

Air Cargo Challenge 2017

Zagreb



REPORT

Team 07





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List of symbols

A_R	Aspect ratio
a_T, b_T	Experimentally derived constants
A_T	Tail surface
A_W	Wing surface
b	Distance, along the y -axis, between the aerodynamic centers of wing and horizontal stabilizer
c	Wing chord
c_c	Control surface average chord
$C_{D,0}$	Fuselage drag coefficient of resistance
$C_{D,2D}$	2D profile coefficient of drag
C_D	Drag coefficient
$C_{L,2D}$	2D profile coefficient of lift
$C_{L,\alpha}^T$	First derivative of the lift coefficient for the tail
$C_{L,\alpha}^W$	First derivative of the lift coefficient for the wing
C_L	Lift coefficient
C_M	Pitching moment coefficient
D	Drag
d_e	Beam external diameter
E	Young's modulus
e	Oswald's factor, assumed equal to 0.8
f	Maximum displacement
I_g	Mass moment of inertia with respect to the center of mass of the wing
J	Moment of inertia
k_u	Translational stiffness of the wing
k_ε	Derivative coefficient of wing downwash deflection
k_θ	Torsional stiffness of the wing
L	Lift
l	Beam length
L_c	Control surface average length
M	Bending moment
m	Mass of the wing and of the surrounding air cylinder
p	Applied force
r	Bending radius
s	Beam thickness
S_W	Wingspan
T	Thrust
v	Airplane speed
x_{ct}	Distance of the center of twist from the leading edge
x_g	Distance of the center of mass from the leading edge
y_{pn}	Distance, along the y -axis, of the neutral point from the wing aerodynamic center



Greek symbols

α	Max control surface deflection
α_A	Airplane angle of attack
α_F	Fuselage angle of attack
α_W	Wing angle of attack
β	Max servo deflection
ε	Wing downwash deflection
ρ	Air density
ρ_b	Beam density
σ_r	Beam UTS



1. Introduction

The AeroUD-MET (Aero Udine University – Mechanical Engineering Team) began its activity in 2011 with the aim to participate in the 2013 SAE Aerodesign competition. At the University of Udine there is no degree in aeronautical engineering, so at the beginning it was not sure that a team only composed by mechanical engineering students and professors could have a chance to rank at high position in this kind of competition. Despite this original handicap, the team could finally participate in the 2013 SAE Aerodesign East competition, ranking fourth in the regular class. Strongly motivated by this result, the team decided to go on in this kind of experience and to face a new challenge by design, build and fly a completely new model to participate to the 2015 Air Cargo Challenge (ACC) edition held in Stuttgart (Germany). Again, the airplane and the team behaved beyond any expectation and the 3rd placed was missed by only few points over 260 of total score.

In September 2015 a new team was founded, some students graduated and left, some new ones joined the group and, supported also by the University Rector that sees in the AeroUD experience an unique opportunity for the students, the decision was taken to participate to the ACC2017.

The work that lead to the actual model that will be flying in Zagreb is the result of a development and optimization process of the previous airplane used in ACC2105. The developments cover both design and manufacturing aspects. In particular, the team has focused his efforts on:

- define and program a simulation tool that will help the aerodynamic design of the model targeted to maximize the competition score;
- improve the static stability calculation software previously developed in order to obtain more reliable predictions on which the new design has been based;
- improve the previous design to solve some issues that limit the performances such as overall airplane drag (in other terms max speed) and remarkable inverse yaw during turns;
- improve the manufacturing process by adopting a full composite (carbon/glass/PVC) construction to ensure higher rigidity of the structure and open the possibility to more aerodynamic shapes.

Finally, many training hours at the flight field are foreseen in the next months to carefully tune the model and the flight path.

The following sections report about the most relevant issues of the project.

2. Project management

2.1 Financial budget

The overall budget made available by the University was € 20000. A first estimate of the expenses was based on the statistics available from the participation to the previous ACC2015 and modified accordingly to the new project requirements. The entire budget was initially roughly divided into five categories:

- development and modification of the previous aircraft;
- realization of the new prototype;
- competition aircraft construction;
- travel costs;
- purchase of workshop tools and external services.

Each of these voices have been successively detailed in sub-items at the best of team's knowledge at the beginning of the project, being conscious that the most relevant expenses would have been related to (and they turned out to be) the choice to go for a composite construction, i.e. manufacturing of the molds (about 3000 euros) and purchases of the materials (glass/carbon fiber tissues, PVC foam sheets, about 2500 euros). Another important voice of cost known from the beginning, but that allowed the team to reach higher construction standards, was the purchase of a small CNC 3-axes milling machine, which took about 3500 euros of the total budget by itself. Another external collaboration was for the wing spars and tail boom (made in composite carbon fiber tubes), in order to ensure a product as much reliable as possible; this purchase costed about € 1400 (for two models).

Travel expenses have been estimated equal to € 2200 for nine people: a professor and eight students. Since Zagreb is fairly close to Udine and easy accessible by road, the estimated 2200 euros seem to be a reasonable foresight.

Table 1. Financial budget forecast

Development and modification of the previous aircraft	Construction materials	€ 300
	TOTAL	€ 300
Realization of the new prototype	Construction materials (glass/carbon fabric, tissues, PVC foam sheets, Epoxy resin, plywood, MDF, etc..)	€ 1400
	Electronics and propulsion systems (servos, cables e plugs, battery, etc..)	€ 300
	TOTAL	€ 1700
Competition aircraft construction	Construction materials (glass/carbon fabric, PVC foam sheets, Epoxy resin, etc..)	€ 1100
	Electronics and propulsion systems (servos, cables e plugs, battery, motor, propellers, landing gears, etc..)	€ 500
	TOTAL	€ 1600
Travel costs	Competition fee	€ 900
	Transport and transfert	€ 2100
	TOTAL	€ 3000
Purchase of workshop tools and external services	Composite carbon fiber tubes	€ 1500
	Manufacturing of aluminum molds	€ 3500
	Purchase of a small CNC, 3-axes milling machine;	€ 3500
	TOTAL	€ 8500
TOTAL EXPENDITURE		€ 15100

Moreover, it is important to note that it was possible to contain some expenses since the team owned already some of the equipment and materials from the previous edition (e.g., battery charger, Tx-Rx system, engine controller, etc.).

Table 1 summarizes the expenses summed up by type. Up to now, this forecast has been respected.

2.2 Time schedule

A detailed view of project timing is provided in Fig. 1. In order to ensure the success of the project and to meet the defined project's milestones, the team was organized in two working groups with specific assigned tasks:

1. aircraft design: simulations and calculations; with sub-groups dedicated to aerodynamics and structural design;
2. drawings: 2D-3D construction drawings, CAD-CAM for CNC milling.

Conversely, construction and flight tests activities were carried out by the totality of the team members.

The project, which began in the fall of 2015, has a total duration of almost two years. The first year was devoted to test different solutions (such as flaps, new tail and cargo-bay desing) to improve the performances of the previous model (named FB2) used in ACC2015. These modifications did not allow to improve significantly the performances of the aircraft. Consequently, the design of a new aircraft started with a first phase devoted to performance simulation according to the new ACC2017 rules.

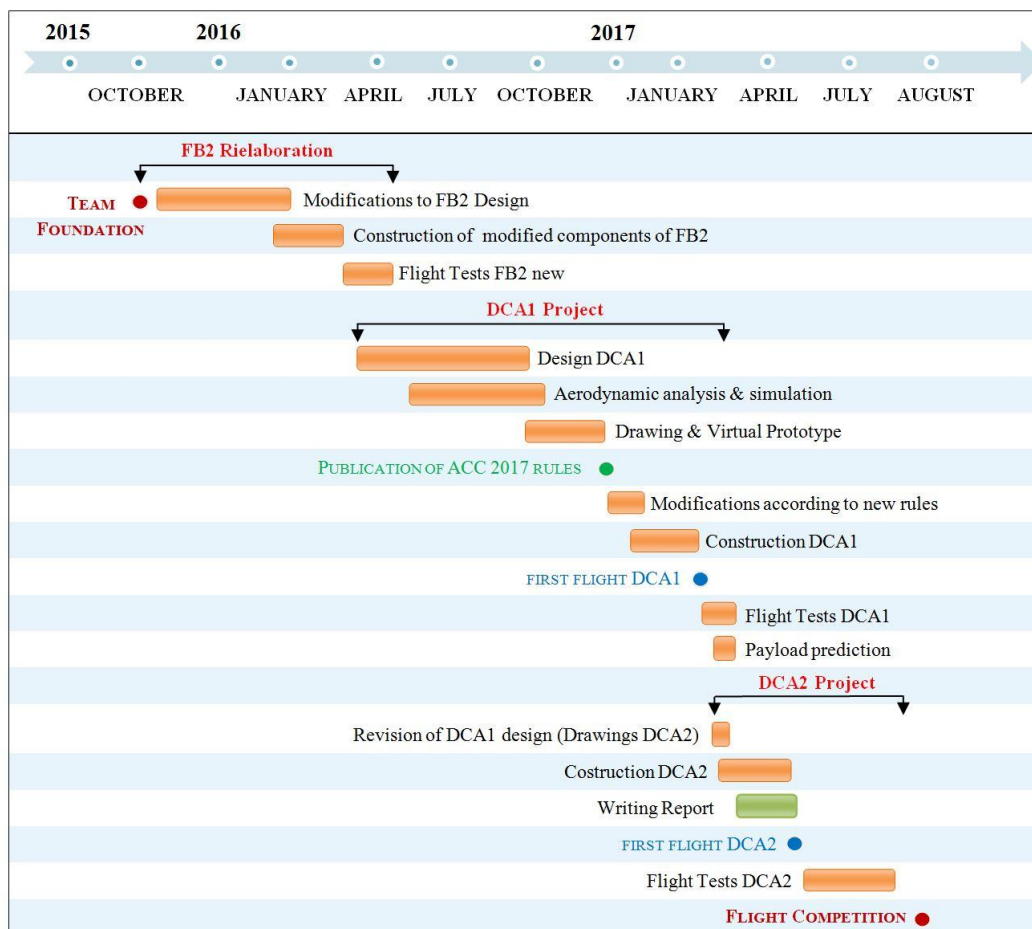


Figure 1. Project timeline



The new prototype, named DCA1, made its maiden flight in February 2017. Multiple flight tests demonstrated the efficiency and the reliability of the design, meeting the target performances. Therefore in March 2017 the competition model was defined with minimal changes to the prototype.

At the end of April the construction of a first version of the competition airplane, the DCA2, was completed. The remaining time before the competition will be used for further testing and to provide to the pilot the time needed to train.

3. Aerodynamic Design

Starting from the configuration adopted in the FB2 model, several improvements were made in order to increase the performances and the maneuverability during flight. Design targets were fixed at 15 kg of maximum take off weight and average flight speed of 15 m/s.

