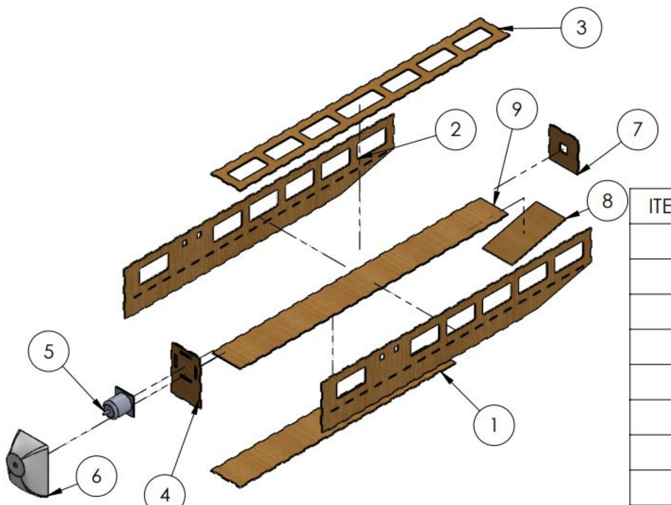
FUSELAGE DESIGN: GEOMETRY

- Inverted camber shape
- Cross-sectional dimensions: 4.125” x 6.25”
- Length:
  - Main Fuselage: 37.5”
  - Boom: 3.095” (exposed)
- Divided into upper and lower compartments for avionics and payload.
- Payload space: 348.75 in^3
- Nose-cone to house motor and reduce drag.



FUSELAGE DESIGN: MANUFACTURE PLAN

- Wooden panels of fuselage will be laser cut from ⅛” plywood and assembled with glue in a “jigsaw” fashion.
- Nose cone will 3D printed and attached with screws.
- Fuselage will be coated in UltraCote



ITEM NO.	PART NUMBER	QTY.
1	fuselageBottom	1
2	fuselageSide	2
3	fuselageTop	1
4	fuselageFront	1
5	EFliteMotorLowFidel	1
6	noseCone	1
7	fuselageBack	1
8	fuselageHatch	1
9	fuselageMiddle	1

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ONE PLACE DECIMAL ±0.2
TWO PLACE DECIMAL ±0.01
THREE PLACE DECIMAL ±0.005

INTERPRET GEOMETRIC TOLERANCING PER: ASME Y14.5

MATERIAL

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Q.A.		

COMMENTS:

ND ENGINEERING

TITLE:

Fuselage Exploded

SIZE A | DWG. NO. FUSELAGE-2 | REV

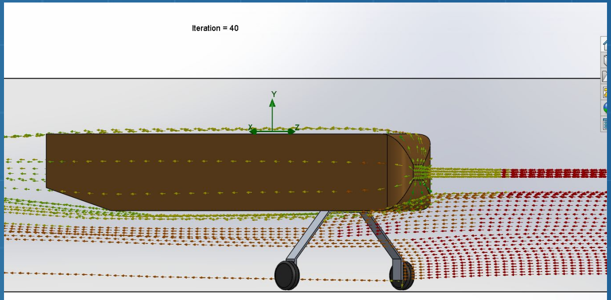
SCALE: 1:32 | SHEET 1 OF 1

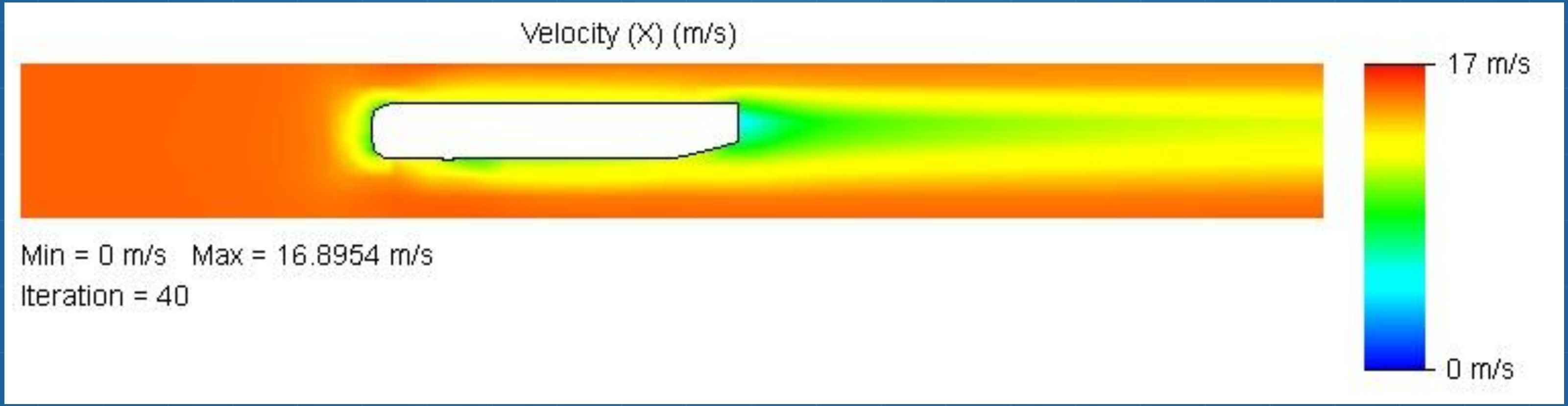
FUSELAGE DESIGN: DRAG ANALYSIS

A Solidworks Flow simulation was conducted to calculate the drag and coefficient of drag due to the fuselage and landing gear.

Drag = 2.176 [N]

$C_{D0} = 0.0284$





TAIL DESIGN: VERTICAL TAIL SIZING

Givens:
$b_W = 70.0\ in^2$
$S_W = 816.67\ in^2$
$C_{VT} = 0.04$
$\Lambda_{wLE} = 4.336$ deg

Calculations:
$ybar_W = 15$ in
$MAC_W = 12.38$ in

Using approximation that $l_{VT} \sim 0.55L$:
$l_{VT} = 33.825$ in

Results:
$S_{VT} = 67.6\ in^2$
$AR_{VT} = 1.6$
$\forall_{VT} = 0.040$

$$\bar{y} \cdot tan\Lambda_{LE} + \frac{MAC}{4}$$

$$S_{VT} = C_{VT} \frac{b_W S_W}{l_{VT}}$$

	C_{VT}
Sail Plane	0.02
Homebuilt	0.04

Normalized spanwise location of MAC from wing center: $\frac{\bar{y}}{b} = \frac{1}{6} \cdot \frac{1+2\lambda}{1+\lambda}$

