



UNIVERSITY OF UDINE

Technical Report



AIR CARGO CHALLENGE 2024

Team 14

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List of Acronyms and Symbols

ACC : Air Cargo Challenge;	$g = 9.81m/s^2$: gravity acceleration;
CAD : Computer Aided Design;	G_c : controller transfer function;
CFD : Computational Fluid Dynamics;	H : system transfer function;
CG : aircraft center of gravity;	I : electric current;
CNC : Computer Numerical Control;	K_F, K_M, K_P : gain (constant);
ESC : Electronic Speed Control;	l : wing loading per unit span;
FEA : Finite Element Analysis;	L : lift force;
FDM : Fused Deposition Modeling;	m : aircraft mass;
GPS : Geometric Positioning System;	M : feedback loop transfer function;
IPE : I-beam Parallel flange European;	mac : mean aerodynamic force;
MDF : Medium Density Fiberboard;	M_b : bending moment;
MTOW : Maximum Take Off Weight;	P : payload mass;
PETG : PolyEthylene Terephthalate Glycol;	R_a : absolute roughness;
PVC : PolyVinyl Chloride;	Re : Reynolds number;
RC-TX : Radio Control Transmitter;	SM : static margin;
TPU : Thermoplastic PolyUrethane;	T : thrust;
	v : aircraft velocity;
	W : aircraft weight;
AOA : Angle of attack;	X_{cg} : CG position with respect to wing leading edge;
Ar : Wing aspect ratio;	X_{np} : neutral point position with respect to wing leading edge;
Aw : Wing surface;	α : servo characteristic angle;
b : wing span;	β : servo characteristic angle;
b_T : tail span;	δ : dihedral angle;
c : wing chord;	ε : aerodynamic efficiency;
C_d : drag coefficient per unit span;	ρ : air density ($1.16kg/m^3$ when not otherwise specified);
$C_{D,0}$: drag coefficient at zero lift;	Π : power;
$C_{D,3D}$: total drag coefficient;	τ : torque;
C_l : lift coefficient per unit span;	ω : angular velocity.
$C_{L,3D}$: total lift coefficient;	
D : drag force;	
e : Oswald factor;	
emf : electromotive force;	

Introduction

This document describes the main steps taken by the *AeroUD* team to design, build and develop the model aircraft *LEObeta* to compete in the next Air Cargo Challenge (ACC 2024).

The *AeroUD* team has a long history, having started its first project in 2011. Since then, the team has taken part in five competitions and has always finished in the top positions.

The present team was formed in November 2022 with a solid knowledge of composite materials and simplified aerodynamic design, based on the experience of the previous teams, whose members were available for an effective transfer of know-how.

The ambitious challenge of building a completely new model from scratch was therefore taken on. The main efforts were concentrated on:

- finding the best propeller for the given motor;
- developing new electronic components to fly under the new regulations;
- finding a new, thinner wing profile that performs better in the new mission with respect to the high-lift airfoil used in the 2022 ACC edition;
- identifying an optimum wing planform geometry given the new size constraints;
- improving the design of the wing spar with the aim of withstanding the new higher loads while being as light as possible;
- designing a new fuselage for the new type of payload;
- designing and testing a new lightweight composite landing gear.

This report describes the main issues of above in detail, and will give little, but sufficient, information about all the other aspects of the project to give the reader an overall view of the whole process.

1 Project management

This section is devoted to a brief description of the team management in terms of time scheduling and activity planning and it will also provide an overview of the team budget.

1.1 Team structure and activity scheduling

The recruiting of the team started in October 2022 and 6 students accepted the challenge: 5 mechanical engineering students and 1 electronic engineering student. It was therefore decided to organize the team into three working groups with the following tasks:

- **Aerodynamic division:** design the aerodynamic surfaces of the aircraft;
- **Electronic and propulsion system division:** select electronic components, develop a real time telemetry and design new regulators;
- **Structural division:** model the external surfaces of the plane (excluding wing and tail) and the internal structures of the aircraft's components.

It was then decided that a team leader would be appointed to co-ordinate the work and to manage budget and schedule (Figure 1) as all the team members worked on the flight tests and on the construction of the aircraft's components.

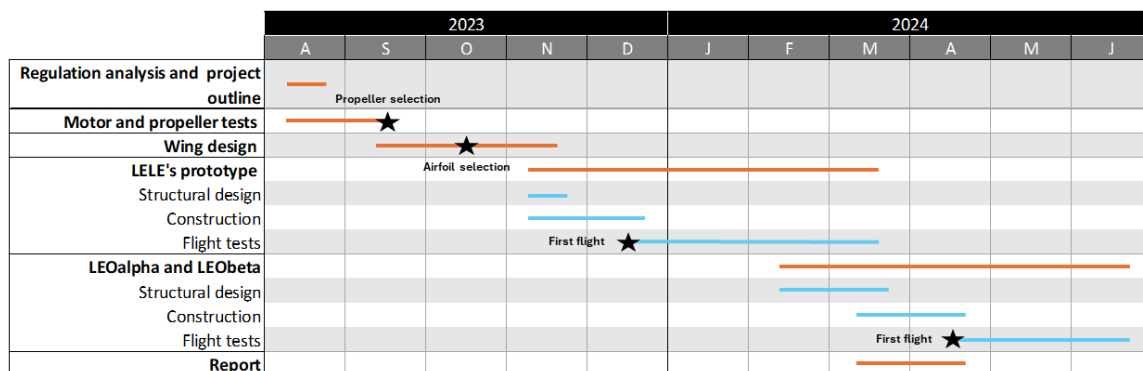


Figure 1: Gantt diagram for activity scheduling

The first months were devoted to the aerodynamic design of a final wing-tail configuration and to the construction of a first prototype (LELE) used to proof the validity of the aerodynamic design. Secondly, the attention was focused on the design of a second aircraft (LEOalpha) with the desired aerodynamics.

This second prototype is used to test performance in race mode and after proof of validity a new final aircraft (LEObeta) is built with the objective of optimizing its weight.

At the present time, the second model (LEOalpha) is validated and will be taken to Germany to be used as spare pieces. The final aircraft (LEObeta) is under construction

and its maiden flight is foreseen for mid-May.

In the remaining time the team will focus its attention on flight tests to define the best race strategy.



Figure 2: *LELE* prototype



Figure 3: *LEOalpha* prototype, top and lateral views

1.2 Budget management

Although the budget provided by the University was substantial (10'000€), careful budgeting was essential, also in view of the extensive use of expensive construction materials (i.e. high modulus carbon fiber) and high quality electronics (servos, ESC, LiPo batteries)

that the team intended to adopt. A first estimation of the costs was based on the financial budget of the previous competitions.

Considering that the first prototype (LELE) was completely built using leftovers from last years competitions, it won't affect the total budget.

A simplified table of costs is shown in Table 1:

	Quantity	Price [€]	Subtotal [€]
Electronics			
Motor	2	85	170
Flap servos	8	55	440
Receivers	2	143	286
Batteries	10	59	590
Others	-	-	450
		Total	1'936 €
Building materials			
Carbon fiber fabric ($2m^2$)	2	500	1'000
Epoxy resin	3	32	96
Sandwich materials (PVC foam, <i>Nomex</i>)	-	-	300
Carbon tubes	-	-	320
MDF for the molds	5	160	800
Other	-	-	350
		Total	2'866 €
Fees and travel			
Competition fee	6	300	1'800
Travel costs	-	-	2'000
		Total	3'800 €
		Total	8'602 €

Table 1: Overview of the project economical budget.

The budget spent is less than what provided the university. This is due to the fact that the first prototype, part of the second, and part of the race plane were built using leftover materials and components from previous competitions.

The remaining money will be used for the new team uniforms and to cover unforeseen expenses.

2 Aerodynamic design

This paragraph is dedicated to the description of the methodologies that are at the core of the aircraft aerodynamic design. Being the team composed by student with no specific background in aerospace engineering, a very simple approach has been pursued. However, satisfactory performances were reached in a first prototype design, leaving further optimizations based on the experimental evidence to flight tests.

Before going into details about the aerodynamic surfaces design, some preliminary considerations must be made:

- the aircraft has to fit into a box of limited size, the total dimensions of which must not exceed 1400mm;
- the aircraft wing must have the highest aspect ratio as possible to reduce induced drag;
- to increase span reducing occupied box volume the wing is divided into 4 sections.

To simplify the team's work, a rough process was first sketched out, dividing the project into 3 main steps, shown in Figure 4:

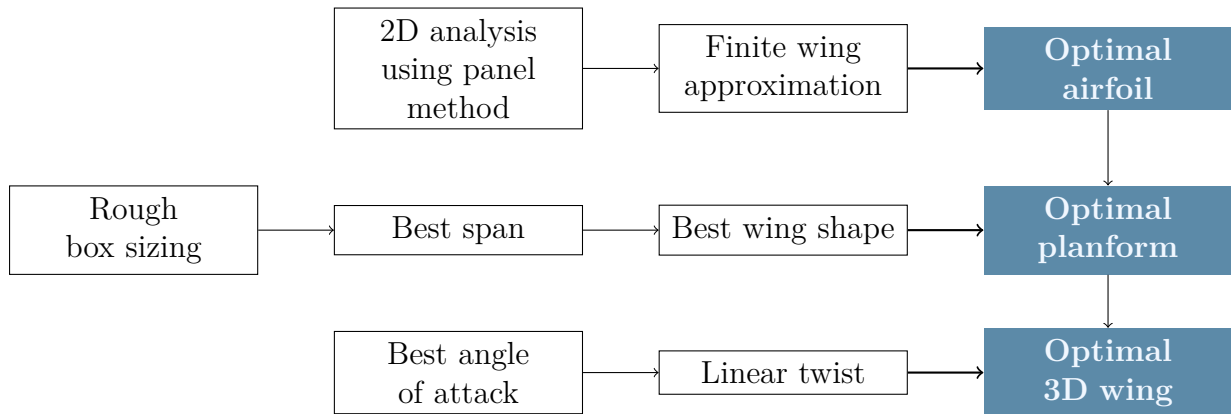


Figure 4: Wing optimization process

2.1 Wing 2D optimization

The design objective was maximizing the points associated with the desired mission:

$$SCORE_{flight} = \frac{P_{team}}{P_{max}} + \frac{D_{team}}{D_{max}} + \frac{E_{team}}{E_{max}}$$

Assuming straight level flight, MTOW and aircraft speed can be easily quantified. On the other hand to compute the efficiency factor, instantly knowing aircraft speed and

thrust would be needed, a task that no simple approach could accomplish.

