ANNEXURE A – REPORT FRONT PAGE

2021 SAEISS AERO DESIGN CHALLENGE

DESIGN REPORT

TEAM PEGASUS | ALPHA

ADC20210118

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30 May 2021

SAEINDIA SOUTHERN SECTION

ANNEXURE B

2021 SAEISS AERO DESIGN CHALLENGE

STATEMENT OF COMPILANCE

CERTIFICATION OF QUALIFICATION

Team Name	PEGASUS | ALPHA
Team ID	ADC20210118
College/University	BITS PILANI K.K. BIRLA GOA CAMPUS
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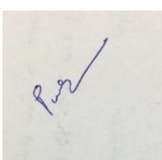
As Faculty Advisor, I certify that the registered team members are enrolled in collegiate courses. This team has designed, constructed and/or modified the radio-controlled airplane they will use for the SAE Aero Design Challenge 2021 competition, without direct assistance from professional engineers, R/C model experts or pilots, or related professionals.



Signature of Faculty Advisor

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Aerodynamic Design and Analysis

Design Summary

This aircraft is designed to accommodate a variable lifting regime, depending on the situation of use. In the application of search and rescue, our aircraft has the capability to fly in multiple configurations. The Morphing Wing design allows for increased lift without much change in other characteristics.

Primary use case: The aircraft is a Search and Rescue drone, needing to fly for a long period of time over a disaster-struck area, locating trapped people.

Design Requirement: The endurance is high for the aircraft, i.e., power consumption of the aircraft is minimal.

Secondary use case: The aircraft needs to fly and drop heavy relief packages, sometimes in far, remote locations.

Design Requirement: The same airframe needs to lift a heavier load without compromising on endurance.

Keeping these use cases in mind, the following design for the airframe was developed:

Airframe Details

Wing Design:

- Base Airfoil Chosen: Wortmann FX 63-110 Airfoil
- General Wing Shape: Rectangular
- Mean Chord: 140 mm
-
- Wing span: 1000 mm
- Wing Area: 140,000 mm^2 = 0.144 m^2
- Wing Tips shape:

Morphed Wing Characteristics:

- Most affected characteristic: Wing camber and angle of attack
- 0 morph state camber: 4.36%
- Full Morph state camber: 12.79%

- Angle of attack increase due to full morph: 13 degrees

Empennage Design:

- Horizontal Stabilizer Chord: 100 mm
- Horizontal Stabilizer Span: 280 mm
- Vertical Stabilizer Chord: 100 mm
- Vertical Stabilizer Span: 120 mm
- Tail Lever Arm: 660 mm

Operating Conditions:

Operating Angle of attack: 3.35 degrees

Operating speed: 16 m/s

Determining Wing Loading and Power Loading

Estimation of Wing Loading from Stall Velocity

Initially, we set a fixed stall velocity of about 12.5m/s for further calculations and analysis. We obtained the wing loading through the following numerical calculation:

$$\left(\frac{W}{S}\right)_{V_s} = \frac{1}{2}\rho V_s^2 C_{L_{\max}}$$

Putting in values for our primary use case,

Air density (ρ) = 1.22 kg/m^3

C_{Lmax} = 1.45

V_{stall} = 12.5 m/s

This calculation gives us the **wing loading as 140 .**

Estimation and Maximization of Power Loading

We get wing loading to be approximately 60 N/m^2 from the stall velocity.

Based on the design requirements of our UAV, we determined the minimum turning radius to be 15m at a cruising speed of 10 m/s. We can write the velocity V and load factor n at the minimum turn radius as:

We can write the minimum turn radius as:

$$R_{\min} = \frac{(V_\infty)^2_{R_{\min}}}{g\sqrt{n^2_{R_{\min}}-1}} = \frac{4K(W/S)}{\rho_\infty (T/W)} \frac{1}{g\sqrt{2-\frac{4KC_{D_0}}{(T/W)^2}-1}}$$
$$= \frac{4K(W/S)}{g\rho_\infty (T/W)} \frac{1}{\sqrt{1-4KC_{D_0}/(T/W)^2}}$$

The power loading comes out to be:

$$\left(\frac{W}{P_{SL}}\right)_{turn} = \frac{(\eta_{pr}\sigma)/V_{turn}}{\left\{\left(\frac{V^4_{turn}}{(Rg)^2}+1\right)\frac{K(W/S)}{0.5\rho V^2_{turn}} + 0.5\rho V^2_{turn}\frac{C_{D_0}}{W/S}\right\}}$$

Here, V_{Turn} is the velocity of the turning radius, and η_{pr} is the propeller efficiency. This equation gives us the relation between the wing and power loading at a particular turning radius.

For a velocity of 16 m/s and a turning radius of 15m, we get the value of power loading to be around **0.833 N/W** from the above relation.

Weight Estimation of the UAV

In this phase of the design process, we will estimate the overall weight of the aircraft. We split the overall weight of the UAV into four parts: payload weight ($W_{Payload}$), empty weight (W_{Empty}), battery weight ($W_{Battery}$), and motor (W_{Motor}). (*Refer Appendix for derivations*)

We can determine the battery weight from the endurance requirements of the aircraft and choose the motor weight from our estimated net drag of the aircraft. The final weight equation comes to:

$$W_{Total} = \frac{W_{Payload}}{1 - \frac{\rho_{Empty}}{\left(\frac{W_{Total}}{S}\right)} - T_{Endrance} * \left(\frac{1.1\sqrt{\frac{2W_{Total}}{\rho S}}\left\{\frac{1}{4}\left(\frac{3}{KC_{D_0}^{\frac{1}{3}}}\right)\right\}}{\eta_{motor}\eta_{prop} f_{DOD}\eta_{Discharge}\rho_{Battery}} \right) - \frac{\left(\frac{P_{req}}{W_{Total}}\right)}{0.6 * PWR}}$$

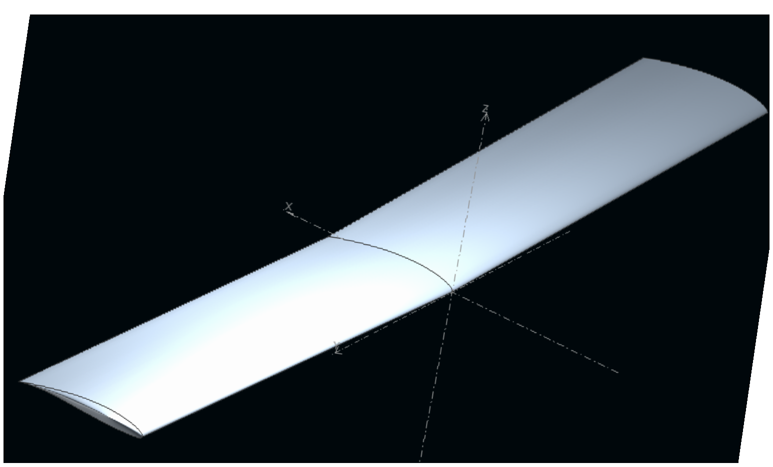
ρ_{Empty}	39 N/m^2
$\rho_{Battery}$	140 Wh/kg
η_{motor}	0.8
$\eta_{Discharge}$	0.9
PWR	3.75 kW/kg
$T_{Endurance}$	200 s
f_{DOD}	0.9

From the above equation, we have a payload weight of **1.7 kg** and need an endurance of **180s**. The total weight of the aircraft comes out to be **2 kg**, with the empty weight being **546 grams**, with the battery weight being **125 grams** and motor weight being **90 grams**.

Airfoil Choice

The FX 63 family of airfoils is a common choice within the team for various projects due to their:

- A good balance of high lift and high aerodynamic efficiency
- Low Critical Reynolds Number
- High stall angle
- Provides the necessary CL and CD requirements



Final (Wing-only) design and important relations

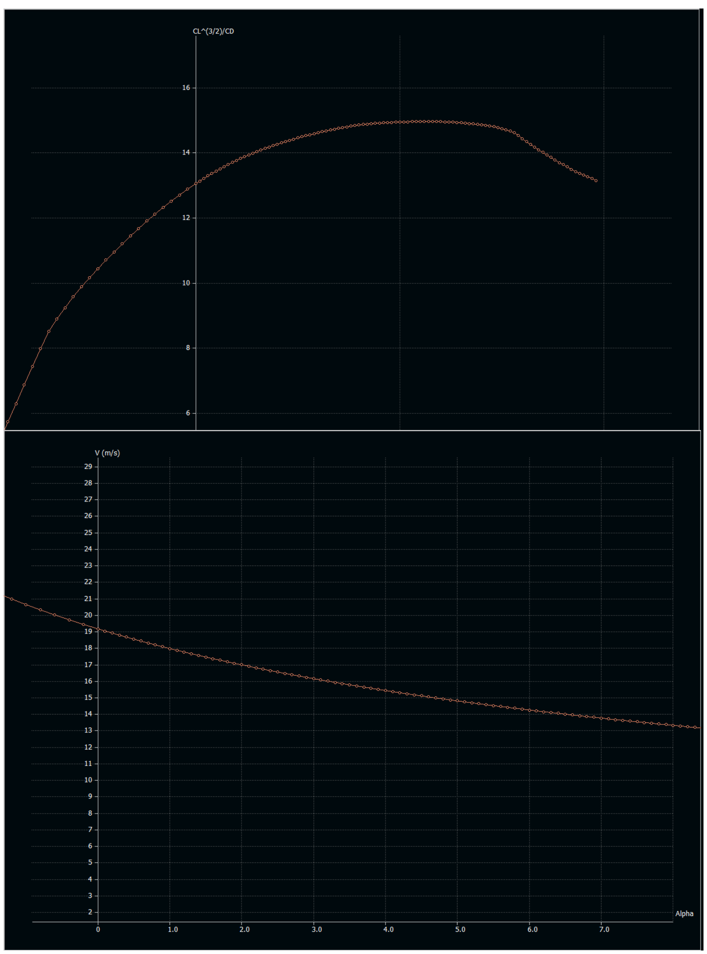
From our given wing and aspect ratio, our wing is expected to perform in a certain regime. That regime can be determined by applying the following principles.