	Aft-horizontal		Vertical	
	AR	λ	AR	λ
Combat	3-4	0.2-0.4	0.6-1.4	0.2-0.4
Sail Plane	6-10	0.3-0.5	1.5-2.0	0.4-0.6
Other	3-5	0.3-0.6	1.3-2.0	0.3-0.6
T-Tail	-	-	0.7-1.2	0.6-1.0

TAIL DESIGN: HORIZONTAL TAIL SIZING

Givens:
$S_W = 816.67\ in^2$
$C_{HT} = 0.5$
$\Lambda_{wLE} = 4.336$ deg

Calculations:
$ybar_W = 15$ in
$MAC_W = 12.38$ in

Using approximation that $l_{HT} \sim 0.55L$:
$l_{HT} = 33.825$ in

Results:
$S_{HT} = 149.45\ in^2$
$AR_{HT} = 0.5 * AR_W = 3$
$\forall_{HT} = 0.50$

$$\bar{y} \cdot tan\Lambda_{LE} + \frac{MAC}{4}$$

$$S_{HT} = C_{HT} \frac{\bar{c}_W S_W}{l_{HT}}$$

	C_{VT}	C_{HT}
Sail Plane	0.02	0.5
Homebuilt	0.04	0.5

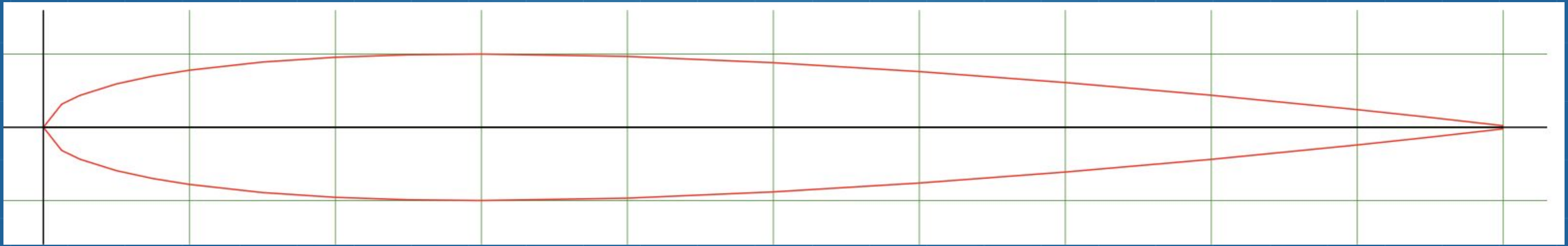
Normalized spanwise location of MAC from wing center: $\frac{\bar{y}}{b} = \frac{1}{6} \cdot \frac{1+2\lambda}{1+\lambda}$

	Aft-horizontal		Vertical	
	AR	λ	AR	λ
Combat	3-4	0.2-0.4	0.6-1.4	0.2-0.4
Sail Plane	6-10	0.3-0.5	1.5-2.0	0.4-0.6
Other	3-5	0.3-0.6	1.3-2.0	0.3-0.6
T-Tail	-	-	0.7-1.2	0.6-1.0

TAIL DESIGN: GEOMETRIC SUMMARY

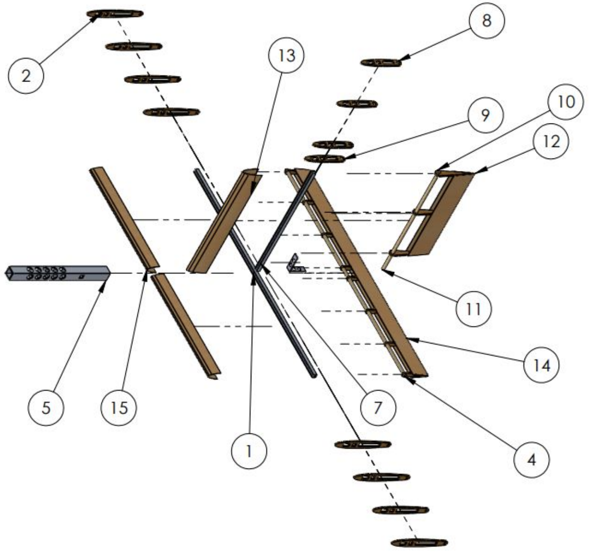
Vertical Tail	
Parameter	**Value**
Airfoil	NACA 0010
VT Area, S_{VT}	67.6 in^2
VT Span, b_{VT}	10.4 in.
VT Aspect Ratio, AR_{VT}	1.6
VT Chord, c_{VT}	6.5 in.
$\bar{x}$	33.825 in.

Horizontal Tail	
Parameter	**Value**
Airfoils	NACA 0010
HT Area, S_{HT}	149.45 in^2
HT Span, b_{HT}	21.17 in.
HT Aspect Ratio, AR_{HT}	3.0
HT Chord, c_{HT}	7.05 in.
$\bar{x}$	33.825 in.



TAIL DESIGN: MANUFACTURE PLAN

- The tails will be manufactured by connecting ribs with a spar.
    - The ribs will be cut from ⅛” thick plywood, and the main structural spar will be aluminum.
- The rib-spar structure will then be wrapped with UltraCote film.
- Additionally, balsa sheeting will be added to the leading and trailing edges for the UltraCote to attach to.
- The tails will be connected to a boom attached to the fuselage.



ITEM NO.	PART NUMBER	QTY.
1	9001K747	1
2	HTRibSplit	8
3	HTRibElevator	8
4	elevatorRod	1
5	boom	1
6	VTBracket	1
7	9001K747	1
8	VTRibSplit	3
9	VTRibSplitRoot	1
10	VTRibRudder	3
11	rudderRod	1
12	rudderSheeting	1
13	VTLESheeting	1
14	elevatorSheeting	1
15	HTLESheeting	2

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	NAME	DATE
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ENG APPR.		
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Q.A.		
COMMENTS:		

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TITLE:

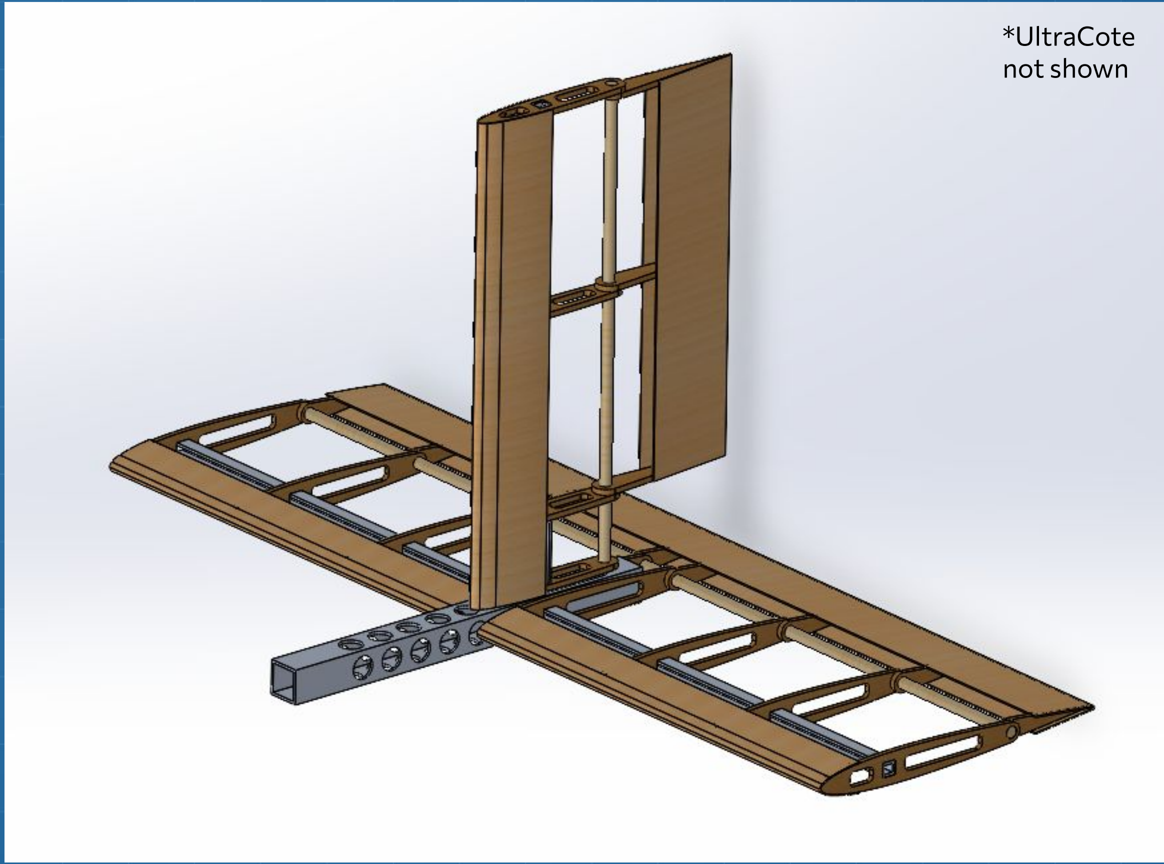
Tail-Exploded

SIZE A | DWG. NO. TAIL-2 | REV

SCALE: 1:24 | SHEET 1 OF 1

TAIL DESIGN: FINAL

- Final CAD of tail is shown.
- Horizontal and vertical stabilizer will be connected to the boom via the geometry displayed.
- Control surfaces will rotate about the rods shown.



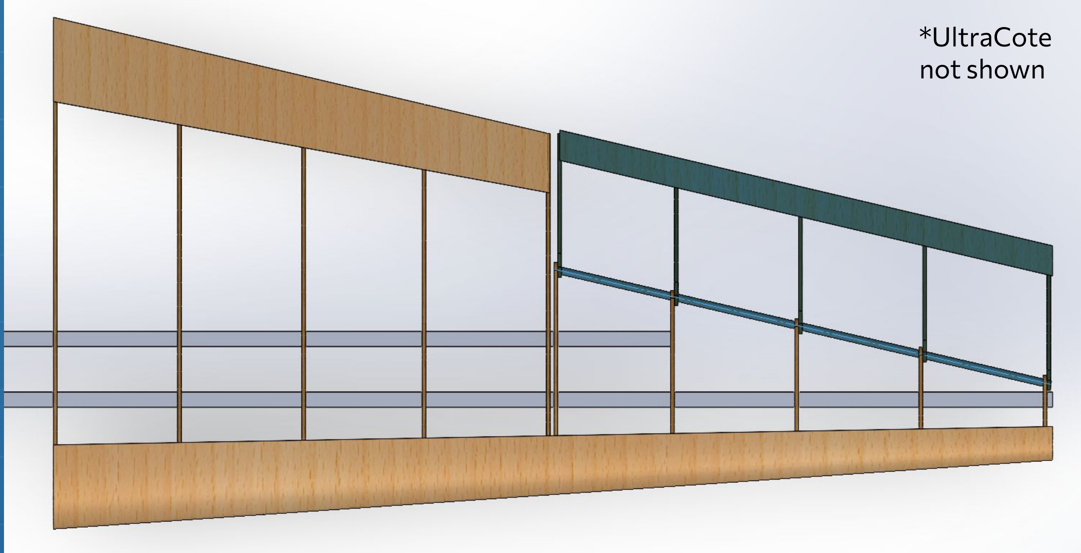
CONTROL SURFACES: FLAPERONS

DBF Historical Sizing
$S_f \sim 0.25 * S_W$
$b_f \sim 0.5 * b_W$

Givens:
$S_W = 816.67\ in^2$
$b_W = 70\ in$

Calculations:
$S_f = 204.167\ in^2$
$b_f = 35\ in$
$c_f = 5.833\ in$

Modifications for our Plane
$c_f = 4.5\ in$
$b_f = 33.1\ in$
$S_f = 148.95\ in^2$



CONTROL SURFACES: ELEVATOR FLAPS

Hand Calculations

Givens:

$c = 12.38$ in

$S_W = 816.67\ in^2$

$l_{HT} = 33.825$ in

$S_{HT} = 149.45\ in^2$

XFLR5:

$C_{m,0}=-0.13$

$C_{L,\alpha}=0.08$

Calculations:

$\tau=0.13 \rightarrow S_e/S_{HT}=0.3 \rightarrow S_e=44.835\ in^2$

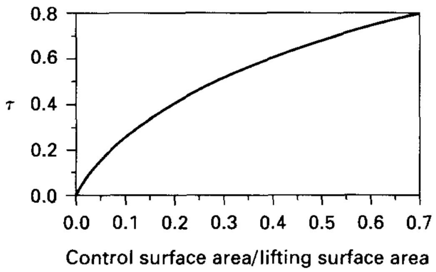


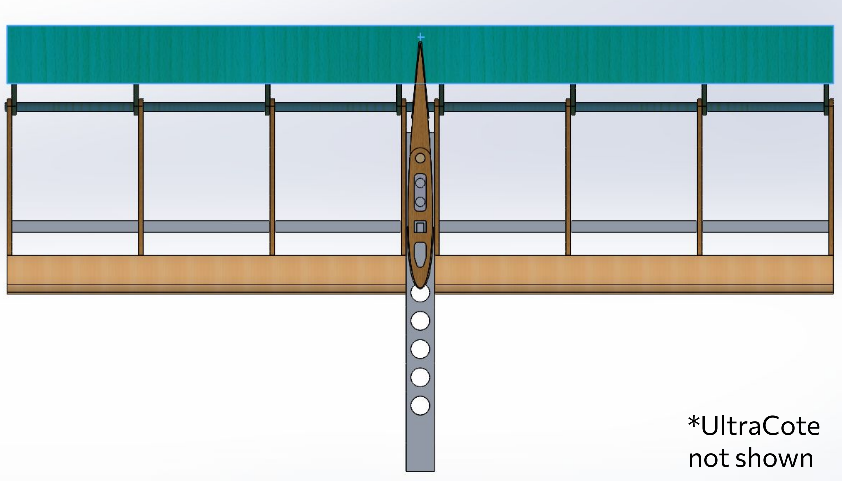
FIGURE 2.21
Flap effectiveness parameter.

DBF Historical Sizing

$S_e \sim 0.3*S_{HT}$

Calculations:

$S_e=44.835\ in^2$



*UltraCote not shown

CONTROL SURFACES: RUDDER FLAP

Hand Calculations

Givens:

$c = 12.38$ in

$S_W = 816.67$ in^2

$l_{VT} = 33.825$ in

$S_{VT} = 67.6$ in^2

XFLR5:

$C_{m,0}$=-0.13

$C_{L,\alpha}$=0.08

Calculations:

τ=0.287 → S_r/S_{VT}=0.47 → S_r=31.772 in^2

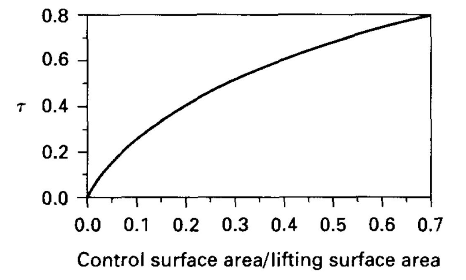


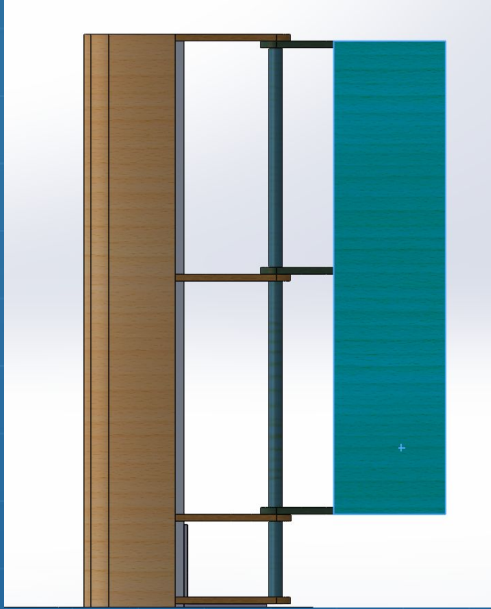
FIGURE 2.21
Flap effectiveness parameter.

DBF Historical Sizing

S_r~0.3*S_{HT}

Calculations:

S_r=44.835 in^2



AERODYNAMIC ANALYSIS: LIFT AND DRAG

Lift

$C_{L0} = 0.391$

$(C_{L,\alpha}) = 4.584$ /rad

$C_{Lmax} = 1.48$

$(C_L/C_D)_{max} = 9.8$

Drag

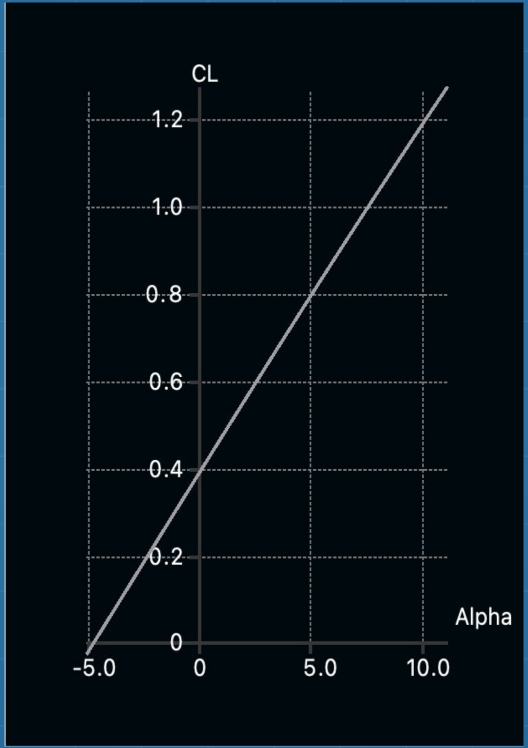
$C_{D,0,VT} = 0.00088$

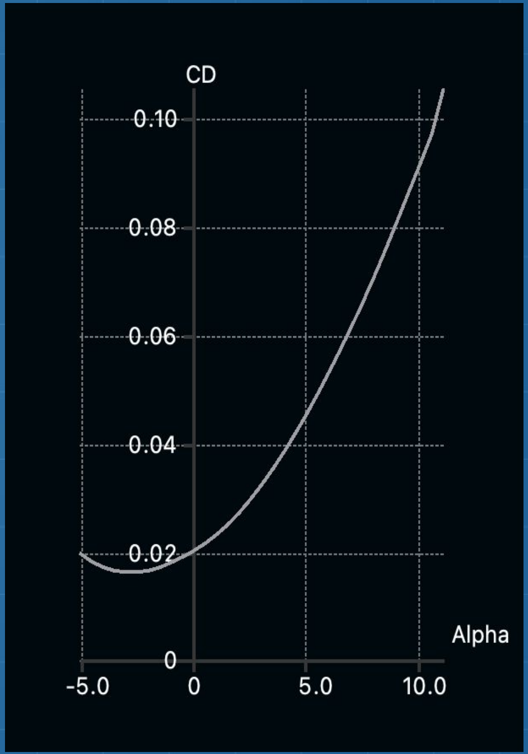
$C_{D,0,HT} = 0.00572$

$C_{D,0,f} = 0.02537$

$C_{D,0,w} = 0.04267$

$C_{D,0,total} = 0.07464$

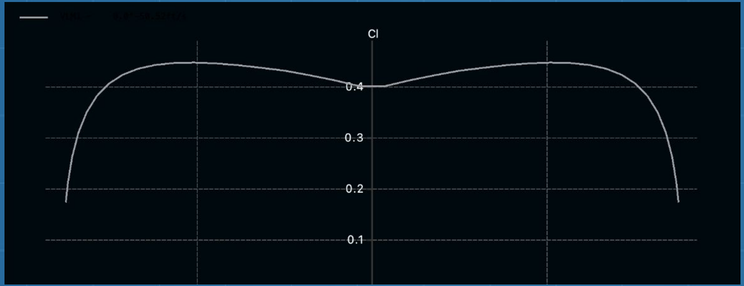


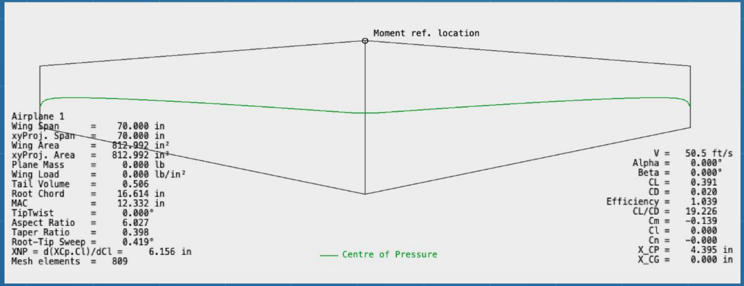


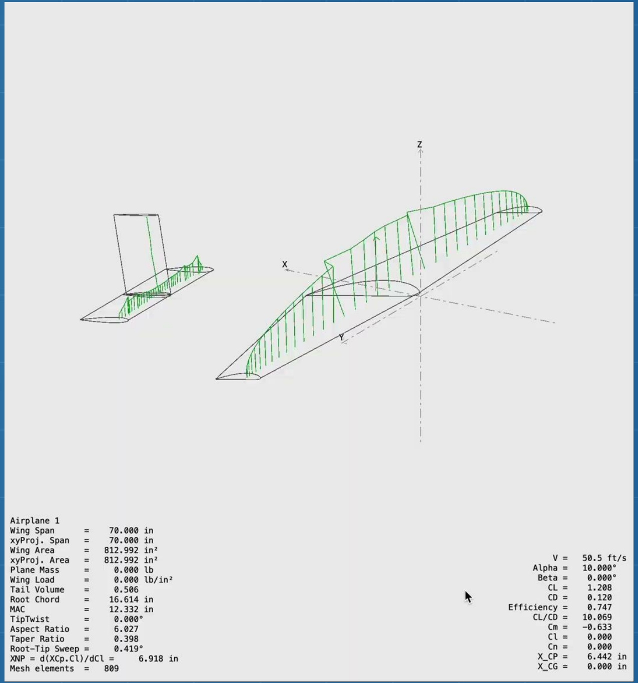
AERODYNAMIC ANALYSIS: LIFT DISTRIBUTION

The lift distribution obtained from XFLR5 is depicted here.

It follows a more elliptical distribution that it would have had the wings been rectangular.



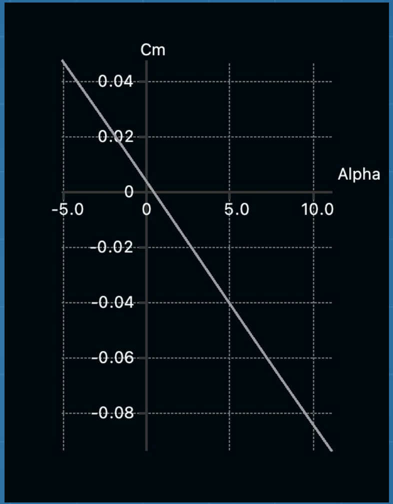




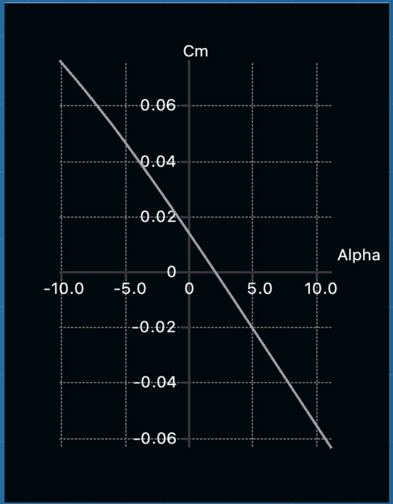
STABILITY ANALYSIS: DIRECTIONAL

Static Stability

Directional Static Stability		
Parameter	Unloaded	Loaded
Neutral Point	6.792 in	6.805 in
Center of Gravity	5.667 in	5.809 in
Static Margin	0.106	0.081
Trim AoA	0.5°	2°
$C_{m,\alpha}$	-0.0089	-0.0067



UNLOADED



LOADED

CG and NP were determined by combining CAD and XFLR5 results through an iterative process, adjusting payload and internal component placement. A zero slope on the Cm vs α curve indicated CG equals NP. The final configuration ensures a negative slope, trimming to a positive α for stability.

STABILITY ANALYSIS: STABILITY MODES

Dynamic Stability

- XFLR5 was used to calculate the time constants, damping frequencies, and damping ratio of the five relevant stability modes.

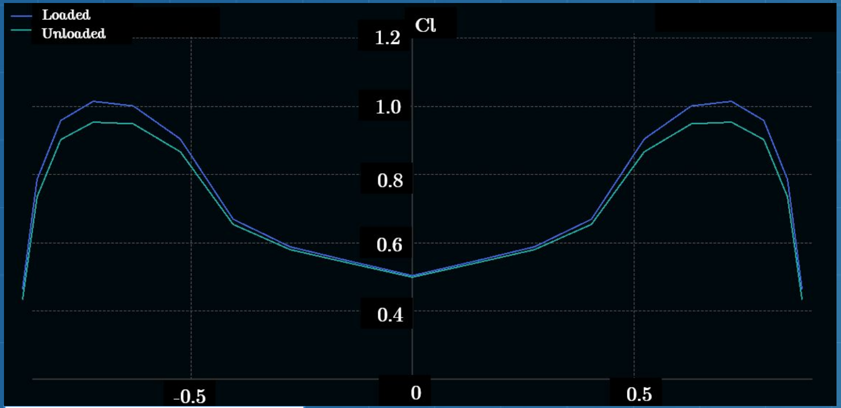
Mode Analysis Unloaded

Parameter	Phugoid	Short Period	Roll	Spiral Div.	Dutch Roll
Time to Double (non-oscillatory)	--	--	0.055 s	3.319 s	--
Damping Frequency (oscillatory)	0.097 Hz	0.792 Hz	--	--	0.816 Hz
Damping Ratio (oscillatory)	0.01	0.770	--	--	0.226