It was therefore decided to replace the last term of the fitness function with the aerodynamic efficiency of the airfoil $\varepsilon = \frac{C_l}{C_d}$. The fitness of a given airfoil was thus chosen to be:

$$fitness = v \cdot m \cdot \frac{C_l}{C_d} \quad (1)$$

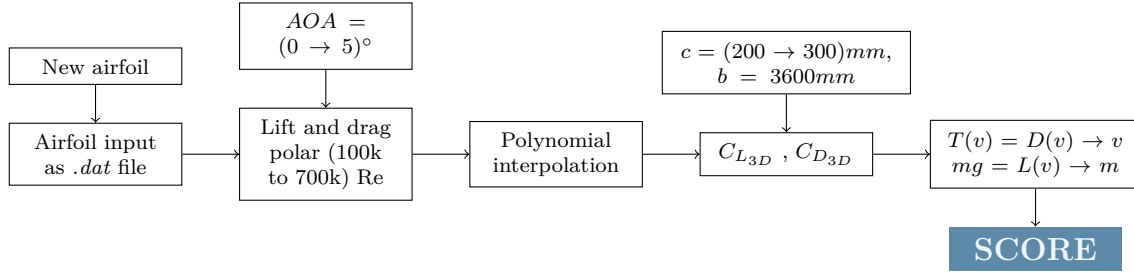


Figure 5: Airfoil selection process

As shown in the above diagram (Figure 5), for a given airfoil, at a given angle of attack, 2D polar are obtained using *XFOIL* by Mark Drela. The polar then can be used to compute lift and drag coefficients of the airfoil at different Reynolds conditions. Introducing finite wing effects from Prandtl's lifting line theory (Equations 2 and 3) it's possible to approximate the total lift and drag coefficients of a rectangular wing. The Oswald factor $e = 0.7$ and a wing span $b = 3600mm$ (supposing 4 sections of $900mm$) are estimated while the chord varies in the range $c = 200 - 300mm$.

$$C_{L,3D} = \frac{C_l}{1 + \frac{2}{A_{Re}}} \quad (2)$$

$$C_{D,3D} = C_{D,0} + C_d + \frac{C_{L,3D}^2}{\pi e A_R} \quad (3)$$

Given the experimental thrust curve of the engine-propeller (Section 4.1), supposing straight level flight the system of equation (4) is obtained:

$$\begin{cases} T = (-0.0043 \cdot emf - 0.2288)v + 1.008 \cdot emf + 6.25967 = \frac{1}{2}\rho v^2 C_{D,3D} A_W = D \\ W = mg = \frac{1}{2}\rho v^2 C_{L,3D} A_W = L \end{cases} \quad (4)$$

In the first equation the only unknown is the velocity which can be computed through iterations, in the second one the lifted mass is exactly computed as $m = 0.5v^2 C_{L,3D} A_W / g$.

This procedure was automated in a python script and ran for over 1600 airfoils from *UIUC airfoil database* (https://m-selig.ae.illinois.edu/ads/coord_database.html) for angles of attack in range $AOA = 0 - 5^\circ$.

The airfoil that got the maximum score was the *Eppler E216* at angle of attack $AOA = 3^\circ$ (Figure 6), which proved to perform very well in the required conditions (Figure 7).



Figure 6: Eppler E216 geometry

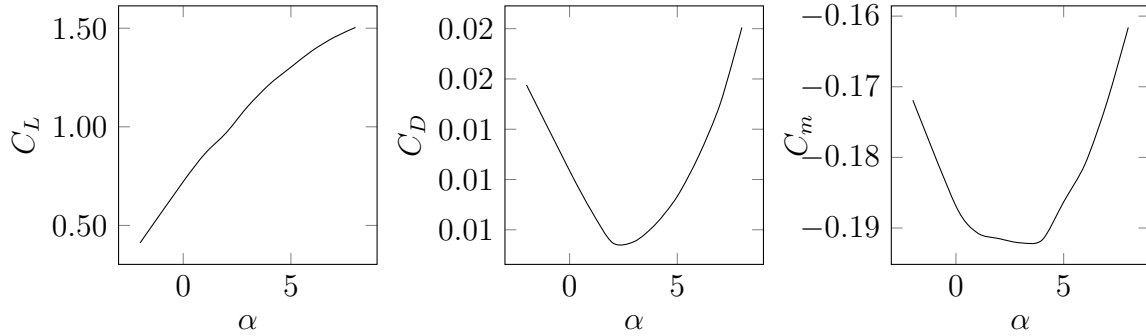


Figure 7: Eppler E216 aerodynamic properties at $Re = 300k$

2.2 Wing 3D optimization

After the first optimization step, the airfoil and a first try angle of attack value were determined. The second optimization step has the goal to find the best wing planform for the given conditions in order to reduce induced drag. The data in this step are obtained using 3D panel method in *XFLR5*.

First elements to be decided were wing span and airfoil chord.

Supposing again a rectangular planform, and using the same fitness function as before (Equation 1), different chords were tested (again in range $200 - 300mm$) getting the best score for $c \simeq 250mm$.

Given the chord, a first draft of the transportation box could be obtained. Using the CAD software *Fusion360* after various testing, a section span of $b = 850mm$ was decided to be the greatest value that allowed all the aircraft components to fit into the box.

Given the rectangular shape of the wing, a further geometry optimization was mandatory trying to obtain an elliptical load distribution. Different wing planforms were thus evaluated, in particular an elliptical geometry and some of its variations. Again *XFLR5* and the fitness function (equation 1) were used until the geometry sketched in figure 8 was identified which resulted the best.

Given the wing planform, a new best angle of attack for the wing was mandatory (since the first angle was obtained using just the airfoil geometry).

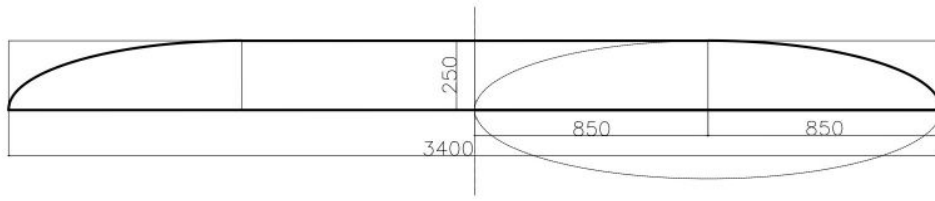


Figure 8: Shape and dimensions of the wing planform in [mm].

Again the same fitness function was used and the optimal value obtained was $AOA = 1^\circ$.

Finally $AOA = 2^\circ$ at the root was chosen, taking into account the twist and a possible forward tilt during flight.

A twisted wing was mandatory to achieve maximum efficiency, but in order to reduce the number of molds needed the decision to make the two root sections identical was taken. A linear twist was thus used just for the tip sections in an attempt to achieve an elliptical load distribution according to the Schrenk criterion.

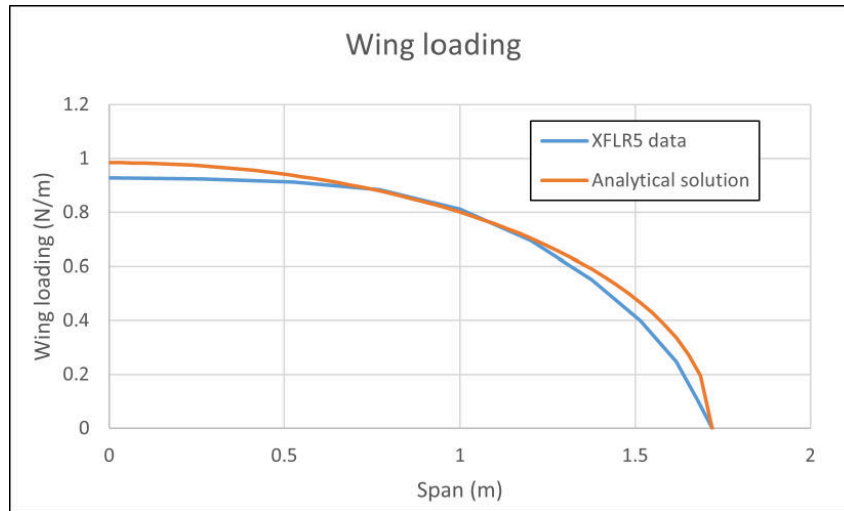


Figure 9: XFLR5 wing loading vs analytical solution according to Schrenk criterion

The wing loading is defined as:

$$l(y) = c(y) \frac{C_l(y)}{mac}$$

where y is the span coordinate, l is the loading, c the chord and mac mean aerodynamic chord.

According to the Schrenk criterion, an elliptic loading distribution is obtained as:

$$l_e(y) \simeq \frac{C_{l,mac}}{mac} \frac{4A_w}{\pi b} \sqrt{1 - \left(\frac{2y}{b}\right)^2}$$

Because of the decision to have symmetrical root sections the obtained loading distribution was not perfectly elliptical but a satisfactory result was obtained, shown in Figure 9.

2.3 Tail design, ailerons and flaps

2.3.1 Tail design

The capabilities of the free software *XFLR5*, which can also perform static stability calculations, were again exploited, this time to correctly size the aircraft's tail.

The team opted for a V-tail full-flying configuration (Figure 10) because of its excellent maneuverability and efficiency, not only in terms of control but also in terms of possible stability control trimmers when using the flaps; a reduction in the overall drag of the tail is possible compared to a more conventional T and flapped configuration. In addition, a V-tail configuration is easier to handle in turbulent conditions, and it's simpler to build.

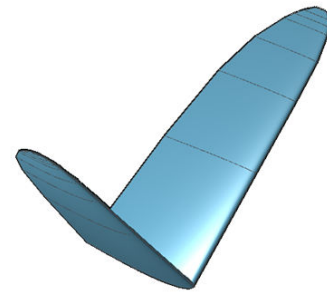


Figure 10: Tail geometry

The capabilities of the free software *XFLR5*, which can also perform static stability calculations, were again exploited, this time to correctly size the aircraft's tail.

The symmetrical *NACA0009* airfoil was chosen as it has a very low drag coefficient while maintaining a significant thickness per quarter chord. This feature was fundamental to the design in order to have enough space for the installation of both servos and spars. The tail pitch angle was set at zero degrees to minimize its contribution to drag.