The final configuration adopted is then based on the assumptions recapped below:

- monoplane architecture;
- rectangular wingplant with airfoil transition at the wingtips;
- high wing configuration;
- tractor propeller configuration;
- tricycle landing gear;
- T-tail configuration with rectangular and trapezoidal plant for the vertical and horizontal stabilizer, respectively.

Monoplane architecture was chosen because is the simplest possible configuration, that could guarantee better efficiency with respect to other architectures, and also can simplify the desing and construction processes. The same considerations mentioned above were applied to the choice of a rectangular wingplant. Wingtips designed with specific profiles have been included in the project in an attempt to reduce the induced drag and improve aircraft speed.

High wing and tractor propeller configurations were selected because of their intrinsic higher stability. The same consideration was made in chosing the landing gear layout. With the T-tail configuration the horizontal stabilizer is kept well out of the wing and fuselage wakes, resulting in improved pitch control and aerodynamic performances.

Another fundamental issue concerns the position of the payload with respect to the wing. Stability considerations would probably suggest putting the payload in a lower position with respect to the wing. On the contrary, a higher payload position (i.e. closer to the wing pressure surface) allows a better overall maneuverability especially when the plane rolls. Furthermore, this solution improves the aerodynamic thanks to the reduction of the frontal section.

The ESC and the batteries were placed in a pod located closer to the electric motor in order to reduce as much as possible the length of the power cables.

3.1 Wing incidence angle

The chosen wing airfoil was the S1223, the same used by the ACC2015 team. This solution was the best choice for the purpose of the competition: the target is to carry the maximum payload as fast as possible and therefore wing efficiency is one of the main variable.

For the previous competition, the chosen wing angle of incidence was 5°, with the aim of operating, during flight, around the point of maximum efficiency of the S1223 airfoil. This time the design target was to search for an incidence angle that could maximizes the competition score.

The score equation defined in the regulation is:

$$S = 2 P \left(\frac{1000}{t} + 0.5 \right) \quad (3.1)$$

where t [s] is the time spent to travel 10 legs (each 100 m long) and P [kg] is the payload weight. The above equation can be arranged in terms of parameters that directly depends by the incidence angle, as reported below:

$$S2 = 2L(v + 0.5) \quad (3.2)$$

It is clear that S_2 value is not the exact scoring, but it can be profitably used to evaluate the effect of different incidence angle on the variation of the score, once v and L are expressed as a function of α_w (with a fixed wing area). The procedure used is described below:

1. in literature, profile polar data are available only for Reynolds numbers lower than the design one, that is fixed at 400.000. Therefore, profile performances were obtained at the design point by means of XFLR5 software, previously calibrated on the lower Re experimental data;
2. corrections on the lift and drag coefficients were made to the data obtained by simulations in order to account for 3D effects. The drag coefficient is:

$$C_{D,3D} = C_{D,0} + C_{D,2D} + \frac{C_{L,3D}^2}{\pi A_R e} \quad (3.3)$$

The lift coefficient is

$$C_{L,3D} = \frac{C_{L,2D}}{1 + \frac{C_{L,2D}}{\pi A_R e}} \quad (3.4)$$

where

$$A_R = \frac{S_w^2}{A_w} \quad (3.5)$$

It is necessary to say that it is difficult to estimate $C_{D,0}$ but, at the same time, it has been noticed that even significant differences of its value ($0.01 < C_{D,0} < 0.05$, a very broad range) do not lead to trends considerably different from the ones which will be shown in Fig. 2;

3. by using data already available from the previous ACC2015 project about the propeller thrust, and by applying the equations of leveled flight, the velocity and the lift force for different angles of attack were found. The propeller thrust can be approximated as a linear function of flight speed:

$$T = a_T v + b_T \quad (3.6)$$

while the airplane drag is

$$D = \frac{1}{2} \rho v^2 A_w C_D(\alpha_w) \quad (3.7)$$

By equating the two expressions, an estimate of the speed can be found, hence an estimate of the load carried is given from the following equation:

$$L = \frac{1}{2} \rho v^2 A_w C_L(\alpha_w) \quad (3.8)$$

These estimates are all dependent from the angle of attack of the wing as shown in the Fig. 2.

As it can be seen, the position of the maximum is significantly different passing from the infinite wing, which means 2D airfoil, to the finite wing. In the case of infinite wing, the maximum is further on the right (at higher angles of attack) compared to the finite wing and close to the point of maximum 2D efficiency. Considering a finite wing, the maximum moves to lower angles, close to -1° .

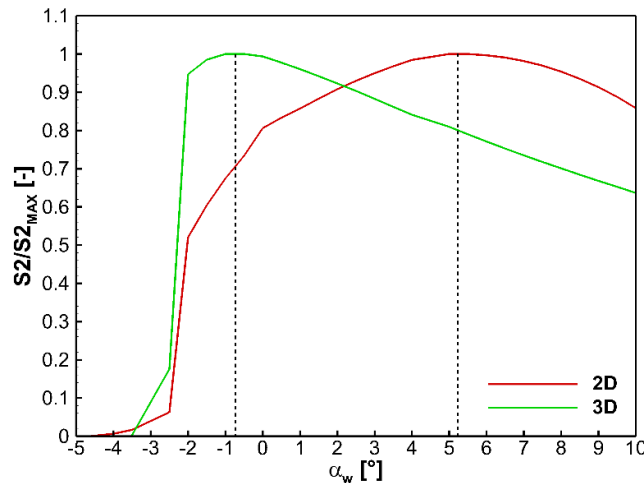


Figure 2. Plot of the function $S2$ for infinite wing (blue line) and finite wing (red line)

In other words, it is more convenient to sacrifice high incidence angles (and so high lift coefficients) to achieve higher speeds which overcome the loss of lift.

To verify the validity of the calculations, several flights were performed with the FB2 model. Flight data acquired by a gyroscope mounted on the plane, highlighted that the better airplane performances occurred when the wing incidence angle were approximately 0° . An example of the sampled data is shown in Fig. 3, where the wing incidence angle measured in horizontal levelled flight is reported (remind that the wing incidence angle of FB2 was $+5^\circ$).

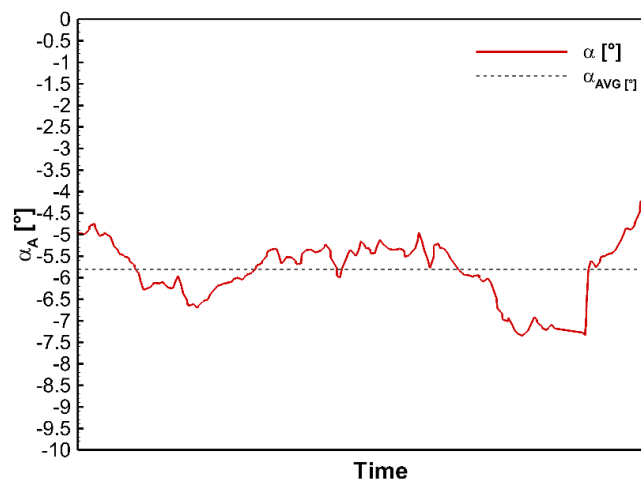


Figure 3. A few sampling seconds of the wing incidence angle of the FB2 airplane (red line) and its mean value on the same interval (dotted line)

Given the results shown above, the wing angle of incidence was set to 0° , in order to keep a safety margin from the abrupt loss of performance that occurs around -2° .

3.2 Wing plant

Assuming to fly averagely with an angle of incidence of about 0° at 15 m/s (a speed which has been measured multiple times on the FB2) and with the S1223 airfoil, the surface needed to lift a total weight of about 15 kg (team goal) would be of about 1.1 m^2 .

Having defined the wing area, the design choices must look for the definition of the suitable aspect ratio taking into account the ACC2017 rules.

A wide wingspan, hence a smaller chord, would imply:

- the need to design a more rigid and so heavier structure in order to pass the static load test;
- problems to fit the entire wing in the transportation box;
- high rotational inertia, hence an increase of the time required to perform a turn;

The design that best fits with the requirements is the one with the wingspan equal to 3.205 m, therefore the wing chord was 380 mm. The final design of the wing plant is shown in Fig. 4. The wing plant is rectangular with a 2.8 m span and wingtips located at the edges.

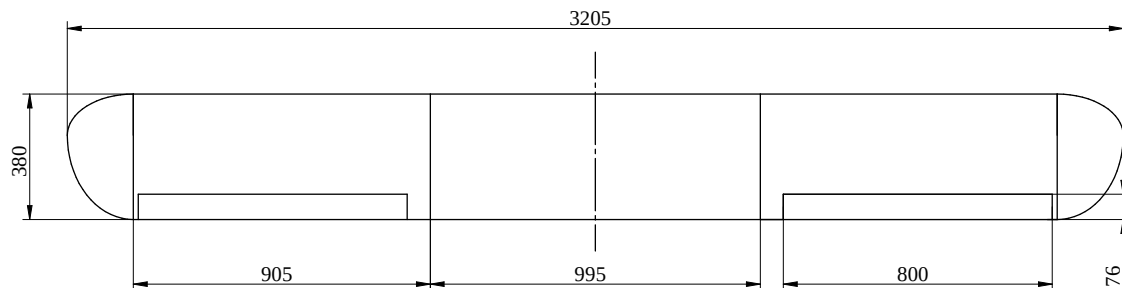


Figure 4. Wing plant

The introduction of the wingtips was due to the need to limit induced drag losses hence enhance the wing efficiency. They were designed with a smooth transition from the S1223 airfoil to a much lower lift airfoil GOE-796, which has similar thickness and similar distance of the maximum camber point as the S1223. The tip plant is a combination of two ellipses as shown in Fig. 5.

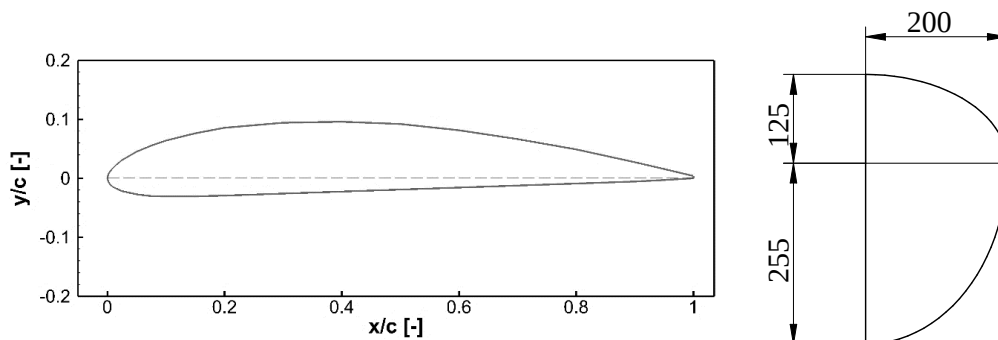


Figure 5. Low lift airfoil GOE-796 used for the tip of the wings (on the left) and top view of the tip (on the right)

3.3 Empennage plant

A symmetric NACA0009 airfoil was chosen for both horizontal and vertical stabilizers.