Mode Analysis Loaded

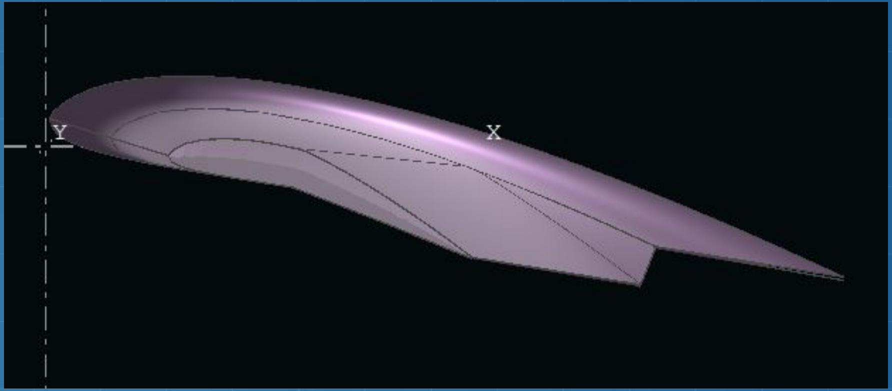
Parameter	Phugoid	Short Period	Roll	Spiral Div.	Dutch Roll
Time to Double(non-oscillatory)	--	--	0.049 s	3.405 s	--
Damping Frequency (oscillatory)	0.091 Hz	0.895 Hz	--	--	0.965 Hz
Damping Ratio (oscillatory)	0.001	0.732	--	--	0.223

AIRCRAFT PERFORMANCE: T/O VELOCITY

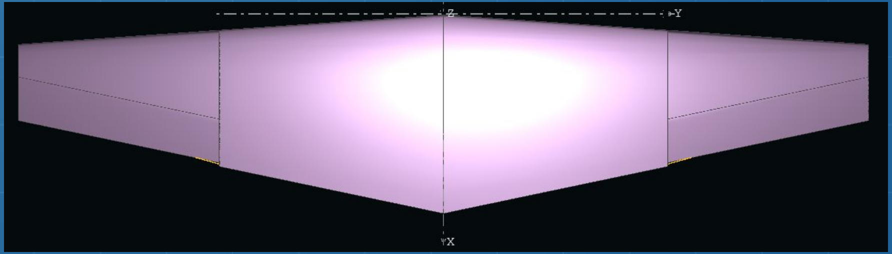


$$u_s = \sqrt{\frac{2W}{C_{L_{TO}} \rho S}}$$

C_L distribution for both loaded and unloaded Takeoff configuration



Take-off configuration with extended flaperon aerodynamic control surfaces.



Aircraft Performance		
Parameter	Unloaded	Loaded
Takeoff Velocity	7.662 ft/s	9.018 ft/s

AIRCRAFT PERFORMANCE: T/O DISTANCE

$$k_{eff} = \frac{1}{\pi e AR_{eff}}$$

$$f_1 = \left(\frac{T}{W} - \mu\right)$$

$$f_2 = \frac{g\rho}{2\left(\frac{W}{S}\right)}\left[\mu C_{LG} - C_{D0} - k_{eff} C_{LG}^2\right]$$

$$s_R = (3\ [\mathrm{s}])V_{TO}$$

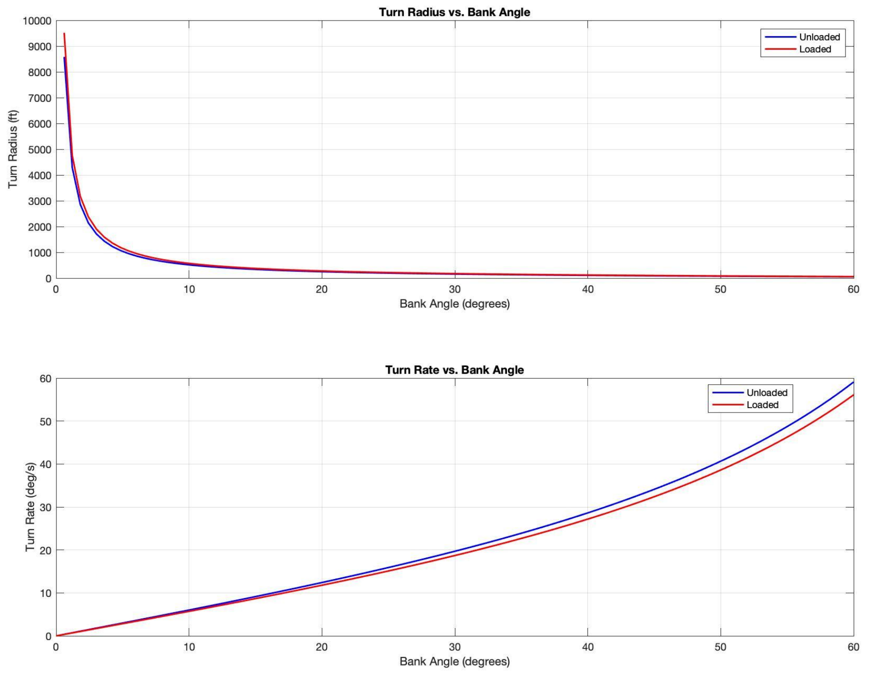
$$s_G = \frac{1}{2f_2}\ln\left(1 + \frac{f_2}{f_1}V_{TO}^2\right)$$

- Calculated takeoff distance from ground roll and rotation equations
- Used XFLR5 to calculate $C_{L,max}$ and C_{D0} from drag study

Unloaded → s_{TO} = 24.2503 ft

Loaded → s_{TO} = 29.6228 ft

AIRCRAFT PERFORMANCE: TURNING



Parameter	Loaded	Unloaded
Cruise Speed	56.9 ft/s	54.7 ft/s
Max Turn Rate	67.09 deg/s	62.99 deg/s
Min Turn Radius	49.15 ft	48.60 ft
Max Bank Angle	61.56 deg	64.23 deg

