AERO2579 Aerodynamic Design Report for Group Number 11

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Abstract

Explored within this report is the validity of designing a low-speed, small scale glider. Specifically considered are the aerofoils viable for such a task and their subsequent performance. Drawing comparisons between Lift Coefficients and Drag Coefficients and employing boundary layer and Lifting line analysis to further determine the performance of the glider. The paper is structured to step through the problem linearly, starting with an assessment of viable aerofoils continuing onto estimating the performance of those air foils when constructed in a wing, stability and material design are evaluated next based on the wing dimensions and weight requirements. Finally the testing and prototyping process is outlined, and the final design is described. It is clear from our findings that for a small-scale project like the one being considered in this paper a simple to manufacture flat plate is more than viable to get sufficient lift, control and stability for a simple glider.

Introduction

Investigated in the following report is the problem of optimising and designing a simple, small scale, unpowered glider. The primary metric considered when measuring the success of the design is its glide ratio. The areas of significant import are therefore the structure of the main wing regarding lift and drag and the corresponding tail plane to ensure stability. As the glider operates autonomously all stability features need to move the glider towards cruise conditions without intervention.

It is accepted that an unpowered aircraft such as we are designing needs to minimise drag as to avoid losing airspeed and maximise the available lift. This is necessary as extra energy cannot be introduced due to the absence of continuous propulsion and therefore the initial propulsive energy is all that is available to the glider. Full scale gliders can take advantage of natural thermal updrafts to increase flight time, our glider however will not take advantage of such effects and will rely only on the initial energy made available at launch.

The importance of reducing drag and increasing the lift area can be summed up with a consideration of Lift Coefficient:

$$C_L = \frac{2L}{\rho v^2 A}$$

Clearly drag which induces a deceleration would rapidly decrease the lift available to the wing, as lift is proportional to the velocity squared. Further an increase in Area acts to increase the total lift available to the wing in general which is advantageous to a glider which needs to minimise the rate altitude is lost.

This report will therefore be taking the above into account when considering a selection of aerofoils that could suit the application. The aerofoils will be evaluated using boundary layer analysis and computational fluid dynamics; the results can then be evaluated and compared to assess the viability for our application. The viability of any individual airfoil however will be contingent on manufacturing requirements, as the theoretical performance of a complex design is likely to be more sensitive to manufacturing errors and imprecision.

The aircraft dimensions and weight will be designed around the most viable airfoil, the report will outline the material selection and main wing design parameters. Further stability and lifting line analysis can be completed to assess the performance of the complete design.

Testing of prototypes will allow for the evaluation of the design and will confirm our theoretical findings. An iterative design mentality will be used to further improve upon the design and will culminate in a final design that serves to satisfy the requirements outlined in the design brief.

Airfoil Analysis

Lift Considerations

Chord	Span	Velocity	Density	Viscosity	AR	Re
m	m	m/s	kg/m^3	mPa s		
0.1	0.4	10	1.225	0.01813	4	67567.57

Figure 1 ~ Airfoil Qualities and Environmental Assumptions @ Standard Temp and Pressure

Flat Plate

The airfoil of primary consideration for our design is a flat plate airfoil. Due to the ease of manufacturing and relatively good performance regarding the application it seemed like a solid and safe candidate for the glider's main wing. Outlined below are the performance metrics for the airfoil, considering the lift generation, drag and estimated glide ratio.

AoA	AoA	Cl	Lift	Mass	Mass
Deg	Rad		N	Kg	g
1	0.0174533	0.109662	0.268673	0.027388	27.38762
2	0.0349066	0.219325	0.537345	0.054775	54.77524
2.5	0.0436332	0.274156	0.671681	0.068469	68.46905
3	0.0523599	0.328987	0.806018	0.082163	82.16286
4	0.0698132	0.438649	1.07469	0.10955	109.5505
5	0.0872665	0.548311	1.343363	0.136938	136.9381
6	0.1047198	0.657974	1.612035	0.164326	164.3257
7	0.1221730	0.767636	1.880708	0.191713	191.7133
8	0.1396263	0.877298	2.149381	0.219101	219.101
9	0.1570796	0.98696	2.418053	0.246489	246.4886

Figure 2 ~ Relating AoA to C_L and subsequently Lift and Weight

In considering the viability of a flat plate airfoil referring to figure 2. At low angles of attack and taking into consideration the constraints outlined in figure 1 the airfoil at a 2.5 Degree AoA can theoretically lift 68.5 Grams. With our max weight being constrained by design at 50 Grams this leaves a buffer of 37 Percent to account for theoretical error and design inefficiencies.

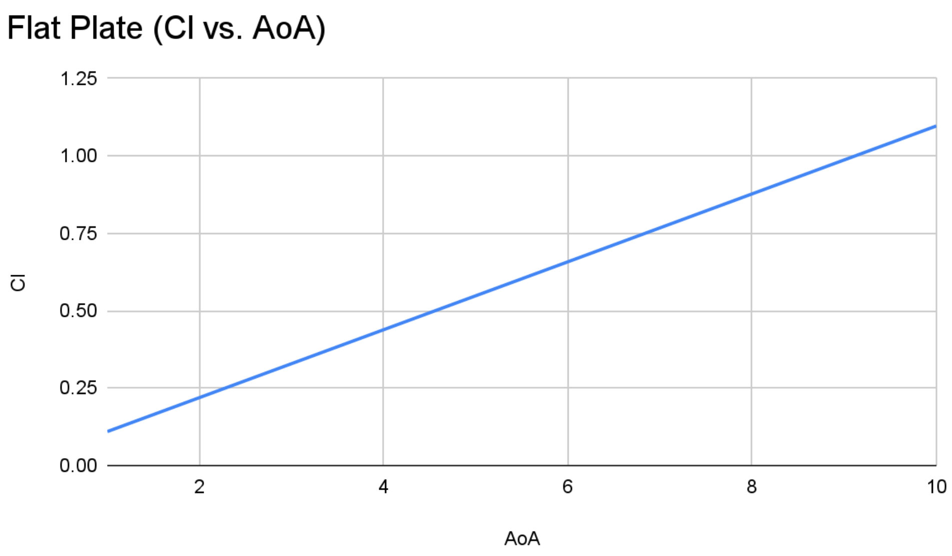


Figure 3 ~ Flat Plate Cl vs AoA using $C_L = 2\pi\alpha$

NACA 4412

The advantages of this airfoil over a flat plate come with the introduction of camber and the generation of lift at an AoA of 0. However as the lift at 0 AoA is not enough to carry the aircraft sufficiently the wing would either have to be mounted at the desired AoA or launched at the AoA. The Lift coefficient is nearly double that of the flat plate while also maintaining a better drag profile than the flat plate. The Airfoil is relatively simple to manufacture although still much more involved when compared to a flat plate.