A first prototype was designed with the characteristics reported below:

- T-tail;
- rectangular plant of the vertical stabilizer with the same surface of the FB2 model;
- trapezoidal plant of the horizontal stabilizer with a surface reduction of 20% with respect to the previous model.

A feedback of the pilot after flying test performed with this new tail design highlights that the horizontal stabilizer was oversized. Conversely, the rudder appeared insufficient to prevent reverse yaw during tight curves. Obviously, the reduction of the horizontal stabilizer had to take into account the requirements of the static stability, which will be discussed in the following section. The final planform of the horizontal stabilizer is reported in Fig. 6. The resulting area is about 50% of the one of the FB2.

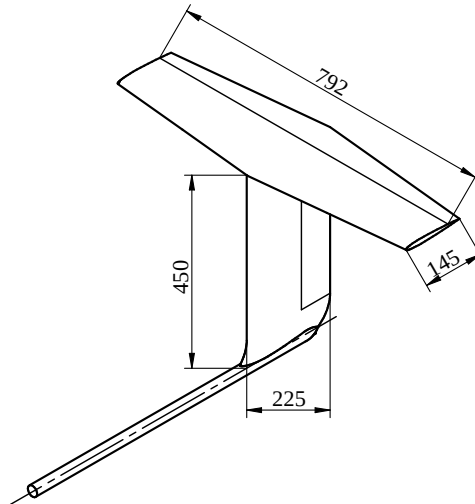


Figure 6. Tail sketch

In order to facilitate the tail construction, the vertical stabilizer chord was chosen equal to the horizontal stabilizer root chord. The height has been maximized up to the dimension allowed by the transportation box. The resulting rudder area has been judged sufficient to ensure better yaw control.

3.4 Stability Analysis

Once the geometries of both wing and tail were defined, the last parameters that had to be defined were the angle of incidence, the location of the horizontal stabilizer, and the location of the center of gravity of the payload. The evaluation of these parameters was based on static stability considerations. Two approaches were used in order to perform the stability analysis of the model: the first exploit the results given by XFLR5 and the second one is based on the results of a previously in-house developed Matlab® routine. The development of an algorithm to perform the stability analysis was useful to reduce the computational time during the design process, where several configurations had to be evaluated. In this way a rough model was used in order to limit the configurations that in a second phase were computed with the XFLR5 software.

The main assumptions at the base of the developed model are:

- 2D model;
- airplane center of gravity located on the y-axis (see Fig. 7);
- turbulence and transition are not considered;
- linear evolution of the lift and momentum pitching coefficients;
- 3D effects evaluated with the corrections already used for wing angle of incidence choice;
- negligible momentum generated by the tail drag.

Once the aerodynamic centers were determined, the following reference systems were defined:

- *airplane reference system*, in which the y -axis, with the origin in the aerodynamic center of the wing, is defined by the line which connects that point with the aerodynamic center of the horizontal stabilizer, while the z -axis is orthogonal to it and points upwards;
- *wind reference system*, in which the x -axis has the origin in the aerodynamic center of the wing and direction parallel to the flow, while the t -axis is orthogonal to it;
- *body reference system*, in which the axis d is the line which passes through the center of the boom tail, while the other h -axis is orthogonal to it and passes through the leading edge of the airfoil.

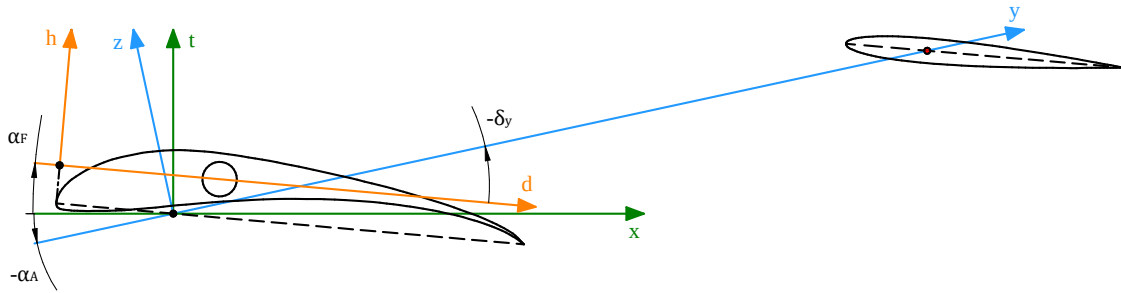


Figure 7. Reference systems used for the stability analysis

The T-tail configuration requested an improvement of the previous Matlab® routine, because that version assumed that α_A was small and, with this assumption, many simplifications could be made. By making the appropriate corrections to the previously developed theory, it was possible to obtain the following formula:

$$y_{pn} = \frac{(1 - k_\varepsilon) C_{L,\alpha}^T A_T}{C_{L,\alpha}^W A_W + (1 - k_\varepsilon) C_{L,\alpha}^T A_T} b \quad (3.9)$$

where k_ε is defined as

$$k_\varepsilon = \frac{d\varepsilon}{d\alpha_W} \quad (3.10)$$

The main difficulty using this formula is the correct determination of the k_ε coefficient. An experimentally derived formula has been employed to define k_ε for the simulations; the simulated results in straight flying conditions have been then compared with the flight results. This has been achieved by measuring the air speed with a Pitot tube and by looking at the angle of incidence of the horizontal stabilizer in the flight tests.

The common parameter that is used for the evaluation of static stability is static margin defined as reported below:

$$m_s = \frac{x_{pn} - x_{cg}}{c_w} \quad (3.11)$$

where

$$x_{pn} = y_{pn} \cos \delta_y \quad (3.12)$$



In order to have as little downforce as possible on the horizontal stabilizer and, therefore to carry as much as possible payload, the center of gravity should be near the center of the pressure of the wing. On the other hand, as the center of gravity is closer to the wing pressure center the static margin decreases. In order to regain static margin the tail boom could be extended, but this would require a heaviest structure, introducing also problems to get an acceptable rotating angle during take-off.

It is clear that the parameters we are looking for are related. Matlab® and XFLR5 were used to find the best trade-off among the following parameters:

- center of gravity position;
- distance between the leading edge of the wing and the leading edge of the horizontal stabilizer;
- horizontal stabilizer angle of incidence;
- static margin.

Finally, the configuration selected is characterised by a distance of 1500 mm between the two leading edges (wing and horizontal stabilizer), an horizontal stabilizer with 0° of angle of incidence, and with the possibility to vary the payload position in order to guarantee a minimum value of the static margin of about 8% (control and manoeuvrability considerations), see Fig. 13.

4. Structural Design

4.1 Wings

The FB2 aircraft (ACC2015) wings were designed with a lightweight wing skin supported by an internal structure composed by ribs and a single wing spar. Depron® sheets kept in shape by ribs of the same material made the wing skin. The wing spar was made by straight carbon fiber reinforced tubes. A rigid constrain between skin, ribs and spar coupled with the fact that the skin has a lower stiffness with respect to the spar, allowed to assume that the whole wing load is transferred to the spar. This wing construction allowed us to obtain a particularly light wing. The experience acquired during the numerous flight tests, highlighted that the attended performances were not achieved. This fact was attributed principally to the high flexibility of the Depron® skin and to the poor surface finishing guaranteed by this material. It was observed that skin deformations modify the wing airfoil especially in the trailing edge region (thinner thickness), changing in fact the airfoil behavior. Starting from this experience, the team decided to improve this wing design finding new solutions for stiffen the wing skin.

4.1.1 Wing skin

A good compromise between lightweight and stiffness was found in composite materials. A campaign of experimental tests was performed to define the appropriate configuration, in terms of number and type of layers of this composite material. The results of the tests led to the definition of the sandwich configuration composed by GF49-PVC-GF49 (plain glass fabric 49 g/m² - PVC foam sheet 1.2 mm - plain glass fabric 49 g/m²). The construction technique used (see Chapter 5) allowed to achieve the desired standards in terms of wing stiffness and surface finishing, against an increase of weight of about 100% with respect to the Depron® configuration.

As previously mentioned the wing skin has not structural functions. However, a determination of deformation failure criteria is needed in order to avoid buckling phenomena. If the radius of curvature r of the wing skin is lower with respect to the critical radius of curvature r_{cr} , the wing skin fails, hence the failure criteria adopted is:

$$r < r_{cr} \quad (4.1)$$

Since an analytical characterization of a composite material is a hard challenge, an experimental procedure was adopted.



Figure 8. Static load test for the determination of critical deformation (radius of curvatre).

Tests was performed on two sample wings made of GF49-PVC-GF49 with 450 mm wingspan in order to evaluate the critical radius of curvature of the composite sandwich.

The wing skin was supported on the tips and loaded in the center in correspondence of the cutting center of the section in order to avoid torque stresses. The central displacements were measured for each load configuration making use of a displacement comparator placed as reported in Fig. 8. The evaluation of the radius of curvature is performed with the strong assumption that the wing skin behavior is like an isotropic and homogeneous beam.

The analytical displacement solution for supported beams loaded in the center is reported below:

$$f = \frac{pL^3}{48 EJ} \quad (4.2)$$

Knowing the displacement f , the bending radius can be calculated by the formula:

$$r = E \frac{J}{M} = \frac{pL^3}{48 f J} \frac{J}{p \frac{L}{4}} = \frac{L^2}{12 f} f = \frac{pL^3}{48 EJ} \quad (4.3)$$

Figure 9 reports the measured displacement and the relative computed bending radius for both wing skins resulting from multiple tests at increasing or decreasing load conditions. Although the data have a remarkable scatter in the region of small loads, they allow to determine the minimum sandwich bending radius that turned out to be about 3.75 m.

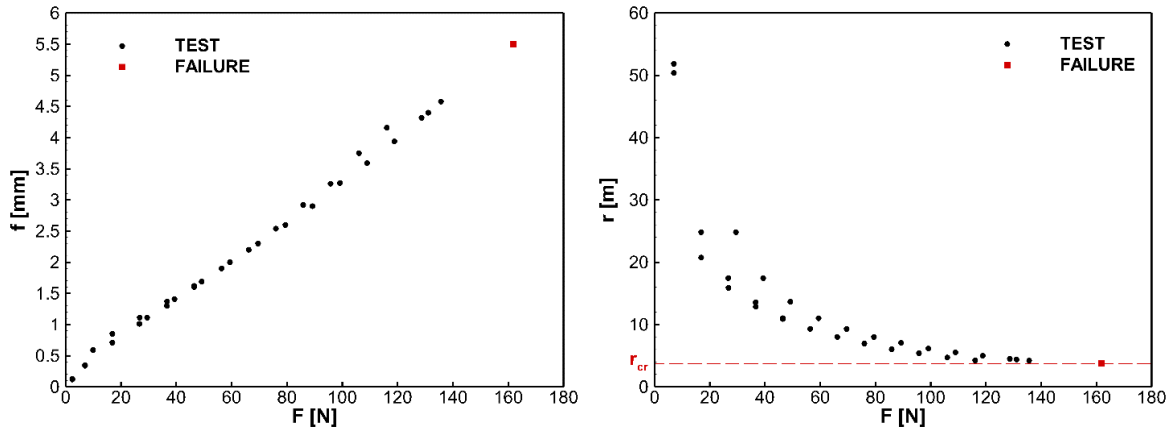


Figure 9. Results of the load tests on the wing skin.