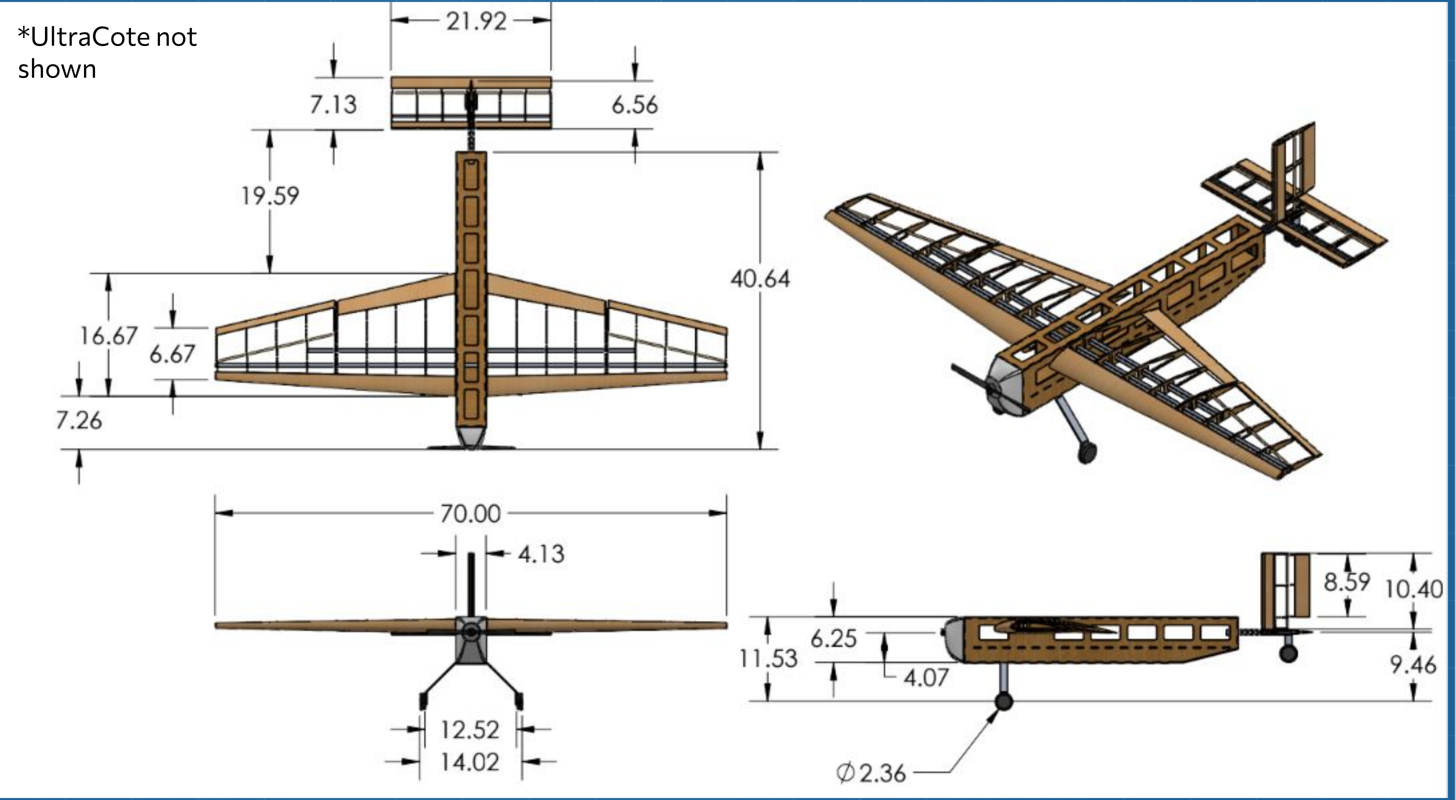
Found using matlab code that iterated through bank angles from 0 to 60. Used the loaded and unloaded speed and max load factors to find values.

$$R = \frac{V_\infty^2}{g * \sqrt{n^2 - 1}}$$

$$\psi = \frac{g * \sqrt{n^2 - 1}}{v}$$

FULL DESIGN SUMMARY: CAD

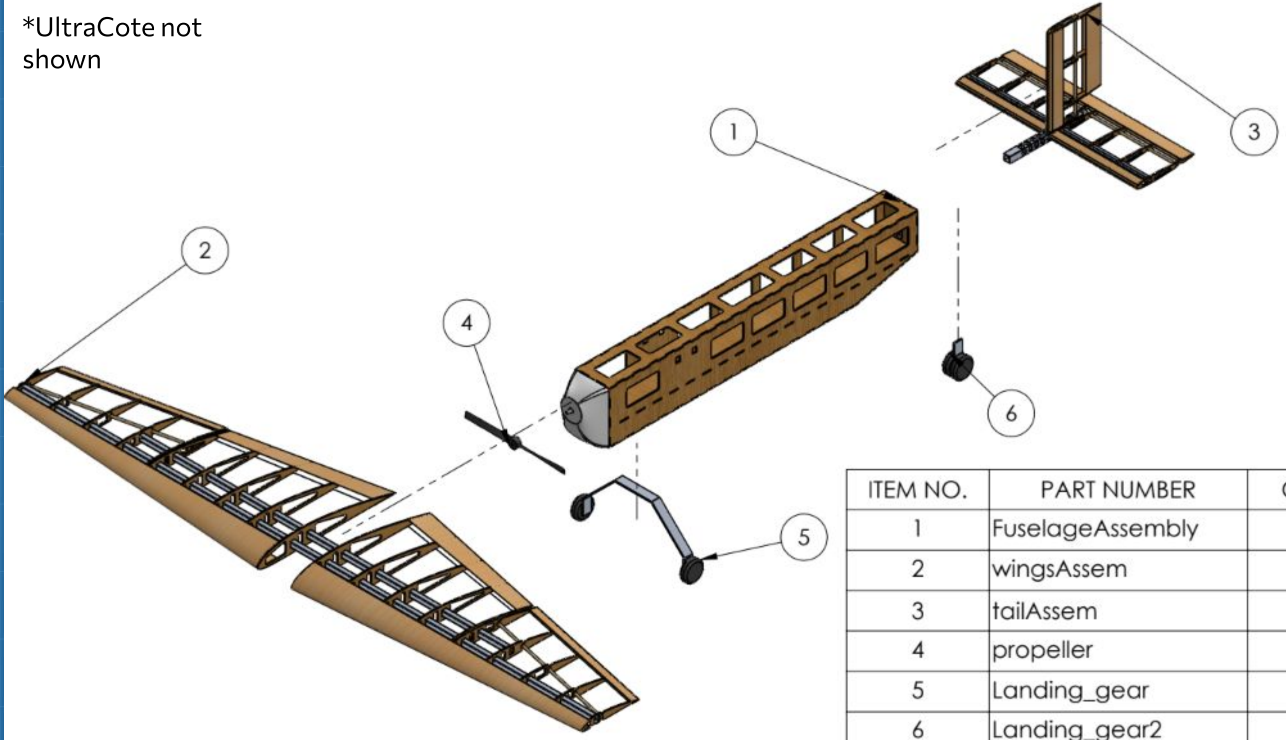
- Fully dimensioned plane is shown for design and construction reference.