The geometry chosen for the tail was determined from the dimensions of the main wing using correlations from the literature (20–30% of the main wing area), (Daniel P. Raymer, *Aircraft Design: A Conceptual Approach*, 4th edition).

This analysis gave a total surface area of $S_T = 0.149m^2$ that can be obtained with a trapezoidal planform with a chord of $c_{T,r} = 202.5mm$ and $c_{T,t} = 130.5mm$ at the root and tip respectively and a length of $b_T = 445mm$. The dihedral angle was chosen to be $\delta = 45^\circ$. The tail was redesigned to reduce the induced drag without losing stability and maneuverability, and a 10% reduction in surface area to $S_T = 0.1374m^2$ was adopted.

2.3.2 Stability analysis

Once the tail geometry was determined, it was necessary to check the static stability, which is ensured as long as the static margin is positive.

The static margin is a dimensionless coefficient defined as:

$$SM = \frac{X_{np} - X_{cg}}{mac}$$

where X_{np} is the position of neutral point and X_{cg} the position of the center of gravity, both calculated with respect to the wing leading edge at the root. The latter was unknown at this stage of the design, thus deciding to impose a target SM and to calculate the value of X_{cg} that would give the desired static stability.

The static margin was set to 10%, which was considered to be a good compromise between stability and maneuverability. The results of the analysis using *XFLR5* are summarized in Table 2:

X_{np} [mm]	X_{cg} [mm]	mac [mm]	SM [%]
136.2	112.6	232.2	10

Table 2: Static stability results

The value of X_{cg} in the table was therefore chosen and the wing and tail design was considered as concluded. However, the choice of a variable tail boom length that can be elongated up to 140mm over the above reported design length (as shown in the drawing n.1) was made to allow for a greater static margin in case of unexpected problems with the control of the final aircraft and also to allow the setting of the design CG.

The fuselage design had to respect the given position for the center of gravity, in particular, no variation of the CG is allowed with variable payload.

2.3.3 Flap sizing

At an early stage in the design, the introduction of flaps for take-off were considered as unnecessary due to the large size of the wing. However, later they were still included in the first experimental prototype to avoid any eventuality during the first flights. The flaps later proved to be very useful, especially for landing, so the decision to include them in the final wing was taken. The decision whether using them or not is left to the flight strategy definition.

The flap size was determined using relations from literature, giving 25% of the chord to be flapped getting a surface of $S_F = 850 \times 60mm$.

2.4 Fuselage, cargo bay and engine pod

The design of an aircraft fuselage is a crucial aspect that directly affects performance and efficiency. The main goal was a design that could accommodate the regulatory payload,

minimizing the wet surface and the presence of sharp edges. This approach aims to improve the aircraft's overall aerodynamics, reducing drag and improving flight efficiency.

To achieve these goals, a different approach was adopted, integrating the cargo bay and engine pod directly into the fuselage. The outer geometry was obtained from literature correlations and confirmed using CFD analysis (Figure 11):

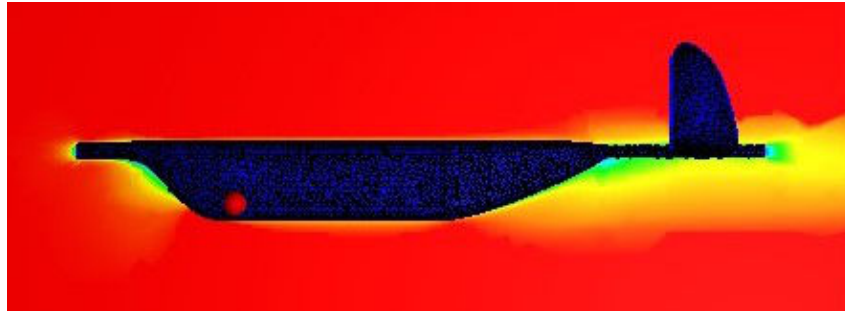


Figure 11: Velocity field outside of the fuselage, obtained using CFD techniques

This solution not only reduces drag, but also optimizes internal space, allowing for more efficient use of space and better weight distribution. A critical aspect of this design was to ensure that the payload center of gravity (fixed for each payload configuration and determined by static stability analysis) was aligned with the overall fuselage center of gravity. This ensures that the payload does not affect the stability and control of the aircraft in any way during flight.

The final cargo bay configuration is capable of holding 32 billiard balls for a total payload of $P = 5024g$. The number of billiard balls to be lifted during flight was again calculated using the *XFLR5* panel method, considering the wing geometry obtained in sections 2.1 and 2.2, giving an upper MTOW ($= L/g$) of about $10kg$. Due to the box limitations, the largest fuselage size that fit in the box accommodates 8 rows of 4 balls (see drawing n.3).

Since the empty weight of the plane was set to be a maximum of $W_{e,max} = 2.5kg$ and the maximum accommodating payload of about $P = 5kg$ i.e. for a total MTOW of $7.5kg$. The size of the payload resulted in slightly less than maximum predicted lifted weight and was anyway accepted with a safety factor because of the inaccuracies of the approach.

The fuselage design integrates not only the cargo bay and engine pod but also directly accommodates the main and the nose landing gears, actuated by a servo mechanism. More details of the fuselage and cargo bay internal structures are provided in the related section (Section 3.1).

Particular attention was paid to the aft fuselage, where experimental relations from *Airplane Design, part II* by Jan Roskam were used.

3 Structural design

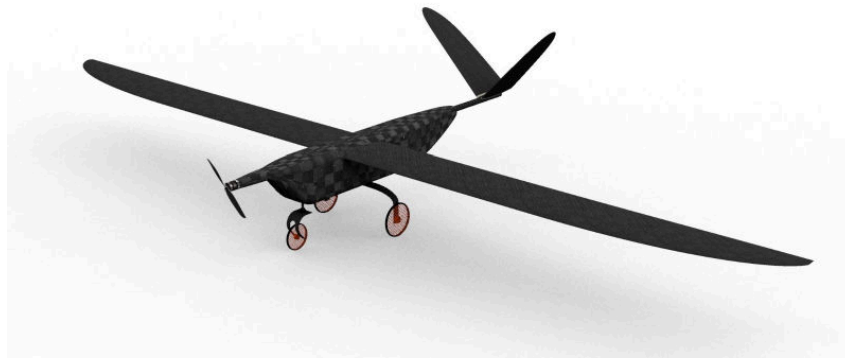


Figure 12: Final airframe

This section describes the internal structure of the airframe as defined in Chapter 2 (Figure 12) and the decisions that led to the final results.

The team decided to use composite materials for the following reasons:

- highest resistance-per-weight ratio, fundamental for maximum performance;
- excellent surface finishing for laminated components, essential for aerodynamic efficiency;
- ease of production of complex surfaces through proper machining of the molds using CNC technologies.

Despite all the advantages, a downside to this choice was that an accurate structural analysis of the highly anisotropic materials such as the composite sandwiches intended to be used is not an easy task to accomplish. A numerical approach by means of FEA software is required but none of the team members have sufficient background on the subject.

However, the expected mechanical loads on the structures are not so high to oblige for a careful design of the components. Indeed, considering that:

- the box size limits the wing span to a maximum of $b = 3400mm$;
- the box size also limits the maximum fuselage dimension with a cargo bay that can host up to 32 pool balls, i.e. about $5kg$;
- the choice of using composite materials should guarantee an aircraft empty weight of about $2.5kg$, therefore a MTOW of about $7.5kg$.

The aircraft structure was therefore designed assuming a maximum bending moment

for the wing of $M_b = 70Nm$, suggesting that an accurate design could be omitted due to the very low mechanical loads and rely on composite structures used in the past ACC competitions and that proved to be able to withstand much higher loads (the team had experience of aircrafts with MTOW up to $17kg$ with a wingspan of $5m$, generating a bending moment of $M_b = 210Nm$).

Consequently, the structural design was mainly focused on creating a lightweight and aerodynamically efficient airframe, while the mechanical strength was verified by experimental tests on the finished model.

3.1 Wing and tail structure

The wing is made up of four sections, each $850mm$ long, connected by short carbon tubes inserted into cylindrical pockets also made from bigger tubes.

The carbon tubes are selected to meet two criteria:

- fitting into the wing profile;
- being sufficiently strong to withstand bending moments.

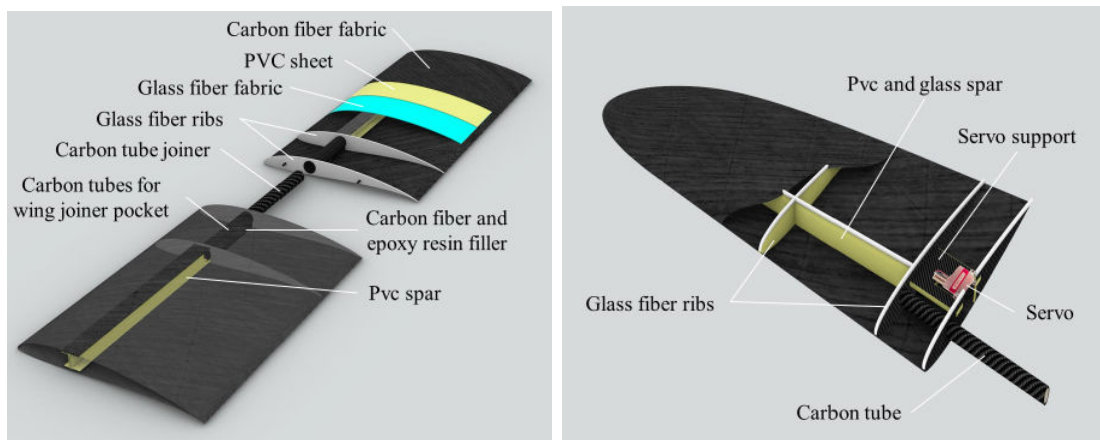


Figure 13: Internal structure of wing and tail

A bending test was carried out to simulate the static load that the wing would experience during the competition. The four sections of the wing, connected by the tube, were suspended at both ends and a load of $8kg$ was applied in the middle. In the end, the choice went for commercial pultruded carbon tubes of $2mm$ thickness and an outer diameter of $18mm$ and $16mm$ for root and tip sections, respectively.

To increase the contact area and transfer the bending moment between the pockets and the inner surfaces of the wings, the space between them was filled with resin mixed with carbon fiber reinforcement.