4.1.2 Wing spar design

From the collaboration between the previous team and an external firm, operating in the fiber reinforced polymer composites field, it was gained the knowledge necessary to design and choose the laminating layers for the carbon fiber tubular beams.

In Tab. 2 the selected layer configuration is reported. The spar is made up of 5 layers of carbon fiber for a theoretical total thickness of 1.2 mm. Mechanical properties of the tubes were characterized as described in the following.

Table 2. Wing spar tubes characteristics

Nr.	Type and Orientation	Purpose
1	HS CF 150 μm @ 90°	to avoid ovalization
2	HM CF 300 μm @ 0°	for bending
1	HS CF 150 μm @ 0°	for bending
1	Biax CF 300 μm @ $\pm 45^\circ$	for torsion

The Young's modulus was evaluated by suspending one end of the tubes to a wire and measuring the natural bending frequencies with an impact test. The analytical solutions are available for the natural bending frequencies of uniform beams in free-free boundary conditions. Boundary conditions closer to the ideal can be achieved hanging the tubes to a long wire.

The shear modulus was determined with a similar impact test: two equal, known and large rotatory inertias were fixed to both ends of the tubes. The tubes were suspended to a wire located in correspondence of the tube mid span. The natural torsional frequencies allowed to easily determine the G modulus.

The ultimate strength of the tubes was measured in simply supported boundary conditions by loading them at the mid span. The property of the tubes are listed in the table below:

Table 3. Wing spar tubes properties

E	82 000	MPa
G	7 850	MPa
σ_r	250	MPa
ρ	1 600	kg/m ³

The dimension of the transportation box and the designed wingspan, forced us to choose a 5-tubes telescopic solution and constrained the length of tubes: a larger diameter main spar (central core), two spars for the wing section and other two section for the tips.

The ACC2017 rules impose a structural validation test for safety reasons to simulate a load during flight (simulating $\sim 2.5 g$ during tight turns). The structural test consists of lifting by hands from the tips the fully loaded aircraft before each flight. Therefore, for the design of the dimension of the tubes, the wing spar was modeled like a supported beam as shown in Fig. 10.

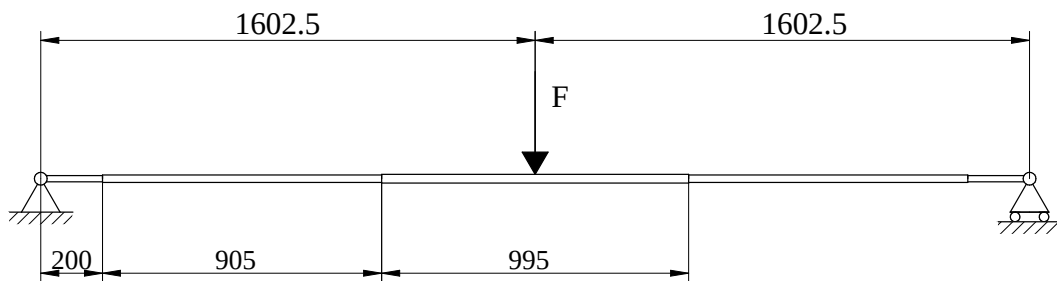


Figure 10. Wing spar static load test

By using a finite element model implemented in Matlab®, the characteristic dimensions of the tubes were defined. The program provided bending radius, tensile stress and safety factor of each part of the spar sections. The diameter of the main tube was selected to allow a safety factor of 1.5, then the other tubes diameter were verified in consequence. All tubes respect the condition $r > r_{cr}$ and consequently the buckling of the wing surface is avoided.

Table 4. Wing spar tubes dimensions

	Tip	Wing	Central Core
Length [mm]	200	905	995
External diameter [mm]	21.4	24.4	27.4
Bending radius [m]	23.9	6.5	6.5
Safety Factor	6.8	1.6	1.5

A confirmation of the validity of the design hypothesis adopted is given by the comparison of the stiffness of the wing skin sandwich and beam. It resulted that the ratio between the stiffness of the sandwich and the spar is below 0.1.

4.1.3 Wing Flutter Analysis

The flutter validation of wing spar is performed by means of a Matlab® code. This code is based on the verification criteria found in J.P. Den Hartog "Mechanical vibrations" (Dover Publications Inc. 1985), where the limit flutter velocity can be computed as:

$$v^2 < \frac{2 k_{\theta} + 2 k_u \left[I_g/m + (x_g - x_{ct})^2 \right]}{C_{L,\alpha}^W A_W \rho (x_g - c/4)} \quad (4.4)$$

With some simplifications ($c/4 < x_g < c/2$ and $I_g/m \cong c^2/24$) we came to the formula reported below:

$$v^2 < \frac{8 k_{\theta} + 8 k_u \left[c/24 + (x_g - x_{ct})^2 \right]}{C_{L,\alpha}^W A_W \rho} \quad (4.5)$$

In the configuration adopted, the computed maximum flutter speed is about 36 m/s. Experimental tests were performed during flight sessions. Diving paths were repeated in order to fly at about the double of the maximum model speed that can be reached during a competition flight. No flutter of the wing structure occurred.

4.2 Tail

A full carbon configuration was chosen for the tail. Due to the lower loads with respect to the wing, and in order to guarantee the best surface finishing, the following layers configuration was adopted:

Table 5. Tail skin characteristics

Nr.	Type and Orientation	Purpose
1	CF 65 μm @ $\pm 45^\circ$	for torsion
1	CF 65 μm @ 90°	for bending
1	GF 49 μm @ $\pm 45^\circ$	for filling

The only carbon skin is not enough to guarantee the expected stiffness, so a spar and three ribs made up of a CF65-PVC-CF65 sandwich (Aero carbon fabric 65 g/m²- PVC foam sheet 3 mm - aero carbon fabric 65 g/m²)structure were added to increase the moment of inertia of the section, providing better stiffness to bending and torque. The spars are made of a CF commercial tube with 6 and 8 mm internal and external diameter, respectively. Once manufactured, both stabilizers were verified with a static load test.

The dimension of the transportation box and the position of the tail with respect the wing, forced the choice of a telescopic solution with 3 tubes.

As already mentioned about the wing spar, the carbon fiber layers configuration adopted is the one reported in Tab. 6.

Table 6. Tail boom tubes characteristics

Nr.	Type and Orientation	Purpose
1	HS CF 150 μm @ 90°	to avoid ovalization
1	HM CF 300 μm @ 0°	for bending
1	Biax CF 300 μm @ $\pm 45^\circ$	for torsion

Table 7 reports the lengths and diameters of the tubes chosen and the relative safety factor.

Table 7. Tail boom tubes dimensions

Parameter	Cross fuselage	First tail tube	Second tail tube
Length [mm]	350	420	750
External diameter [mm]	28	26.5	25
Safety Factor	2.9	3.3	4.6

The safety factors were calculated assuming a cantilever model, as shown in Fig. 11, where the force F is the maximum estimated lift force generated by the elevator (about 25 N).

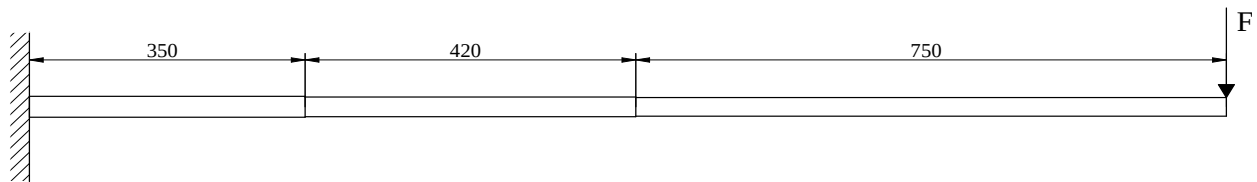


Figure 11. Tail boom cantilever test

An experimental verification about possible flutter instabilities of the tail was performed. The tail was located at the outlet of a wind tunnel as reported in Fig. 12. The tail boom is fixed in such a way to replicate the cantilever constrain. Different flight velocities were replicated in the range within 2 and 36 m/s. These tests showed that no flutter of the tail structure occurred.

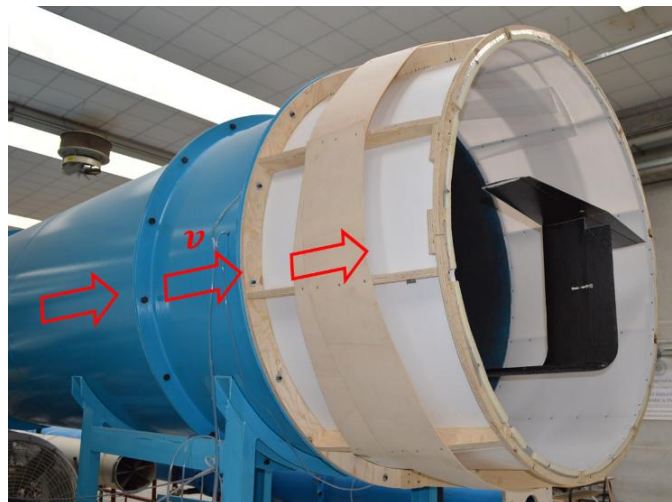


Figure 12. Tail flutter test

4.3 Fuselage, cargo-bay and engine pod

The fuselage of the aircraft is made up of two structures linked by a cross joint between wing spar, engine boom and tail boom (see Fig. 13). The cross joint is an additional construction complication but it was chosen with the aim to not overlap engine, tail and wing beams, thereby minimizing the frontal section of the aircraft. The cross joint is also responsible for distributing the loads from the cargo-bay to the wing beam and ensures the coupling between cargo-bay and wings.

The cargo-bay structural design and consequent construction must fulfill many requirements:

- minimal dimensions required by the ACC rules;
- allow an easy and fast payload loading;
- it must be able to withstand the acceleration forces on the payload during the flight;
- in the present configuration, it must be able to house the landing gear and therefore resist to the landing loads;
- minimal cross section and aerodynamic shape in order to limit drag.

With these requirements in mind, the final design of the cargo-bay is essentially made by two flat walls emerging from the wing pressure side and a horizontal bottom plate joined together by means of bulkheads and cross bars, Fig. 13.