First, we need the plot of CL^1.5/CD for the wing at different angles of attack. Since

$$P_{req} = T_{req}V = DV = \frac{WV}{\frac{C_L}{C_D}} = \sqrt{\frac{2W^3C_D^2}{S\rho C_L^3}}$$
$$\Rightarrow P_{req} \propto \frac{1}{\frac{C_L^{\frac{3}{2}}}{C_D}}$$

Use case 1: Aircraft Weight = 2 kg

The angle at which the value of CL^1.5/CD is maximum, is the point at which the airframe will consume minimum power. Minimum power consumption will correspond to a longer flight time. The graph once computed at a constant amount of lift is seen as:

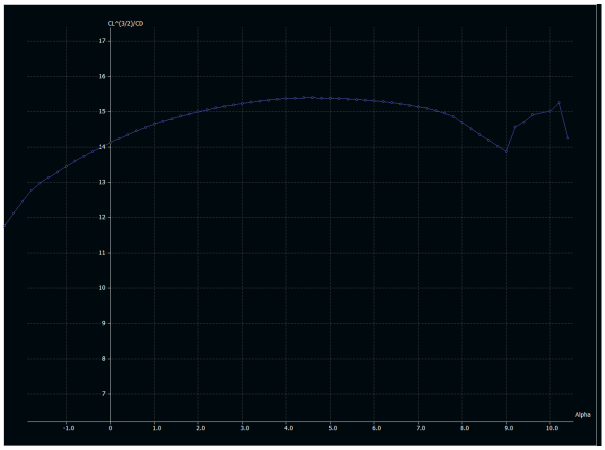
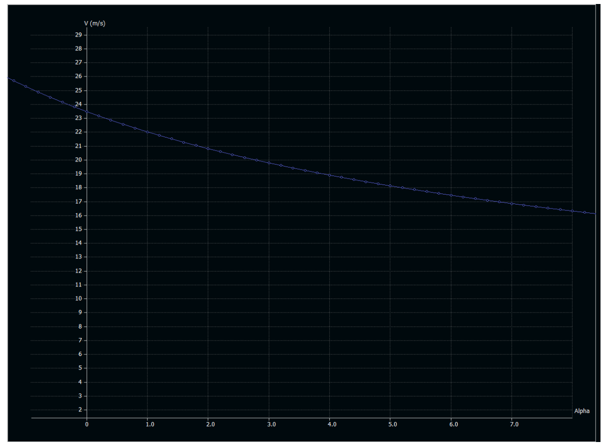


From this graph, it can be seen that the wing needs to operate in the range of 3-6 degrees angle of attack in order to be efficient. To make sure it is feasible, let us check the velocity at the operating range.

The velocity can be seen varying between 17 m/s at 6 degrees AoA and 14 m/s at 3 degrees AoA. This gives us the estimate that we can fly the aircraft at an approximate 15 m/s, which is a good number for the scale of the aircraft and the materials used.

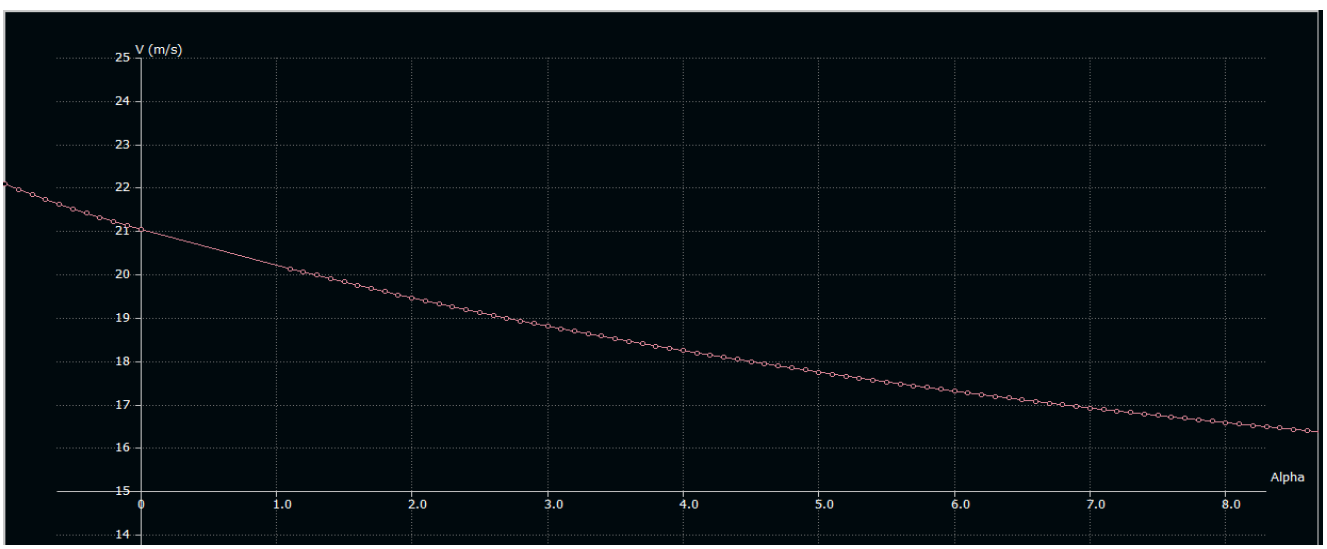
Use Case 2: Aircraft Weight = 3 kg

Let us see the same curves for the wing at the new value of lift.

A quick comparison of the curves will show that the aircraft needs to operate at a higher velocity to achieve the same efficiency. Dynamic thrust of a motor/propeller combination dictates that, faster the aircraft, lesser the thrust is generated by the power plant at the same RPM. Hence, the power plant has to draw more energy to keep up with the pace and a higher drag, leading to drastic loss in flight time. This is where our morphing wing comes into play.

Let us compare our morphing wing's performance at the same load. The velocity is the key factor here, let us use the 75% morph state to compare here.



Immediately we can see that the velocity has dropped significantly. Since dynamic thrust is not affecting the aircraft as much as the previous case due to a similar velocity regime as the primary use case, drag is the only component that the powerplant needs to fight and thus greatly saves on energy, increasing flight time.

Wing Sizing

We follow an iterative approach towards determining the size of the wing. We have the wing area from the weight estimation and wing loading analysis. From our set of airfoil choices, we can determine the $C_{l\alpha}$ of the airfoil and use the approximation: $\frac{dC_L}{d\propto} = C_{L_\propto} = \frac{C_{l_\propto} AR}{2+\sqrt{4+AR^2}}$ to estimate the lift data for a 3D wing.

By iterating over the variables $C_{l\alpha}$ and AR, choosing an airfoil first and identifying if the aspect ratio falls within the limits of our aircraft, we finally chose the FS 63-110 airfoil and an aspect ratio of 7 for our baseline configuration.

Now that we have the wing aspect ratio and airfoil used, we need to vary the camber to achieve a higher lift for carrying a heavier payload. The C_{lmax} of the new airfoil will need to be higher than that of our original configuration since we need a higher value of C_L for the same aspect ratio.

We iterate the airfoil to find the max camber which will provide the necessary lift for our new payload.

Longitudinal Static Stability

Static stability was achieved in the aircraft by manipulating the moments created by just the tail, and trimming the aircraft at an angle of attack we wanted. Moments (or torques) on an aircraft can be stabilizing or destabilizing, and some general guidelines were followed to make sure moments were, at the very least stabilizing, if not trimmed to a particular AoA, like maintaining a nose heavy aircraft, keeping inline motors and choosing a high wing configuration.

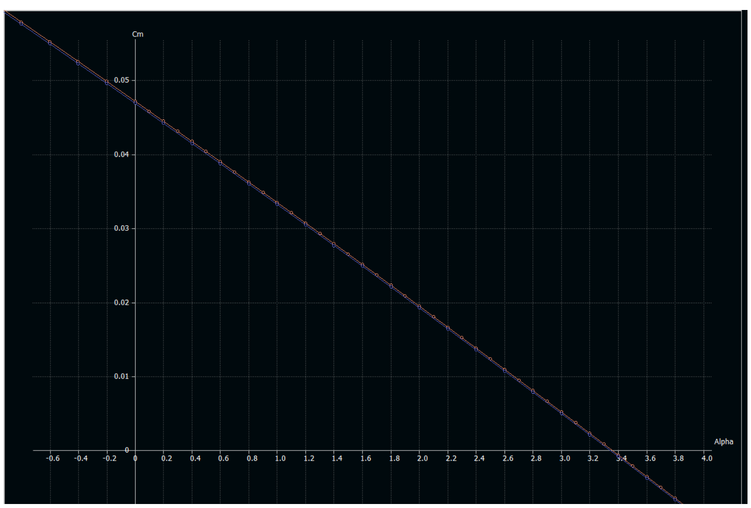
In order to trim the aircraft at a particular point, the manipulation of the wing and tail's Coefficient of Moment (Cm) is necessary. Cm can be defined by the following formula:

$$C_m = C_{m_{ac\,W}} + C_L \cdot \frac{l_{cg}}{c} - C_{L_H} \cdot V_H$$

Where Vh is the Horizontal Tail Volume Ratio. For an Aircraft to be statically stable in the longitudinal direction, the variation of Cm with respect to the AoA must be negative and its x intercept at the operating angle of attack range we want, i.e. 3-6 degrees.