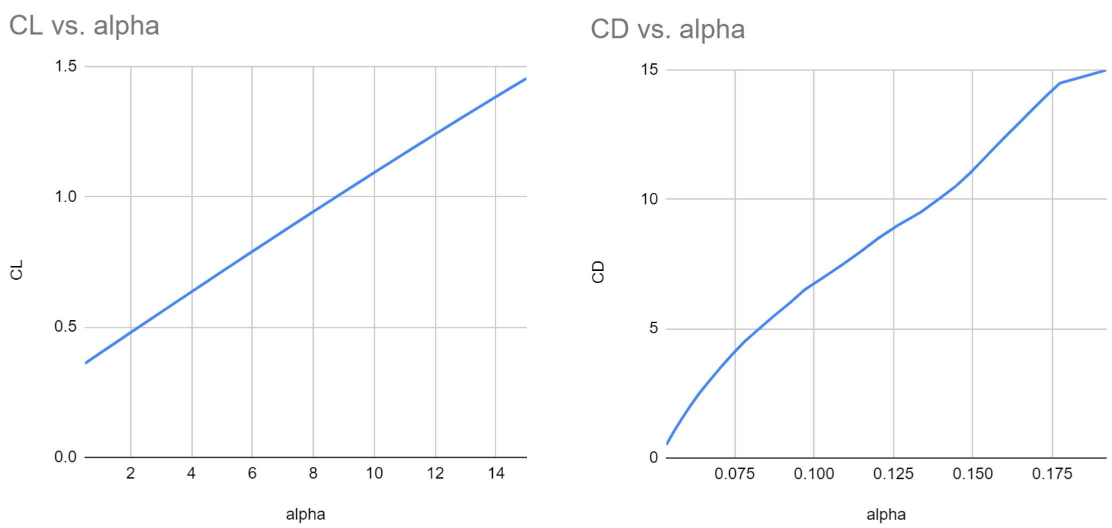


Figure 4 ~ NACA 4412 Cd and Cl vs AoA from CFD using Xflr5

Drag Considerations

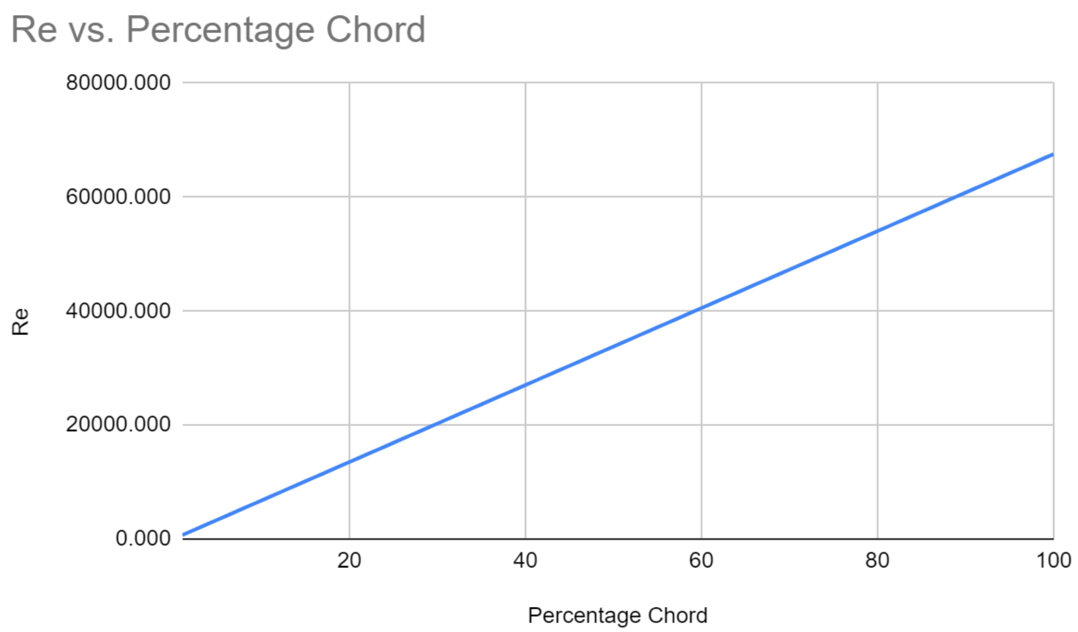


Figure 5 ~ Reynolds Number vs Percentage Chord

Referring to Figure 5. The Airfoil experiences a short laminar flow regime remaining below a Re of 4000 up to 6 Percent of the Chord length. Therefore 94 Percent of the airfoil boundary layer is turbulent or transitional as such the boundary layer calculations will assume turbulent flow.

$$C_{d,skin} = \frac{0.074}{Re^{0.2}} \tag{1}$$

$$C_{d,skin} = \frac{0.074}{67567.57^{0.2}} \tag{2}$$

$$C_{d,skin} = 0.008 \tag{3}$$

$$C_{d,skin,total} = 0.008 * 2 = 0.016 \tag{4}$$

The above coefficient of skin friction calculation can be applied to all airfoils considered.

