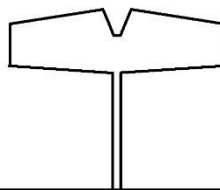
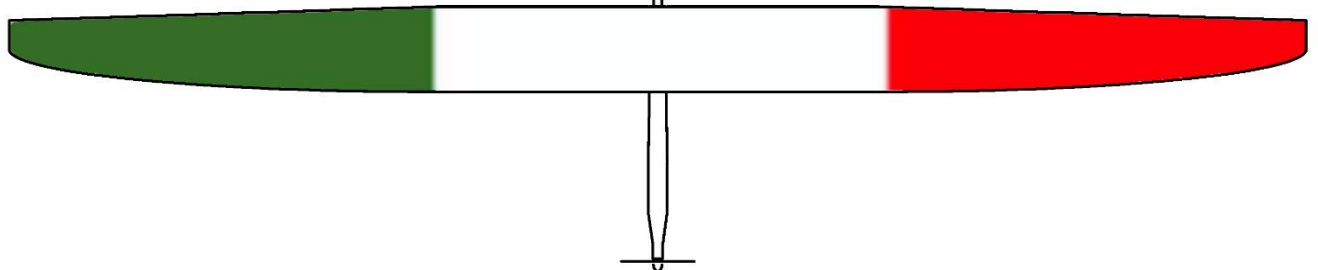


AIR CARGO CHALLENGE 2019 STUTT GART



REPORT TEAM 02



AerouD ... *AND THE BEAST*



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List of symbols

a	Acceleration [m/s ²]	L_{12}	Length of the link between hinge and servo motor in the tail [m]
a_c	Adherence coefficient [-]	L_{23}	Length of the link between servo motor and hinge in tail boom [m]
b	Wingspan [m]	MS	Static margin [-]
C	Battery capacity [mAh]	M_f	Bending moment [Nm]
C_{brake}	Brake torque [Nm]	m_{plane}	Mass of the plane [kg]
C_{aero}	Pitching moment acting on the tail [Nm]	mac	Aerodynamic medium chord [m]
C_{servo}	Torque required to the servo [Nm]	P	Payload [kg]
c_c	Control surface average chord [m]	r_{wheel}	Radius of the wheel [m]
C_{d0}	Fuselage, tail and landing gear drag coefficient [-]	r_{pulley}	Radius of the brake's pulley [m]
C_d	Drag coefficient [-]	S	Wing area [m ²]
$C_{d,tot}$	Airplane total drag coefficient [-]	T	Propeller traction [N]
$C_{d,3D}^{xflr5}$	Calculated drag coefficient [-]	V	Battery voltage [V]
C_m	Pitching moment coeff. [-]	v	Airplane speed [m/s]
C_l	Lift coefficient [-]	v_i	Airplane speed before landing [m/s]
$C_{l,mac}$	Lift coefficient at the aerodynamic medium chord [-]	X_{np}	Distance of the neutral point from the leading edge [m]
D	Drag [N]	X_{cg}	Distance of the centre of gravity from the leading edge [m]
EI	Spar's bending stiffness [Nm ²]	y	Semi-wingspan distance from the wing root [m]
F	Force [N]	α	Angle of attack [rad]
f	Deflection [m]	γ	Maximum control surface deflection [rad]
I	Current [A]	β	Maximum servo deflection [rad]
L	Lift [N]	ρ	Air Density [kg/m ³]
L_c	Control surface average length [m]	μ	Kinetic friction coefficient [-]
L_s	Distance amid supports [m]	ϑ	Belt wrapping angle [rad]
L_{servo}	Servo's arm length [m]	Δt	Flight time [s]
L_{stop}	Distance required to stop the aircraft [m]		



INTRODUCTION

The present document describes the main steps that were taken by the team “AeroUD...and the beast” to design, build and develop the model plane “Titti” to participate at the next Air Cargo Challenge (ACC) 2019 edition.

The AeroUD-MET has quite a long history, since it started its first project in 2011. Since then, the team has taken part in three competitions, always ranking 4th. This time, the University of Udine will participate with two different teams, the youngest of which is the one in charge of this document.

The team was founded in September 2018 with a solid knowledge about composite materials and simplified aerodynamic design, grounded on the previous team’s experience, whose members were available for an effective know-how transfer. The ambitious challenge to build a completely new model starting from a white sheet was therefore taken. The main efforts were focused on:

- finding a new wing profile that would perform better in the flight mission required by the regulations than the S1223 used by the previous AeroUD models.
- identify an optimal design for the wing planform;
- develop a new V-tail, full flying to improve model manoeuvrability;
- improve wing spar design with the aim to reduce weight.

This report describes in detail the main issues of above and will give few but however sufficient information about all the other aspects involved in the project, in order to provide to the reader an overall view of the entire process.

CHAPTER 1: PROJECT MANAGEMENT

This section briefly describes the management of the team in terms of time and activity planning and will also provide an overview of the overall budget.

1.1 Time schedule

The project GANNT is reported in *Fig. 1.1*, and the main phases are described in the following.

The team was recruited at the beginning of September 2018 with the aim of building an aircraft with a new profile, different from the S1223 used by the AeroUD team in the previous ACC editions. The simplified model used for the aerodynamic design needed some experimental validation. Therefore, it was necessary to build an early and simplified prototype, named IKEA, in the fastest and cheapest way. Simultaneously, the design and experimental validation of new and lighter wing spars was taking place. To achieve these two tasks, the team was divided in two groups: aerodynamics and structural design. This division was maintained until the final aircraft design was defined. Anyway, construction was performed by all team members.

After the regulations were published, the team started designing the new aircraft (Titti) using the analytical models developed and the knowledge gained in the preliminary part of the project. The prototype took off for the first time on March the 21st. By the end of April, the design details were defined. In the remaining time before the competition, we will build the final aircraft (reusing some parts from the prototype) and refine building techniques in order to minimize weight; simultaneously we will use the prototype to improve piloting skills.

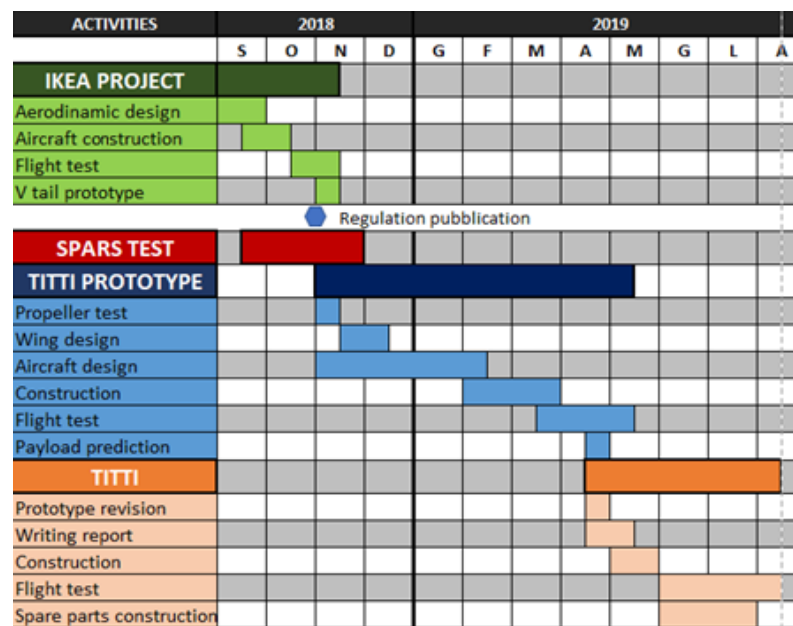


Figure 1.1: Gantt chart of the project.

1.2 Financial budget

Although the budget made available by the University was significant (15000 euros), a careful planning of the expenses was mandatory, also in view of the extensive use of expensive construction materials (i.e. high modulus carbon fibers) and high quality electronics (servos, ESC, LiPo batteries) that the team intended to adopt. A first assessment of the expenses was based on the financial budget of the previous ACC2017. We divided it into five categories, which follow in temporary order, except for the last voice:

- Development of IKEA (first prototype)
- Development of first Titti prototype
- Development of the competition aircrafts (two identical models)
- Travel costs
- External services

For the ‘IKEA development’ we purchased the most important electronic components like servos, receivers and motors, all elements that we would also use in the final aircraft. Despite that, in the following aircrafts the most expensive item is related to materials (carbon/glass fibre, epoxy resin, PVC foam sheets). However, the second competition aircraft will be used as spare parts, so it will not be equipped with electronic components. Other expensive items are the carbon tubes (used for the tail boom, V-tail spars, landing gear, wings junctions) and the manufacturing of wings moulds.

Due to the reasonable distance between Udine and Stuttgart, we intend to join the competition by car, so we estimate the travel expenses to be around 2000 euros for nine people.

At the present time, the budget has been respected, and leaves more space for the coverage of unexpected events, as it is possible to observe in *Tab. 1.1*.

Development of IKEA	Construction materials	€ 200
	Electronics	€ 1.000
	Subtotal	€ 1.200
Realization of prototype	Construction materials	€ 1.000
	Electronics	€ 300
	Subtotal	€ 1.300
Competition aircraft	Construction materials	€ 2.000
	Electronics	€ 300
	Subtotal	€ 2.300
Travelling costs	Competition fee	€ 2.220
	Transfer	€ 2.000
	Subtotal	€ 4.220
External services	Carbon tubes	€ 1.500
	Manufacturing of MDF moulds	€ 1.600
	Subtotal	€ 3.100
TOTAL		€ 12.120

Table 1.1: Overview of the project economical budget.

CHAPTER 2: AERODYNAMIC DESIGN

The team decided at first to maintain a classical high wing airplane configuration with a conventional tricycle landing gear. By the experience of other teams and based on the previous competitions, this configuration has proven to be the most reliable. Successively, we engaged in the choice of the wing profile: that was made consulting the available literature, focusing our attention on designs suited for a low Reynolds number application, as in the present case. The chosen profile is the Wortmann Fx 63-137 (Fig. 2.1) because of its remarkable lift performance associated with acceptable drag (Fig.2.2) which should allow us to reach the desired speed with the available propeller traction.

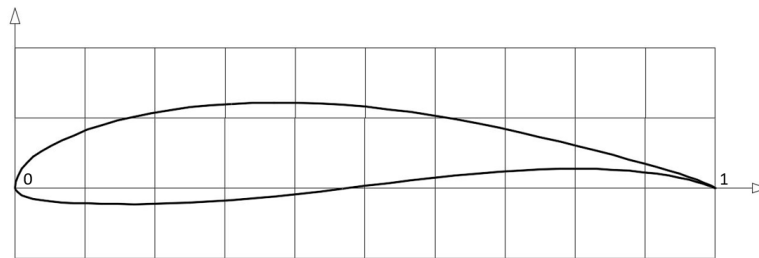


Figure 2.1: The chosen profile Wfx 63-137.

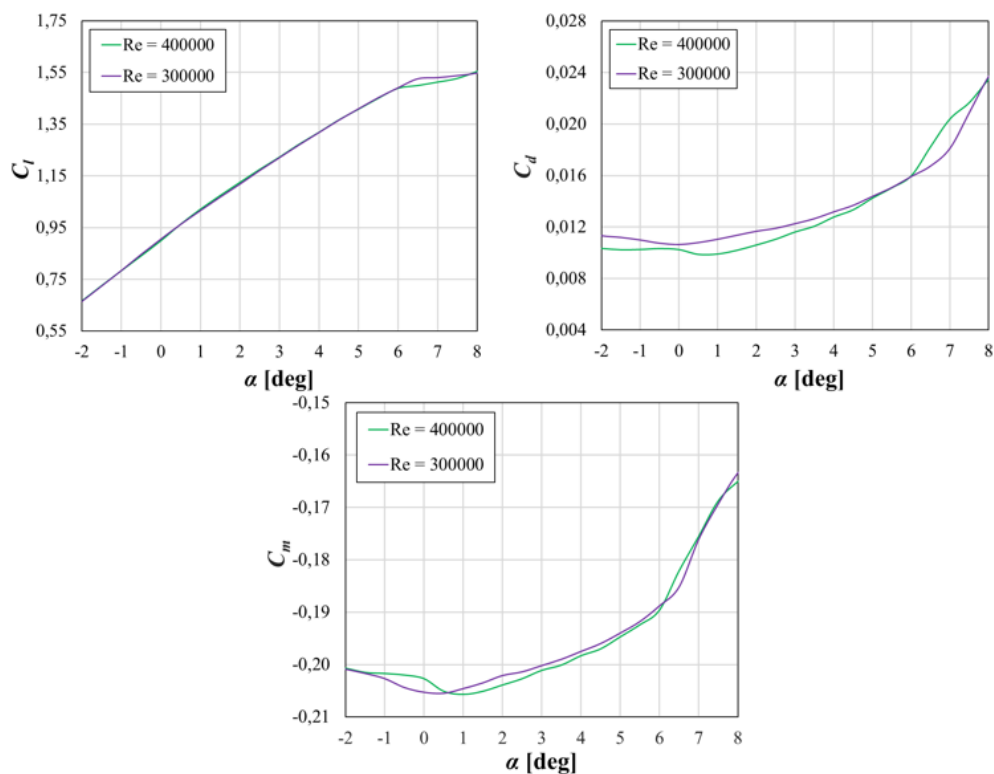


Figure 2.2: Lift (top left), drag (top right) and pitching moment coefficients for the FX63-137 airfoil, 2D case.