The sandwich structure of the wing skin has to withstand the torsional forces of the wing and retain sufficient stiffness to maintain the correct aerodynamic profile under load, it consists of three layers: carbon fabric (*carboweave*, $43\text{gr}/\text{m}^2$, $+/- 45^\circ$), PVC foam (1.2mm) and glass fiber ($25\text{g}/\text{m}^2$) (Figure 13).

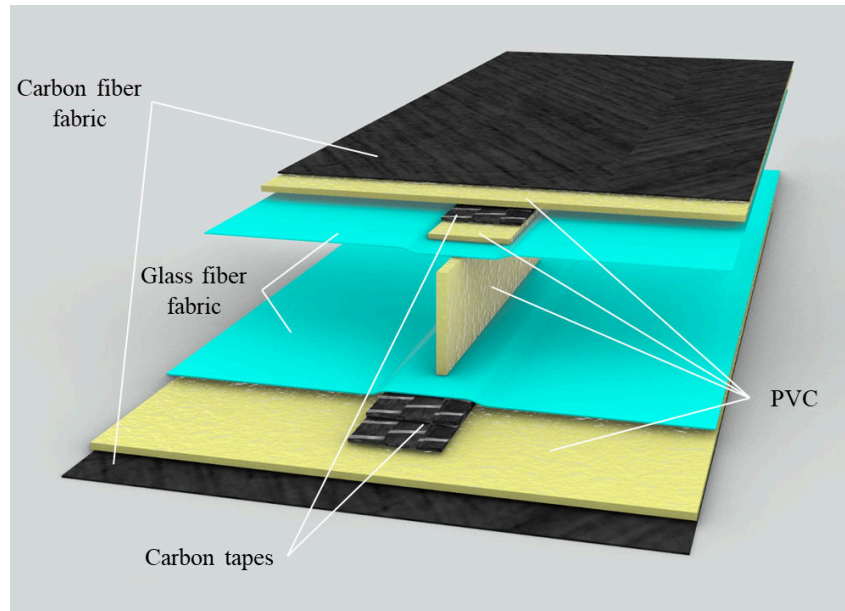


Figure 14: Internal structure of the wing resembling an IPE section

The wing's load distribution on the other hand is primarily managed by the internal spars, which are constructed using an IPE section. The horizontal part is integrated into the wing sandwich and consists of unidirectional carbon tapes ($125\text{gr}/\text{m}^2$, 25mm width) and PVC, while the vertical consists of a sandwich made by 3mm thickness PVC foam and 2 layers of $80\text{gr}/\text{m}^2$ of glass fabric (Figure 14).

A similar process was followed for the tail, which has the same structure as the wing for the external skins but being lighter since the spars are no longer necessary because of the lower loads acting on the tail.

3.2 Fuselage structure

The wing is attached directly to the fuselage, while the cargo bay and engine pod are integrated into the fuselage itself. The fuselage must fulfill many requirements:

- allow easy and quick payload loading;
- withstand the acceleration forces on the payload during the flight;
- accommodate the landing gear and therefore withstand the landing loads;
- have minimal frontal cross section and wet surface in order to limit drag.

Composite technologies allows the fuselage to be manufactured in one piece, giving the aircraft a streamlined design and improved aerodynamic efficiency. This feature means that the fuselage is self-supporting, offering a robust yet lightweight structure requiring fewer internal reinforcement frames. Such an approach allows the overall weight of the aircraft to be optimized, thereby increasing payload capacity.

The upper section of the fuselage provides just enough space for the receiver, its battery, the engine battery and the Automated Measuring Equipment, while the custom ESC is located in the engine pod (see Drawing 3). The payload is loaded from the rear, allowing the positioning of two trays to hold and separate the billiard balls. The trays are machined out from polystyrene slabs of $30\text{kg}/\text{m}^3$ density, getting a light but still easy to handle support for the payload (Figure 15). The fuselage is vacuum-formed from a sandwich of $60\text{g}/\text{m}^2$ carbon fiber, 1.2mm thick PVC foam and $25\text{g}/\text{m}^2$ glass-fiber fabric. The rear fairing is made from a single sheet of carbon fiber, as its sole purpose is to improve aerodynamics.



Figure 15: Cargo bay trays

The landing gears are a sandwich of carbon fiber and *Kevlar*, with a 3mm *Nomex* core, providing robust support for the aircraft during take-off and landing and minimizing additional weight. Its mechanical resistance was proved by loading it with about 300N (i.e. more than 3 times the expected MTOW).

The wheels are 3D printed using FDM technology and their design is the result of both FEA simulations and airfield trials. The inner part is made with an optimized structure of polyethylene terephthalate glycol (PETG) that also houses the bearing, while the outer part is printed in thermoplastic polyurethane (TPU) a softer plastic to get more grip and simulating a tire.

3.3 Construction process

In order to produce all the composite components of the aircraft, the vacuum molding technique was chosen. To achieve this, the team had to carry out several operations to enable the production of the aircraft's structural components.

The first step is to create the molds in MDF panels. A 3-axis CNC milling machine is used meaning that the team members had to acquire the skills to use the machine already present in lab, such as the ability to draw complex 3D surfaces and generate the resulting milling paths. This took some time and several attempts were made due to the complexity of the machine, but allowed to produce all the components in-house, giving more flexibility and saving money.

Once machined, the molds have to undergo surface treatment, they are first painted and then polished several times. This process greatly improves the surface finish (final roughness $R_a = 20\mu m$, based on the fact that the final step is polishing with P1000 grit paper). At this point, the molds are treated several times with a release wax and finally covered with a thin layer of PVA release agent. The final result is shown in Figure 16.



Figure 16: Wing and fuselage molds

At this point it's possible to start working on the composite components. A high quality Epoxy-F resin was chosen as matrix. Once the sandwich is impregnated with the resin, the molds are kept under vacuum in a warm room ($24^{\circ}C$ or more) for at least 24 hours to ensure complete curing of the polymer before any manipulation. The same technique was used to obtain the laminated panels for the internal structural components. In this case, no molds are needed, as the sandwiches are left on a glass plate under vacuum and then cut with the milling machine. The lamination process is shown in Figure 17.

Once all the components are manufactured, it's time to assemble. Some important



Figure 17: Carbon fiber fabric placed in the mold during lamination

things the team has to take care of are the correct alignment of the internal structure and interference checking. These steps are necessary to ensure perfect closing of the various parts to obtain the correct (previously calculated) angles of attack between the wings and the fuselage. Epoxy-F resin mixed with glass microspheres are used to seal the various parts.

4 Electronics and propulsion systems

This section is dedicated to the aircraft electronics, in particular the development of sensors used to optimize aerodynamic efficiency and the sizing of all the electronic components needed for flight.

4.1 Propeller test and selection

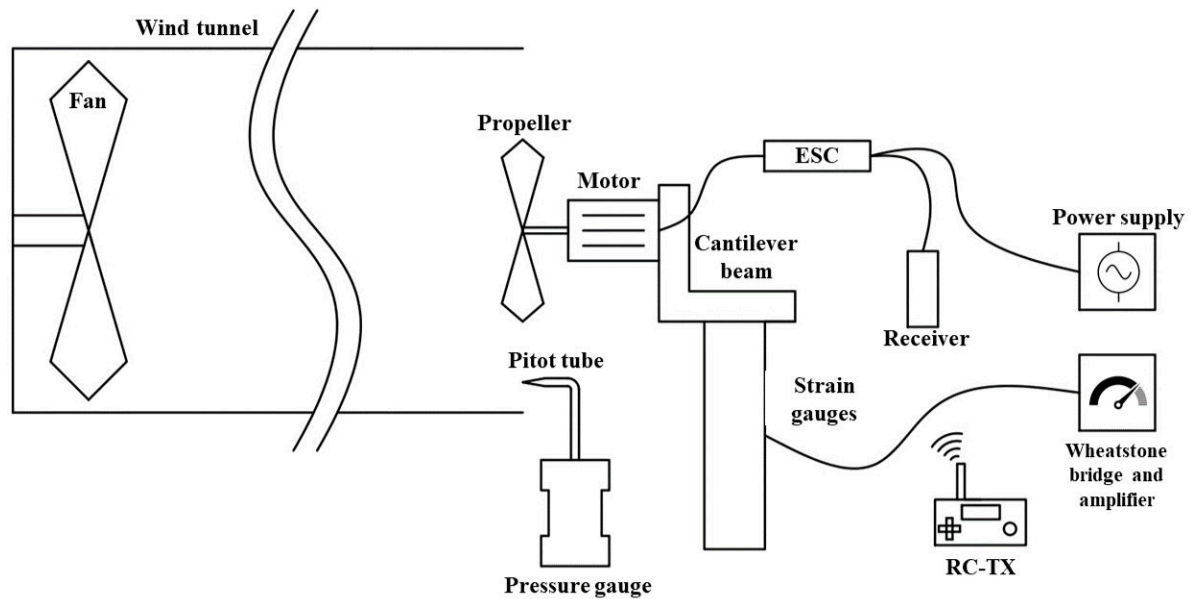


Figure 18: Propeller test bench simplified diagram

The main input for the aerodynamic design is the available propeller thrust, therefore a series of tests were carried out to evaluate the performance of propellers from two renowned manufacturers, *Aeronaut* and *APC*. These tests were also driven by the need to select the best propellers under the new regulatory restrictions that penalize currents above 30A.

The tests were carried out on a test stand already available in our lab and a schematic of which is reported in Figure 18. The motor is installed at the free end of a cantilever beam. The propeller thrust causes the beam deflection which can be measured by a 4 identical strain gauges forming a Wheatstone bridge (calibration with known force applied is necessary). The whole assembly is placed at the outlet of an open loop wind tunnel the velocity of which can be regulated to simulate different flight speeds. The flow velocity is measured with a Pitot tube connected to water column micro-manometer.

During the tests, different voltage levels were examined, corresponding to battery charge states, and different flight speeds ranging from $0m/s$ to $25m/s$ are simulated. Analyzing the data collected gave a clear performance overview of each propeller under different

conditions and also made clear that the implementation of a current control and limiter device was mandatory.

After careful analysis of the results, the *Aeronaut* 13×6 propeller was selected as the most suitable choice for the aircraft.

The data obtained from the tests was used by the aerodynamics team to select the airfoil, but first an interpolation of the data obtained for different flight speed and battery voltages was essential. First an interpolation of the type:

$$T = av^2 + bv + c \cdot emf + d$$

was found, however, it was quickly clear that the test case required a different type of function due to the new regulations, which place an upper limit to the current; in particular, the thrust is not limited by the maximum motor power as expected, but by the electrical power supplied.