Wing Analysis

Main Wing

Airfoil selection, Dimensions and Considerations

Airfoil Selection

Option	Description
Flat Plate	**An easier option with similar performance at a low Reynold's number**
NACA Airfoil	An option for increased lift performance, however at the expense of weight and manufacturing simplicity
Blended Wing-body	Similar to a B-2 bomber 'triangular' style. Highly unstable and with no control surfaces. Hard to recover from instability and sudden gusts as it is only a glider

Wing Parameters

Angle of Incidence
Set at zero for ease of manufacturing - so that the connecting fuselage rod can be attached flat for the entire chord length.

Dihedral
Improves the stability in terms of roll control, with a value of 5 degrees being enough to satisfy the stability requirements.

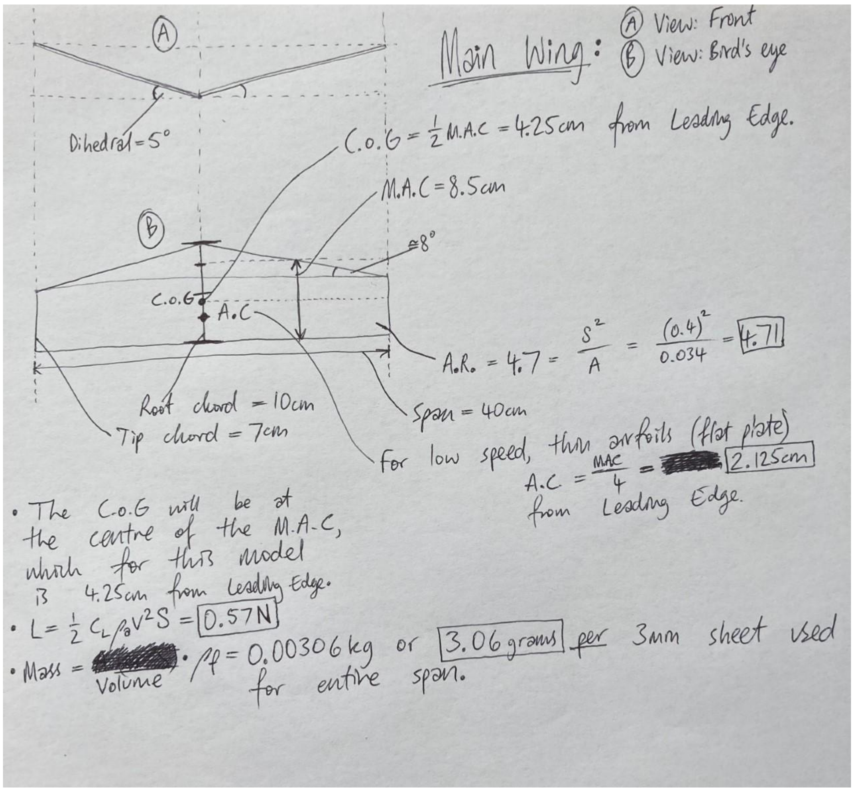
Taper ratio
The ratio between the chord length at the tip to the chord length at the root for a rectangular wing, or the span to the power of two to area ratio (S^2 / A) which can be used to reduce induced drag developed at the wingtips. We will go with a set trailing edge taper of 8 degrees which makes it more stable and less maneuverable, which means it will theoretically fly further by sticking to a straighter flight path.

Sweep ratio
Set at 0 for ease of manufacturing as it doesn't assist our performance in low speed flight.

Aspect ratio
The desired aspect ratio for a glider is anywhere between 4.5 and 7.5. A high aspect ratio is desired because it gives a lower drag, resulting in a more efficient glide angle. We are setting ours at a value of 4.7 (REF - Glider Design! – Design Hints & Tips).

Material Selection

Material	Description and Justification
Balsa wood	Lightweight and a density of only 0.11-0.14g/cm3
Expanded polystyrene (EPS)	Has a high strength to weight ratio and is very lightweight with a density of 0.011-0.032g/cm3 However if it has to be cut to shape it might add too much drag through surface roughness unless we came up with a process to smooth the surface out consistently, or buy pre cut sheets of it. Ended up purchasing 3mm thick sheets which have a density of 0.03g/cm3
Extruded polystyrene foam (XPS)	Has less surface roughness and offers higher stiffness compared to EPS, however it is more dense at 0.028-0.045g/cm3. It is also far more expensive and difficult to find suppliers that sell in small quantities
Glad wrap over bamboo skewers	Too much weight in the large structure and potential to puncture unless we wrapped several layers, which would make it far too heavy



Tail Plane

Airfoil selection, Dimensions and Considerations

Airfoil Selection

Option	Description
Flat plate	
NACA 4412	
NACA reflexed airfoil	Could be used to satisfy longitudinal stability requirements if the A.C is behind the C.o.G. However in our model it will be designed so that this isn't the case
A symmetrical NACA airfoil	Wouldn't be useful in application as we won't use it to control pitch through producing both positive and negative angles of attack as required in flight. Hence no airfoil analysis was considered

Style

Option	Description and Justification
Forewing 'canard' configuration	A small wing placed forward of the main wing. Makes it look a lot like the configuration of a lot of jets, which would make it look quite stylish for a balsa wood construction. This has the benefit of avoiding deep stall. This is because it is located forward of the main wing and so if the AoA becomes too great: it stalls first, causing it to begin to fall out of the stall. However, a downside to this is that the main wing won't achieve its maximum lift potential. This means it isn't the most efficient as its trade off is for increased safety in the form of greater stability
Reflexed trailing edge	They have a camber line that towards the trailing edge is concave in the upwards direction. This change in camber helps stability as it helps the aircraft avoid sudden pitch up manoeuvres caused by instability. One of the issues preventing us from going with this option is that it is difficult to manufacture and an airfoil would add significant weight. It would also be fragile and if it was broken in testing would be time

	consuming to replace
V-tail	A way to include both horizontal and vertical stability – through having the surfaces angled on a diagonal and therefore using the same surface for both. Can still achieve longitudinal stability with the benefit of a weight reduction. However, this design is more susceptible to lock into a dutch roll compared to a conventional configuration
Regular aft configuration of a flat plate horizontal stabiliser	Most convenient and performs all necessary tail functions

Tail Parameters

Optimum tail arm
This is a tail parameter that must be determined to satisfy the longitudinal stability requirements, as it performs the role of controlling the magnitude of the tail pitching moment.