In addition, the selected profile has a low momentum coefficient, which limits the request for large tail surfaces or long tail booms, thus reducing the overall drag and weight of the airplane. Flight tests made with the IKEA prototype confirmed the potentiality of the profile and validated the predictions made by our developed simplified aerodynamic model. This encouraged the team to proceed with an optimization process in order to choose the wing planform design that should achieve the highest competition score possible.

The following paragraphs describe the tools, the adopted procedure and the assumptions made to reach a satisfying aerodynamic design.

2.1: Design criteria and methodology

The design goal was to obtain a wing planform configuration that maximizes the score associated with the requested flight mission. According to the ACC2019 rules, the formula used to generate the score is:

$$Score = \left(\frac{flown\ weight}{flown\ time} 2000 + a + b + c \right) d + Timebonus \quad [2.1]$$

Where the terms a, b, c, d , and $Timebonus$ do not depend by the performances that the plane has in flying the 100 m legs. Consequently, we can assume that the score varies mainly according to two parameters: the payload at take-off and the average speed of the aircraft. Therefore, the score function variation can be approximated by the product of lift and speed in horizontal flight:

$$Score \propto L v \quad [2.2]$$

Thus, maximizing this function means maximizing the actual score.

Wing performances in terms of 3D polars, necessary to estimate the score as explained above, have been computed by means of XFLR5, an open-source software developed by Mark Drela that uses a numerical panel method. This method is able to generate two-dimensional and three-dimensional polars with excellent approximation for Reynolds numbers higher than $3 \cdot 10^5$.

The hypothesis on which this simplified analysis is based is that in-flight speed can be estimated by balancing the propeller traction with the overall drag of the airplane:

$$C_{d,tot} \frac{1}{2} \rho v^2 S = D = T \rightarrow v \quad [2.3]$$

To do this, the propeller traction variation with the flight speed must be known, and it has indeed been object of a dedicated experimental campaign as described in paragraph (4.3).

The airplane's total drag coefficient, $C_{d,tot}$, is assumed to be the sum of the wing 3D drag coefficient computed by the software and an additional term associated with all the aerodynamic losses of the aircraft except the wing (i.e. the drag of the tail surfaces, fuselage, landing gears, etc.)

$$C_{d,tot} = C_{d,3D}^{xflr5} + C_{d0} \quad [2.4]$$

The latter has been experimentally estimated as will be clarified later in the discussion. The design procedure followed the subsequent steps:

- 1) Wing planform sizing: a first analysis is performed on a rectangular planform wing, varying in wingspan and chord at a fixed angle of attack close to the one for the airfoil's maximum efficiency, i.e. 0 deg.
This first step allowed us to define the range of wing dimensions inside of which to look for the optimal design and to identify a first geometry.
For a given wing geometry, a first attempt flight velocity is defined and 3D polars are computed by XFLR5. With this data, the eq. [2.3] returns a velocity that is used for a second run of simulations: the iterative procedure is then repeated until a convergence and a simplified score (eq. [2.2]) is computed.
- 2) Wing planform optimization: the wing planform is optimized in terms of wing load following the Schrenk criterion.
- 3) Angle of attack: the last step concerns the definition of the angle of attack at which the optimized wing output of step 2 must fly.

As previously mentioned, we performed an experimental activity to estimate the value of C_{d0} . We did this by recording the flight velocity at different take-off weights with a GPS system installed in the IKEA prototype. The recorded data during an extensive flight test campaign, allowed us to define an estimate of the C_{d0} value by comparing predicted and lifted payloads. This kind of approach is obviously much simplified and can only provide a rough estimation of the drag coefficient. However, the team decided to proceed in this way rather than, as for example, go for a fully 3D viscous simulation of the entire aircraft, which would've been time consuming and need validation.

2.2: Wing planform sizing

At first, it was necessary to identify for which range of chord and span it made sense to investigate. Concerning the wingspan, the main limitation was given by the dimensions of the transportation box. Since the maximum size of the box is 750 mm, the wing is composed by sectors 740 mm long. From a two-dimensional CAD design, we concluded that the maximum number of sectors that the box can contain is 7. This implies a maximum limit on the wingspan, which will be 5180 mm. Obviously, the structural constraints regarding wing resistance has been also taken into account. We questioned the model for a wingspan ranging from a minimum of 1480 mm to a maximum of 5180 mm, and for a chord value in the range 250÷400 mm, i.e. width and depth of the box.

In *Tab. 2.1*, the values of the computed score for the investigated wing (rectangular planform) are reported normalized on the maximum value. In *Tab 2.2* reports the same data but normalized for the maximum score obtained for each different value of the wing span.

Chord [mm]	Wingspan [mm]					
	1480	2220	2960	3700	4440	5180
250	0,344	0,509	0,657	0,780	0,893	0,995
275	0,354	0,518	0,664	0,788	0,897	0,999
300	0,359	0,524	0,670	0,791	0,900	1,000
325	0,364	0,527	0,673	0,789	0,897	0,993
350	0,367	0,527	0,671	0,784	0,890	0,981
375	0,369	0,527	0,668	0,777	0,876	0,970
400	0,369	0,524	0,662	0,769	0,867	0,956

Table 2.1 Score distribution at variable wing span and chord for a rectangular wing planform, data normalized with respect to the maximum.

Chord [mm]	Wingspan [mm]					
	1480	2220	2960	3700	4440	5180
250	0,933	0,966	0,977	0,987	0,992	0,995
275	0,960	0,983	0,987	0,997	0,997	0,999
300	0,972	0,994	0,996	1,000	1,000	1,000
325	0,985	0,999	1,000	0,998	0,997	0,993
350	0,995	1,000	0,998	0,992	0,989	0,981
375	1,000	0,999	0,994	0,983	0,974	0,970
400	0,999	0,993	0,984	0,973	0,963	0,956

Table 2.2: Distribution of the score normalized by maximum value obtained for each wingspan.

From these matrices, it is possible to observe that:

- the maximum score can be obtained with the largest wingspan (see Tab 2.1), this is obviously associated to aspect ratio effects;
- for each wingspan, a chord value that maximizes the score can be identify (see Tab 2.2).

With this in mind, and with the constraints posed by the structural design, the team decided to go for a wingspan of 4440 mm. The analytical model was used again to refine the selection of the best chord, which turned out to be 295 mm.

2.3: Wing planform optimization

The design goals of this phase were two:

- limit the induced drag effects localized at the wing tips;
- obtain an elliptical load distribution, according to the Schrenk criterion.

The wing loading is defined as:

$$Loading(y) = Chord(y) \frac{C_l(y)}{mac} \quad [2.5]$$

According to the Schrenk criterion, the wing loading can be described by the following equation:

$$Elliptic\ loading(y) = \frac{C_l(y)}{mac} \frac{4S}{\pi b} \sqrt{1 - \left(\frac{2y}{b}\right)^2} \simeq \frac{C_{l,mac}}{mac} \frac{4S}{\pi b} \sqrt{1 - \left(\frac{2y}{b}\right)^2} \quad [2.6]$$

In the calculation of the elliptic loading, $C_l(y)$ can be approximated by $C_{l,mac}$ in view of the small variation of the profile polar with the Reynolds number, see *Fig. 2.2*.

To obtain a load distribution as much as possible according to the Schrenk criterion, an elliptical taper for the wing planform was adopted. The leading edge moves backwards according to the equation of an ellipse, while the trailing edge is kept as a straight line, to facilitate the installation of both ailerons and flaps.

Multiple attempts were made, and the wing loading distribution computed with XFLR5 was compared with the theoretical one until a satisfactory geometry was identified. The resulting geometry is sketched in *Fig. 2.3* while the resulting wing loading is shown in *Fig. 2.4*.

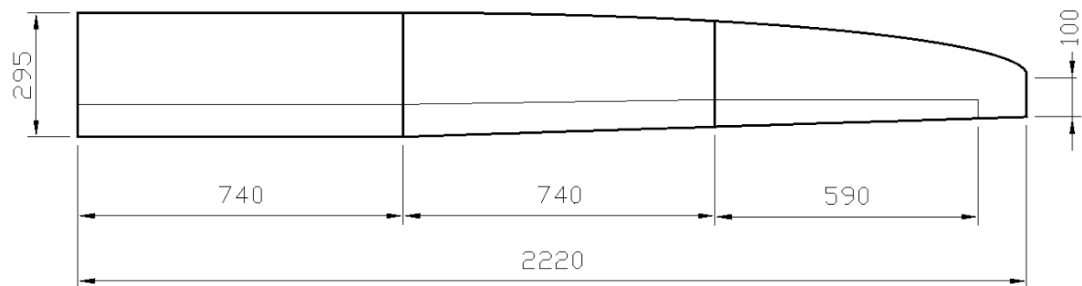


Figure 2.3: Shape and dimensions of the wing planform in [mm].

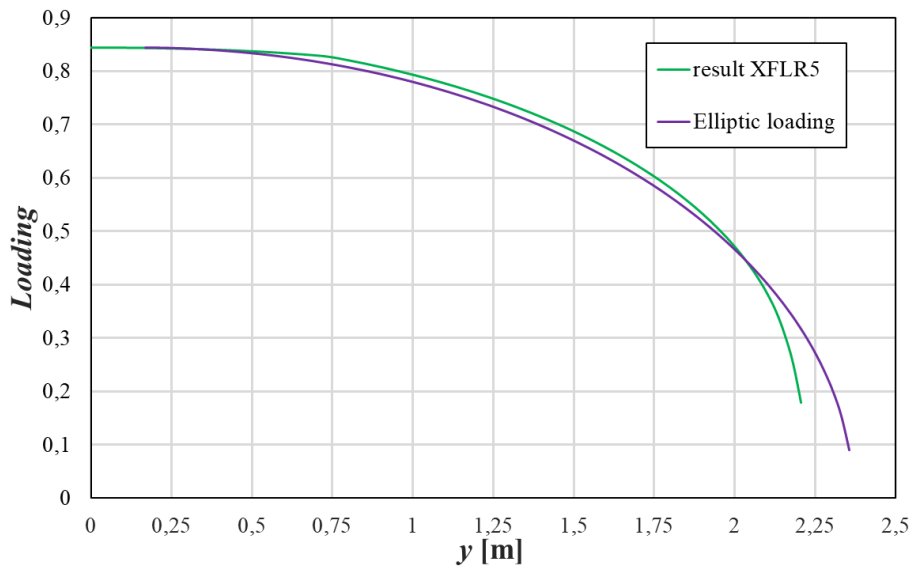


Figure 2.4 Comparison of wing loading distributions.

2.4: Angle of attack

To conclude the sizing of the wing we searched for the wing angle of attack value that maximizes the score. To do this, we used XFLR5 software to compute the 3D polar of the final wing geometry. From these values, the score was determined for each angle of attack using the standard formulas for lift and drag and the usual balance between drag and propeller traction to define flight velocity. The results are plotted in Fig. 2.5, where it's possible to observe that the maximum score is achievable for an angle of attack close to 0 deg. We consequently chose this value for the design.

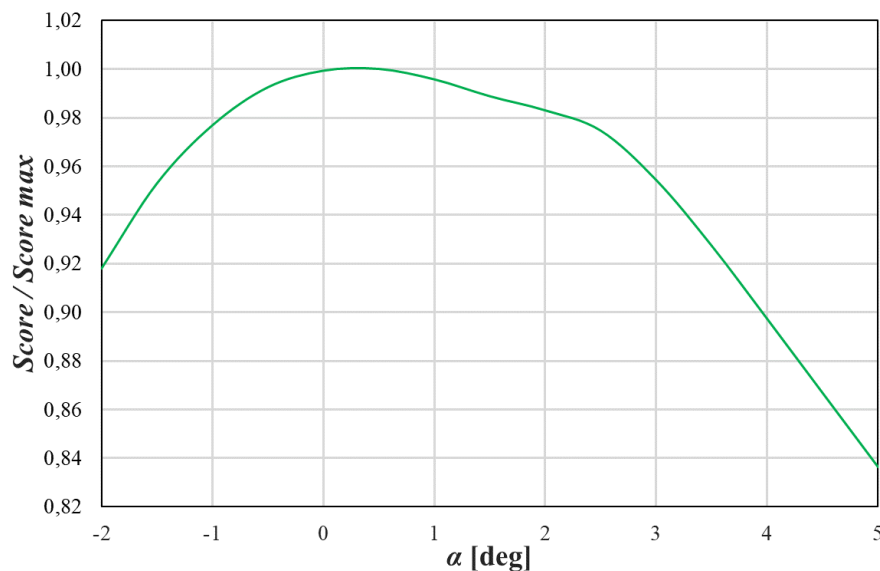


Figure 2.5: Distribution of the dimensionless score with the wing angle of attack.