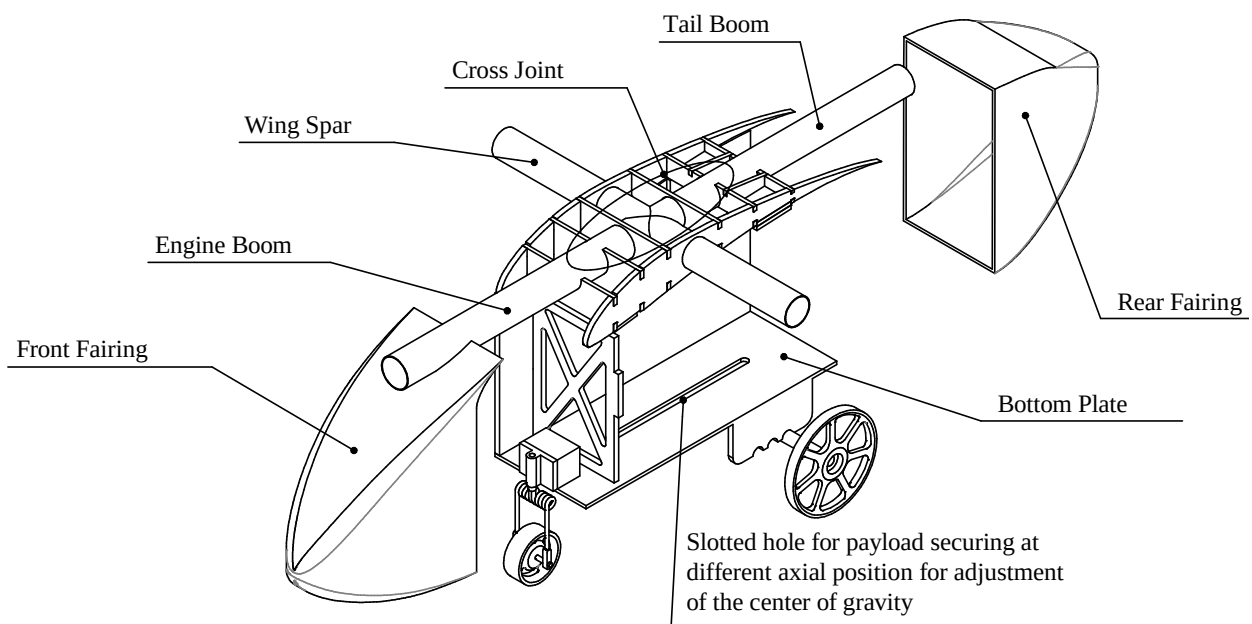


Figure 13. Cargo-bay and Cross Joint detail

The whole structure is made by a CF160-PVC-CF160 (Aero carbon fabric 160 g/m²- PVC foam sheet 3 mm - aero carbon fabric 160 g/m²) sandwich. Only some components that are critical from a structural point of view (e.g. main landing gear supports, connection to the fuselage) are additionally reinforced by 4 mm thick birch plywood.

The outer portions of the cargo-bay (frontal and rear fairing) have only an aerodynamic function and for this reason are made of a very thin vacuum molded layer of glass fiber and epoxy resin.

The payload loading will be done by the rear of the cargo-bay, where the fairing is fixed with four magnets making easy and fast the loading operations. The bolts that keep in position the payload grips on the bottom plate and can be moved along the airplane longitudinal axis inside a slotted hole, so allowing to set the payload position as requested by stability considerations.



The engine boom is composed by a pod and a carbon fiber tube. To avoid the occurrence of resonance phenomenon, the boom dimensions and structure were selected in such a way to ensure that its natural frequency is much lower than the frequency induced by propeller and motor spinning.

The engine pod must ensure the accessibility to a load bay where engine regulator, engine and RX system batteries have to be allocated. The only significant load that the engine pod must be able to sustain is the propeller torque. It was therefore decided to manufacture this component again with the vacuum molding technique applied on a layer of CF (93 g/m²) and epoxy resin. A 3 mm thickness sandwich of CF93-PVC foam-CF93 is used for internal bulkheads that are adopted to improve torsional stiffness, to support the engine and to connect the engine pod to its tube.

5. Airplane construction technique

As already commented in the previous chapters, the team decided to build the aircraft model by using composite materials based on the following considerations:

- higher rigidity of the wing that allows to keep shape stability of the aerodynamic profile, especially close to the thin trailing edge region;
- excellent surface finishing are achievable;
- possibility to manufacture shapes of multiple or complex curvatures in order to obtain aerodinamically smooth coupling surfaces between the different airplane components (e.g. coupling of wing with tail boom or cargo-bay)

In this chapter, some details about the construction technique or solutions that have been adopted are briefly described.

Wing molds are made from aluminum ingot machined by a 4 axis CNC milling machine in the Manufacturing Techniques Laboratories of Udine University. The choice of aluminum as material was necessary because the molds had to be used many times without damaging. The team practiced on the vacuum molding technique and tested different types of composite structures on this molds before to gain sufficient confidence and identify the optimal layout of the composite structures. Beside these aspects, aluminum as material for molds can guarantee other advantages: dimensional stability and a very good surface finishing of the molded object.

The two sandwiches that made the wing (pressure and suction side of the airfoil) are joined together with the use of a limited number of ribs. These latters are made of 6mm thick Depron®, with the exception of the ribs at the edges of the three wing sections that are made of a sandwich of CF (93 g/m²) and PVC foam (3 mm). The two wing parts are glued together along the leading and trailing sides and on the ribs thickness by means of epoxy resin mixed with glass hollow microspheres in order to increase the volume of the resin while maintaining practically unchanged the weight.

The use of this mixture ensures also the required stiffness of the trailing edge portion of the wing, where a layer of CF65 is also inserted in the sandwich for the same purpose.

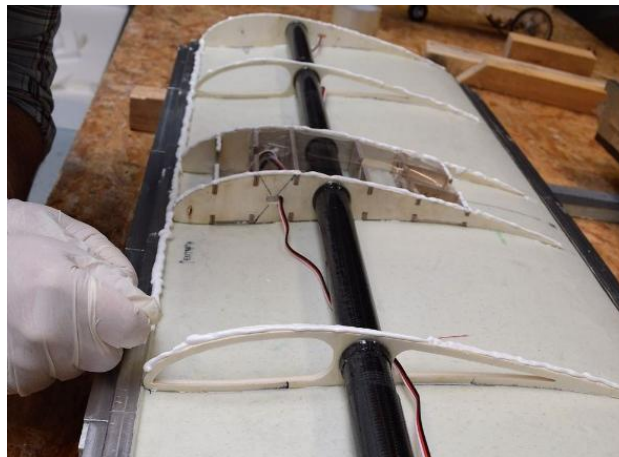


Figure 14. Wing internal structure

Another detail of the construction that deserve some attention is the technique that has been used to realize the ailerons hinges, a skecth of which is reported in Fig. 15.

The hinges are made using a peel ply strip inserted in the most external layer of the wing suction side sandwich. Peel ply is a special fabric that remains flexyble at the end of the molding process.

A small gap is opened underneath the hinge on the pressure side. By doing so, the surface is kept aerodynamically smooth on the suction side of the profile that is the most critical for flow transition and separation.

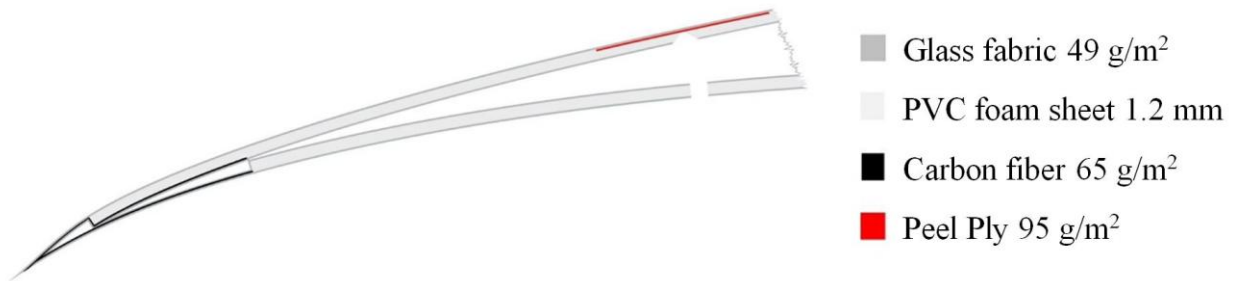


Figure 15. Hinges and trailing edge detail

Wingtips are manufactured as a separated part from the outer wing sections. Since they have substantially no aerodynamic load they are made by a composite CF93-epoxy skins coupled with an internal structure of ribs and spars made out of Depron®. In order to be able to perform the required structure validation test, dedicated supports to be placed under the wingtips have been designed and manufactured.



Figure 16. Vertical stabilizer internal structure

The molds to manufacture both empennages have been machined out from blocks of MDF since this material is easily available at a limited cost and it is also easy to mill with the small CNC machine that has been acquired for the purpose. In order to accomplish the task to realize the molds, the team had to gain the required skill in 3D surface CAD modelling and CAM conversion to generate the required CNC path programs.

6. Electric and electronic components

6.1 Battery capacity

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

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

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

$$C = 1000I \frac{\Delta t}{60} \quad (6.1)$$

where $C [mAh]$ is the battery capacity, $I [A]$ is the current to the engine and $\Delta t [min]$ the flight time.

Assuming to run the engine at max throttle for all the flight time and considering an average current of 50 A (as a precaution 10% higher than the value measured at the propeller thrust test bench) the resulting battery capacity tuned 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 50C Gold V2 (continuous discharge: 50 A, peak discharge 80 A).

Concerning the battery that supply the on-board RC system, the regulation requires a battery capacity of at least 600 mAh. This imposed lower bound is largely sufficient to cover the estimated flight time.

6.2 Servomotors sizing

The calculation of the required torque for each of the servomotor that move the aerodynamic control appendices (i.e. rudder, elevator and ailerons) was based on the following assumptions:

- the angle of incidence of the wing, stab, or fuse is zero (relative to the airflow) and the airflow is considered as non-compressible flow;
- control linkages have zero offset at hinge line and are perpendicular to horns at neutral;
- the pushrods are longer than the servo and control horns;
- control mechanisms are frictionless and surfaces are mass-balanced;

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

$$M = \frac{1}{4} \rho v^2 c_c^2 L_c \sin \alpha \frac{\tan \alpha}{\tan \beta} \quad (6.2)$$

where we considered $\alpha = 35^\circ$ and $\beta = 60^\circ$.

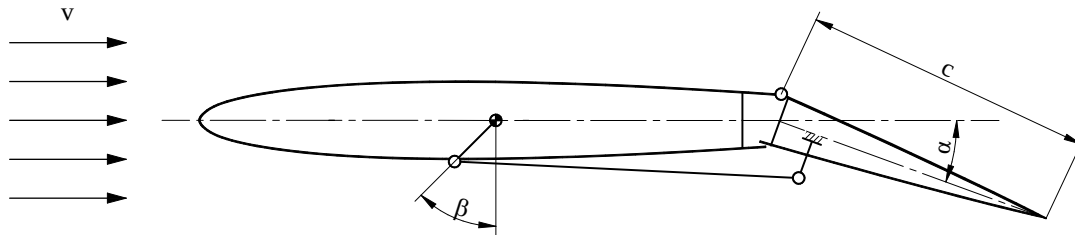


Figure 17. Sketch of the parameters considered for the servo torque calculation

This equation under the assumptions above listed, should generally yield conservative predicted torques (except the last assumption). The resulted required torques are reported in Tab. 8 for a flight speed of 20 m/s (assumed prudently higher than the design speed of 15 m/s).