Fixed, all-moving or adjustable
Our design will be fixed as we don't want to include any mechanism to control movement of its angle of attack because it would add significant weight and unnecessary technicality.

Tail incidence
Set at zero, following the main wing's design. The glider will be more sturdy if the entire tail surface is attached to the fuselage rod instead of just at a point, making it less susceptible to breakages while test flights are carried out.

Aspect Ratio
The ratio between the tail span and the tail mean aerodynamic chord (MAC). The tail's lift curve slope can be increased through increasing the aspect ratio. Typical value range is somewhere around ⅔ of the A.R of the main wing, so we'll set it at a value of 3.14.

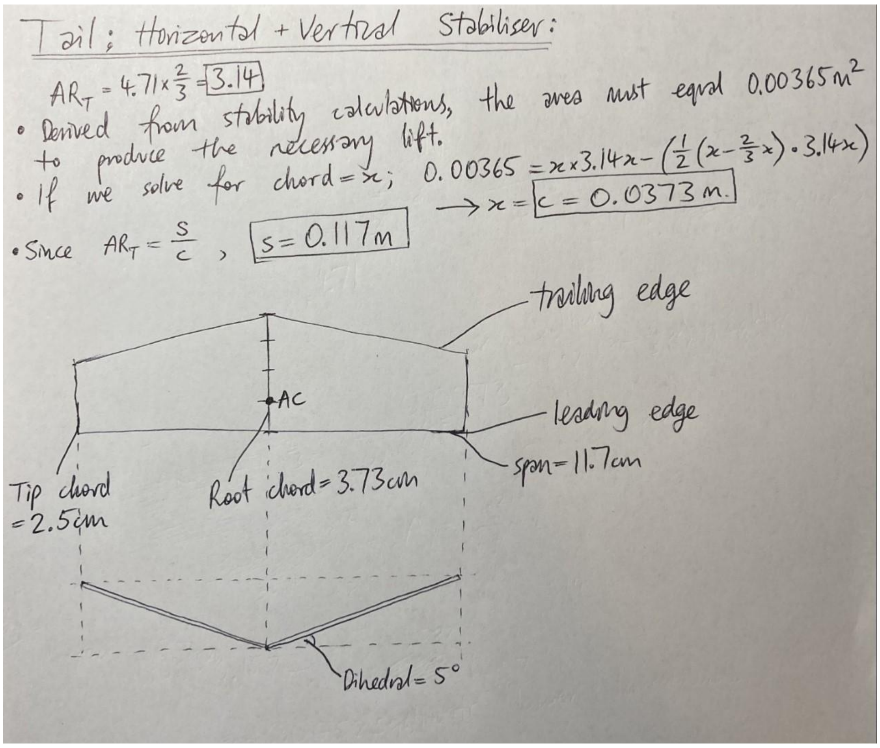
Taper ratio
Ratio between the chord length at the tip to the chord length at the root. Set at the same value as the main wing's which is 8 degrees of leading edge taper.

Sweep ratio
The sweep ratio is usually that of the main wing for the tail also, set at 0 for ease of manufacturing as it doesn't assist our performance in low speed flight.

Dihedral angle
Designed to be the same value as that of the main wing, it is set at a value of 5 degrees.

Material Selection

Option	Description and Justification
Balsa wood	Lightweight and a density of only 0.11-0.14g/cm^3
Expanded polystyrene (EPS)	Has a high strength to weight ratio and is very lightweight with a density of 0.011-0.032g/cm3. However if it has to be cut to shape it might add too much drag through surface roughness unless we came up with a process to smooth the surface out consistently, or bought pre cut sheets of it
Extruded polystyrene foam (XPS)	Has less surface roughness and offers higher stiffness compared to EPS, however it is more dense at 0.028-0.045g/cm3. It is also far more expensive and difficult to find suppliers that sell in small quantities
Glad wrap over bamboo skewers	Too much weight in the large structure and potential to puncture unless we wrapped several layers, which would make it far too heavy



Lifting Line Analysis

Table 1: Input Parameters used for Lifting Line analysis

Parameter	Value	Unit
2D lift curve slope (a_0)	2π	1/rad
Reference angle of attack (α_{ref})	2.5	deg
Twist factor (C)	0	deg
Velocity (V)	10	m/s
Mean chord ()	0.085	m
wingspan (b)	0.4	m
Taper ratio (λ)	0.07	
Air Density (ρ)	1.225	kg/m^3

Table 2: Lifting Line metrics

Y_{CPj}	Value	Y_{ri}	Value	b_{ri}	Value	$Chord_{CPj}$	Value	$ageom_{CPj}$	Value	Q_{ij}	Value
YCP1	0.18	*Yr1*	-0.19	*br1*	0.38	*ChordCP1*	0.073	*ageomCP1*	2.5	*Qij*	
YCP2	0.16	*Yr2*	-0.17	*br2*	0.34	*ChordCP2*	0.076	*ageomCP2*	2.5		
YCP3	0.14	*Yr3*	-0.15	*br3*	0.3	*ChordCP3*	0.079	*ageomCP3*	2.5		
YCP4	0.12	*Yr4*	-0.13	*br4*	0.26	*ChordCP4*	0.082	*ageomCP4*	2.5		