2.5: Flaps design

The take-off run of the aircraft can be roughly described by the following momentum equation:

$$m_{plane}a = T - D \quad [2.7]$$

where m_{plane} is the mass of the aircraft, T the propeller traction, D the Drag of the aircraft, a the acceleration; the friction of the wheels is discarded for simplification. This equation allows us to compute the velocity that the aircraft can reach in the available 60m take-off run. It turned out that the take-off velocity could not be reached and therefore the adoption of flaps was mandatory.

We decided to build flaps at 25% of the chord and with maximum excursion of 10 deg. We ran simulations by means of XFLR5 to determine how much flap surface would be necessary to guarantee the desired lift. We chose to apply the flaps along the entire length of the two central sectors of the wing.

This choice increases the available surface and consequently it increases the lift value more than necessary but, at the same time, it allows to obtain the desired lift by decreasing the flap travel angle, making it more efficient.

2.6: Final considerations

The aerodynamic design methodology of above is obviously quite simplified and does not take into account a number of variables and their interaction and effects on the final performances. The team was therefore conscious that the developed wing design couldn't have been the best possible. Considering also that the wingspan selected is quite demanding in terms of structural resistance, we decided to investigate experimentally another wing configuration, identical to the one before but with a shorter wingspan. This was possible and easy to achieve thanks to the fact that the wing planform has a rectangular shape in the centre, that can be conveniently replaced by a longer or shorter section, while keeping unmodified the tapered ones, see *Fig.2.6*. A prototype capable of flying with the two different wingspans was built and the two configurations were compared in both race simulations and wing static structural tests. From the static loading tests, we were able to understand, besides the real resistance, also the real deformation, dangerous for any flutter instability. From the race simulations, we obtained estimates for the score achieved by the two possible configurations.

It was therefore decided to build the final aircraft with wing span of 4440 mm.

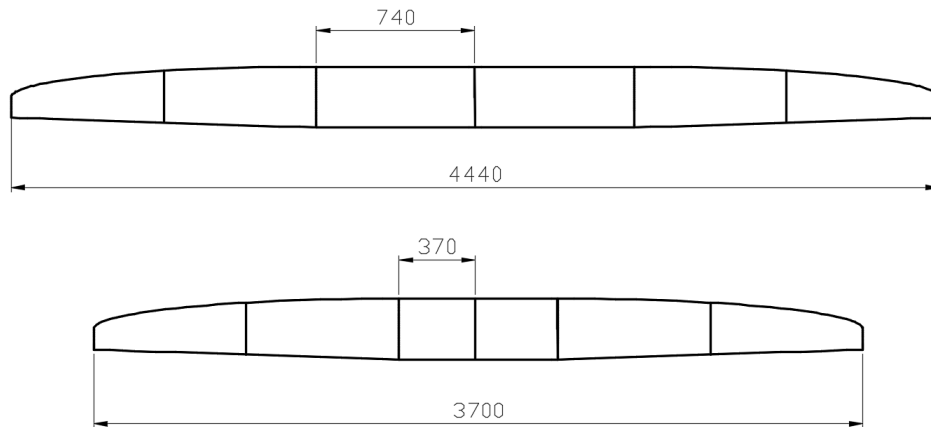


Figure 2.6: Representation of the two possible wing plans configurations with their relative measures in [mm].

2.7: Empennage design

To correctly size the tail of the plane, XFLR5 program was used once again, since it can also develop static stability simulations. The profile chosen from literature is the symmetrical NACA 0009 profile, it's characterized by a very low drag coefficient, maintaining a significant thickness per quarter of chord. This feature has been fundamental in terms of construction, in order to have enough space for both servos and spars installation. The tail stagger angle was set at zero degrees, to limit its contribution to drag as much as possible.

The reason behind the adoption of a V-shaped full flying tail is twofold:

- have an excellent manoeuvrability and efficiency, not only in the controls but also in possible stability control trimmers, during the use of the flaps;
- reduce the overall drag of the tail.

The geometry chosen for the empennages was found starting from the dimensions of the main wing through correlations from literature (Daniel P. Raymer, Aircraft Design: A Conceptual Approach, 4th edition). This analysis provides a total surface of 0.182 m², a chord of 225 mm and of 145 mm at the root and tip, respectively, and a length of 500 mm. The dihedral angle was chosen at 45 deg.



Figure 2.7: The new V-tail.

The distance between the leading edge of the tail and the wing has been chosen to satisfy a static margin of more than 10%. The static margin is a dimensionless defined as:

$$MS = \frac{X_{np} - X_{cg}}{mac} \quad [2.8]$$

where X_{np} is the neutral point and X_{cg} is the centre of gravity. Their determination was made once again by exploiting the XFLR5 program capabilities on a number of configurations. With reference to the leading edge of the wing at the root, the results obtained are:

$$\begin{cases} X_{np} = 170.7 \text{ mm} \\ X_{cg} = 142 \text{ mm} \end{cases} \quad [2.9]$$

$$mac = 266.14 \text{ mm} \quad [2.10]$$

$$MS = \frac{X_{np} - X_{cg}}{mac} = 10.5\% \quad [2.11]$$

and the distance chosen between the leading edge of the wing and the tail was 1350 mm. In Fig. 2.8 we can see the representation of the aerodynamic bodies of the aircraft.

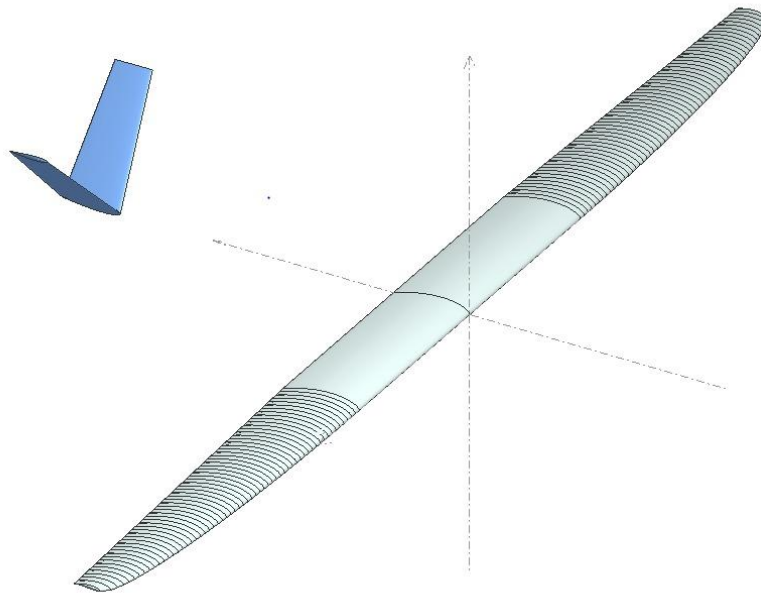


Figure 2.8: Schematic representation of the aerodynamic bodies of the aircraft.

2.8: Fuselage and cargo-bay

The fuselage has to be large enough to host the cargo bay and landing gears. The general criterion that drove the design of the fuselage was to keep as low as possible its aerodynamic drag and the negative interaction with the wing at the junction. To minimize in particular this effect, we decided to connect the fuselage to the wing by means of a pedestal, *Fig. 2.9*. The latter is shaped externally as a NACA 0022 profile, which is thick enough to allow the installation of the internal structure that is necessary to withstand the mechanical loads, mostly associated to the payload inertia.

At the same time, we put particular care in the design of the external surfaces in order to keep them as much as possible aerodynamically smooth. We also considered to install a fairing on the main landing gear, but this was not possible due to the constraints associated with the installation of the mechanical brake (see Paragraph 4.4).



Figure 2.9: Prototype pedestal, fuselage and cargo bay.

CHAPTER 3: STRUCTURAL DESIGN

In this section the structural design will be discussed with particular emphasis on the wing. This indeed was the most challenging mechanical design problem that the team had to face in view of the large selected wing span and consequent mechanical loads, in contrast with the need to keep the structure as light as possible.

3.1: Wing structure

The wing is realized in six sections of 740 mm length. The sections are joined together by means of short carbon tubes inserted in cylindrical pockets, also made with carbon tubes, *Fig. 3.1*. The external surface of the wing is made of a composite sandwich that sees, from the external to the internal surface, a layer of carbonweave 40g/m^2 with fibres oriented at $\pm 45^\circ$ respect to wing aperture direction, a PVC foam sheet of 1.2 mm thickness and a final layer of glass fabric of 25 g/m^2 , again at $\pm 45^\circ$. This sandwich has to withstand the wing's torsional moment and be sufficiently rigid to keep the correct aerodynamic shape under load. The wing's loading must be therefore handled mostly by the internal spars, the design of which is described in the following.

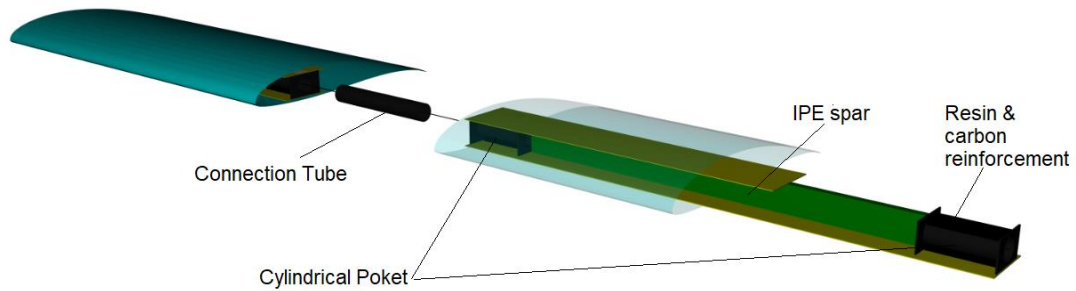


Figure 3.1: Detail of the internal section of the wing.

Given the short time left to complete the aircraft and the complexity of the subject, the team opted for an empirical trial-and-error approach to the spar sizing problem. Several different spar solutions were put under load test, until the optimal was identified. The spars were evaluated under two aspects: maximum bending moment and rigidity. For reasons of brevity, only the final solution will be discussed, a schematic of the structure is shown in Fig. 3.2, it resembles an IPE beam.

The first two layers of glass fabric and PVC foam guarantee a good interface with the wing's internal surface. The carbon tapes vary from 2 to 6 in function of the rigidity (an assumption verified during the test, was that the rigidity of the wings is mainly dependent on the number of carbon tapes). The upper carbon strip is then covered with a PVC foam sheet and then a final layer of glass fabric completes the sandwich. The vertical element is composed by an overlay of glass, PVC and glass; several configurations were tested, but at last the choosen combination has been: glass fabric $49 \text{ g/m}^2 \pm 90^\circ$; PVC 1.2 mm ;glass fabric $49 \text{ g/m}^2 \pm 90^\circ$. The carbon tapes are 25mm wide of weight equal to 125 g/m^2 .

The first problem we encountered was to guarantee the mechanical continuity between the connection tube and the IPE spar: after some tests, the best solution was found by filling the space between them with resin mixed with carbon fibre reinforcement.

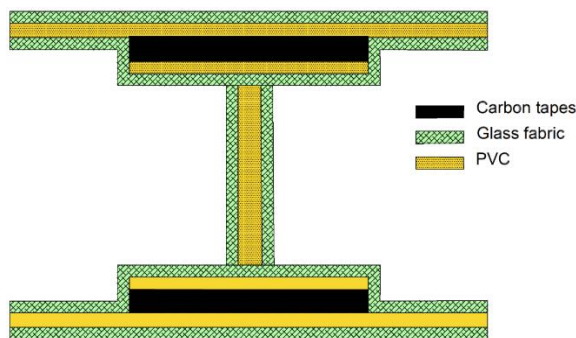


Figure 3.2: Schematic representation of the solution adopted for the wing spars.