Because of energy conservation: $\Pi_E + \Pi_A - \Pi_R = \Pi_M$ where $\Pi_E = I \cdot emf$ is the electrical power given by the battery, Π_A is the power given to the propeller by the incoming air, Π_R is the power lost to friction and heat generation and Π_M is the total power of the motor.

Knowing that torque and power are related because $\Pi_M = \tau \cdot \omega$ and $T \propto \omega^2$, made clear that a quadratic interpolation no longer fitted the experimental data, in fact the motor power Π_M increases up to a point where the contribution of Π_A gets to maximum motor power. Once this point is reached ($v \simeq 30 \frac{m}{s}$) the electric energy absorbed by the motor Π_E decreases requiring less power from the battery.

A bi-linear interpolation is therefore applied as follows

$$T(v, emf) = (-0.0043 \cdot emf - 0.2288)v + 1.008 \cdot emf + 6.25967$$

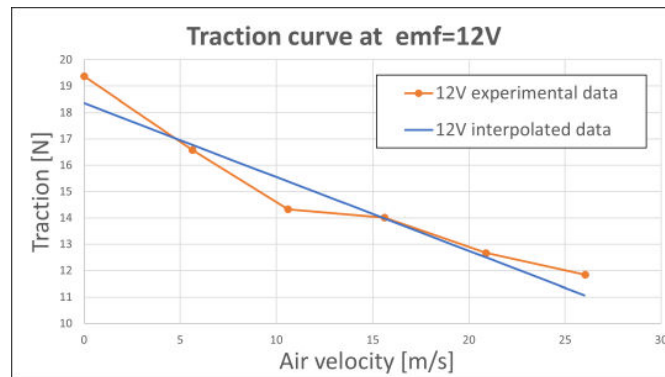


Figure 19: Propeller experimental thrust data and its interpolation at $emf = 12V$

This function has the double advantage of fitting the experimental data very well and being very simple to implement (Figure 19).

4.2 Telemetry

With the aim to verify and improve aircraft design and performance the team needed several experimental data from flight tests:

- both air and ground speed;
- wing angle of attack during flight;
- vertical speed (i.e. variometer);
- GPS position;
- current absorption.

This data were collected by different sensors, namely:

- a commercial GPS and current sensor as those that will be used in the competition;
- a Pitot tube connected to a pressure transducer linked to the GPS module;
- a custom made electronic board that integrates a commercial inclination sensor.

All these devices were connected to an ESP32 micro-controller used to collect all data on a common time base and store them on a micro-SD memory card to be later analyzed using a spreadsheet (Figure 20).

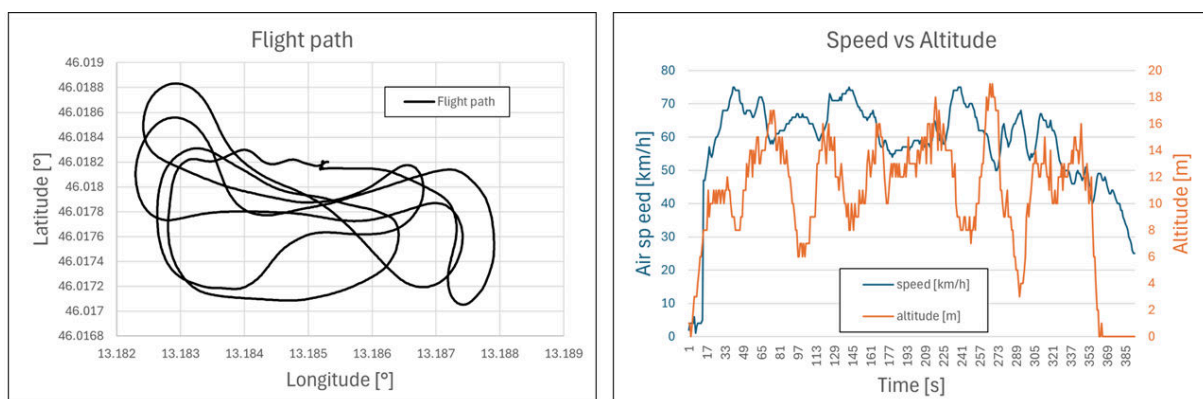


Figure 20: Example of data acquired during a test flight

Several test flights were performed both with first and second prototype and the gathered data were analyzed to verify the consistency with the design of wing and tail aerodynamics and to identify a flight mode optimized for the efficiency task request by regulation.

4.3 Current regulation system

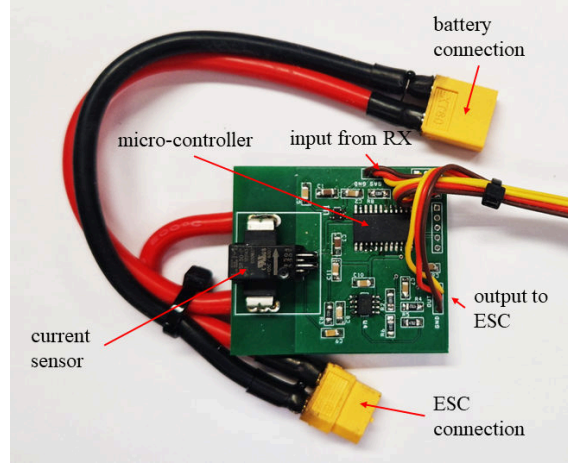


Figure 21: Picture of the current regulator

For the Air Cargo Challenge 2024, it was essential to develop a current regulator that was both robust and reliable in limiting the current absorption to 30A in any flight condition. The device must receive in input the TX throttle position and provide an output to the motor ESC (commercial from *Jety model*) under a current limit that can be manually adjusted by setting a dedicated channel on the RX-TX system. This single feedback system is based on reading the current as the output value and sending a block sum difference to the TX input. The project began by modeling a block diagram for the regulator (Figure 22), which featured a two-pole complex conjugate transfer function, ideal for ensuring adequate transient response and optimum settling time. The feedback current sensor, with its single pole transfer function, was tasked with monitoring the current and providing accurate feedback to the regulator.

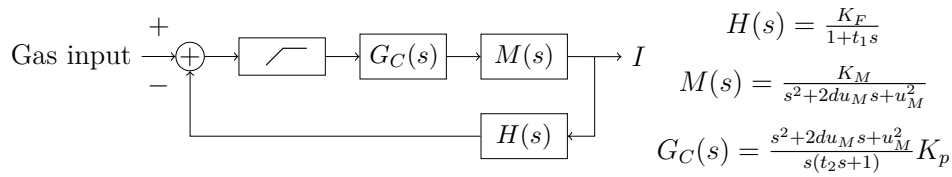


Figure 22: Current regulator block diagram

The most innovative part of the project was the modeling of a micro-controller that introduced two complex conjugate zeros, in addition to a pole at the origin and another pole, to compensate for the dynamics of the motor and sensor, thus stabilizing the system. The transfer function introduced was designed to respect the Nyquist theorem, thus ensuring the stability of the system. During the design phase, particular attention was paid to the choice of micro-controller parameters to ensure stability and the desired performance. Once the control system had been implemented, the ideal gain was sought

manually through various tests until the value that allowed the controller to operate at an optimum frequency was found, in accordance with the restrictions imposed by the competition rules.

The current regulator (Figure 21) proved to be effective, meeting the competition requirements and providing the necessary flexibility to quickly adapt to race conditions. Moreover, the ability to manually adjust the gain provided a significant competitive advantage, allowing real-time setting of engine performance in compliance with regulatory requirements without compromising flight quality. Peak currents above the limits of 30A could be profitably used in some flight phases, e.g. to allow take off at full payload, even arising a penalty. The decision to exploit this kind of opportunity will be part of the team strategy during the competition.

4.4 Battery

The choice of an appropriate battery was extremely important for the project, it needed to be light to not be a burden to the plane, but it also needed to be electrically adequate to guarantee the expected performance during the flight. Additionally, it had to respect the specifics from the race rules. Those last included having a maximum capacity of 100Wh, a maximum voltage of 12.6V.

The first step for the choice of the battery was to calculate its capacity, to do that an approximated time of flight, and the maximum current absorbed by the motor during that time are required.

The time was calculated in the eventuality of the most challenging scenario, wherein the total flight duration amounts to 4 minutes, comprising:

- take-off phase, anticipated to consume approximately 1 minute, factoring in multiple attempts for optimal lift-off;
- 90 seconds each for the efficiency task and the speed task;
- final 45 seconds for the landing maneuver.

Obtaining a total of $T = 240s$. Through analytic computation, a minimum required capacity to successfully navigate through each phase of flight was estimated. Using the formula:

$$C = \frac{1000 \cdot I \cdot T}{3600}$$

ensuring that the current I did not exceed 30A and where the factors 1000 and 3600 are used to scale the dimensions in mA and hours. Factoring in the entirety of the 4-minutes flight duration, a cumulative charge of 2000mAh was deemed essential for optimal performance and safe operations.

For the first tests a $3300mAh - 90C$ battery from the manufacturer *Fullpower* was chosen, being extremely over sized but in an attempt to preserve the highest possible battery voltage values until the completion of the flight speed task. The battery proved to work well but at the same time being quite heavy (about $320g$). In the next future, three more batteries will be tested $2600mAh - 60C$, $2600mAh - 90C$ and $3300mAh - 60C$, in an attempt to identify a lighter but still performing solution.

4.5 Servomotors for ailerons, flaps and V-tail

4.5.1 Flaps and ailerons

The servos have to be as light as possible, yet to provide enough torque to move the heavily loaded surfaces while fitting into the thin wing structure.

Knowing that the wing is divided into 4 sections, it was decided to have 1 flap at each roots, used for take-off and landing, and 1 aileron at the tips, used in combination with the tail to control the aircraft during flight. As a rule of thumb 4 identical servos are used since the forces acting on the flaps and ailerons are not very different. At the same time, knowing that the flap servos are slightly more stressed, are the ones chosen for the sizing procedure.

As a first sizing stage, the following relation was used:

$$\tau = \frac{1}{4} \rho v^2 c_F^2 L_c \sin \beta \frac{\tan \beta}{\tan \alpha} = 2.6 kg \cdot cm$$

where c_F is the flap chord, b is the flap span and α and β are characteristic angles shown in Figure 23.