YCP5	0.1	*Yr5*	-0.11	*br5*	0.22	*ChordCP5*	0.085	*ageomCP5*	2.5		
YCP6	0.08	*Yr6*	-0.09	*br6*	0.18	*ChordCP6*	0.088	*ageomCP6*	2.5		
YCP7	0.06	*Yr7*	-0.07	*br7*	0.14	*ChordCP7*	0.091	*ageomCP7*	2.5		
YCP8	0.04	*Yr8*	-0.05	*br8*	0.1	*ChordCP8*	0.094	*ageomCP8*	2.5		
YCP9	0.02	*Yr9*	-0.03	*br9*	0.06	*ChordCP9*	0.097	*ageomCP9*	2.5		
YCP10	0	*Yr10*	-0.01	*br10*	0.02	*ChordCP10*	0.1	*ageomCP10*	2.5		

Table 3: Q_{ji} matrix

Q_{ji}									
-107.3908732	101.577232	31.68630231	17.53990153	11.33214555	7.745540564	5.323298787	3.497150422	1.991710431	0.6474600471
-39.39766934	-112.160829	105.3503043	32.53346561	17.74047099	11.19723234	7.362646063	4.712639873	2.644420593	0.8538194594
-26.0608985	-41.37001714	-117.0612047	108.9680844	33.19335493	17.71186923	10.77706329	6.617495002	3.630775333	1.160606816
-20.56839274	-27.5414885	-43.50235111	-122.1546019	112.3495503	33.55895657	17.30935655	9.870281345	5.220282133	1.642746126
-17.7265678	-21.90275169	-29.22084755	-45.87814229	-127.5513187	115.3454509	33.42253805	16.2338042	8.027705371	2.459667302
-16.1277009	-19.04766359	-23.48771086	-31.21255798	-48.65115313	-133.4654629	117.6473339	32.32069614	13.75098708	4.001609998
-15.24067735	-17.51710887	-20.6901426	-25.48154405	-33.73710311	-52.13915936	-140.3746598	118.4980894	28.96619964	7.448451337
-14.83047278	-16.76898455	-19.32704524	-22.88086358	-28.21135048	-37.28633036	-57.12215594	-149.6056465	115.4100702	17.95267758
-14.78937278	-16.57556849	-18.86094099	-21.89393272	-26.12589604	-32.48001007	-43.22648254	-66.16298349	-166.7307184	92.62817688
-15.07783671	-16.85169986	-19.09859317	-22.03683827	-26.04353614	-31.83098862	-40.9255568	-57.29577951	-95.49296586	-286.4788976

Table 4: P matrix

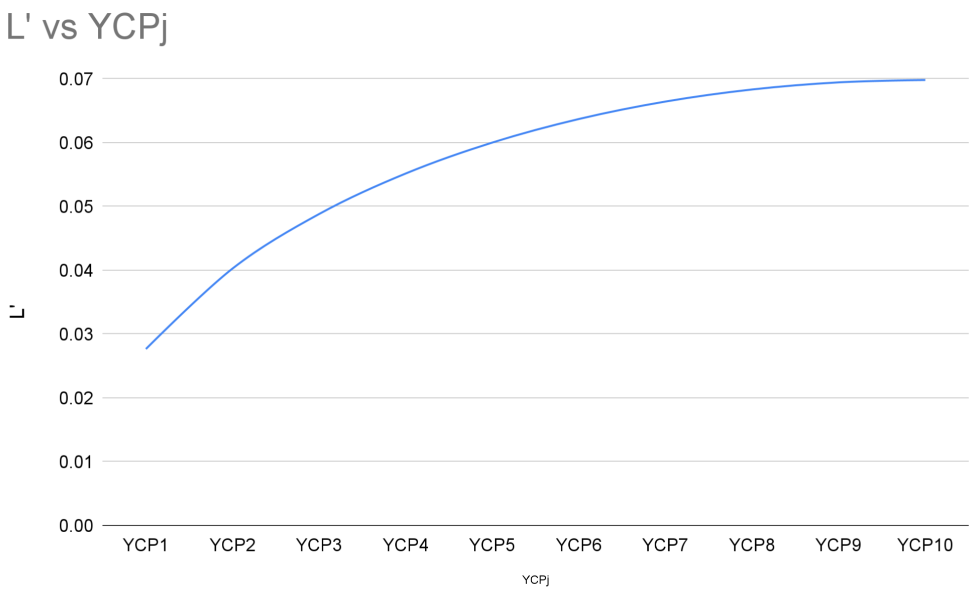
P									
1	0	0	0	0	0	0	0	0	0
1	1	0	0	0	0	0	0	0	0
1	1	1	0	0	0	0	0	0	0
1	1	1	1	0	0	0	0	0	0
1	1	1	1	1	0	0	0	0	0
1	1	1	1	1	1	0	0	0	0
1	1	1	1	1	1	1	0	0	0
1	1	1	1	1	1	1	1	0	0
1	1	1	1	1	1	1	1	1	0
1	1	1	1	1	1	1	1	1	1