To put this in application, we began with a general tail using recommended values for Horizontal Tail Volume Ratio. The airfoil chosen for the purpose was **NACA 63010,** a thin airfoil with low drag characteristics and high stall. The first tail was designed and a Cm value and slope computed. In order to better the design, we can manipulate the **tail area, tail lever arm, angle of incidence of the tail and the position of the aircraft's center of gravity.**

The aim of this manipulation was to bring the Static margin of the aircraft, i.e., $\frac{X_{NP} - X_{CG}}{\bar{c}}$ To a recommended range of 12%-18%. Given below are the results obtained from the finalised design:



This graph clearly shows the trim point of the aircraft and its static margin, mentioned as below:

- Location of Center of Gravity: 70 mm behind wing leading edge
- Tail Lever Arm: 660 mm
- Horizontal Tail Volume Ratio: 0.94
- Static Margin: 15.47%

Lateral Static Stability

To maintain yaw and roll stability, a few key guidelines were followed to ensure stabilizing moments:

- The Center of Gravity is on the centerline of the aircraft
- A small dihedral of 2 degrees has been used
- A vertical stabilizer has been used

- The motor is kept in line with the center of gravity so as to not create moments due to thrust.

Vertical Stabilizer sizing:

Using the empirical formula: $C_{n_\beta} = 0.0005\left(\frac{W_{Total}}{b^2}\right)^{1/2}$

We can get the required value of C_{n_β} which comes out to be 0.00036 from our design.

The C_{n_β} contributions of the wing and tail can also be calculated through this empirical formula:

we get $\left(C_{n_\beta}\right)_{wing + tail + fuselage}$ to be 0.00032

$$C_{n\beta v} = V_v\ \eta_v\ C_{L\alpha v}\ \left(1+\frac{d\sigma}{d\beta}\right)$$

$$\left(C_{n_\beta}\right)_{wing} = C_{D_{wing}} \frac{\bar{y}}{b} \frac{1}{57.3} \sin^2\Lambda_{wing}$$

Subtracting the value of $\left(C_{n_\beta}\right)_{wing}$ calculated we get

$$\left(C_{n_\beta}\right)_{vertical\ tail\ required} = 0.0001212$$

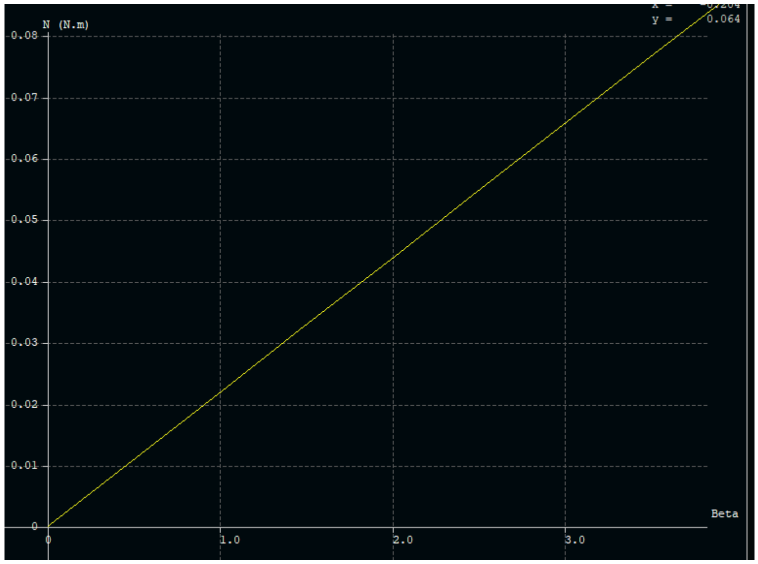
Iterating over Vv and Sv we get the desired tail size:

Vv : 0.94 cmsq Sv : 63.75 cmsq span : 90 mm Root chord : 100 Tip cord : 100mm

Airfoil **NACA 63010** for the same reasons as the horizontal stabilizer.

In order to trim an aircraft at a particular β angle (sideslip), an equation of Coefficient of Moment of various surfaces needs to be devised. That equation is complex because of several factors, but the key point to enforce in the solution is to **the slope of Cm w.r.t. Sideslip angle must be positive, and Cm must be 0 at sideslip = 0.**

Keeping these factors in mind, the Vertical tail behaves with the aircraft as follows:



Control Surface Sizing

Elevator

Purpose of the elevator is to provide extra moment to obtain different trim points in a range of velocity , i.e. (12 - 23 m/s) to match the desired lift of 20 or 30N at that trim point. The following equations help us to determine the elevator area:

$$\Rightarrow C_{m_{\delta e}} = -V_H \eta C_{L_{\alpha,t}} \tau$$

Where V_H is the tail volume ratio, and η is the dynamic pressure ratio. **τ** is the elevator effectiveness and is proportional to the size of the flap being used.

Once we obtain the elevator effectiveness using these equations , we use the a standard plot to obtain a relation between **τ** and Elevator to tail Area ratio:

τ =0.218 implies **Se/St=0.125** => **Se=35 sqsm**

Elevator sizing : **span = 14cm, chord= 25% tail chord = 2.5 cm Max deflection = +-10 deg**

Rudder

Spiral Stability Margin

The CLA must be closer to the tail than the CG of the aircraft. The position of CLA w.r.t. to CG determines whether the aircraft is spirally stable, neutral or unstable

To obtain the CLA at 22% of VTMA and behind the CG (for best spiral stability), the VTA is generally found as:

$$VTA = 0.05 * \text{Wing Area} * \text{Wing Span}/VTMA$$

The CLA was observed to be located at approx. 25% VTMA from the tentative CG location for a 200 cmsq tail area.

$$\text{Aspect Ratio} = 1.55 * h2/VTA$$

Rudder sizing : 30% Chord of Vstab = 3 cm Max deflection = +-10deg

Dynamic Stability and stability derivatives

Static stability derivatives :

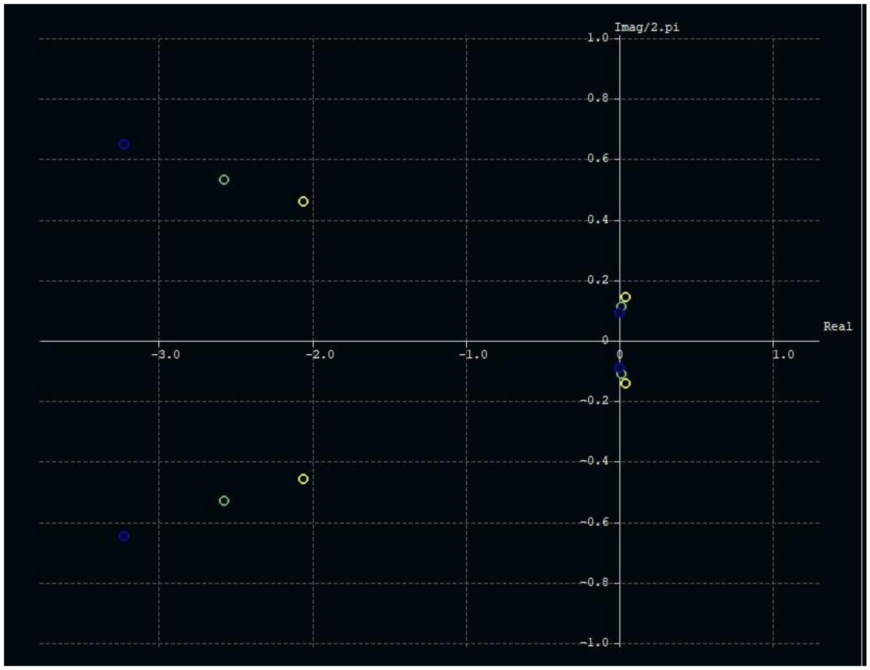
Longitudinal derivatives

Derivative Value : $C_{m\alpha} = -0.78457$ $C_{l\beta} = -0.012676$ $C_{n\beta} = 0.045145$

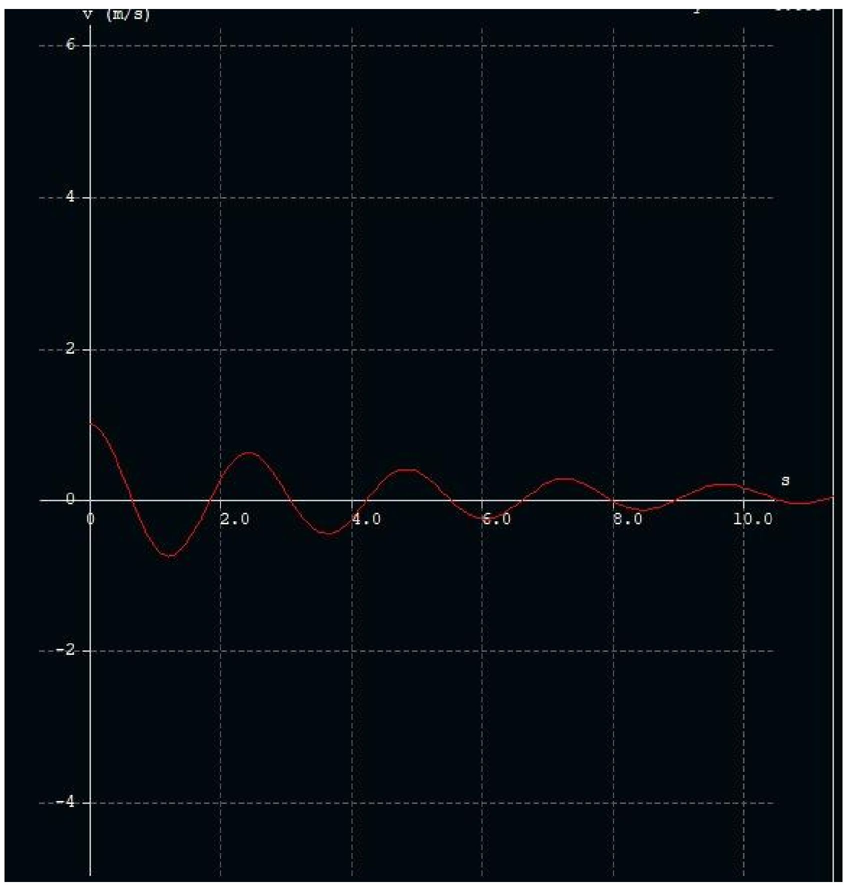
Requirement for longitudinal and lateral static stability : $C_{m\alpha} < 0$, $C_{l\beta} < 0$, $C_{n\beta} > 0$