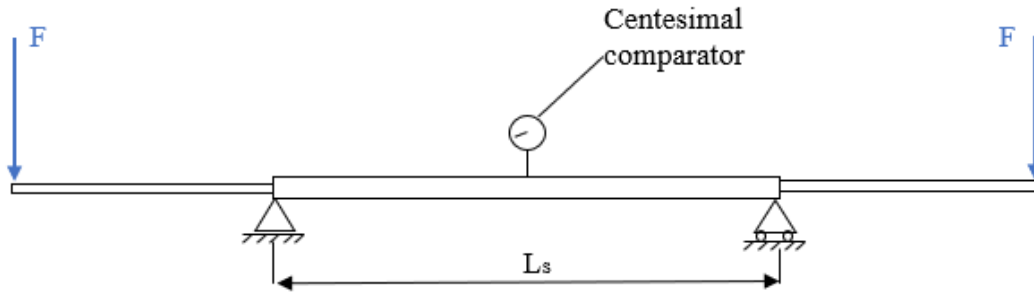


Figure 3.3: Scheme of the set up used for testing the spars.

The mechanical tests consisted in putting the spar on two hinges and enforcing two constant bending moments at its extremities (Fig. 3.3). The resulting deflection f was measured at its centre with a centesimal dial gauge. The moment M_f and deflection values together allowed to compute the rigidity of spar:

$$EI = \frac{L_s^2 M_f}{8 f} \quad [3.1]$$

The design rigidity (i.e. the number of carbon tapes) was chosen in order to be comparable to the one of the aircrafts used in the previous competitions by AeroUD teams. The bending moment at which the mock-up broke is used as a parameter to evaluate the safety coefficient.

Figure 3.4 summarizes the tests' results for the approved solutions (the data are the averaged of multiple tests), while Fig 3.5 shows a picture of one broken mock-up.

As the ultimate moment corresponded to a value 1.5 times greater than the maximum moment of a wing section, the spar was approved for that section.

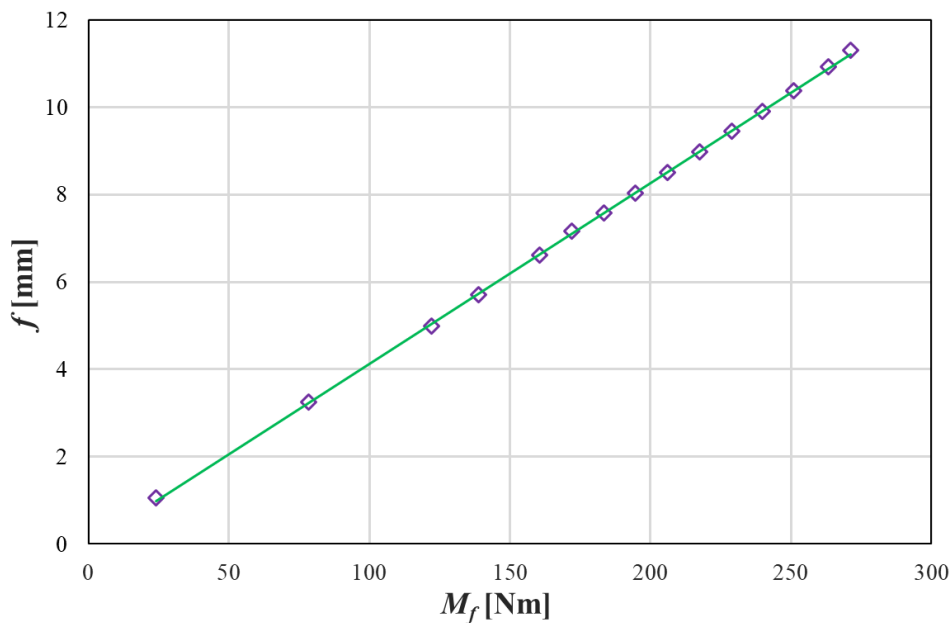


Figure 3.4: Deflection as a function of the applied bending moment for the solution used.



Figure 3.5: Failure of the spar.

The wing made by using this spar solution weights 500g less than the equivalent with a spar made by a carbon tube of equivalent mechanical properties, but unfortunately it was not possible to fully adopt this solution. Indeed, even if the first Titti prototype passed successfully the structural test, during the flight with the maximum payload the wing structure had a critical failure at the junction between wing and fuselage. A deep analysis of the flight dynamics revealed that the aircraft could've been subject to more than 2g of acceleration (conservatively assumed to be replicated in the static structural test), due to the flight speed that turned out to be much higher than expected. Consequently, a home-made telemetry sensor endowed with accelerometers was mounted on the repaired prototype in order to understand which one loaded the aircraft undergoing a real flight. An example of real flight acceleration is provided in *Fig.3.6*. After defining the loads, we adopted a hybrid solution: in the central sections a commercial carbon tube is used as a spar, while the in house developed spar described above is mounted in the middle and tip wing sections. All the spars must have at least a safety factor of 1.5 on the most loaded section (i.e. the interface sections): by doing this we safely guarantee structural integrity for 3g loads.

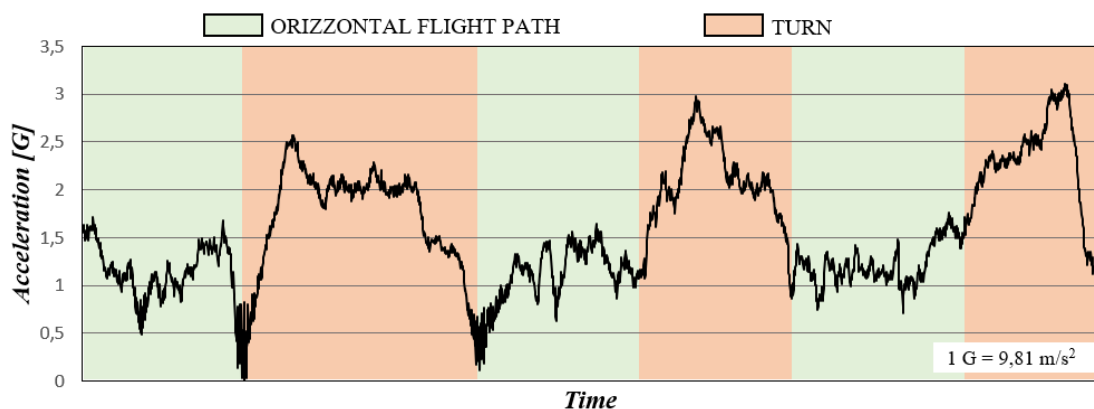


Figure 3.6: Acceleration data acquired during a flight test.

To determine the correct size for the carbon tube used for the junctions and for the central section's spars, we implemented a finite model in MATLAB[®]. The mechanical properties of the carbon tube had instead already been determined by the past AeroUD team and proved reliable, so they have been tested only at the final assembly stage. The scheme that they made is shown in *Tab. 3.1*:

1 x Carbon fabric 150 g/m ² @ 90 deg	To avoid ovalization
5 x Carbon fabric 300 g/m ² @ 0 deg	For bending moment
1 x Carbon fabric 300 g/m ² @ ±45 deg	For torsion

Table 3.1: Layer configuration of central spar.

As a final verification, we put a full wing section under torsional stress evaluation. This was done by building a test bench and comparing the obtained data with the estimated aerodynamic torque. At the test bench, one extremity of the wing section was kept constrained while the other was free to rotate; then at the free side a torque was applied. The torsional rigidity of the wing section was obtained by measuring the wing's rotation with a centesimal dial gauge and combined with the corresponding applied torque. Interpolating the data, a torsional rigidity of 201 Nm² was obtained, a value that broadly exceeds our minimum threshold to withstand the torque due to aerodynamic loads.

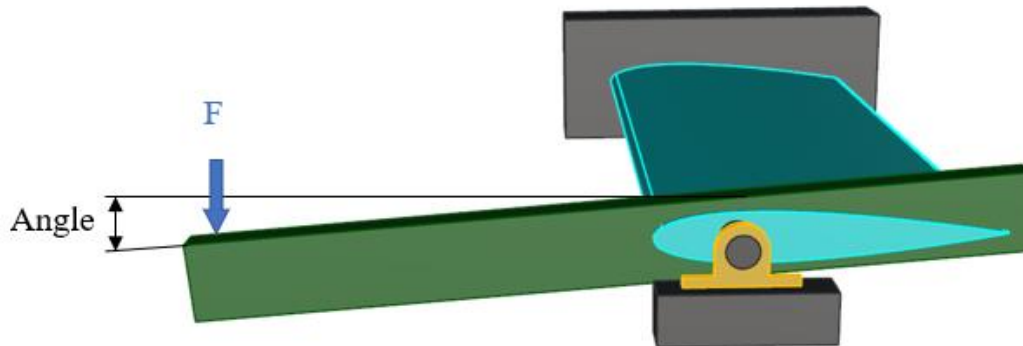


Figure 3.7: Sketch of the set up used for the torsional stress evaluation.

3.2: Empennage and tail boom

The empennage surfaces have been built with a classical structure made of balsa ribs, glued to a carbon tube and covered with monokote. The aerodynamic loads are therefore handled by the carbon tubes which must guarantee high torsional and bending rigidity considering that the surfaces are full flying. In order to save time and budget, we bought the spars from a supplier of composite materials and their mechanical properties verified experimentally.

Instead, the tail boom is composed by three custom made carbon tubes (*Fig.3.8*), produced on our specification by the same supplier of the wing spars, with the layer configuration given in *Tab. 3.2*:

Carbon fabric 150 g/m ² @ 90 deg	To avoid ovalization
Carbon fabric 300 g/m ² @ 0 deg	For bending moment
Carbon fabric 300 g/m ² @ ± 45 deg	For torsion

Table 3.2: Layer configuration of tail's boom carbon tubes.

The design criterion was to consider the tail boom as a cantilever subject to a vertical force at the free end equal to the maximum lift from the empennage. The latter was taken from the XFLR5 simulations and is about 20N. The final tubes dimension and safety factors are specified in *Tab. 3.3*.

Tube identification	Inner Diameter [mm]	Outer Diameter [mm]	Lenght [mm]	Safety factor
A	26,5	28,5	300	4,9
B	25	26,5	550	4,1
C	23,5	25	550	7,3

Table 3.3: Dimension of the used tubes with related safety factors.

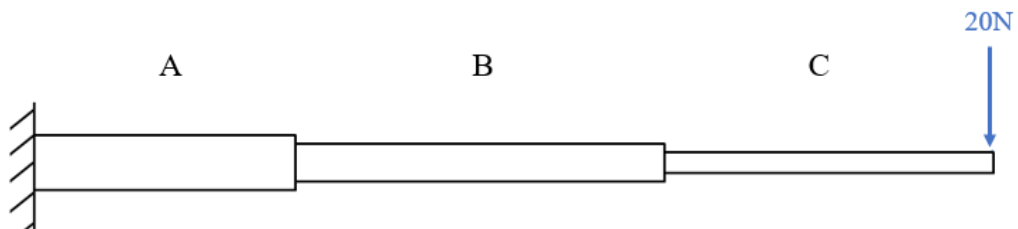


Figure 3.8: Schematic representation of the tail's boom.

The main concern about this tail boom and empennage design was the possibility to encounter the flutter phenomenon. The team had no previous experience with a full flying tail and, at the same time, flutter instabilities are a rather complicated subject to be treated. To validate the design, we tested in flight the tail boom and empennage once again on the IKEA model. Flight speeds as high as 30m/s where reached without any problem. The full reliability of the empennage actuation, which is described in section 4.2.2, was also proved.

3.3: Fuselage, cargo bay and engine pod

As already mentioned on chapter 2, the cargo bay is accommodated inside a pod suspended by the wing by means of a pedestal. The latter is the most critical component since it has to transfer the inertial forces acting on the payload during flight to the wing. Also, it is subjected to impulsive loads at touch down. The internal structure of the pedestal is sketched in *Fig 3.9*. A structure made of ribs and frames in composite sandwich, covered with a layer of carbon fabric resulted the best choice. The pedestal is connected on the upper part to a short fuselage, where the connection between the wings and the cargo bay takes place. The payload is secured to a slit that is then fixed inside the airplane by a pin and a clip. This system should allow a fast payload loading process. On the bottom plate of the cargo bay, the landing gear is also installed. On the main carriage, a mechanical brake is mounted, the details of which are given in paragraph 4.4.

Fuselage, cargo bay and pedestal, form a unique structure and cannot be disassembled. The structure is made with a sandwich of carbon fabric of 49 gr/m^2 and PVC foam. Only the cargo bay fairings are made with a single sheet of carbon fabric, since their function is only to provide better aerodynamics. Both the tail boom and the engine pod connect to the fuselage by insertion in a cylindrical pocket. The engine pod must accommodate also the engine controller (ESC) and battery. To guarantee torsional rigidity, the layers of the engine pod skin are in order: 49 g/m^2 carbon fabric oriented $\pm 45 \text{ deg}$, PVC foam of 1.2 mm thickness and again a carbon layer as the internal one.