FULL DESIGN SUMMARY: CAD

- Components will be assembled in the geometry shown.
- Fasteners are not included in this assembly for clarity of information presentation.

*UltraCote not shown



ITEM NO.	PART NUMBER	QTY.
1	FuselageAssembly	1
2	wingsAssem	1
3	tailAssem	1
4	propeller	1
5	Landing_gear	1
6	Landing_gear2	1

FULL DESIGN SUMMARY: CHARACTERISTICS

Wing	
Parameter	**Value**
Airfoil	N-22
Area, S	816.66 in^2
Span, b	70 in.
Aspect Ratio, AR	6.0
Root Chord, c_r	16.66 in.
Tip Chord, c_t	6.66 in.
Taper Ratio, λ	0.4
MAC, $\bar{c}$	12.38 in.
LE Sweep, Λ_{LE}	4.336°
$\bar{c}/4$ Sweep, $\Lambda_{\bar{c}/4}$	0°

Tails	
Parameter	**Value**
Airfoils	NACA 0010
HT Area, S_{HT}	149.45 in^2
HT Span, b_{HT}	21.17 in.
HT Aspect Ratio, AR_{HT}	3.0
HT Chord, c_{HT}	7.05 in.
VT Area, S_{VT}	67.6 in^2
VT Span, b_{VT}	10.4 in.
VT Aspect Ratio, AR_{VT}	1.6
VT Chord, c_{VT}	6.5 in.

Control Surface	
Parameter	**Value**
Flaperon Area, S_f	148.95 in^2
Flaperon Span, b_f	33.1 in.
Flaperon Chord, c_f	4.5 in.
Elevator Area, S_e	44.835 in^2
Elevator Span, b_e	21.17 in.
Elevator Chord, c_e	2.118 in.
Rudder Area, S_r	31.772 in^2
Rudder Span, b_r	10.4 in.
Rudder Chord, c_r	3.055 in.

FULL DESIGN SUMMARY: CHARACTERISTICS

- The weights found through the CAD in SolidWorks were added to the expected weight of components and glue.
- The result was incredibly similar to the initial weight estimate
    - As such, the initial estimates were very accurate and are improved here slightly

Performance		
Parameter	**Unloaded**	**Loaded**
TO Speed	7.662 ft/s	9.018 ft/s
Cruise Speed	54.04 ft/s	56.91 ft/s
TO Distance	24.25 ft	29.62 ft
Min. Turn Radius	49.15 ft	48.60 ft
Max. Bank Angle	61.56 deg	64.23 deg
Max. Turn Rate	63.04 deg/s	67.15 deg/s

Weights and Loading		
Parameter	**Unloaded**	**Loaded**
Weight	8.744 lb	12.279 lb
Wing Loading	0.0104 psi	0.0147 psi
Max Load Factor	2.1	2.3
Max Gust Load Factor	5.95	5.70

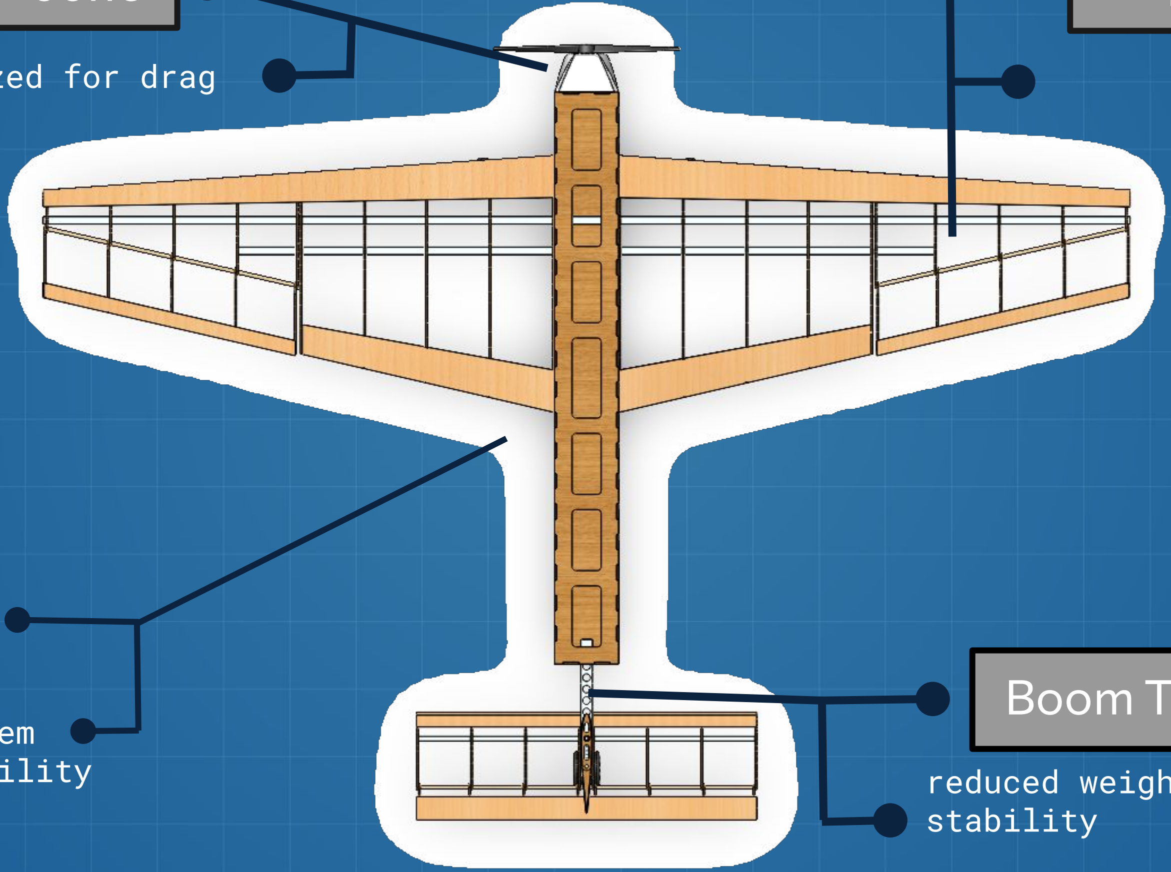
DESIGN UNIQUENESS

Nose Cone

optimized for drag

Tapered Wings

lower drag &
improved
maneuverability
&
unswept at
quarter-chord



SAS System

Looking into system
for improved stability
& control

Boom Tail

reduced weight & increased
stability

QUESTIONS

and

FEEDBACK