Table 8. Servomotors required torque

Control surface	Average chord [m]	Length [m]	Torque [kg·cm]
Aileron	0,080	0,8	1,48
Rudder	0,076	0,347	0,58
Elevator	0,043	0,78	0,42

The chosen servomotor are HS-5125MG (Hitec) with 3.0 kg·cm of torque for the rudder and elevator, while DM4082M (FullPower) with 8.2 kg·cm of torque are used for the ailerons. All the four servos installed in the airplane have a torque far above the estimated ones.

7. Payload prediction

Payload prediction chart is based on data from test flights and propeller thrust 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 airplane take-off weight, allow us to determine experimentally the actual lift coefficient C_L , 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 regulation requires that payload prediction have to be calculated as a function of the air density, with the following linear approximation:

$$P = 17.52\rho - 9.78 \quad (7.1)$$

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

The prediction was also verified in occurrence of flight tests with different density altitude, as shown in Fig. 18.

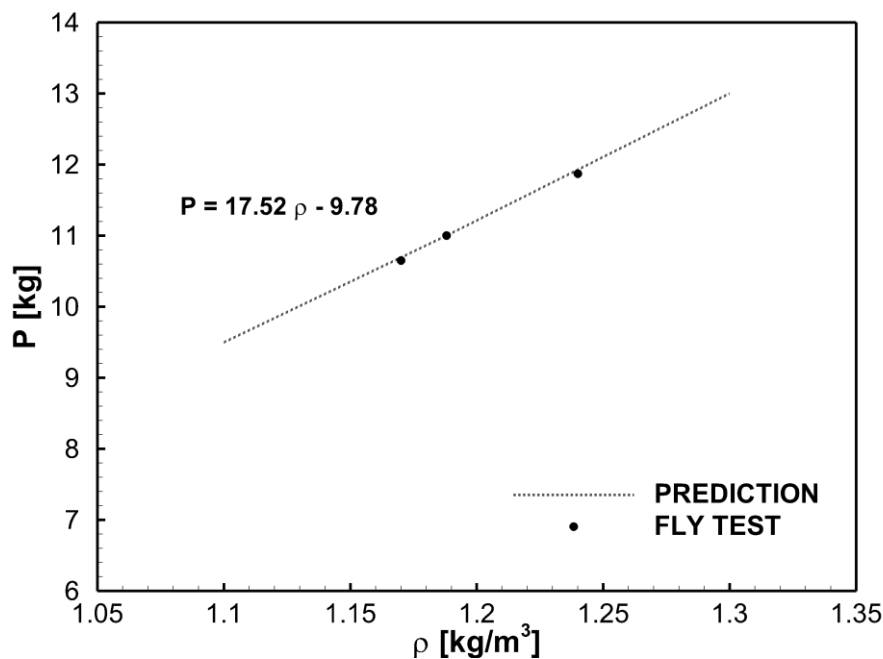


Figure 18. Payload prediction curve as a function of air density



8. Outlook

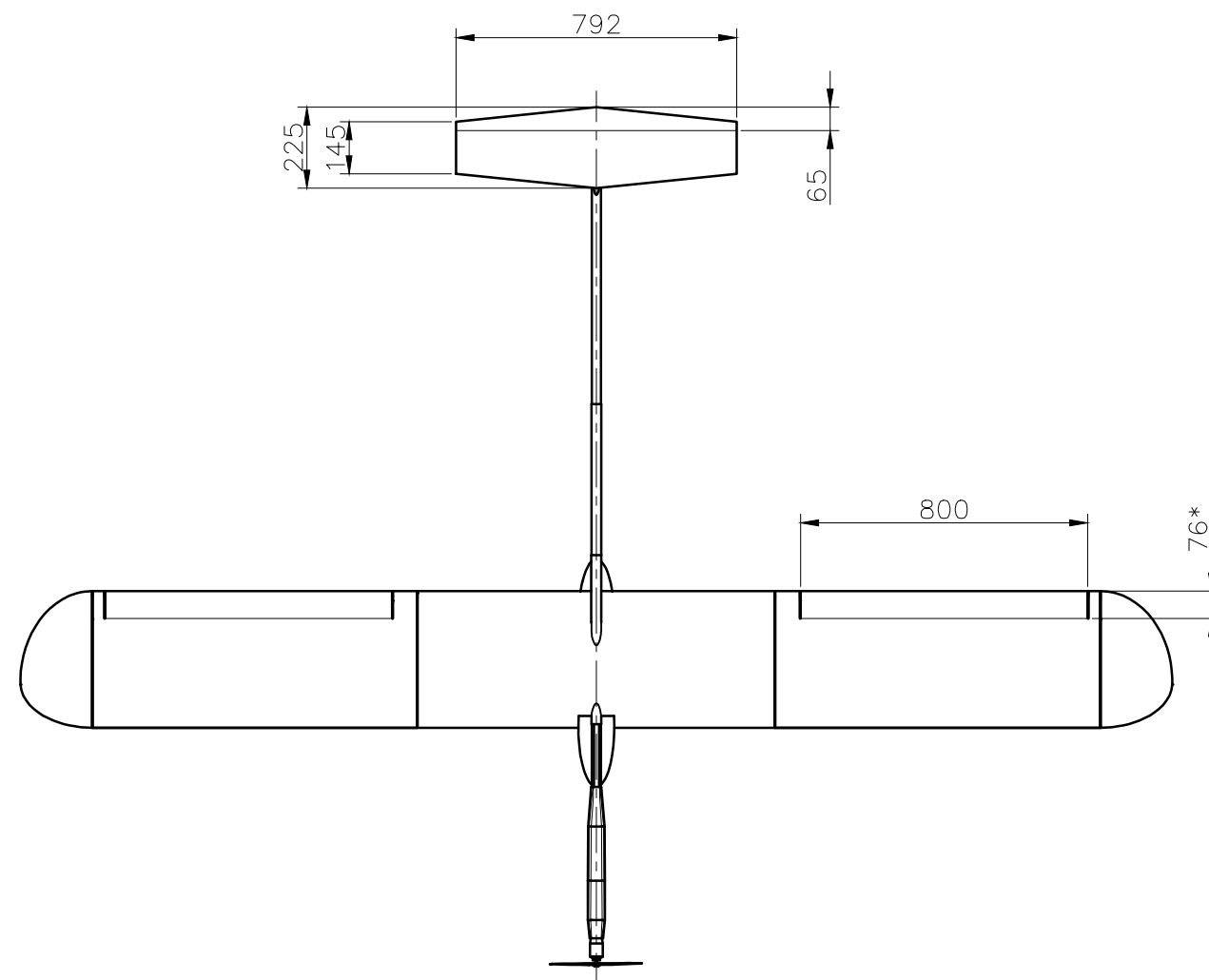
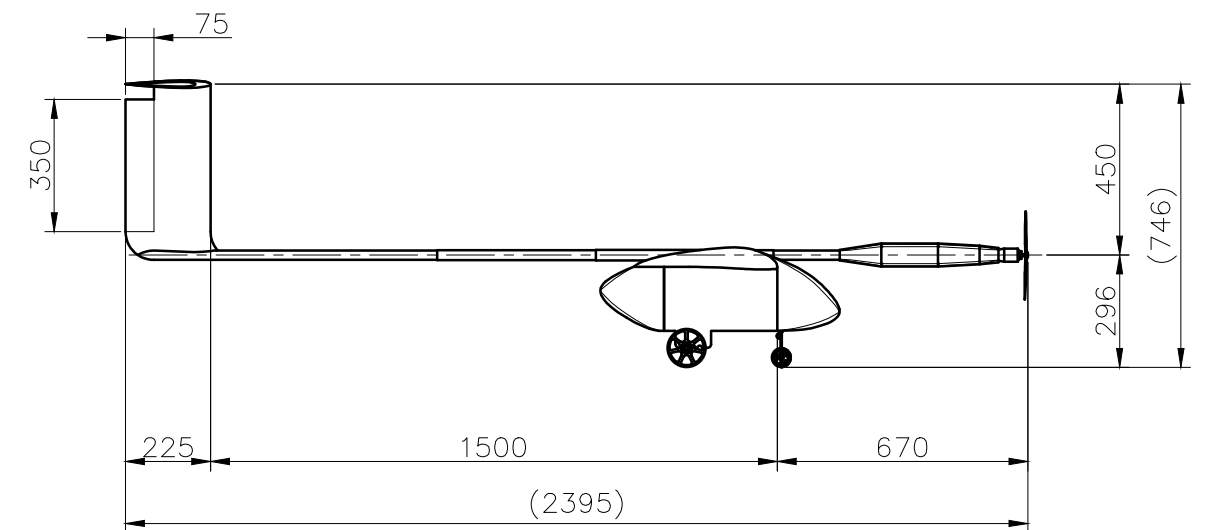
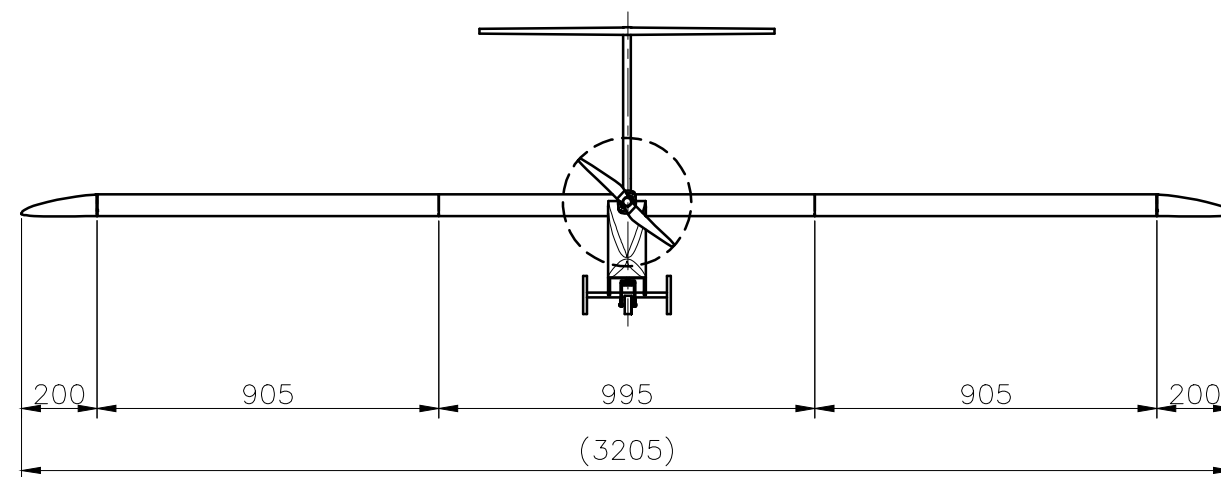
A first version of the competition airplane was already available by the end of April. Flight tests confirmed the design performances while highlights that some improvements are still possible by a fine-tuning of the aircraft.