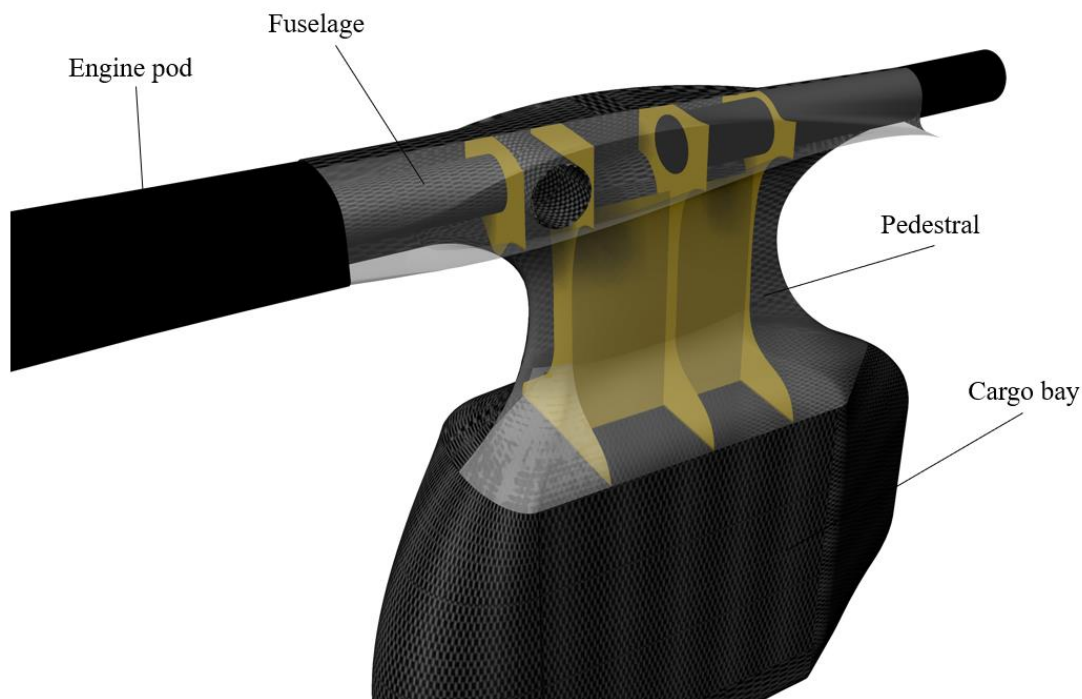


Figure 3.9: Fuselage, cargo bay and pedestal.

3.4: Construction process

The team opted to rely on composite materials for the construction of the whole aircraft, and this is because:

- composite materials ensure some of the lowest weight-per-rigidity ratios, a mandatory characteristic in search for maximum performance;
- lamination of composite materials results in excellent surface finishing;
- through proper mould machining, complex shapes can be manufactured, allowing smooth and aerodynamically sensible couplings of the various components.

These considerations put together would end up in the best result in search of performance.

As the wing is divided in six different sections, we commissioned twelve different moulds to an external subcontractor, as the laboratory milling machine wouldn't be up to a task of such load. In order to speed up the machining while keeping the overall costs low, the moulds were commissioned to be realized in MDF wood.

Once machined, the team put the moulds through a thorough surface treatment. The treatment consisted of several painting and polishing steps, in order to limit wear from usage and increase surface finishing. The mould's ultimate surface roughness is estimated to be around 20 μm , as the final step consisted of P1000-rated sandpaper polishing.



Figure 3.10: A view of some moulds of Titti's model undergoing surface treatment.

Vacuum moulding is the technique chosen to realize all the composite components and required some try-outs before reaching the optimal procedure. As explained in the section about the structural design, three-layer sandwiches are used in the construction. As a core material, we used PVC foam of 1.2 mm thickness while the external fibre reinforcement can vary, depending on the final use. As a matrix we used high quality epoxy-F resin. The moulds were left at least 12 hours under vacuum and in a warm room (24°C or more) for at least 24 hours to ensure complete curing of the polymer before any manipulation.

The same technique was applied to obtain the laminated panels that make up the internal structures.

Eventually the team acknowledged the CAM skills required to generate complex milling paths, in order to machine the several pieces with the laboratory's 3-axis CNC milling machine.

CHAPTER 4: ELETRIC, ELETRONIC AND OTHER COMPONENTS

4.1: Battery capacity

The team decided to use three cells Lithium-Polymer battery (LiPo) to supply the airplane engine. To select the battery size, the minimum capacity required has been calculated. In the worst-case scenario, the total flight time is 4 minutes, composed by:

- take-off, assuming about 1 minute considering several attempts;
- time window for gaining altitude and entering the race circuit: 30 seconds imposed by regulation;
- 1 minute and 45 seconds to fully fly 10 legs;
- 45 seconds for landing.

The following equation has been used to compute the minimum battery capacity:

$$C = 1000 I \Delta t \quad 60 \quad [4.1]$$

where $C[\text{mAh}]$ is the battery capacity, $I[\text{A}]$ is the current that supplies the engine and $\Delta t[\text{min}]$ is the flight time. Assuming to run the engine at max throttle for all of the flight time and considering an average current of 50 A (as a precaution 10% higher than the value measured at the propeller traction test bench) the resulting battery capacity turned out to be about 3350 mAh.

The chosen battery has a higher capacity than the minimum calculated above. This is needed in order to meet a requirement imposed by the regulation: the product of max continuous discharge rate times the capacity has to be at least 45 A. The battery chosen was a FullPower Lipo 3S 4200 mAh 90C PLATINUM (continuous discharge: 90C, peak discharge 180C).

The battery was tested at the test bench that was used to measure the propeller traction (see section 4.3) and a full competition flight was simulated in order to verify the actual energy consumption (mAh) and have a figure for the battery voltage behaviour in time. The test revealed that the battery has sufficient capacity and thanks to high C value, the voltage never drops below 11.6V, which is crucial to maintain an acceptable propeller traction during the whole flight attempt.

Concerning the feed of the on-board RC system (RX and servomotors) we chose a supplementary LiPo 2S battery with a capacity of 850 mAh. The battery is coupled with a voltage regulator to ensure to the RC system a constant voltage of 6V.

4.2: Servomotors sizing

4.2.1: Ailerons and Flaps

In order to minimize aerodynamic drag, the servo-mechanisms which move both ailerons and flaps are fully embedded inside the wing body. These systems can be modelled as a four-linkage mechanism, as schematised in Fig. 4.1.

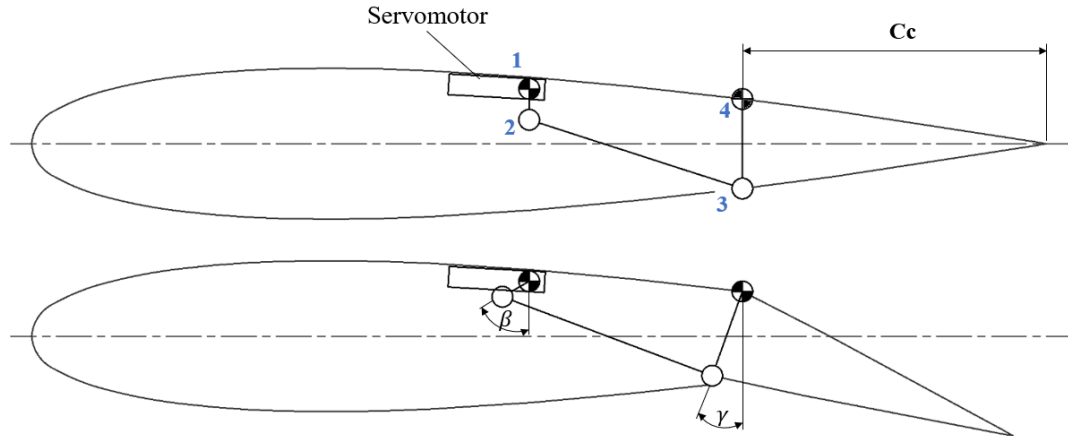


Figure 4.1: Schematic of the aileron and flap actuation system.

In Fig 4.1, joint 1 is the servo motor, joint 4 is on the hinge (pressure side) and joint 3 is on the opposite side, so in other words the length of the bar 4-3 is determined by the wing's dimensions. In order to compute the required servomotor torque, the following assumptions were made:

- the angle of incidence of the wing is zero and the airflow is considered as a non-compressible flow;
- in the neutral position, links 1-2 and 3-4 are parallel;
- friction is negligible.

Starting from the dynamic pressure that acts on the air foil, the servomotor torque can be calculated as:

$$C_{servo} = \frac{1}{4} \rho v^2 c_c^2 L_c \sin \gamma \frac{\tan \gamma}{\tan \beta} \quad [4.2]$$

where $\beta = 60$ deg in the worst case, and γ depends by link lengths, anyway it always between 30 deg and 40deg.

The estimated torques are listed in Tab. 4.1.

Control surface	Average chord (C_c) [m]	Length (L_c) [m]	C_{servo} [kg cm]
Aileron	0,059	0,59	1,1
Flap	0,070	0,74	1,8

Table 4.1: Values of average chord, length and torque of aileron and flap.

The chosen servomotors are 'KST X10' which guarantee a 7.5 kg cm torque at 6 V. 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.

4.2.2: V-tail

We used a similar approach to size the servomotors to be installed in the full-flying V-tail. The schematics of the mechanism are depicted in *Fig. 4.2*. 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.

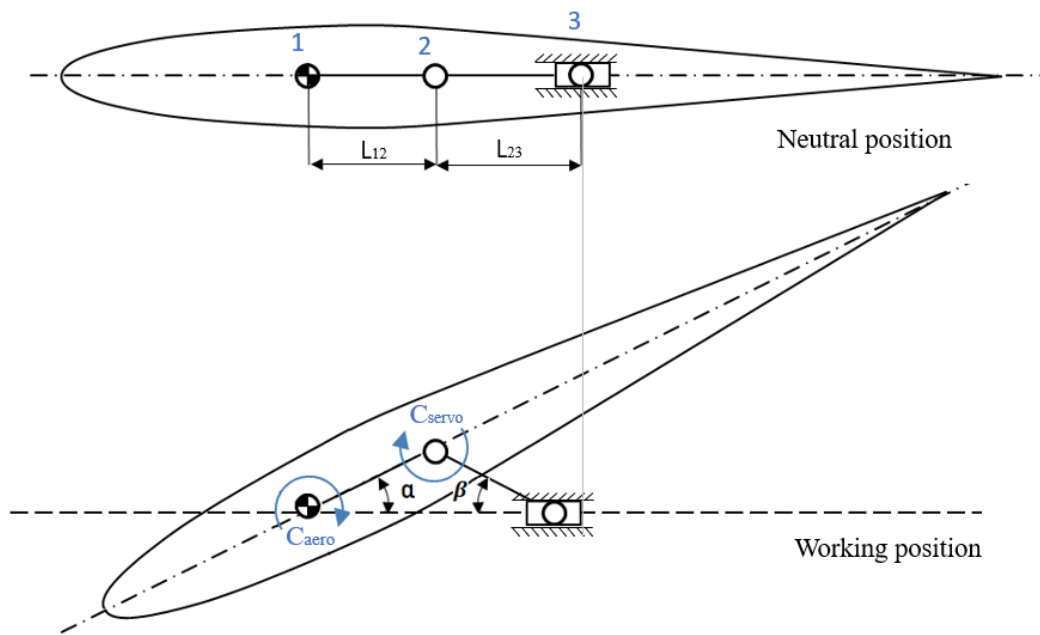


Figure 4.2: Schematic representation of the servo motors mechanism for the V-tail.

The aerodynamic forces that act on the tail can be calculated assuming that:

- air flow is incompressible;
- the angle of incidence is zero at the neutral position;
- the maximum torque is at α max (10 deg.).

Assuming that the aerodynamic forces act at $\frac{1}{4}$ 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 10 deg and at Reynolds number of 200.000.

So the motor torque is:

$$C_{servo} = C_{aero} \left(1 - \frac{L_{12} \cos \alpha}{L_{23} \cos \beta} \right)^{-1} \quad [4.3]$$

	L_{12}	L_{23}	C_{aero}	α	β	C_{servo}
	[mm]	[mm]	[Nm]	[deg]	[deg]	[kg cm]
<i>Tail</i>	28,6	14	0,051	10	36	0,35

Table 4.2: Parameters to compute the servo torque.

The chosen servomotors are again the ‘KST X10’ for the same reason that justify their usage for the wing control surfaces.

4.3: Propeller traction analysis

For the aerodynamic design, the new propeller imposed by the ACC2019 regulations had to be characterized in terms of traction/flight speed. This goal was achieved by exploiting the rig already available at the university’s laboratory and used by the previous AeroUD teams. The engine is mounted at the top of a vertical cantilever beam, *Fig. 4.3*. The small deformation of the cantilever was then associated to the propeller traction by means of a strain gauge placed near the bottom/clamped end of the beam, *Fig. 4.4*. For small deformation, the strain is proportional to the force applied at the free end, hence the traction. With a simple calibration it was possible to calculate the constant of proportionality between the force and the electrical signal given as an output from the amplifier linked to the strain gauge.