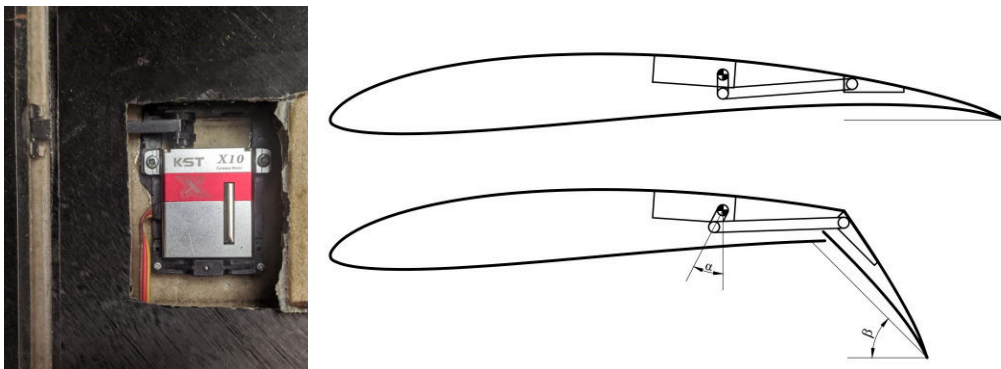


Figure 23: On the left, *KSTX10* mounted on the wing; on the right, flaps and ailerons servo kinematic

The first servo mounted was the *KST X08H Plus*, which guarantees a torque of $3.9Kg \cdot cm$ at $6V$. Although the value provided by the seller is much higher than the value obtained using the empirical formula, during flight the aerodynamic loads generated by the wing

	ρ	v	c_F	b	α	β
Flap	1.2 kg/m^3	30 m/s	60 mm	850 mm	60 deg	40 deg

Table 3: Flap dimensions used to calculate the required torque

caused the flap to rotate upwards. Therefore the decision to fit the larger *KST X10* was taken, which guarantees a torque of $7.5 \text{ kg} \cdot \text{cm}$ at $6V$ (Figure 23).

The available torque is much higher than required but the selected servos are characterized by high precision and stable positioning, essential qualities for a smooth flight path and to keep the surfaces in the neutral position when not activated.

4.5.2 V-tail



Figure 24: Tail servo kinematic

A similar approach to size the servomotors to be installed in the full-flying V-tail was followed. The schematics of the mechanism are depicted in Figure 24. Joint 1 is the hinge, joint 2 is the servo motor that is installed fixed inside the aerodynamic surface, joint 3 is a sliding joint realized on the tail boom.

The aerodynamic forces that act on the tail can be calculated assuming that air flow is incompressible, the angle of incidence is zero at neutral position and the maximum torque is at $\alpha_{max} = 10^\circ$.

Assuming that the aerodynamic forces act at 25% of the mean chord of the tail, by the drawings it is possible to deduce the distance between the first joint and the force application point. The aerodynamic forces (C_{aero}) have been evaluated by the profile polar curves for an incidence angle of $\alpha = 10^\circ$ and at Reynolds number of $Re = 300'000$.

Finally, the servo torque is:

$$\tau = C_{aero} \left(1 - \frac{L_{12} \cos \alpha}{L_{23} \cos \beta} \right)^{-1} = 0.31 \text{ kg} \cdot \text{cm}$$

	L_{12}	L_{23}	C_{aero}	α	β
Tail	28.6 mm	14 mm	0.051 Nm	10°	36°

Table 4: Tail dimensions used to calculate the required torque

The chosen servomotors are again the *KST X08H Plus* that this time proved to be strong enough to withstand the aerodynamic forces.

5 Flight strategy

As soon as the second prototype *LEOalpha* was ready, and while the construction of the final model *LEObeta* was carried on, an intensive flight campaign was undertaken. At the moment (end of April) the campaign is still in an early stage and will be continued during the next months. This section details the procedures employed to acquire data to optimize flight strategy.

Prior to commencing test flights exploring diverse climatic conditions, the pilot undergoes a familiarization phase. This involves a series of circuits around the airfield, allowing the pilot to become accustomed to the aircraft's handling characteristics and refine safe landing procedures.

Each flight test follows a standardized procedure and is performed for different flight conditions such as wind and density altitude. Operations can be resumed as follows:

1. The test aircraft is loaded with different payload configurations to simulate various operational scenarios. For each payload configuration the take off distance is measured also regulating the maximum current absorbed (controlled by the regulator described in Section 4.3) and also considering the bonus that arises for liftoff before 40m.
2. The initial 30 seconds of the flight focus on achieving maximum altitude (limited to 100 meters) while again employing different current settings. This maneuver aims to store potential energy through altitude gain, which can be utilized later to reduce motor power consumption.
3. The following 90 seconds target finding the optimal balance between airspeed and power consumption. During this phase, constant thrust percentages are applied to analyze the aircraft's behavior under different power settings.
4. The final 90 seconds of the flight test prioritize maximizing flight distance. The throttle is set to its limit (i.e. not exceeding 30A), and the aircraft is allowed to fly as far as possible.

Following the completion of all test flights, the collected data will be analyzed to identify performance trends under various conditions (payload, density altitude, wind speed/direction). This analysis will culminate in the creation of a comprehensive strategy matrix.

This matrix will serve as a reference for the pilot, providing the optimal flight strategy (current settings, climb profiles, etc.) for each combination of density altitude and wind conditions encountered during actual operation. This ensures that the aircraft performs at its peak performance in any scenario.

6 Payload prediction

The payload prediction chart is based on test flight data and propeller traction data.

During the flight tests, the average airspeed was measured using the GPS telemetry also taking into account the possible effects of wind, as explained in section 4.2. This data, together with knowledge of the air density on those days and the take-off weight of the aircraft, allows to experimentally determine the actual lift coefficient, which is held constant for the purpose of estimating the payload.

The propeller performance data are scaled by the actual value of the air density, so that a new airspeed can be determined for each air density condition from the dynamic equilibrium (traction propeller = drag). The regulation requires the payload prediction to be calculated as a function of air density using the following linear approximation:

$$P = 2.272\rho + 2.266$$

plotted in Figure 25 where ρ is the density altitude and P is the predicted payload.

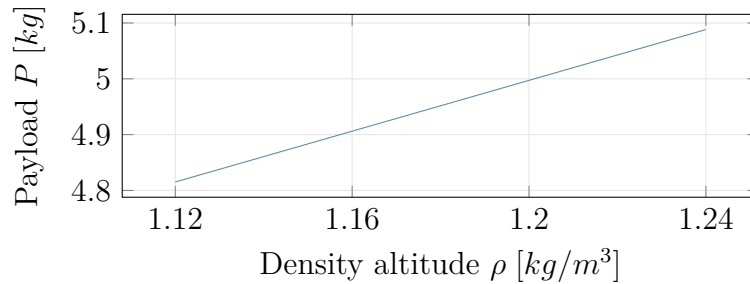


Figure 25: Plot of the payload prediction as a function of air density

The prediction was also verified in occurrence of flight tests with different density altitudes.

Outlook

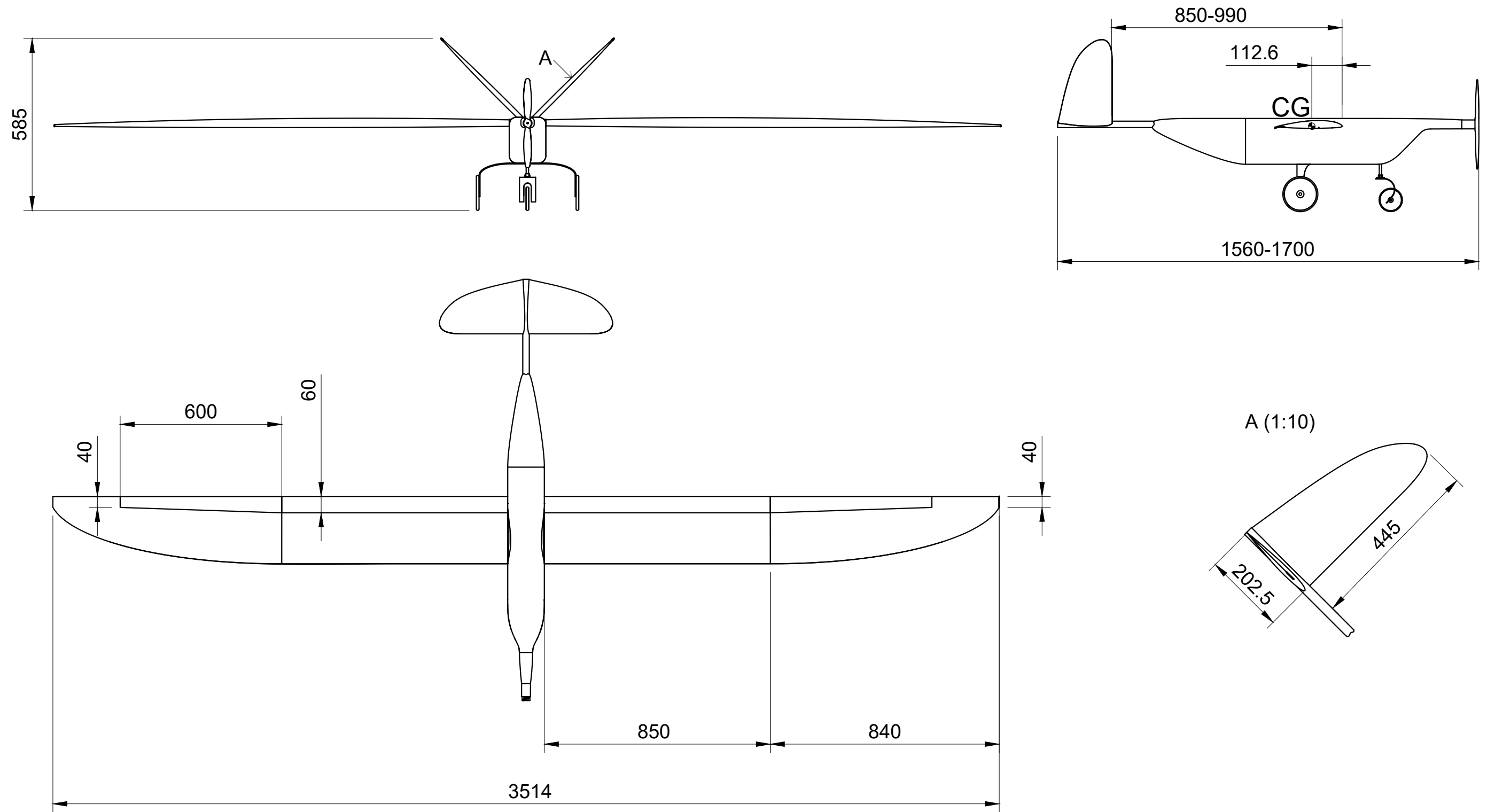
With the present report, the team aims providing to the jury a thorough overview of the challenging path that was undertaken to get ready for the race. For this reason, and within the limits of the available pages, only the most important and qualifying aspects have been reported.