To evaluate the dynamic stability of the aircraft, an analysis was performed for both longitudinal and lateral-directional stability for fully loaded cases. The main parameter of each mode can be seen in :

Mode	Level 1 requirement	Max payload case	Stability level
Phugoid	$\lvert\zeta\rvert > 0.04$	$\lvert\zeta\rvert = 0.045$	Level 1
Short period	$0.35 < \zeta < 1.3$	$\zeta = 0.611$	Level 1
Roll	$\tau < 1$ s	$\tau = 0.168$ sec	Level 1
Dutch roll	$\zeta > 0.08$ $\omega > 0.4$ rad/s $\zeta\omega > 0.15$ rad/s	$\zeta = 0.098$ $\omega = 2.601$ rad/s $\zeta\omega = 0.2549$ rad/s	Level 1
Spiral	Tdouble > 20 s	$T_{double} = 10.413$ s	Level 2



Root Locus Graph



Response to Gust

Theoretical capabilities of Morphing Wing and Future Use

The competition-spec aircraft is limited in its capabilities due to restrictions in available power and materials. As a result, with ample power and strong materials, the aircraft can perform to a much greater extent than we have limited it to.

Let us set some reasonable restrictions on performance like max speed at **20 m/s** and only analyse at **cruise.**

Some interesting results

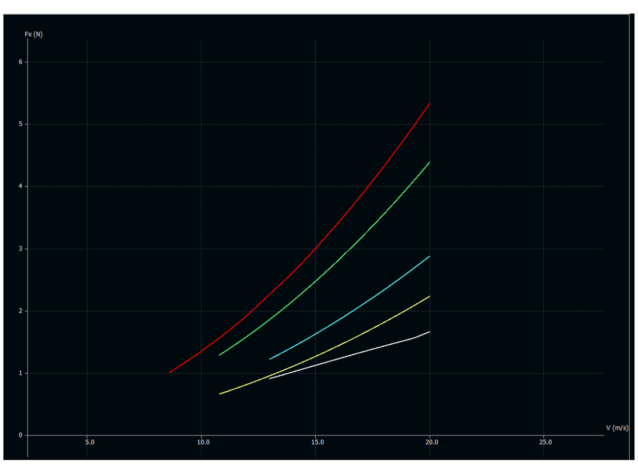
While cruising at 20 m/s with 75% morph, the aircraft can lift a net weight of **nearly 5 kg,** at the expense of high drag and short range. The aircraft operating in this state must pack a bigger battery, bigger motor, and stronger wing spars, preferably carbon fiber.

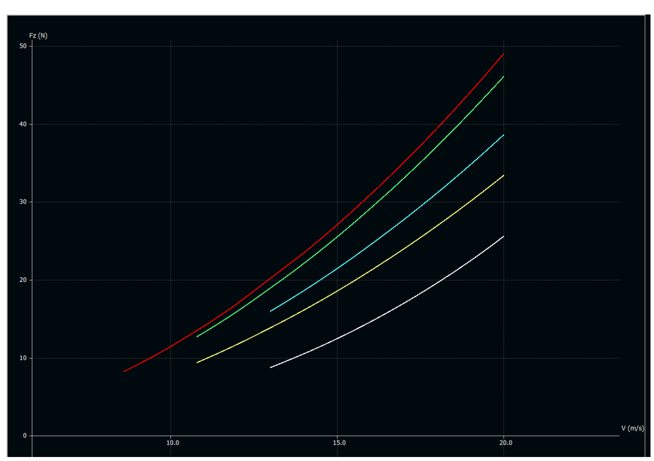
At 75% morph, the aircraft will only stall at **8.6 m/s** while lifting a load of **836 grams.**

For our primary use cases, a 2 kg gross weight aircraft will stall at as low as **12.2 m/s** when operating with **75% morph,** at an AoA of **5.54 degrees.**

How does one configure the aircraft in a specific situation?

The aircraft can demonstrate its ability to be useful in a myriad of situations only when a user is aware of the considerations necessary to choose the appropriate configuration. To that extent, one may use condensed charts like the ones below:





These graphs are the lift and drag produced by the morphed wing for various morph states. Looking at the lift graph, one may choose at what velocity the aircraft is required to fly, and according to the payload to be carried, choose a % deflection for the wing. With that velocity, one may choose a particular motor/propeller/battery/ESC combination for the aircraft, depending on their endurance requirements.

For our design, it was made sure that the aircraft is able to trim at the 2 cases we designed the aircraft for. In a much more varied case, use of gyros for active stability is necessary, else a similar chart is needed that advises on the tail lever arm length. As a part of a professional package to be used by rescuers, basic equipment like these planners are an expected ergonomic requirement. The aircraft is to fly autonomously in the application, hence active stability control in these rare configurations will greatly improve operational efficiency.

Electronics

Battery

The Battery employed is 1500mAh 3S 40C Lithium Polymer Battery. It was chosen keeping in mind the competition restrictions for maximum number of cells allowed and battery weight estimation . This battery weighs 120 gms and can provide an endurance of 3 mins, considering all modes.

Servos

The Torque required, as observed from XFLR analysis for the Rudder and Elevators is about 0.016 Nm. We use the Towerpro SG90 servos in the Tail section, providing a torque of 0.117 Nm. This Servo is readily available in the market at low prices and is lightweight at 9gm/ servo.

SG 90 Servo SG92R Servo

The Wing morphing capabilities of the plane are achieved using Towerpro SG92R servos. The torque requirements from XFLR analysis was about 0.18 Nm on each wing. We use the SG92R, which can provide 0.24 Nm of torque and weighs at only 9 gms. The same servo was not used for elevators and rudder as this much torque was not required and the SG90 comes at lesser cost.

Motor and Propeller

XFLR analyses were done on the different configurations that the plane can fly in depending on wing shape, payload carried and cruise velocity. In each case the drag on the plane was noted along with the cruise velocity to be able to decide the thrust requirements for the motor and propeller.

To simplify the process, we considered 2 configurations, one that will be most commonly used i.e. general cruise conditions and the one with maximum payload lift off creating the most drag. The first configuration has cruise velocity of 16m/s, drag of 1.231 N for the Wing and Tail and 2N for the fuselage. The second configuration, carrying a payload of 2Kg, produced a drag of 3.1 N at the wing and 2N at the fuselage at 16 m/s. These values are limited by the maximum angle of attack that we propose to operate at.

Due to the pandemic, we could not physically test the different configurations that will be optimum for our plane, we have used data provided on websites like T-motors and APC propellers for their products. This along with the below equations was used to compare multiple motor and propeller combinations. Our approach was to analyse a variety of propellers and find corresponding RPM for producing desired dynamic thrust. For doing this we used the following formula

Dynamic Thrust Equation
F = thrust (N), d = prop diam. (in.), RPM = prop rotations/min., pitch = prop pitch (in.), V_0 = propeller forward airspeed (m/s)

Expanded Form:

$$F = 1.225\frac{\pi(0.0254\cdot d)^2}{4}\left[\left(RPM_{prop}\cdot 0.0254\cdot pitch\cdot\frac{1min}{60sec}\right)^2 - \left(RPM_{prop}\cdot 0.0254\cdot pitch\cdot\frac{1min}{60sec}\right)V_0\right]\left(\frac{d}{3.29546\cdot pitch}\right)^{1.5}$$

Simplified Form:

$$F = 4.392399x10^{-8}\cdot RPM\frac{d^{3.5}}{\sqrt{pitch}}(4.23333x10^{-4}\cdot RPM\cdot pitch - V_0)$$

Gabriel Staples, 2013. http://electricrcaircraftguy.blogspot.com/

Solving the simplified form, we were able to get an equation for RPM in terms of F, d, pitch and Vo. By plotting a table for several configurations of motor and propellers, we compared their feasibility.

We finalised the **AS2317 Long Shaft 1400 KV motor with a 9x7 inch propeller**. For cruise conditions, the motor will operate at about ~75% throttle at 9000 RPM. In the maximum liftoff configuration, the motor will be operated at ~95% throttle at 10500 RPM. Due margins were considered in the selection process keeping in mind the variations and errors that might occur in actual flight testing. The motor weighs in at 81gm and the propeller at 22gms.

Electronic Speed Controller

The maximum Currents are expected to reach 40A at maximum throttle. Therefore we have chosen a 50 A ESC for our purposes so that it is not affected by surge in current and there is sufficient margin for operations. The ESC used is the Wolfpack Opto 50Amp ESC.