To perform the measurements at different airspeeds (in order to simulate flight conditions) the entire system is placed in front of the open end of a wind tunnel, where flow velocity can be set and controlled up to 25 m/s.

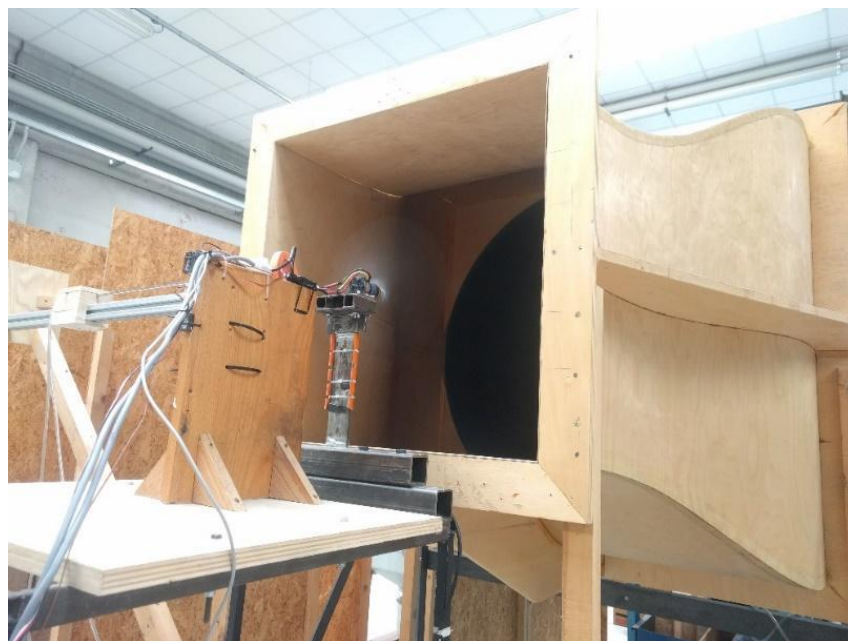


Figure 4.3: Set up used for the characterization of the new propeller.

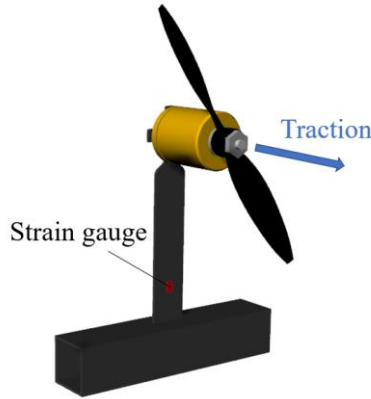


Figure 4.4: Sensorized engine mount.

For the test, the engine was connected to a stabilized power supply. This allowed to perform traction measurement at different levels of voltage (in the range admitted by the batteries) without incurring in the voltage drop of the LiPo cells. The data were then interpolated to obtain a parametric curve that linked traction T [N] with speed v [m/s] and supply voltage V [V]:

$$T = k_1 v^2 + k_2 v + k_3 V + k_4 \quad [4.3]$$

where the parameters k_1 , k_2 , k_3 and k_4 assume the following values:

$$\begin{aligned} k_1 &= -0.00667 & k_2 &= -0.18596 \\ k_3 &= 2.66383 & k_4 &= -8.79332 \end{aligned}$$

This function, together with an estimation of the battery voltage variation in time, were extremely useful to determine the available propeller traction in the different flight phases and therefore guiding the aerodynamic design.

4.4: Brake System

The team decided to adopt a belt brake system for the main carriage wheels. The design is based on the requirements to stop an aircraft of mass $m_{plane}=15$ kg which velocity at landing is $v_i=15$ m/s in $L_{stop}=45$ meters. We made the following assumptions:

- while braking the wheels have a rolling motion;
- the deceleration is constant;
- only the brake system decelerates the wheels (very conservative assumption).

The deceleration is:

$$a = v_i^2 \frac{0.5}{L_{stop}} = 2.5 \left[\frac{m}{s^2} \right] \quad [4.4]$$

It could be assumed that the adherence coefficient of wheels with asphalt is $a_c=0.6$ so any deceleration under $9.81a_c = 6 \text{ m/s}^2$ do not make the wheel slip, so the first assumption is true.

A simplified drawing of the brake system is given in Fig. 4.5. It consists of an aluminium pulley integral with the main carriage axle, which is made by a carbon tube free to rotate inside two bearings located on the sides of the cargo bay. On the pulley, a plastic belt is wrapped and connected to a servo motor to apply the required force to the belt which turns into a brake torque on the pulley.

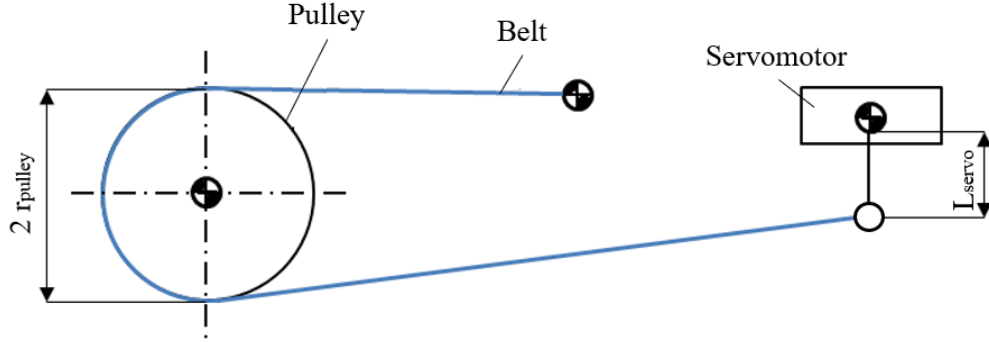


Figure 4.5: Drawing of the braking system.

The brake torque can be calculated as:

$$C_{brake} = a m_{plane} r_{wheel} = 1.875 [Nm] \quad [4.5]$$

being the wheels' radius equal to 50 mm. This braking torque has to be applied by the servo motor

$$C_{servo} = \frac{C_{brake}}{r_{pulley}(1 - e^{-\mu\vartheta})} L_{servo} = 8 [kg cm] \quad [4.6]$$

Where μ is the kinetic friction coefficient (for nylon aluminium is 0.4) and ϑ is the wrapping angle (assumed 180 deg).

With this value of torque, we chose the 'FullPower DM4082M' servo motor.

CHAPTER 5: PAYLOAD PREDICTION

Payload prediction chart is based on data from test flights and propeller traction data. During the flight tests, the average aircraft speed was measured by means of a GPS telemetry module, by taking also into account possible effects of wind. This data, together with the knowledge about air density value at those days and the airplane take-off weight, allowed us to determine experimentally the actual lift coefficient, that is kept constant for the estimation of the payload. Propeller performance data is scaled based on the actual air density value and so, from dynamic equilibrium (traction propeller = drag), it is possible to determine a new flight speed for every air density condition. The regulations required that the payload predictions have to be calculated as a function of air density, with the following linear approximation:

$$P = 8.33 \rho + 1.23 \quad [5.1]$$

where ρ is the density altitude in $[kg/m^3]$.

The prediction, shown in *Fig. 5.1*, was also verified in occurrence of flight tests with different density altitude.

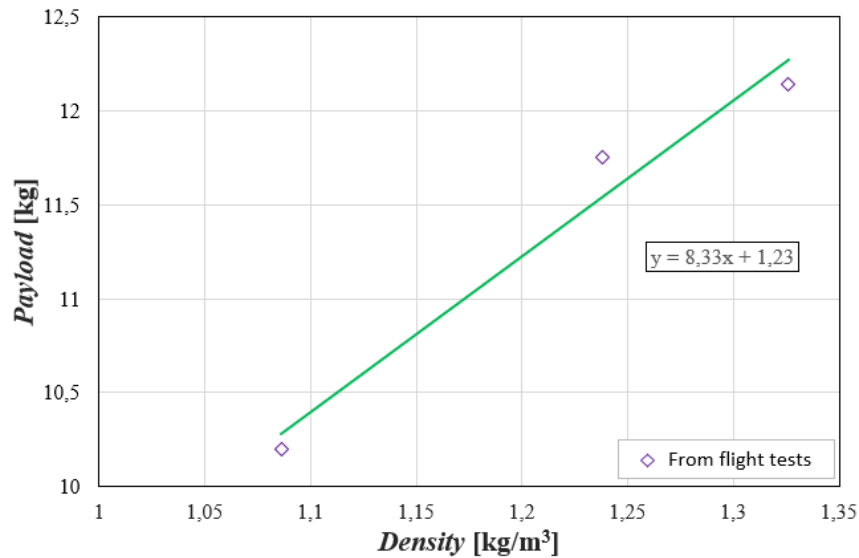


Figure 5.1: Payload prediction.

CHAPTER 6: OUTLOOK

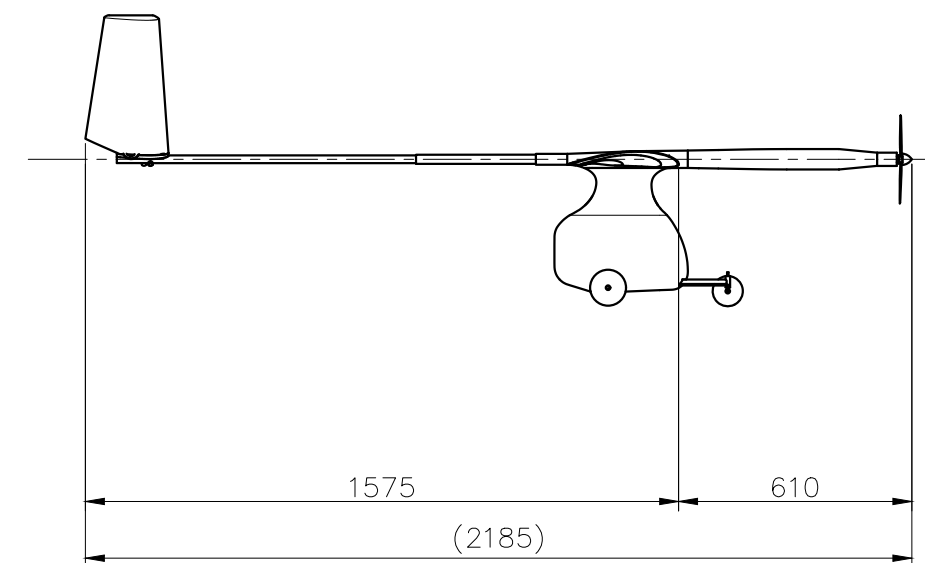
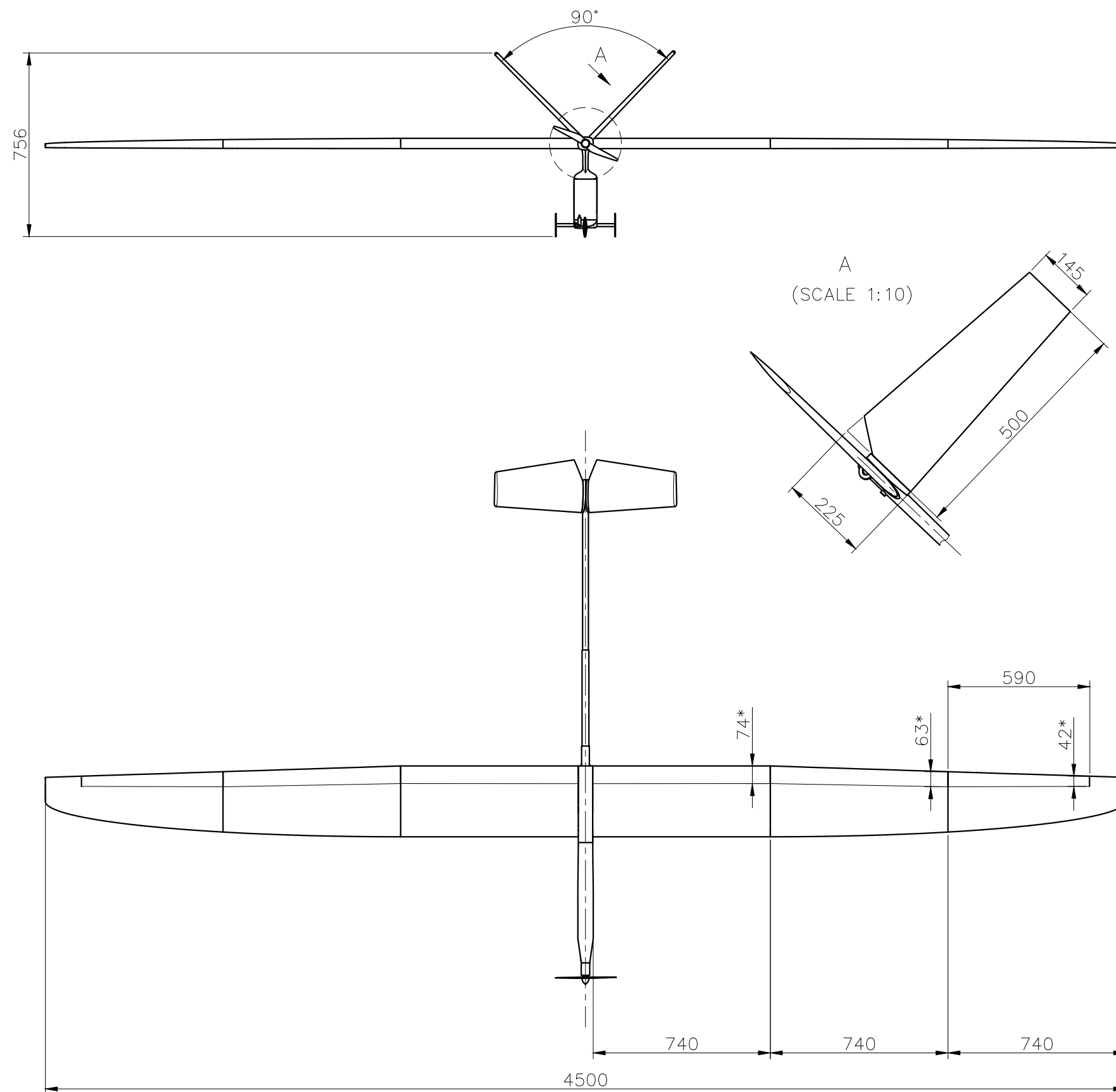
The team is building the final version of the aircraft and the first take off is expected by the end of May. Waiting for the final version, during this period we will use the prototype to improve the pilot's skills. In June and July an intense flight test camping is programmed in order to:

- find the best flight trajectory;
- refine landing phase in order to stop the aircraft in 60 m.

A second competition aircraft will be built in order to be used as spare parts during the competition. A manufacturing revising process is under evaluation to minimize the airplane's weight.

Attachments:

DRAWINGS and APPLICATION FORM

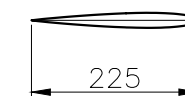


Airfoil – SCALE 1:10

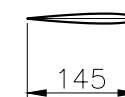
Wing airfoil at the root
(Wortmann Fx 63-137)

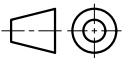


Full-flying airfoil at the root
(NACA0009)

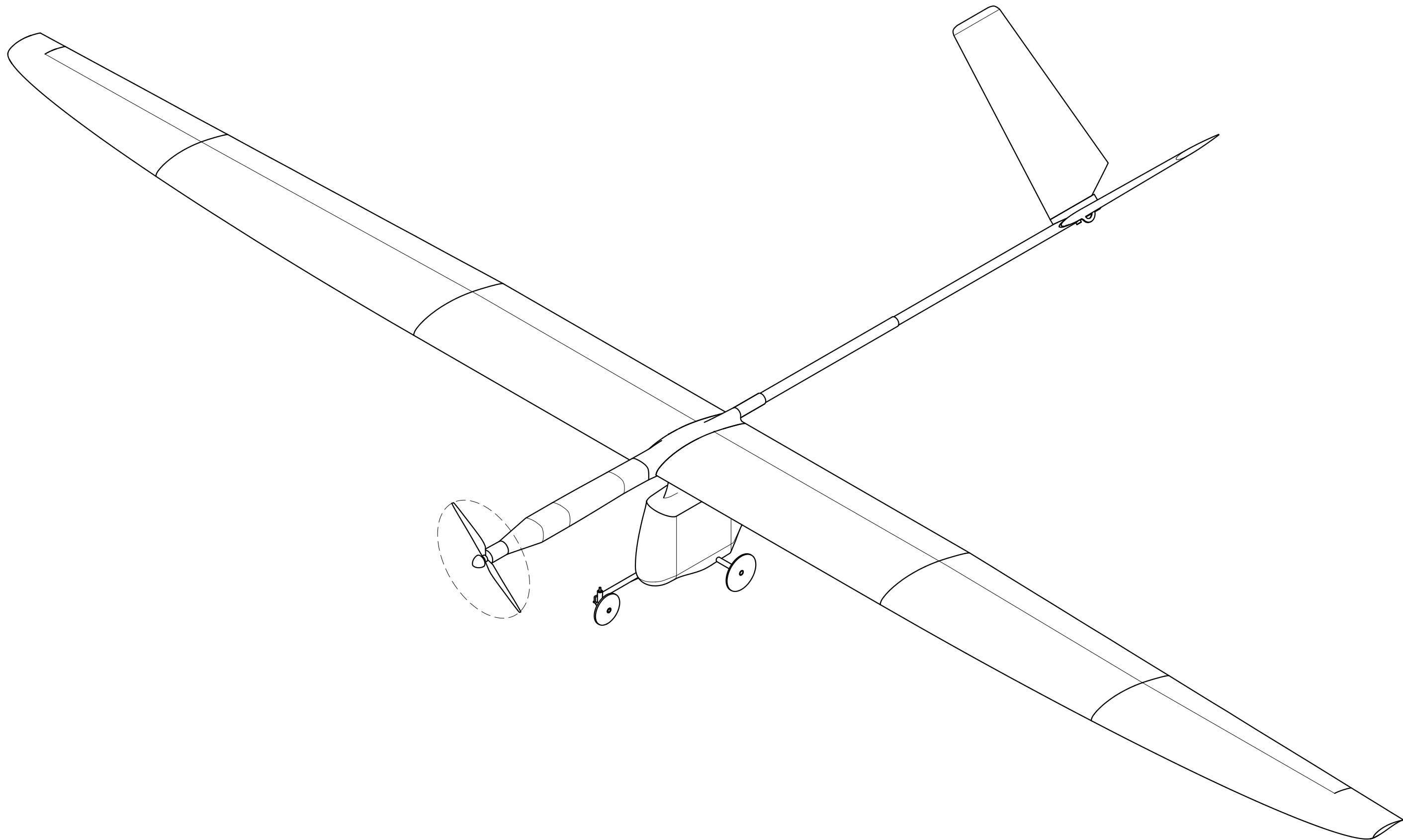


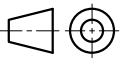
Full-flying airfoil at the tip
(NACA0009)

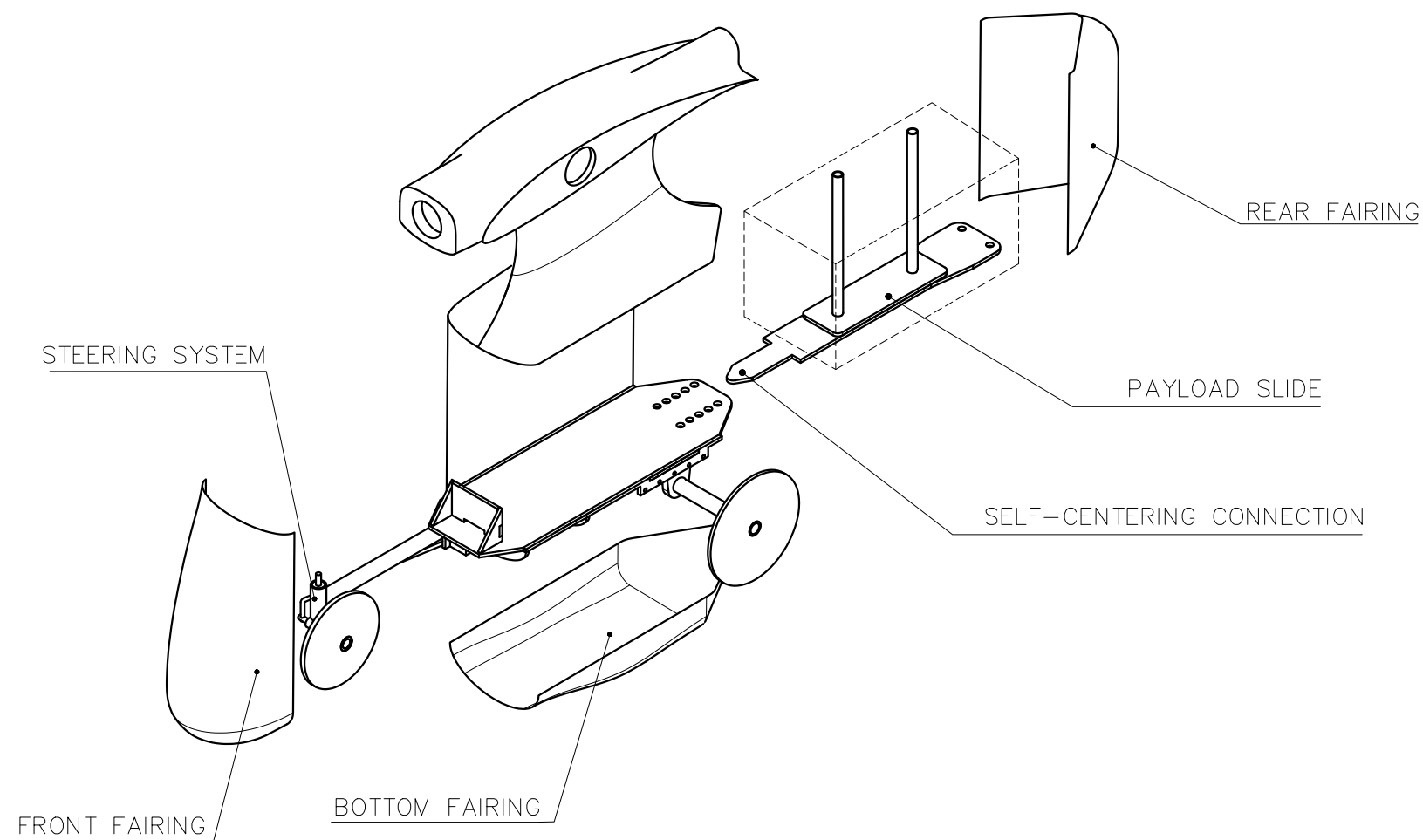
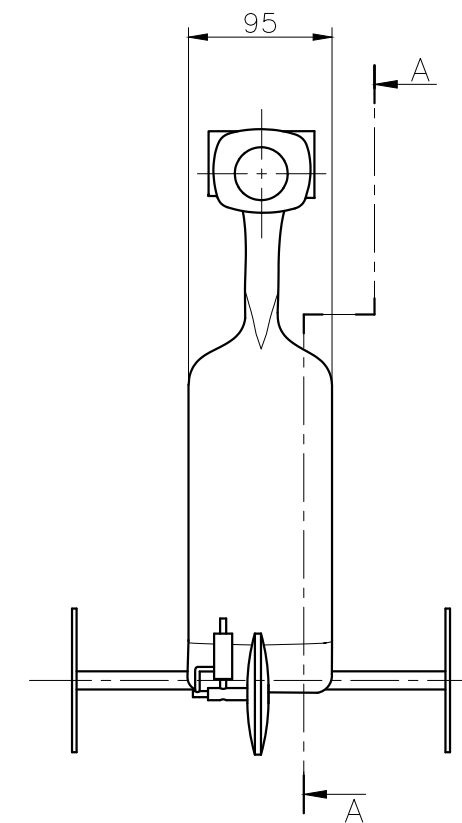
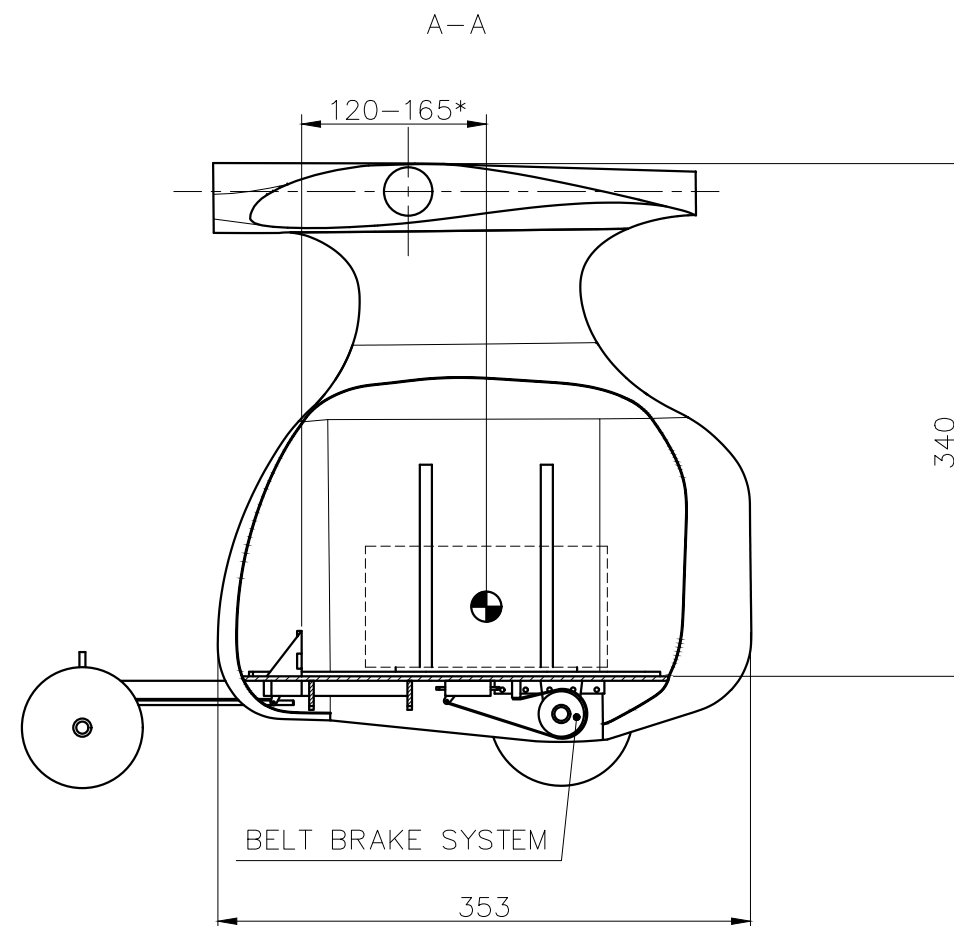