During the design and construction phase the team had to face many difficulties, mostly related to the first approach of almost the whole team to the aeronautical world. In fact, although there are no specific courses offered at the University of Udine, the passion of each team member is deeply rooted in the aeronautical history of our territory. It's been a great challenge for all of us, but at the same time we are very grateful for the opportunity to work on our passions.

Taking into account the above considerations, it became clear that the design approach had to be simplified as much as possible in order to make the project feasible on time and with the available resources. Despite these difficulties, the team believes that the competition aircraft can achieve good performances, thanks to the intense optimization process based mainly on the analysis of the data collected during the flight test campaigns.

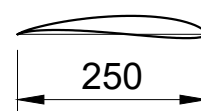
Lot of work still has to be done. The competition aircraft is currently in the final stages of completion. In the remaining months before ACC 2024, the race strategy will be defined and various tests to speed up the loading and unloading process will be made while further optimizing the aircraft improving aerodynamic efficiency and reducing weight.

Drawings

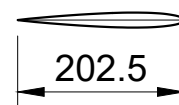


Airfoil - SCALE 1:10

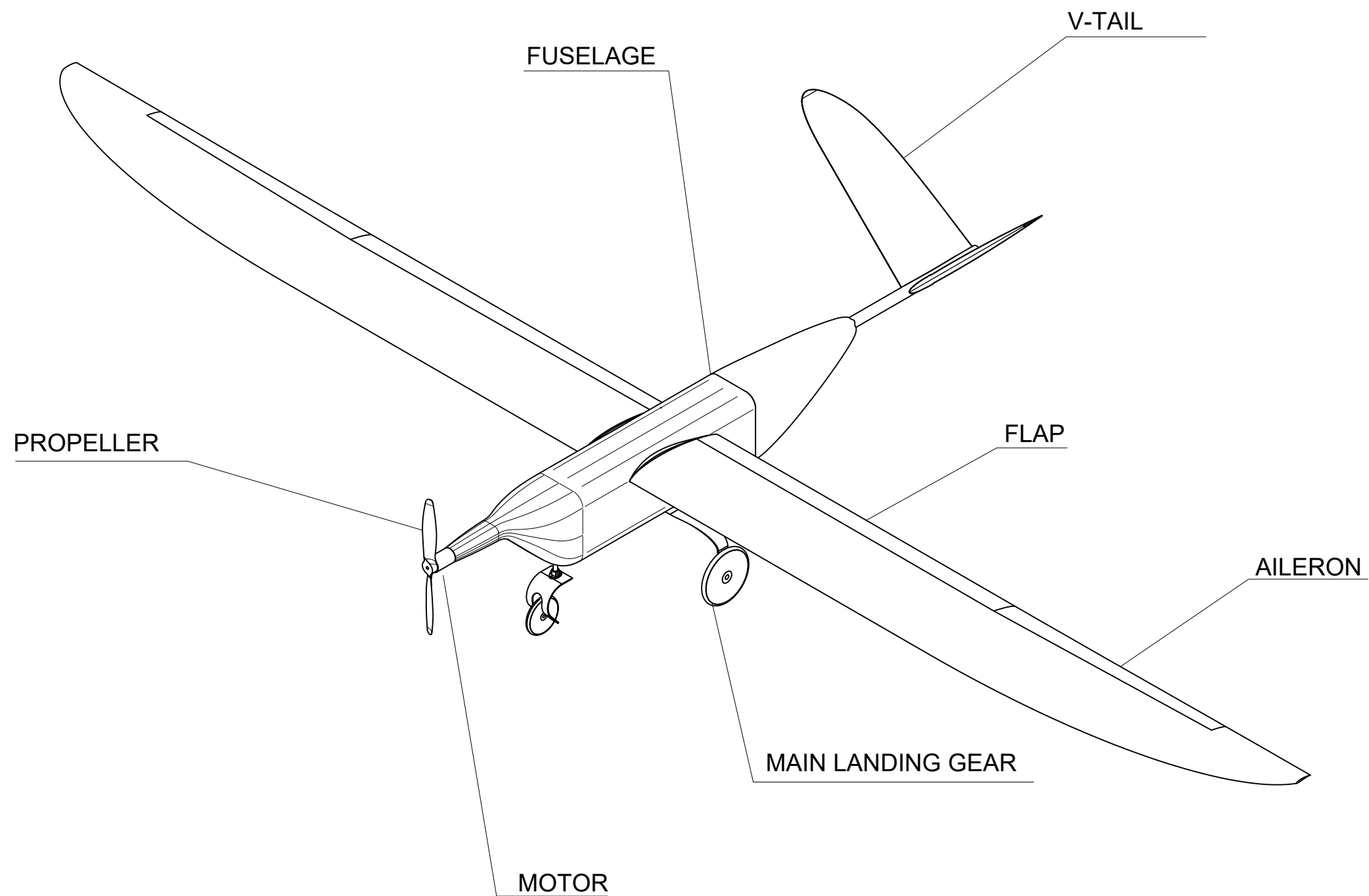
Wing airfoil at the root
(E216 (10.4%))



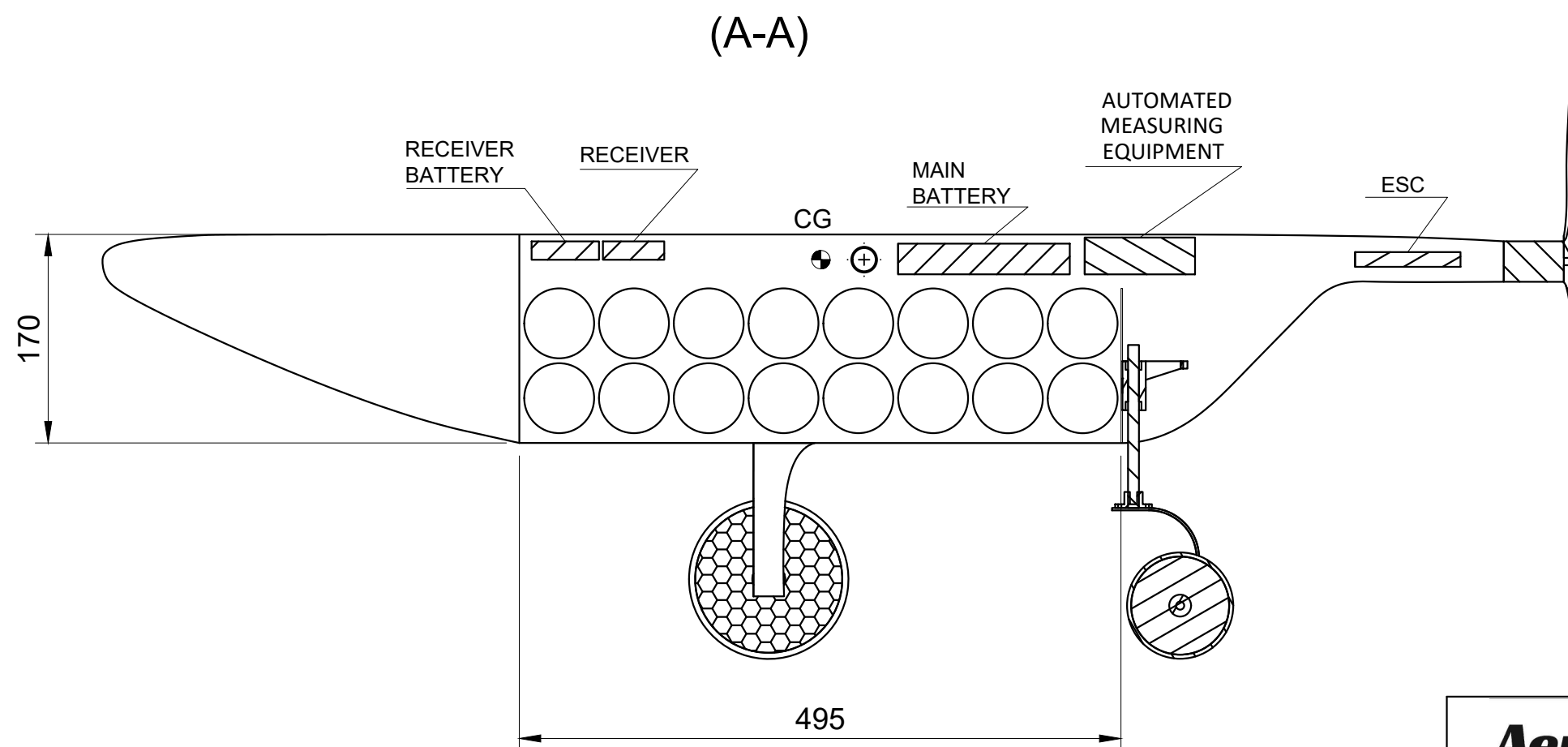
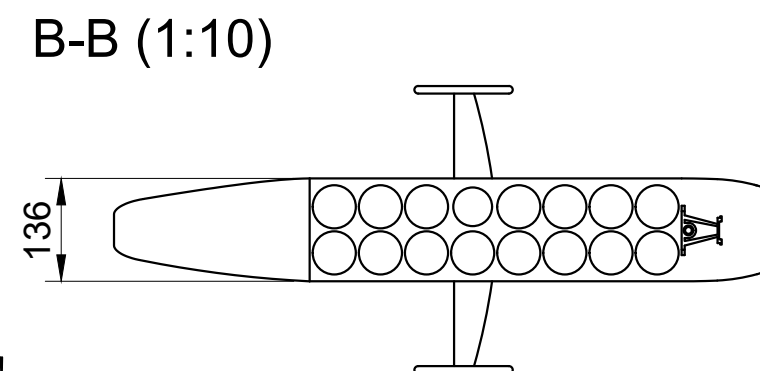
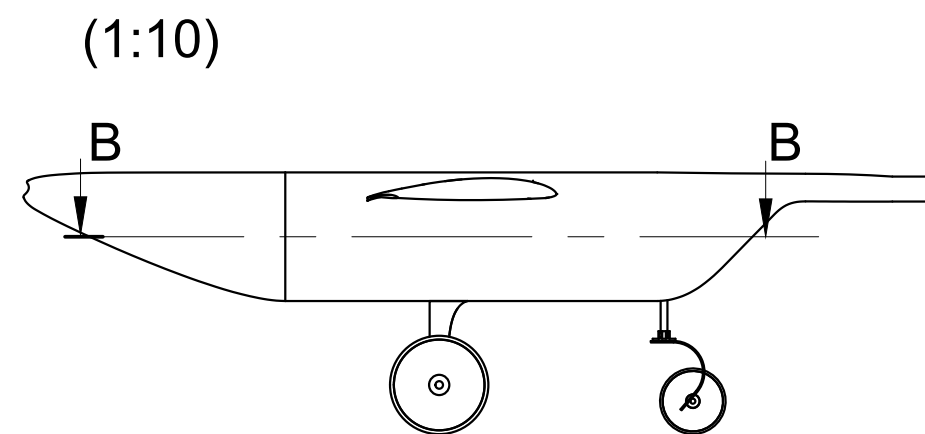
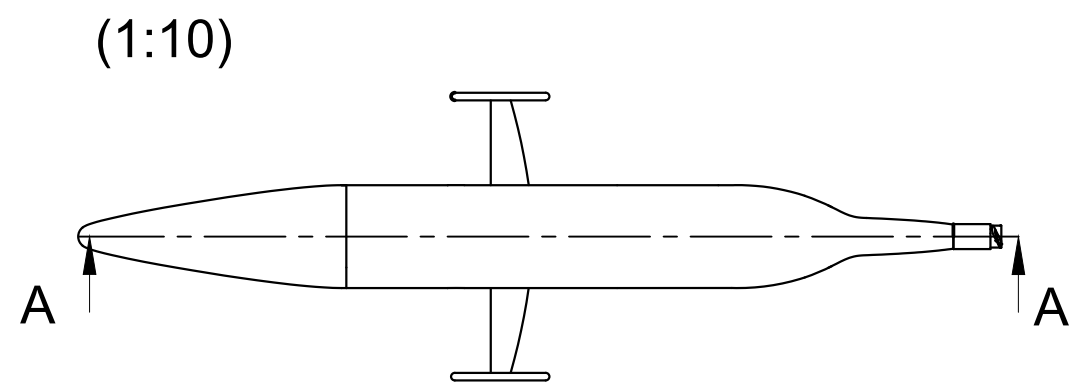
V-tail airfoil
(NACA0009)