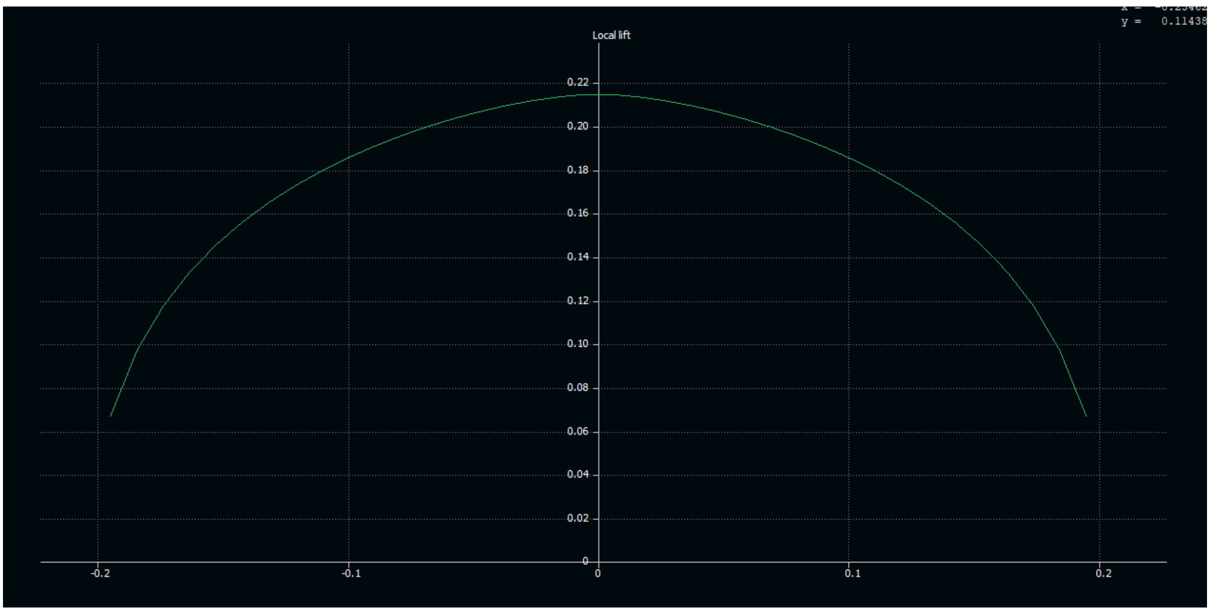
Table 5: Lifting line Matrices

β	Γ_i	Γ_{cpj}	L'_{CPj}	$\alpha_{induced\ j}$	D'_{CPj}
0.1000668224	0.002256135004	0.002256135004	0.0276376538	-0.4264946407	-0.0114331998
0.1041791576	0.001029034538	0.003285169543	0.0402433269	-0.409659326	-0.01602879096
0.1082914927	0.0006977954608	0.003982965004	0.04879132129	-0.3941026427	-0.01873488063
0.1124038279	0.0005182304359	0.004501195439	0.05513964413	-0.3796842533	-0.02043625347
0.1165161631	0.0003958010633	0.004896996503	0.05998820716	-0.3662836326	-0.02148466072
0.1206284982	0.0003016079617	0.005198604464	0.06368290469	-0.3537966906	-0.02206369602
0.1247408334	0.0002232897745	0.005421894239	0.06641820443	-0.3421330634	-0.02228312742
0.1288531686	0.000154395281	0.00557628952	0.06830954662	-0.3312139231	-0.02221366426
0.1329655037	0.00009100511944	0.005667294639	0.06942435933	-0.3209701935	-0.02190250746
0.1370778389	0.00003033792061	0.00569763256	0.06979599886	-0.3113410877	-0.02138099393

Table 6: Total forces computed by lifting line theory

Total Lift Force (N)	Total Drag Force (N)
2.138N	**-0.749N**





Despite both the lifting line analysis and the local lift calculated in xflr5 are following the same trend, there is a major discrepancy between them in the order of magnitudes as xflr5 is unable to simulate flat plates. To counter this limitation we instead settled to use a very thin NACA 0006 airfoil for its mostly flat profile. A major problem with this approach is that it behaves more like an airfoil rather than a flat plate.

Glider Design

Launch configurations

Angle of attack at launch: Set at 2.5 degrees

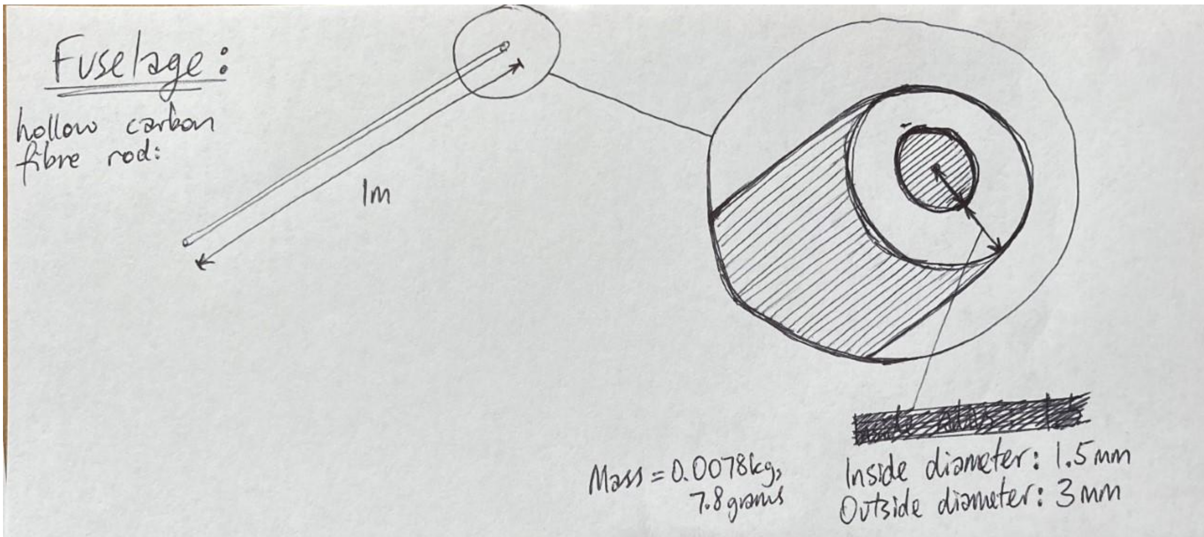
Launch Velocity: 10m/s

Fuselage

Since our design is a glider and it doesn't have any cargo capacity requirements, a standard fuselage is unnecessary as this is its primary function. Therefore, to make it as lightweight as possible we'll use a rod that's either wooden or carbon based (potentially use a section of fishing rod). It needs to be strong yet lightweight.