Surface	Area	Surface	Area
Wing	1.146 m ²	Flaps	0.210 m ²
Full-flying Tail	0.182 m ²	Ailerons	0.0620 m ²
 UNIVERSITA DEGLI STUDI DI UDINE		TITLE 3-VIEW DRAWING	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD – ...and the Beast/02	GENERAL TOLLERANCES UNI EN-ISO 22768-1-v
SCALE 1: 20	DATE 03/05/2019	DESCRIPTION – front view at the top left corner – top view below – starboard view on the right	

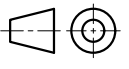
(*) Length measured on the wing chord (25% of the chord at the beginning and end of the section)

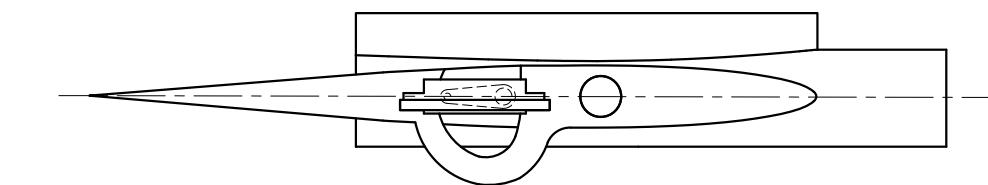
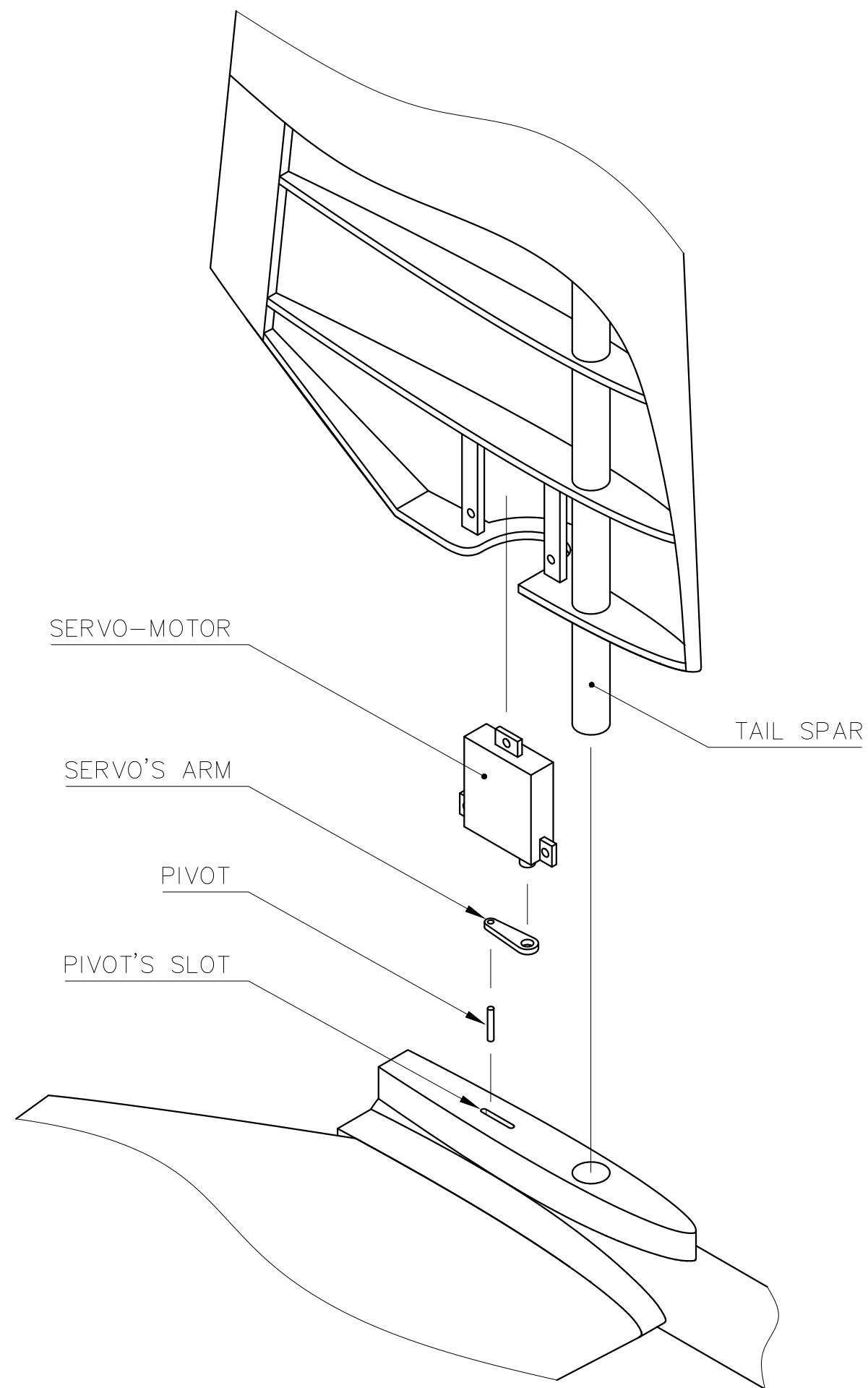


 UNIVERSITA DEGLI STUDI DI UDINE		TITLE ISOMETRIC PERSPECTIVE VIEW	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - ...and the Beast/02	GENERAL TOLLERANCES ---
SCALE 1:10	DATE 03/05/2019	DESCRIPTION Isometric perspective view	

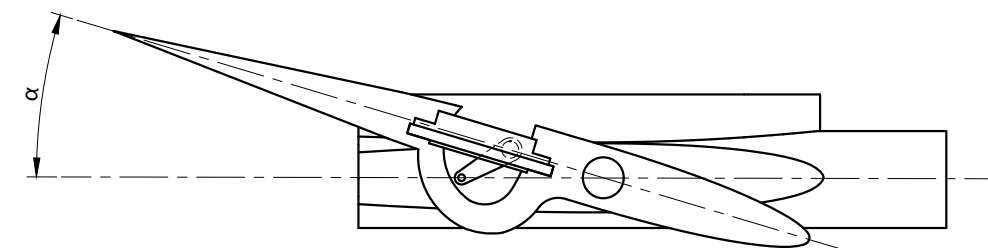


(*) Payload range to align the center of gravity

 UNIVERSITÀ DEGLI STUDI DI UDINE		TITLE CARGO BAY	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - ...and the Beast/02	GENERAL TOLERANCES UNI EN-ISO 22768-1-v
SCALE 1:5	DATE 03/05/2019	DESCRIPTION Cargo bay location and principal dimensions On dashed line: orthogonal volume of 160x80x80 mm for technical inspection	

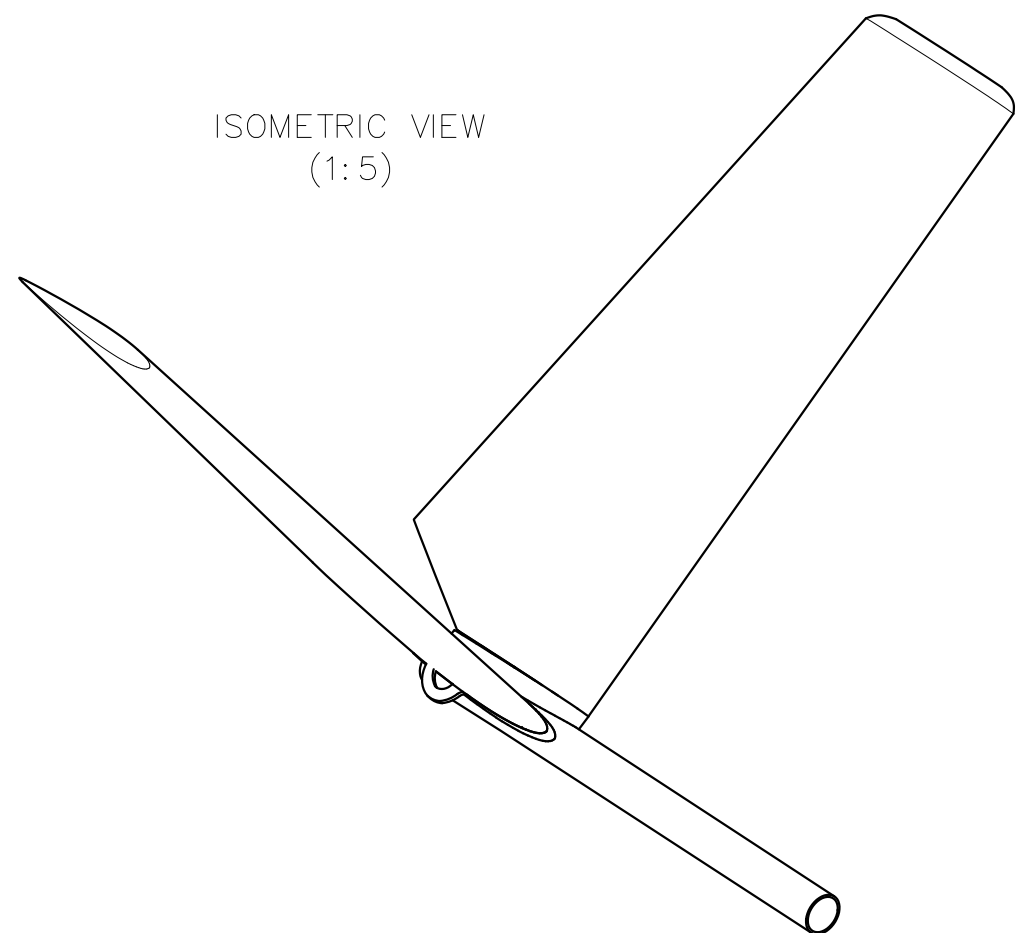


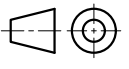
NEUTRAL POSITION



WORKING POSITION

ISOMETRIC VIEW
(1:5)



 UNIVERSITÀ DEGLI STUDI DI UDINE		TITLE FULL-FLYING TAIL	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - ...and the Beast/02	GENERAL TOLLERANCES ---
SCALE 1:2	DATE 03/05/2019	DESCRIPTION Full-flying tail and explanation of how it works	



UNIVERSITÀ
DEGLI STUDI
DI UDINE

AerOUD ... AND THE BEAST

TEAM 02: Participants

NAME	SURNAME	GENDERS	DATE OF BIRTH	T-SHIRT SIZE	TEAM ROLE	SIGNATURE
Massimo	Bosco	male	24/04/1995	L	leader	Massimo Bosco
Luca	Casarsa	male	18/12/1974	M	pilot	Luca Casarsa
Luca	Pupil	male	15/04/1995	M	member	Luca Pupil
Diego	Perissutti	male	26/08/1997	L	member	Diego Perissutti
Matteo	Fornasin	male	26/02/1994	L	member	Matteo Fornasin
Elia	Cucit	male	26/10/1993	XXL	member	Elia Cucit
Pietro	Martincigh	male	02/01/1995	M	member	Pietro Martincigh
Stefano	Sfriso	male	20/02/1998	S	guest	Stefano Sfriso
Dario	Fagotto	male	01/07/1996	L	guest	Dario Fagotto
Davide	Cozzarizza	male	20/09/1996	M	guest	Davide Cozzarizza

Team leader

Massimo Bosco

Professor in charge

Piero Pinamonti