Therefore, in the forthcoming months, an intense flight tests program has been defined in order to find the best settings of a number of possible small adjustments such as aircraft CG, horizontal stabilizer stagger angle. By relying on the pilot skills and feeling about the airplane behavior, and by simulations of the race circuit, the aim is to further optimize the performances. Particular care will be also put in the definition of the best flight trajectories to keep during the turns at the end of each leg.

The team will also revise the manufacturing process to investigate on the possibility to build a second model lighter than the actual one.

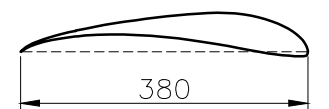
ATTACHMENT – A

Drawings

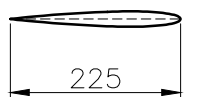


Airfoil – SCALE 1:10

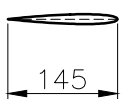
Wing airfoil
(S1223)



Vertical and Horizontal stabilizer
airfoil at the root
(NACA0009)

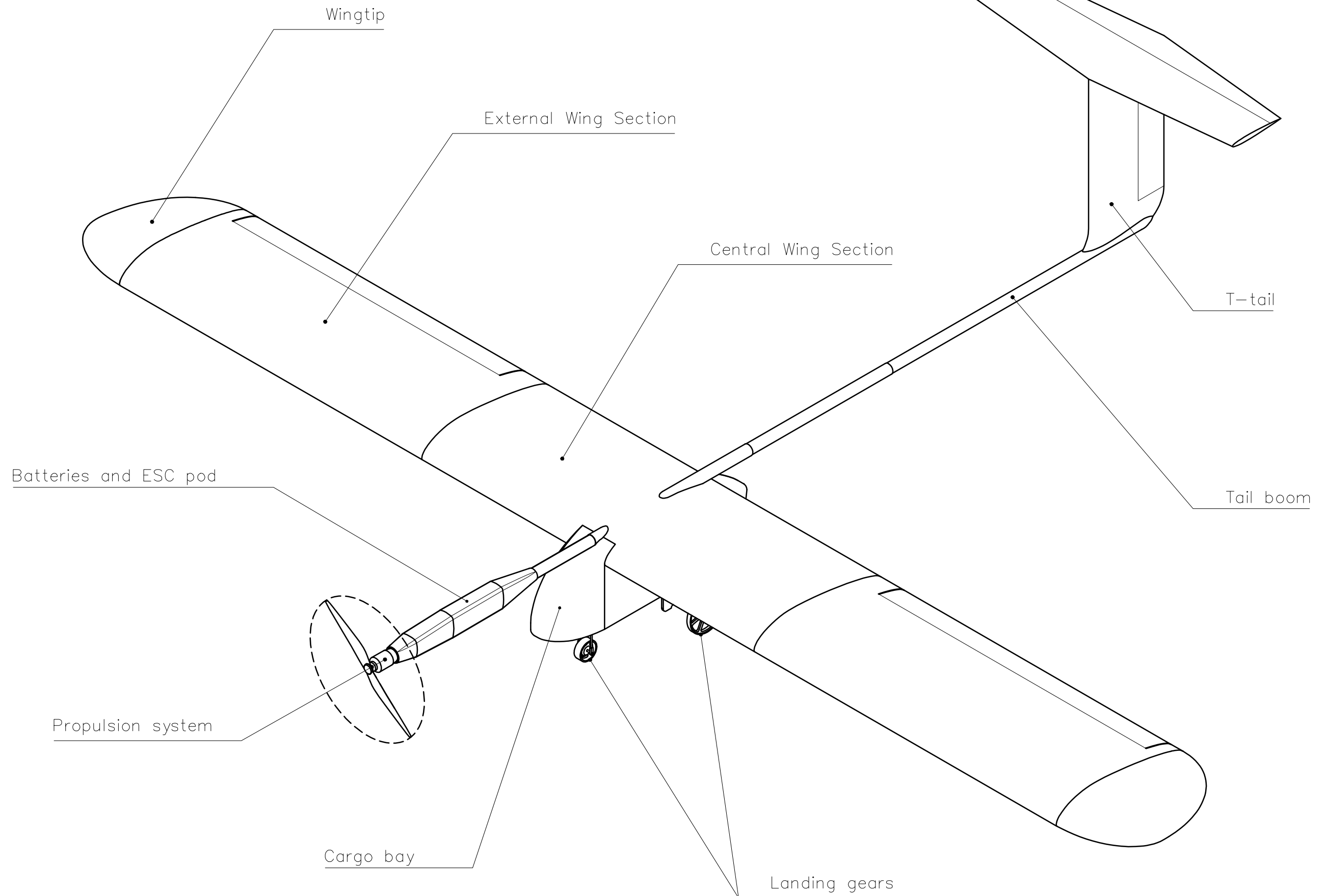


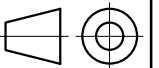
Horizontal stabilizer airfoil at the tip
(NACA0009)

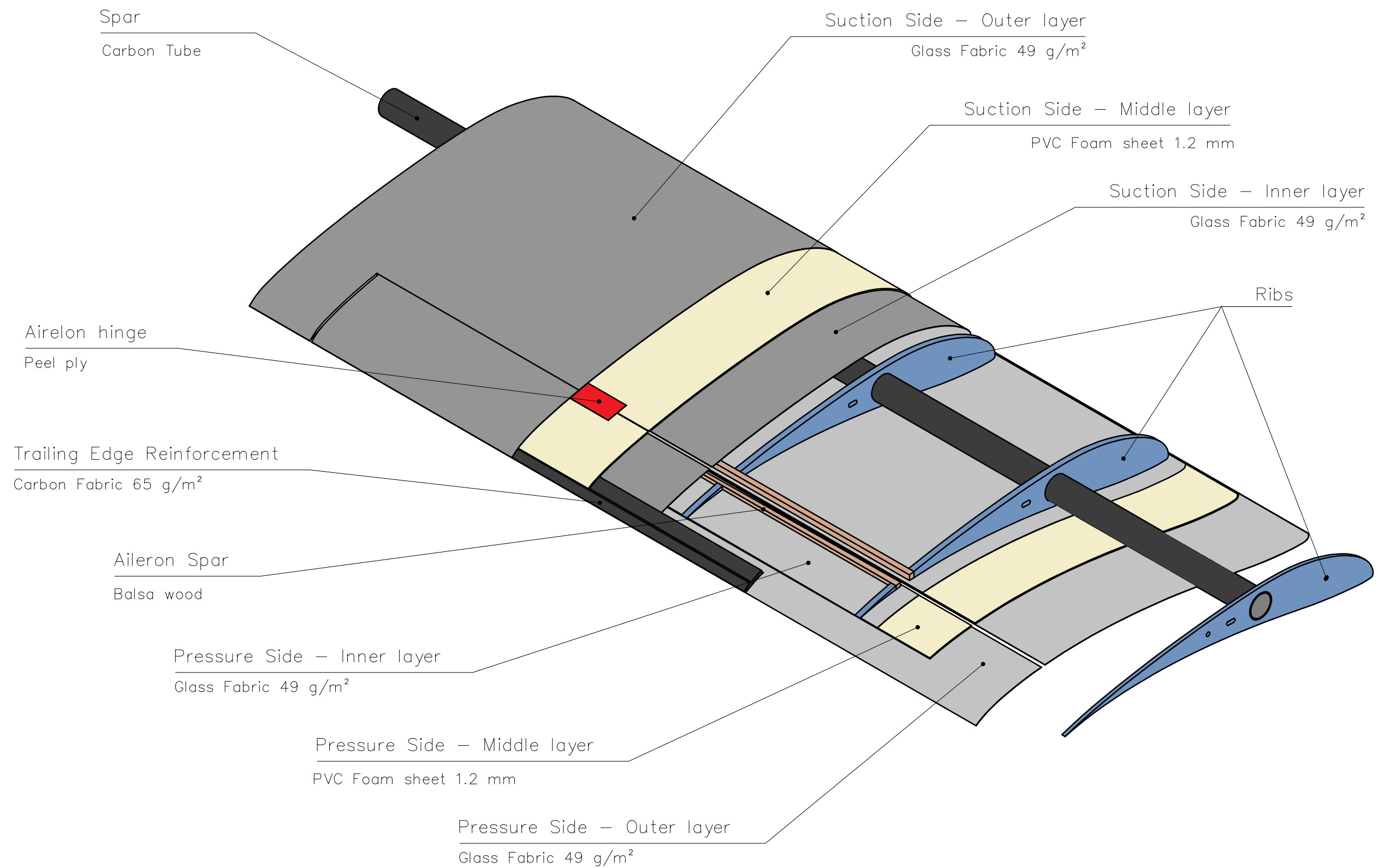


Surface	Area	Surface	Area
Horizontal stabilizer	0.147 m ²	Rudder	0.0263 m ²
Elevator	0.0356 m ²	Wing	1.183 m ²
Vertical stabilizer	0.0985 m ²	Aileron	0.0608 m ²
TEAM'S UNIVERSITY University of Udine		GENERAL TOLLERANCES UNI EN-ISO 22768 c	
		TEAM'S NAME / NUMBER AeroUD-MET / N.07	
		DATE 29/05/2017	
TITLE 3-VIEW DRAWING		SCALE 1:20	
DESCRIPTION - front view at the top left corner - top view below - starboard view on the right			

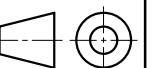
(*) length measured on the wing airfoil chord