<u>Transmitter and Receiver Module</u>

Mechanically, the wing is capable of morphing its shape freely within the given parameters but we are limited by the controller technology that is available. Therefore using a 6 channel transmitter receiver by Aviaonic, we effectively operate our plane in 2 wing morph positions. With the normal payload of 1 Kg, we operate at 0% morph in cruise and 75% morph while landing and takeoff. With the payload of 2Kg, we operate the entire flight with 75% morph.

Out of the 6 channels, 4 channels are used for the standard AETR (Aileron Elevator Thrust Rudder) control. The 6th channel, a binary switch, is used to control the morph settings. The 5th channel, a switch with 3 positions, is left open for future modifications like payload drop, etc. Further Morph states can be achieved, apart from the two in use, by using channel 5 and channel 6 in combination to provide a total of 6 states of wing morph. This was not employed in this design as the extra states did not offer much benefits in our application.

Morphing Wing Mechanism:

The most important distinguishing feature of our morphing wing compared to any other wing with flaps is the maintenance of a smooth, curved surface on either side of the wing. This requirement was met by a few designs elaborated in the literature review.

Literature Review

Morphing mechanisms have been common study topics for several aerospace laboratories around the world. Most notably, engineers at NASA and MIT have tested a shape-shifting wing with tiny panels making up the surface, each of which can be manipulated independently.

Another notable design was the FlexSys Inc. complaint flap, wherein the flap bends with a smooth surface to the required angle, as opposed to a rigidly hinged flap. This design used a complex internal compliant structure, which allowed for a smooth bend without stressing the bending parts too much. Besides these, several ameteur and professional designs were looked at, most of which fell into one of these categories:

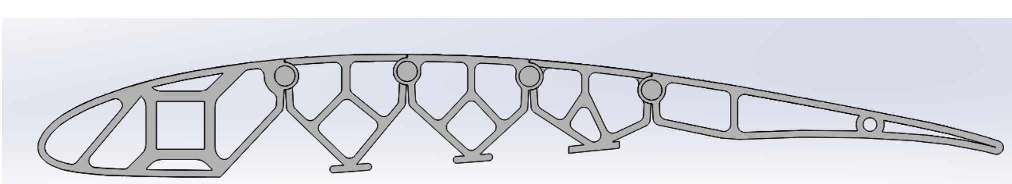
- A bendable surface with a complex inner skeleton that is manipulated to morph the wing.
- Segmented rigid structures with complex wire controls
- Small wings with flexible ribs that can bend under force to increase camber

Methodology

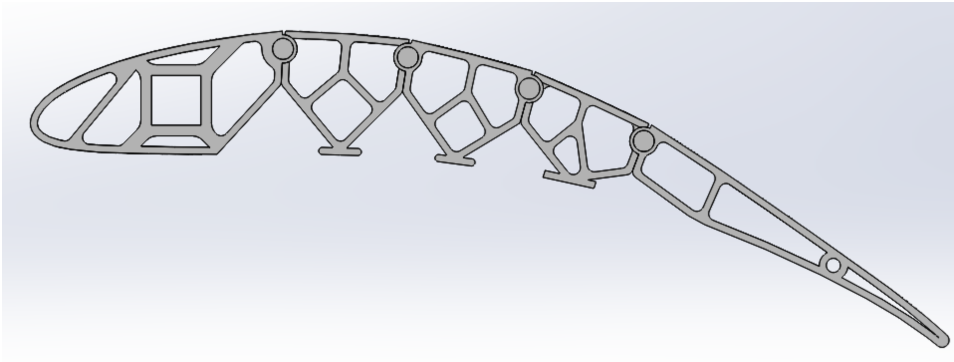
A feature of every design was adopted to design the best and most feasible mechanism for the competition. To decide which feature to adopt, we compared them with our core requirements for the event, them being lightweight design, significant camber change and simplicity.

The use of complex actuated skeletons and wire controls were dropped, and any compliant mechanism that required in-depth study of the part's mechanics and manufacturing processes were dropped.

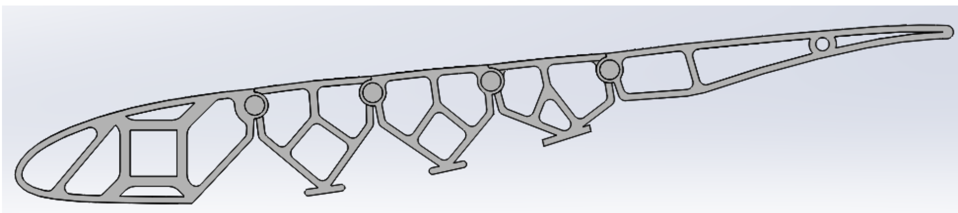
To simplify the design, a multi-section airfoil became our approach of choice. Similar to other segmented designs, we divided a generic wing rib into 4 segments, the first one containing the leading edge to remain fixed with respect to the plane. The 4 other segments are linked in a linear fashion to each other, free to rotate about 4 hinges between every 2 segments. In order to constrain the rotation and maintain a near-continuous curved surface, every segment has extensions that physically restrict rotation about the hinge to a certain angle. The angle is set to 6 degrees downward per hinge and 2 degrees upward per hinge. A full deflection of all segments will lead to a 48.8% decrease in camber when deflected upwards and a 193.3% increase in camber when deflected downwards. Drag and efficiency considerations led to the upper and lower limits mentioned above.



Central Configuration

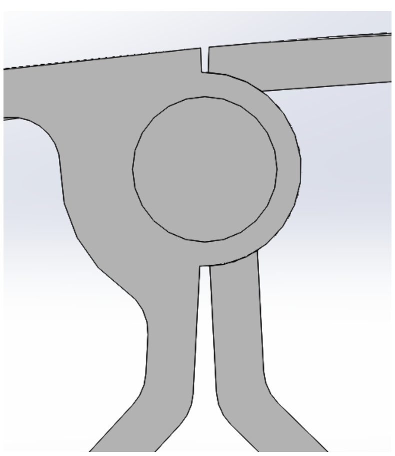


100% Morph Down

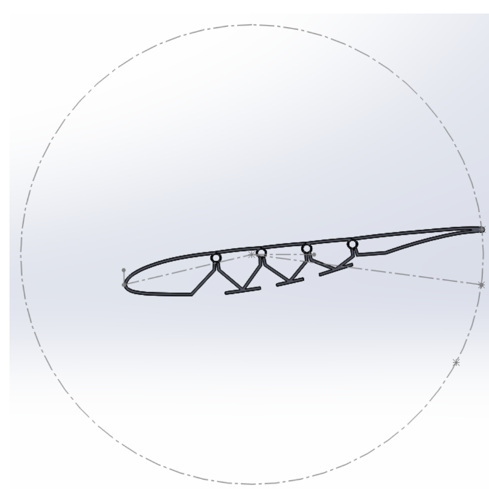


100% Morph Up

This mechanism is suited for the process of 3D printing. Every segment is a small part, and prints of ABS are better equipped for the complexity that balsa ribs cannot produce, the strength that isn't dependent on grain direction like many woods and the low density and voluminous nature to build the entire wing out of, without increasing the weight, unlike aluminium. The wing is divided into convenient sections that can be 3D printed separately, and the use of alternative infills can help reduce the weight further.



Since the mechanism resembles a quadruple pendulum, there is the possibility of the mechanism not returning to the perfect central position or any other between the two extremes. The mechanism will turn different angles at each hinge, which given the very small restrictions, will not significantly deform the airfoil, but may create unwanted moments that may worsen the form. To mitigate this and ensure that the hinge will tend towards its central position, a torsion spring or rod (similar to those used in

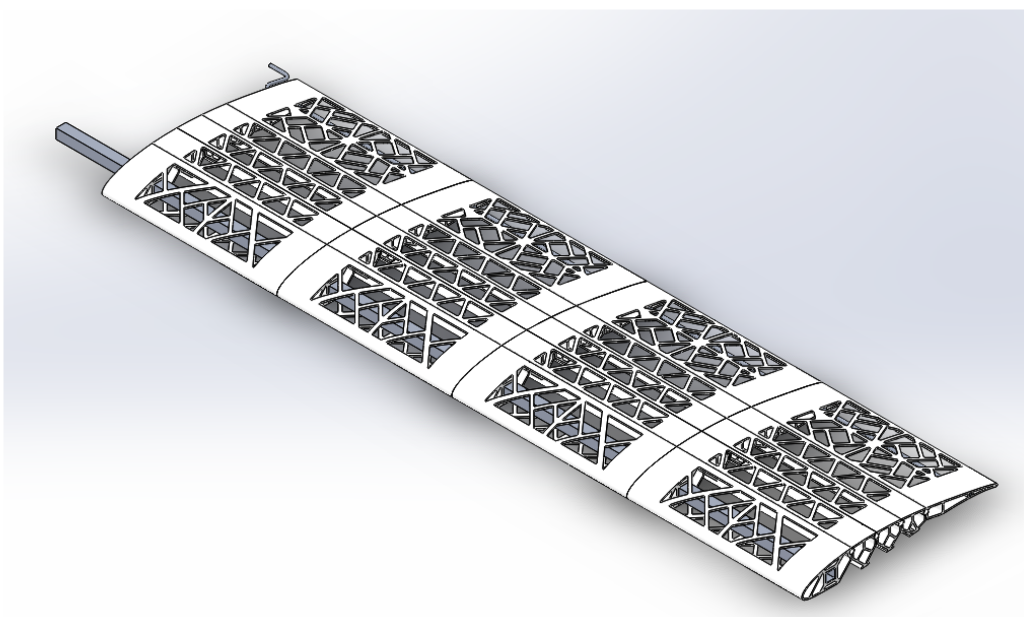


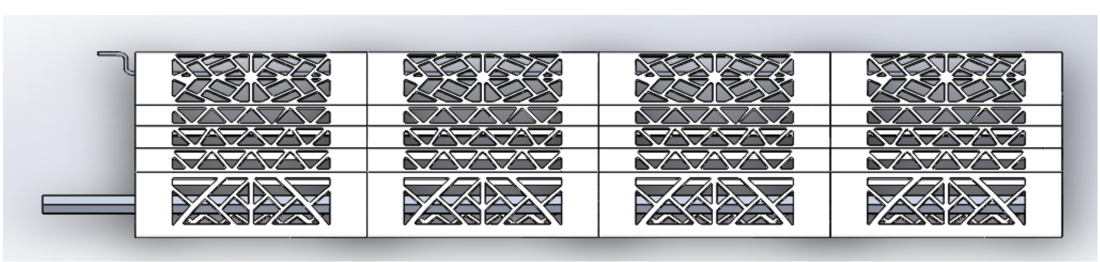
clothesline clips) shall be attached to every other hinge along the span, with its unloaded condition set in the central position.