Option	Description and Justification
Balsa wood rod	Density of only 0.11-0.14cm3 which would be great to cut down on weight but it might not be sturdy enough with a small diameter, potentially breaking if large velocity high impact crashes occur during testing
Plastic rod	We could 3-D print one but it would be too heavy. If we tried to hollow it out at that small of a scale the machines we have access to wouldn't be able to form the structure with enough precision
Radiata pine wood	A heavier wood with a higher specific strength than balsa, with a density of 0.545g/cm3. Can get rods of this from bunnings up to 1.2m, however diameter is set at 12.5mm. Mass for a 1m section = 67g
Tasmanian oak rod	A higher density than radiata pine at 0.78g/cm3, however is available at a thinner diameter of 9.5mm. Mass for a 1m section = 55g
Carbon fibre rod	Carbon fibre has a high strength to weight ratio (specific strength). It has a density of 2.267g/cm3.This seems too heavy in

	comparison to other options that serve the same purpose. It also adds manufacturing difficulty as it would need to be cut to size which could be difficult without the necessary tools. However, a hollow tube of dimensions 3mm outside diameter and 1.5mm inside diameter only weighs 7.8g for a 1m section and would provide less surface drag as it has a far smaller frontal area compared to the solid wood options



Manufacturing considerations

How shall we attach the main wing and horizontal/vertical stabilisers to the carbon rod fuselage?

Superglue - highest strength per amount weight added

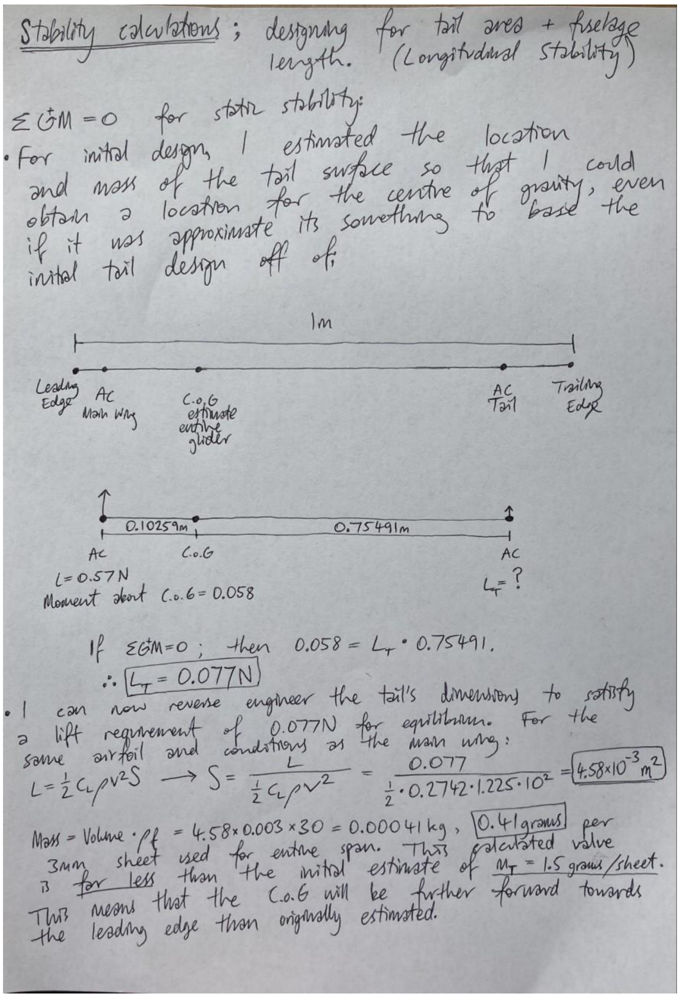
Round off through sanding with sandpaper the leading edge of the main wing and tail in order to create less induced surface drag if made out of balsa wood. If constructed from polystyrene sheets, may need to somehow carefully carve a curvature and smooth down the surface.

Set the angle of incidence of the lifting surfaces to 0 for stability in our structure and ease of assembly.

Stability

In all cases longitudinal trim must hold. We desire the main wing's aerodynamic centre (A.C) to be in front of the centre of gravity (C.o.G) so that the tail can be used as a lifting surface as well, to counteract the moment generated from the main wing's lift about the C.o.G. To begin, I estimated values for the tail to figure out a rough C.o.G location;

Finding the position of the C.o.G				
Dataset 1	Main wing:	Fuselage:	Tail:	
Distance of C.o.G for part from leading edge (m):	0.0425	0.5	0.9	initial estimated value
Mass (kg)	0.00306	0.0078	0.0015	estimated at half of main wing value
Mass x distance (mass distance moment)	0.00013005	0.0039	0.00135	
Averaged mass location (C.o.G)	0.145093042			
Lift generated at 2.5deg AoA (N)	0.57	0	0.077	
Moment generated (Nm)	0.0584763	0	0.05812807	



Stability calculations; designing for tail area + fuselage length. (Longitudinal Stability)

$\Sigma \overset{+}{G}M = 0$ for static stability:

- For initial design, I estimated the location and mass of the tail surface so that I could obtain a location for the centre of gravity, even if it was approximate its something to base the initial tail design off of;

$L = 0.57N$

Moment about C.o.G = 0.058

$L_T = ?$

If $\Sigma \overset{+}{G}M = 0$; then $0.058 = L_T \cdot 0.75491$.

$\therefore \boxed{L_T = 0.077N}$

- I can now reverse engineer the tail's dimensions to satisfy a lift requirement of 0.077N for equilibrium. For the same airfoil and conditions as the main wing:

$$L = \frac{1}{2} C_L \rho v^2 S \longrightarrow S = \frac{L}{\frac{1}{2} C_L \rho v^2} = \frac{0.077}{\frac{1}{2} \cdot 0.2742 \cdot 1.225 \cdot 10^2} = \boxed{4.58 \times 10^{-3} m^2}$$

Mass = Volume $\cdot \rho_f$ = $4.58 \times 0.003 \times 30 = 0.00041 kg$, $\boxed{0.41 \text{grams}}$ per 3mm sheet used for entire span. This calculated value is far less than the initial estimate of $M_T = 1.5$ grams/sheet. This means that the C.o.G will be further forward towards the leading edge than originally estimated.

With these results, to design a more stable and efficient glider we need to change the updated input data for the tail, to calculate a more accurate C.o.G and tail parameters. This iterative process is tedious yet necessary for efficiency and accuracy.