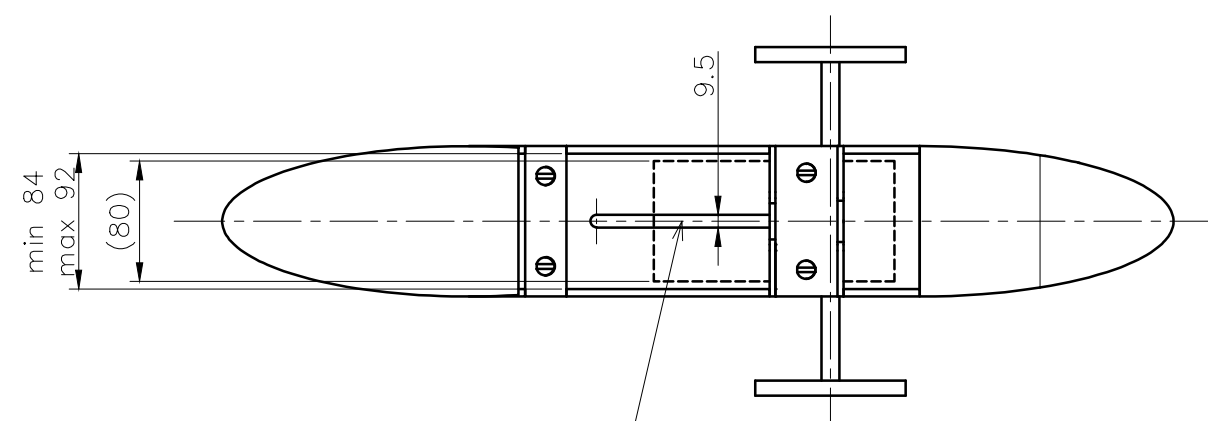
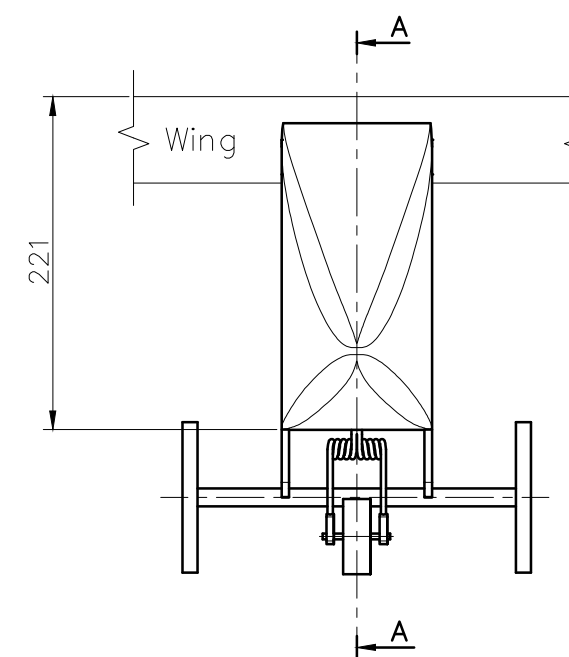
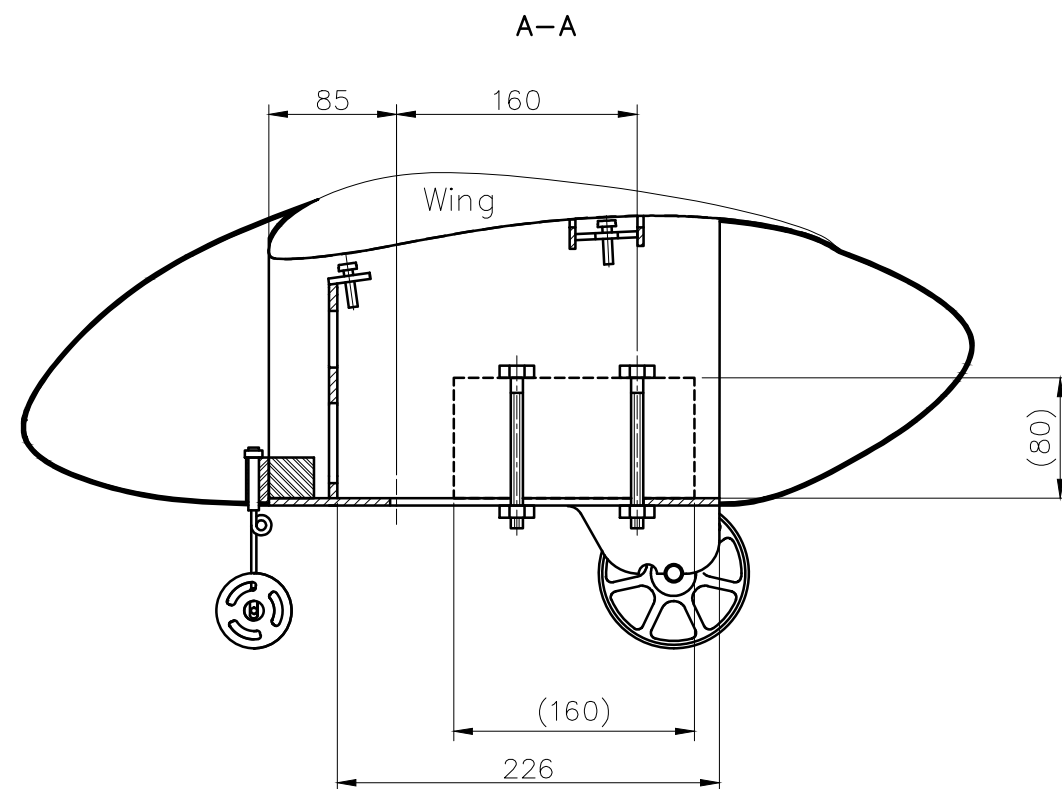


TEAM'S UNIVERSITY University of Udine		GENERAL TOLLERANCES ---	TEAM'S NAME / NUMBER AeroUD-MET	
	TITLE ISOMETRIC PERSPECTIVE VIEW		DATE 29/05/2017	
	DESCRIPTION Isometric perspective view with indication of the main airplane elements		SCALE 1:10	

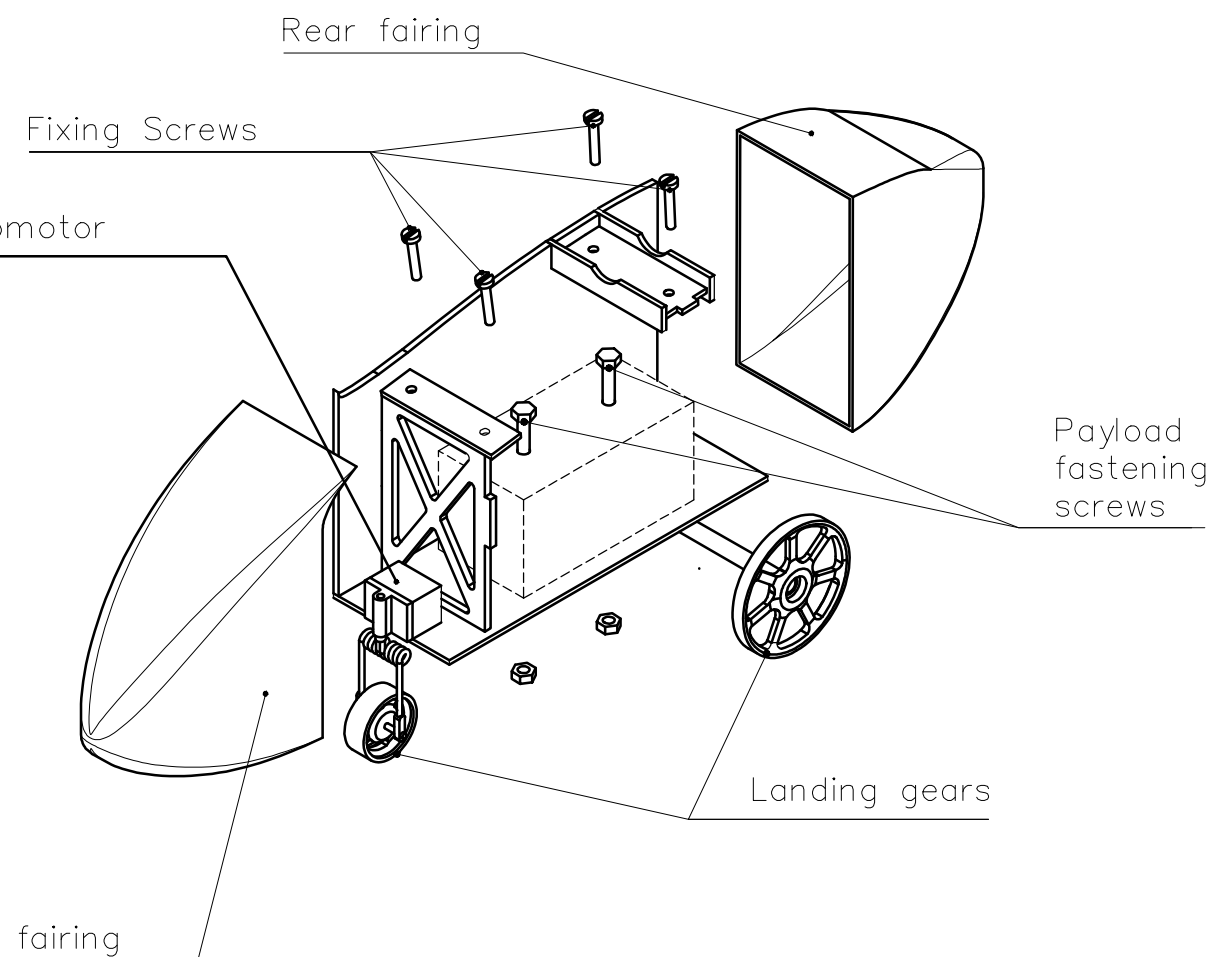


TEAM'S UNIVERSITY University of Udine	GENERAL TOLLERANCES ---	TEAM'S NAME / NUMBER AeroUD-MET
	TITLE Laminate Layers of Wing	DATE 29/05/2017
	DESCRIPTION Wing laminate layers and some glimpses of the internal structure	SCALE 1:5

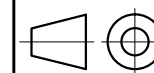




Slotted hole for payload securing at different axial position for adjustment of the centre of gravity.



TEAM'S UNIVERSITY University of Udine	GENERAL TOLERANCES UNI EN-ISO 22768 c	TEAM'S NAME / NUMBER AeroUD-MET / N.07	
	TITLE CARGO BAY - REV.1		DATE 24/07/2017
	DESCRIPTION Cargo bay location and principal dimensions On dashed line: orthogonal volume of 160x80x80 mm for technical inspection		SCALE 1:5



ATTACHMENT – B

Team Members and Guests Info



TEAM 07: Participants

NAME	SURNAME	GENDERS	DATE OF BIRTH	T-SHIRT SIZE	TEAM ROLE	SIGNATURE
Riccardo	Siege Brusattin	male	05/12/1990	XL	leader	<i>Ric. Siege Brusattin</i>
Fabio	Di Lenarda	male	08/07/1993	L	member	<i>Fabio Di Lenarda</i>
Gianluca	Flora	male	27/04/1990	L	member	<i>Gianluca Flora</i>
Cristian	Demarchi	male	15/01/1994	M	member	<i>Cristian Demarchi</i>
Michele	Burella	male	10/10/1990	L	member	<i>Michele Burella</i>
Lucrezia	Vacchiano	female	13/11/1993	M	member	<i>Lucrezia Vacchiano</i>
Luca	Casarsa	male	18/12/1974	M	pilot	<i>Luca Casarsa</i>
Andrea	Lorenzon	male	09/11/1991	S	guest	<i>Andrea Lorenzon</i>
Fabio	Pagnacco	male	06/12/1989	M	guest	<i>Fabio Pagnacco</i>

Team Leader
Riccardo Siege Brusattin

Pilot
Luca Casarsa