The trailing edge is seen to traverse a circle while morphing, and hence can be actuated by use of a single servo motor. A servo motor is placed in the center of rotation, and an actuator arm rotates the trailing edge that is supported on a thin aluminium rod.

Each wing is made up of 4 sets of segments, each set forming 2 ribs, interconnected by a 3D printed surface. The ribs will also be 3D printed, and kept sufficiently thick to prevent deformation under lifting loads. Weight reduction is done by cutting patterns out on surfaces in a way that does not deprive it of bending strength. An aluminium spar will traverse the span of the wing, acting as its main load bearing structure. The dimensions and thickness of the aluminium rod are decided by simulating lifting loads on the bar and maintaining a safety factor over 2.

Final Wing Model Dimensions and Characteristics:

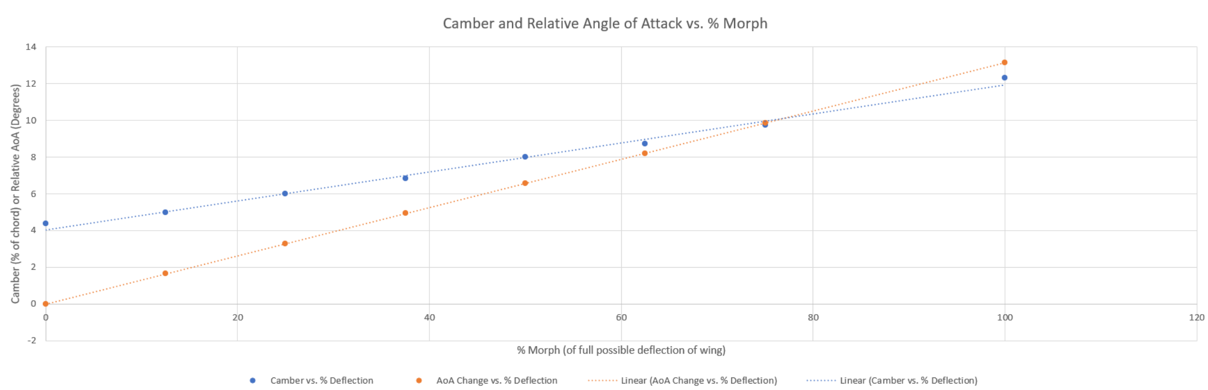




Deflection vs. Camber and Angle of Attack (Relative to 0 deflection state)

While morphing, the wing's camber and relative angle of attack will change. Downward deflection will increase angle of attack and camber, and vice versa with upward deflection. The trend of the two deflections is recorded below, and is an indicator of the change in lifting characteristics of the airfoil while morphing.

At 0% deflection, i.e., in base airfoil configuration, take relative angle of attack as 0 and camber as 4.36% of chord.



The effects of the change in camber are well documented in the previous sections.

Main Spar Load Bearing Characteristics:

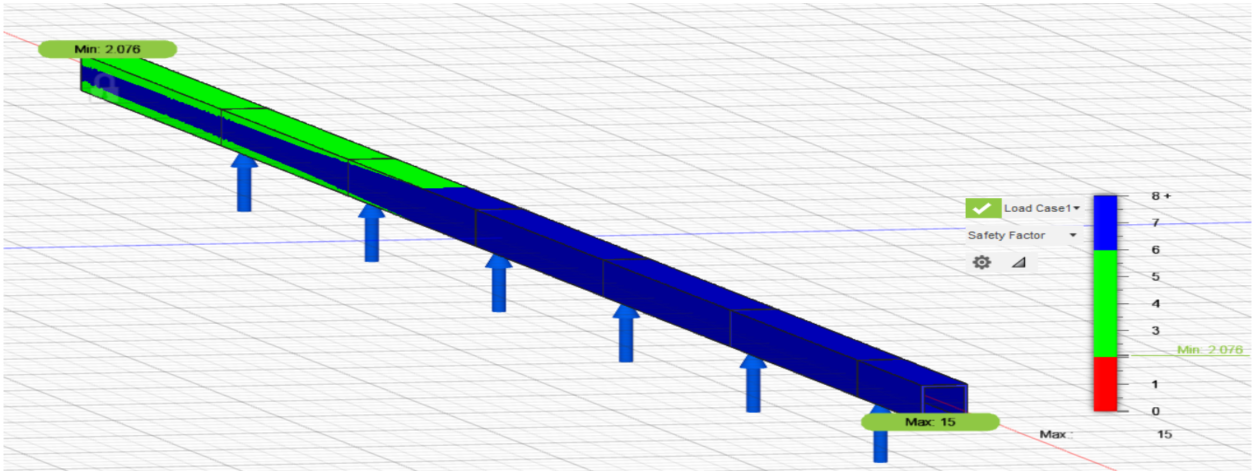
To confirm the load bearing capacity of the spar, the Half wing Spar was divided into 6 parts and force was applied to each of the parts as can be calculated from the CL vs. Span graph in XFLR5. One end of the beam was fixed. The results were satisfactory, maintaining a safety factor of 2 without excessive deflection.

Square beam was chosen so that one spar is sufficient and twisting about the joint to the fuselage is prevented.

Square beam side length: 7 mm

Wall thickness: 0.5 mm

Material: Aluminium 6061



Tail Design:

Keeping in accordance with the full 3D printed wing, and since the fuselage will also be printed in ABS, The tail is also to be printed. Since the loads on the tail are minimal, a simple 3D printed skeleton with few ribs and a patterned surface has been implemented. The support is also 3D printed, designed to attach seamlessly to the stabilizers and the tail boom.

A single aluminium tail boom was selected because it was sufficiently meeting the requirements of the load. An aluminium square beam with side 6 mm and thickness 0.75 mm was used for the boom. The boom was analysed, even though it was very obvious that the load would be handled. A boom of similar dimensions was used to handle larger loads in the wing. The results gave a minimum safety factor of 7.

Fuselage Design

A unique design tool was implemented in order to create the fuselage, called **Generative design.** Generative design is an iterative process by which a software can create a design within some constraints, and simultaneously analyse it to meet structural requirements.

In our design, there were a few constraints that was input into the program:

- Needs to withstand lifting forces on the joints with wing and tail.
- Needs to position, tightly fit and handle the weights of various components in the fuselage, including payload, battery, motor, etc.
- Needs to withstand the thrust generated by the powerplant.

- Needs to have an aerodynamic shape.
- While withstanding all the loads, it must maintain a safety factor over 2.

The result was an iterative process, making the body more streamlined. The results can be seen in the final renders.

Assembly

CG Consideration

From the wing and tail design, we have a range within which the center of gravity (CG) of the aircraft must be, and a well-defined value at which it is most preferable. Since this aircraft comes in various loading conditions, it is possible that the CG will shift when the aircraft is unloaded. It is imperative that the aircraft flies at all load conditions.

Moment of Inertia Consideration

While the CG may be placed at the appropriate position, the distribution of the mass affects the moment of inertia of the aircraft in the lateral and longitudinal directions. This affects the dynamic stability of the aircraft, as a compact mass distribution will lead to a dynamically unstable aircraft.

Payload Loading and Securing

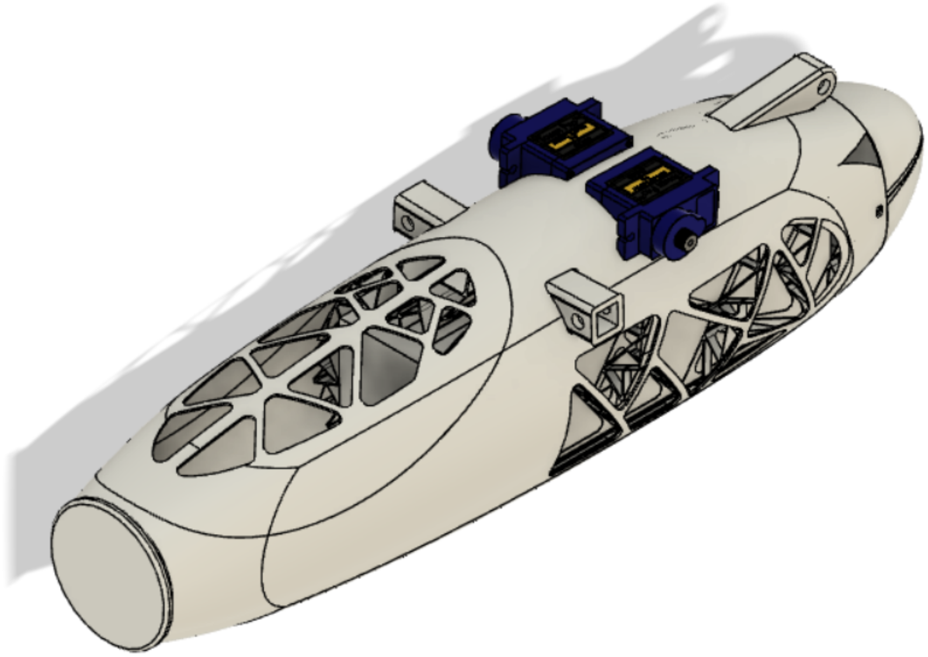
The payload can be placed within a tight-fit container in the fuselage, sliding in from the rear. The rear has a cap that can be bolted on to the fuselage to secure the payload.