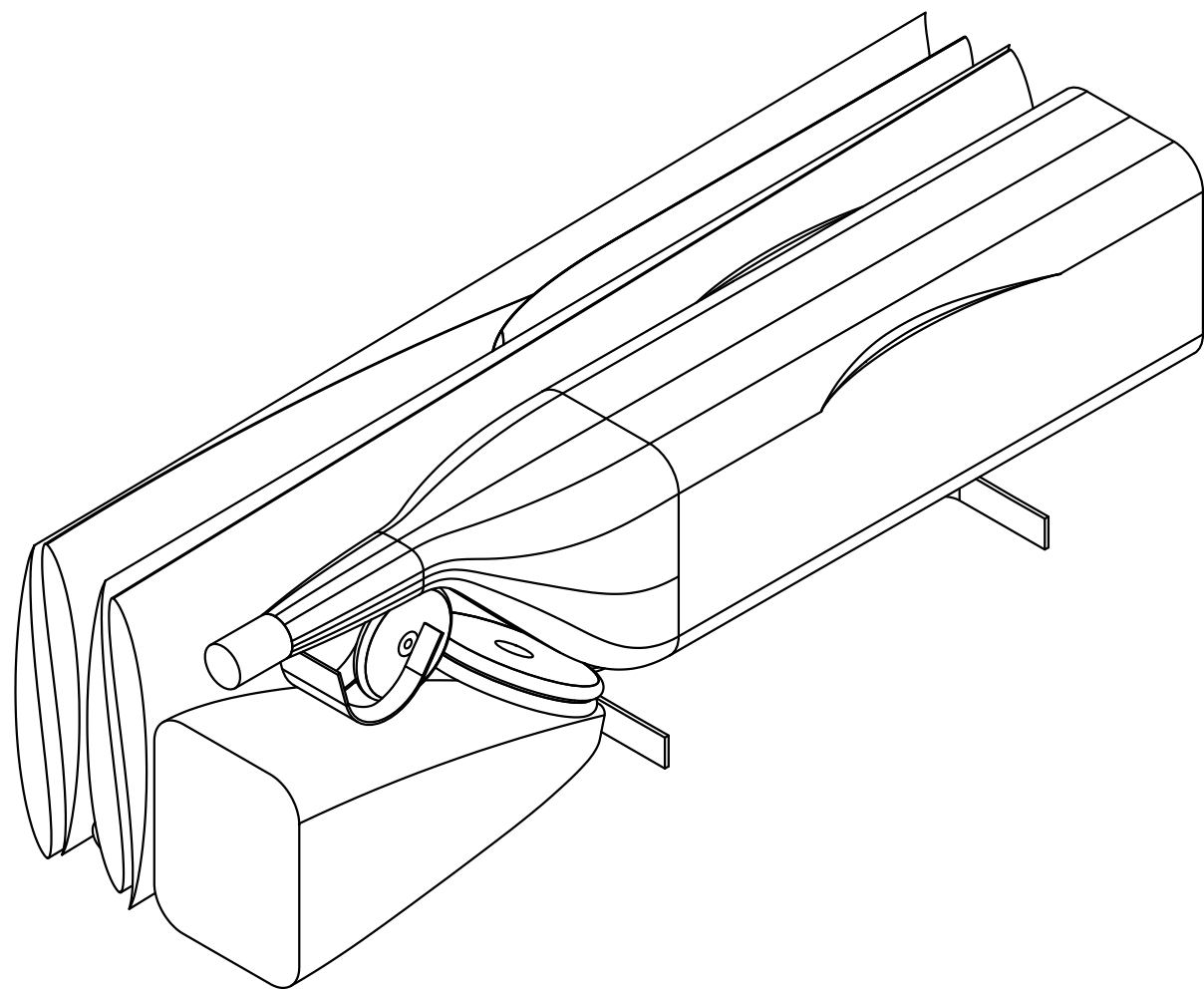
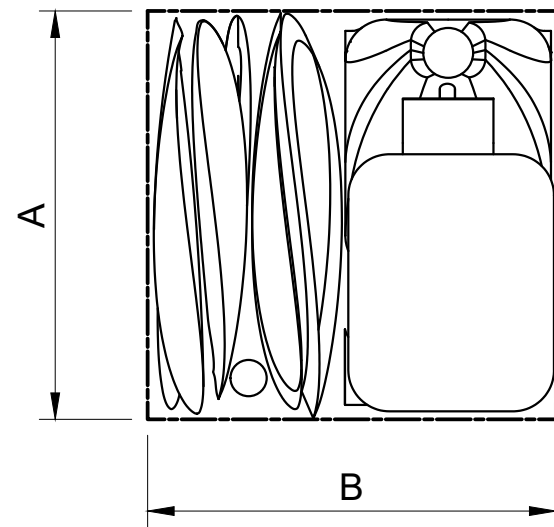
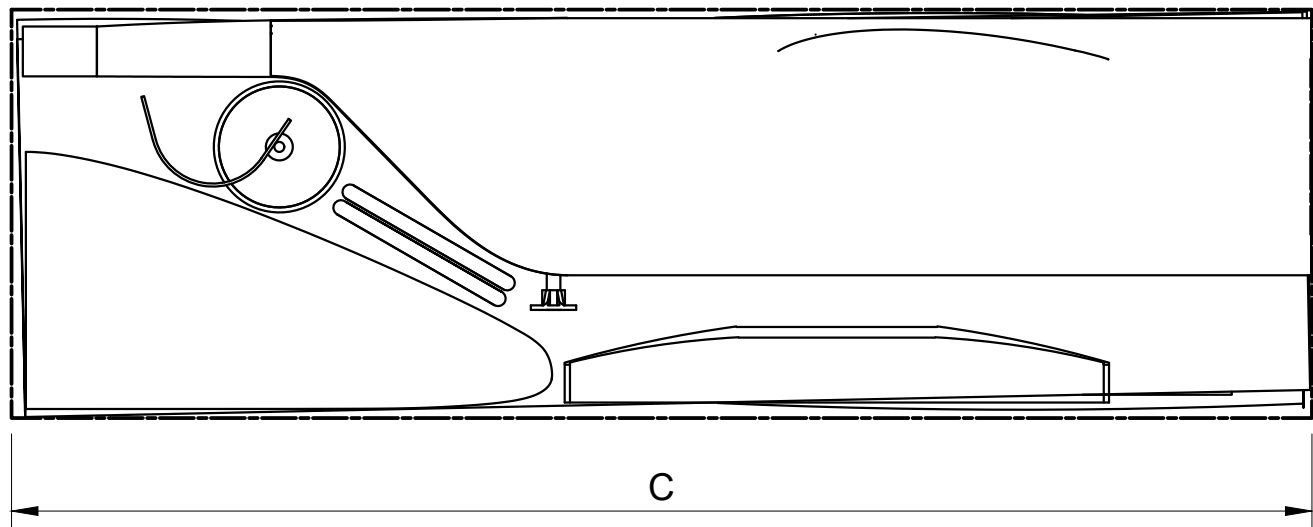
Surface	Area	Surface	Area
Wing	0.757 m²	Flaps	0.051 m²
Full-flying tail	0.136 m²	Ailerons	0.03 m²
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SCALE 1:15	DATE 15/04/2024	DESCRIPTION - front view at the top left corner - top view below - starboard view on the right	



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FORMAT A3		TEAM'S NAME - NUMBER AeroUD - 14	GENERAL TOLERANCES UNI EN-ISO 22768-1-v
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SCALE 1:5	DATE 15/04/2024	DESCRIPTION -Cargo bay location and principal dimensions -Location of flight systems and measuring equipment	



$$A+B+C=1400 \text{ mm}$$

AeroUD		TITLE TRANSPORTATION BOX	
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