Dataset 2 (Updated tail mass and position)	Main wing:	Fuselage:	Tail:
Distance of C.o.G for part from leading edge (m):	0.0425	0.5	0.95
Mass / 3mm sheet (kg)	0.00306	0.0078	0.00041
Mass x distance (mass distance moment)	0.00013005	0.0039	0.0003895
Averaged mass location (C.o.G)	0.130717243		
Lift generated at 2.5deg AoA (N)	0.57	0	0.06137
Moment generated (Nm)	0.05028369	0	0.050279383

$L = 0.57N$

Moment about C.o.G $= 0.57 \times 0.088217 = 0.05028$

$\therefore \quad 0.05028 = L_T \cdot (0.95 - 0.13) \longrightarrow$ New $\boxed{L_T = \cancel{0.041197N}\ 0.06137N}$

$L = \frac{1}{2} C_L \rho v^2 S \longrightarrow S = \frac{L_T}{\frac{1}{2} C_L \rho v^2} = \frac{0.06137}{\frac{1}{2} \cdot 0.274156 \cdot 1.225 \cdot 10^2} = \boxed{3.65 \times 10^{-3} m^2}$

Mass = Volume $\cdot \rho = 3.65 \times 10^{-3} \times 0.003 \times 30 = 0.0003285 = \boxed{0.33 \text{ grams}}$ per 3mm sheet used. This is 0.0000837kg off the predicted value, which is a fine error to work with.

With this calculated we can now go on to calculating the tail dimensions.

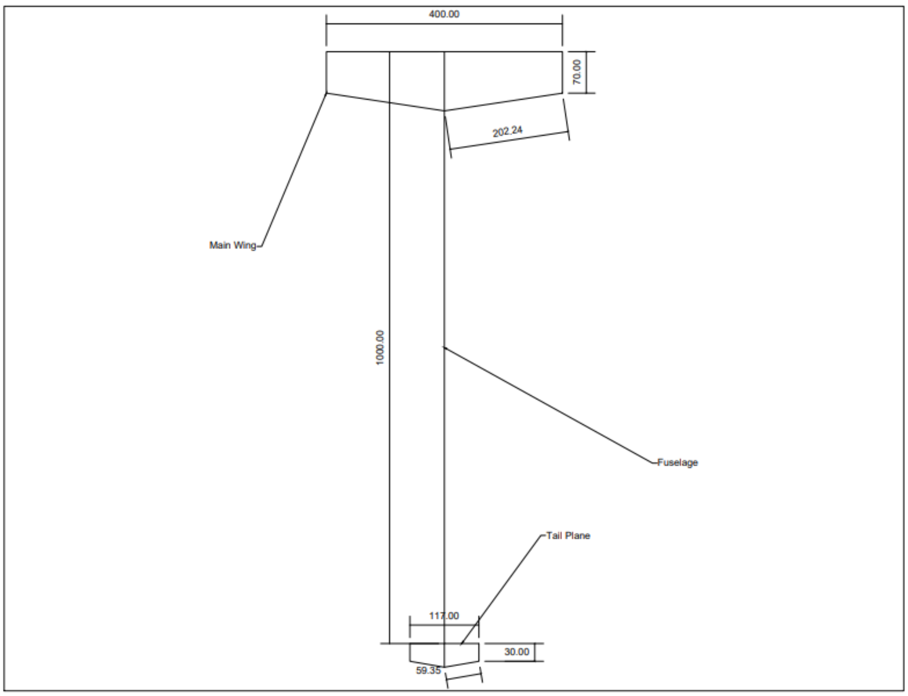
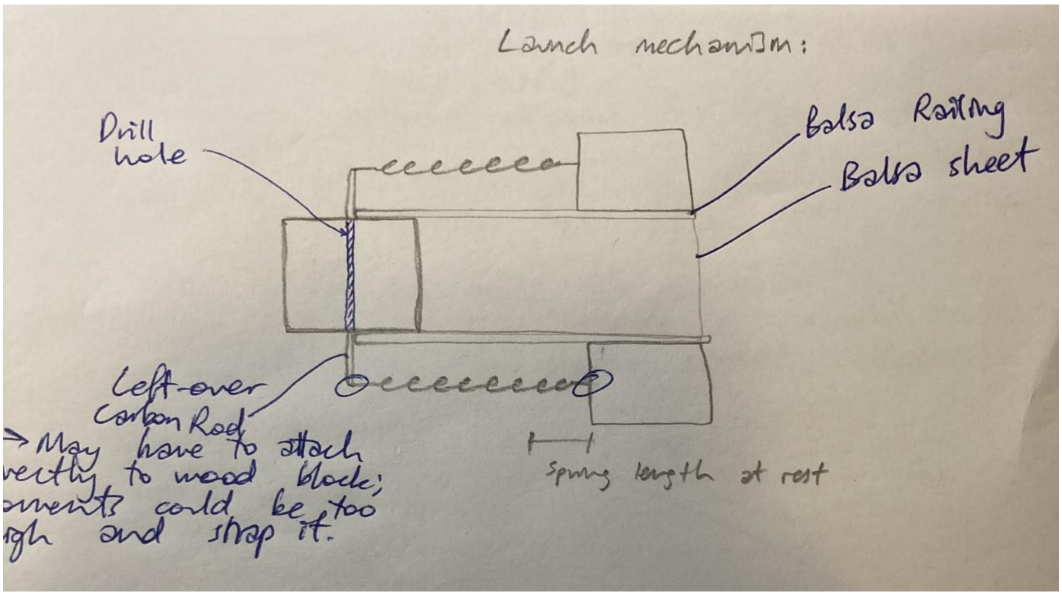


Figure 5 ~ Glider CAD Dimensions (Top Down)

Flight Testing the Prototype

Tested at Princess Park, Carlton North on the grass oval. This is because it's an open area with patches of longer grass, which offer a nice compressive layer for when our glider touches back down.

Launch mechanism



Flight test data

Couldn't construct/test as a group due to Covid-19 lockdowns. Hardware stores are closed to the public and we weren't allowed to meet up to collaborate and test.

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