Joints

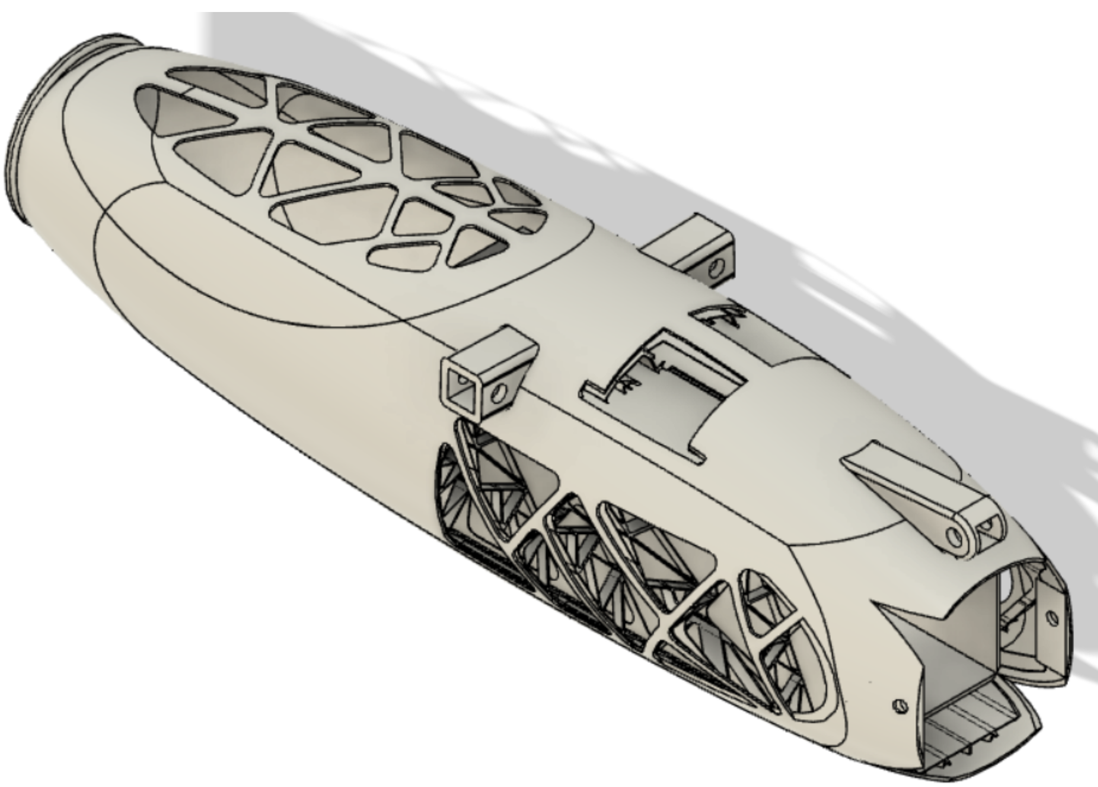
The various components of the aircraft, i.e. the tail boom, wing spars and printed parts of the fuselage are combined together using M4 bolts.

Final Assembly

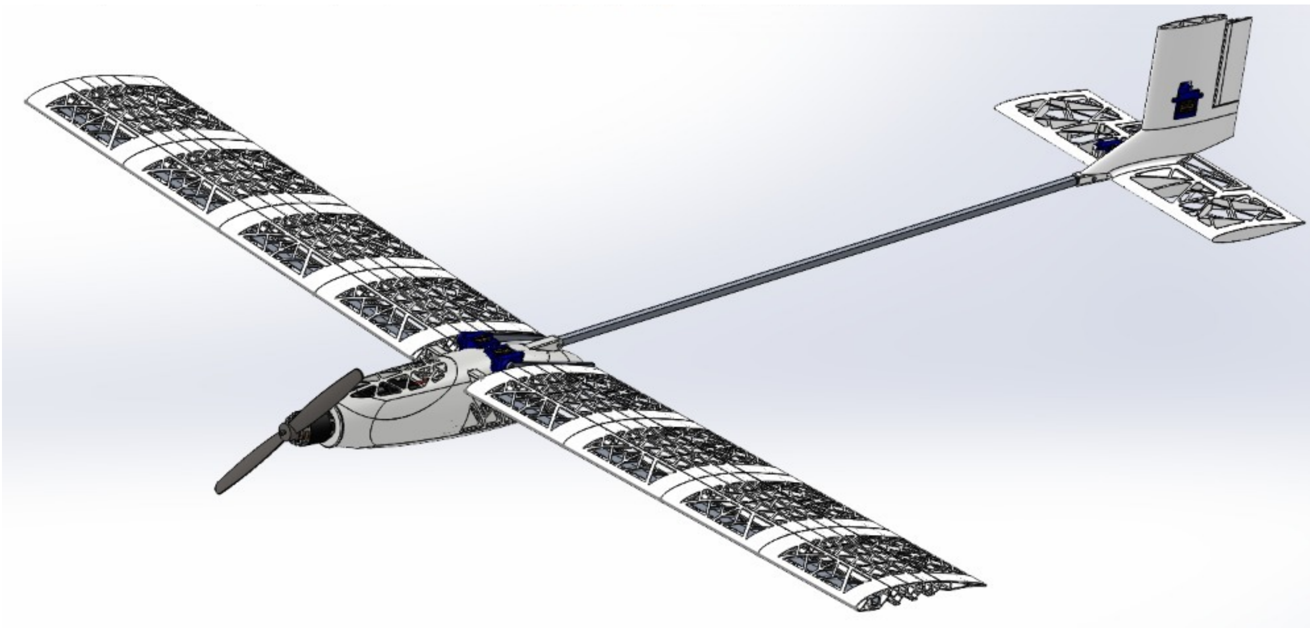
Given below are some renders of the entire aircraft, completed.



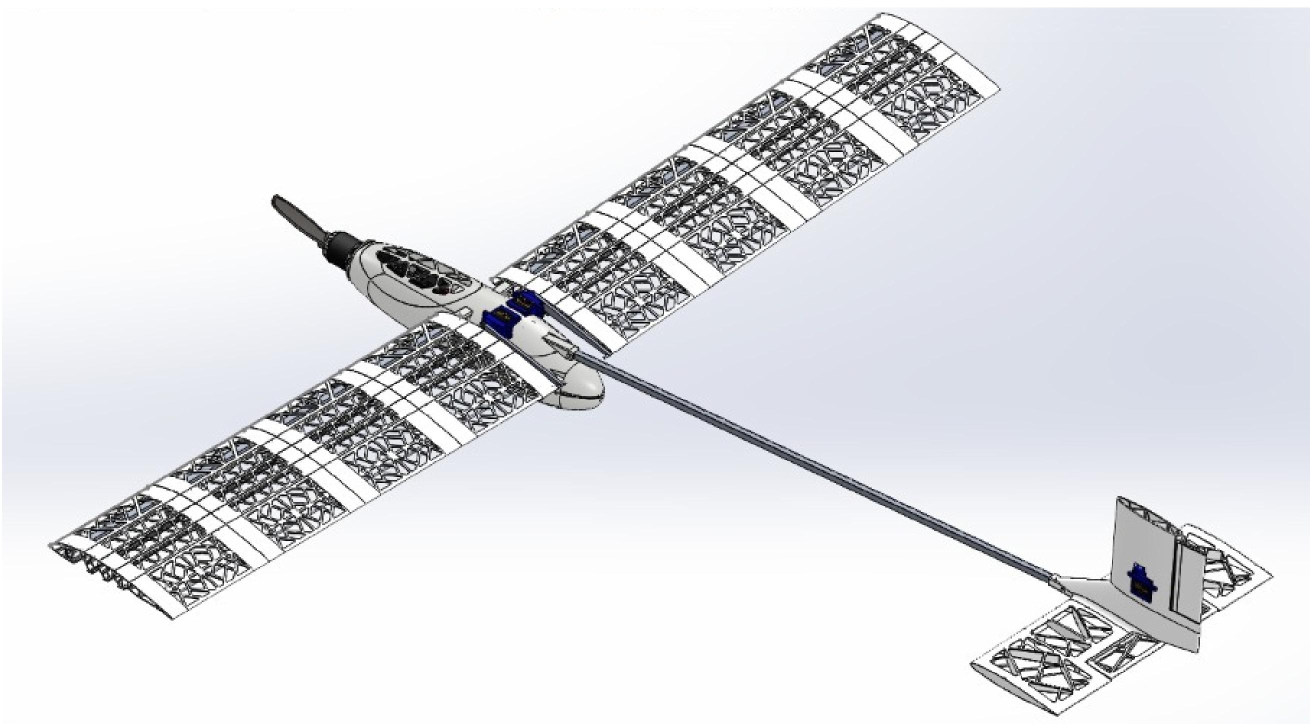
Fuselage



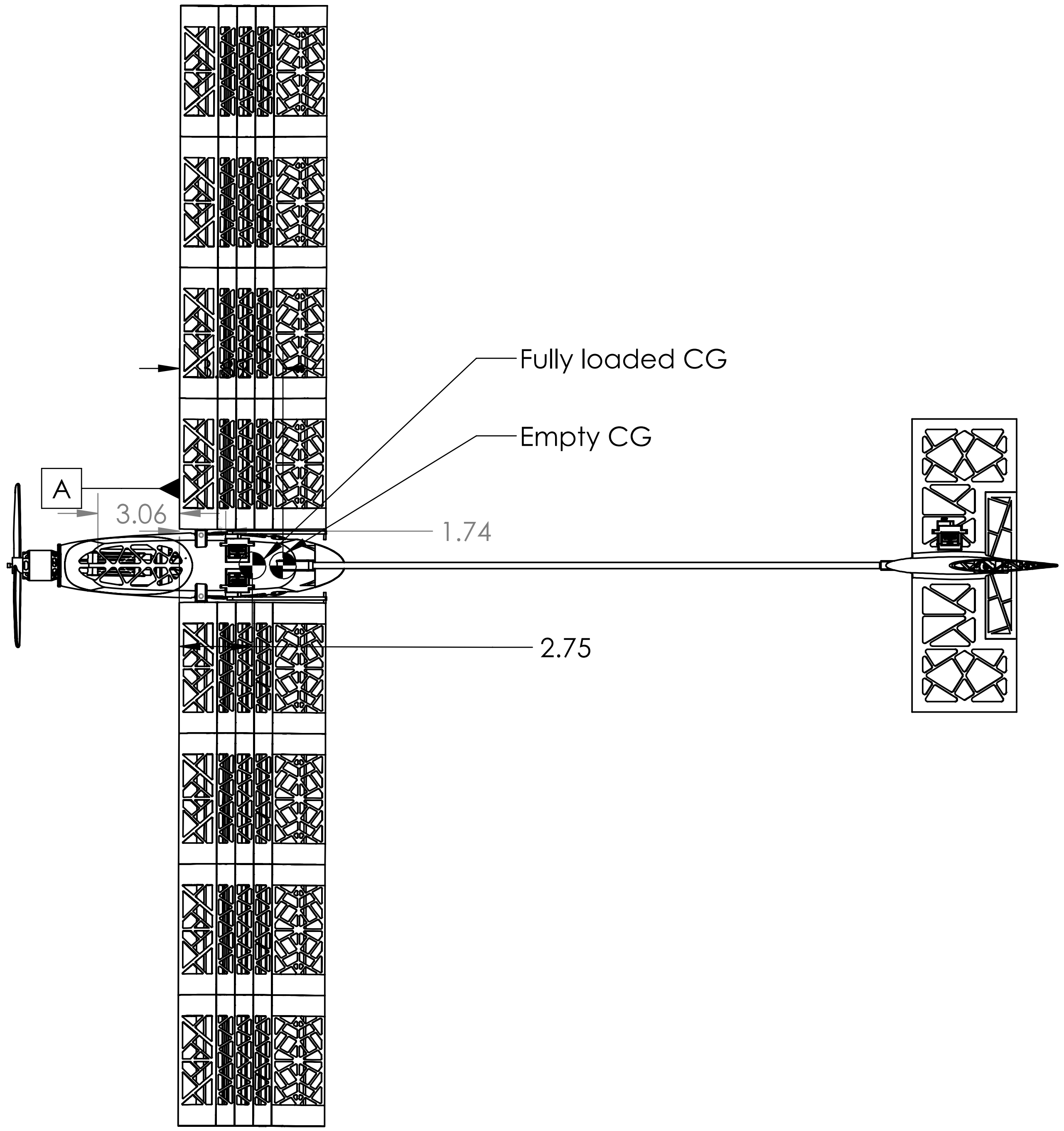
Fuselage back view, with payload door open



Full Plane



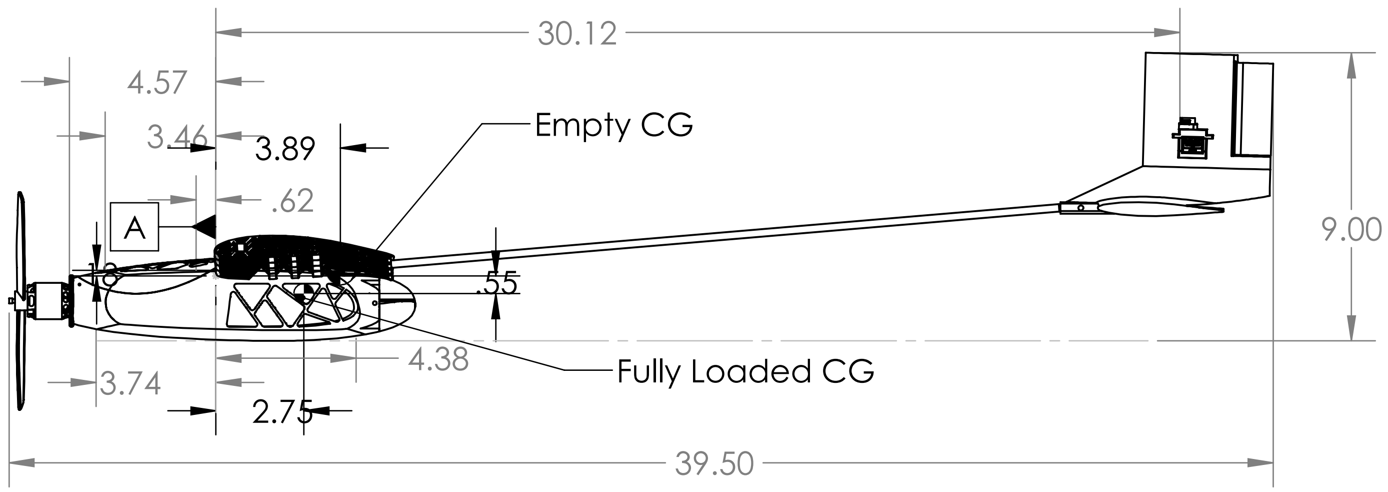
Full Plane (different view)

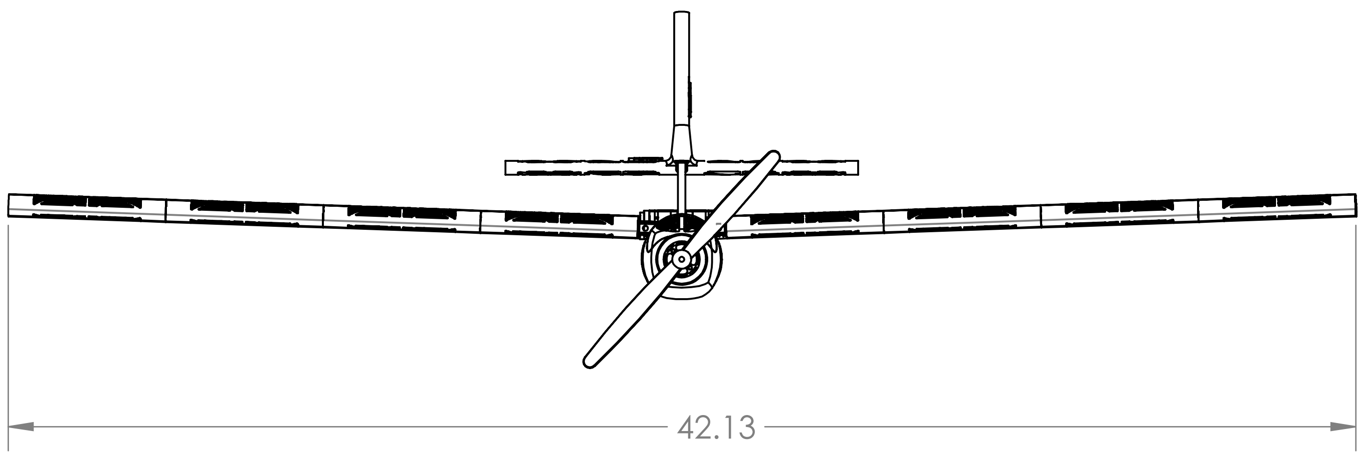


Team No.	ADC20210118
Team Name	PEGASUS | Alpha
University	BITS PILANI K.K.BIRLA GOA CAMPUS

Wingspan	42.13"
Empty Weight	0.731 KG
Motor	TMOTOR AS2317 KV2600
Propeller	9" x 7

Component	Weight(KG)	Location WRT Datum (Wing LE)
Motor	0.081	4.57"
Battery	0.12	3.74"
Payload	1.2	0.62" aft to 4.38" aft
ESC	0.034	3.46"
Servo(Tail)	0.009	30.12"
Servo(Wing)	0.009	1.74"
Reciever	0.025	3.06"





Note: The empty CG is aft of the fully loaded CG

Technical Data Sheet: Weight Build-Up

Component	**Weight (grams)**
Wing Body (ABS) (x2)	237.5
Main Wing Spars (x2)	38.6
Wing Trailing Edge Spars (x2)	9
Fuselage Body (ABS)	81
Tail Body (ABS)	55.2
Tail Boom	27.6
Battery	120
Motor	81
Propeller	22
Servos (x4)	36
Wing Servo Arms (x2)	5
ESC	30
Receiver Module	25
Torsion Springs (x8)	20