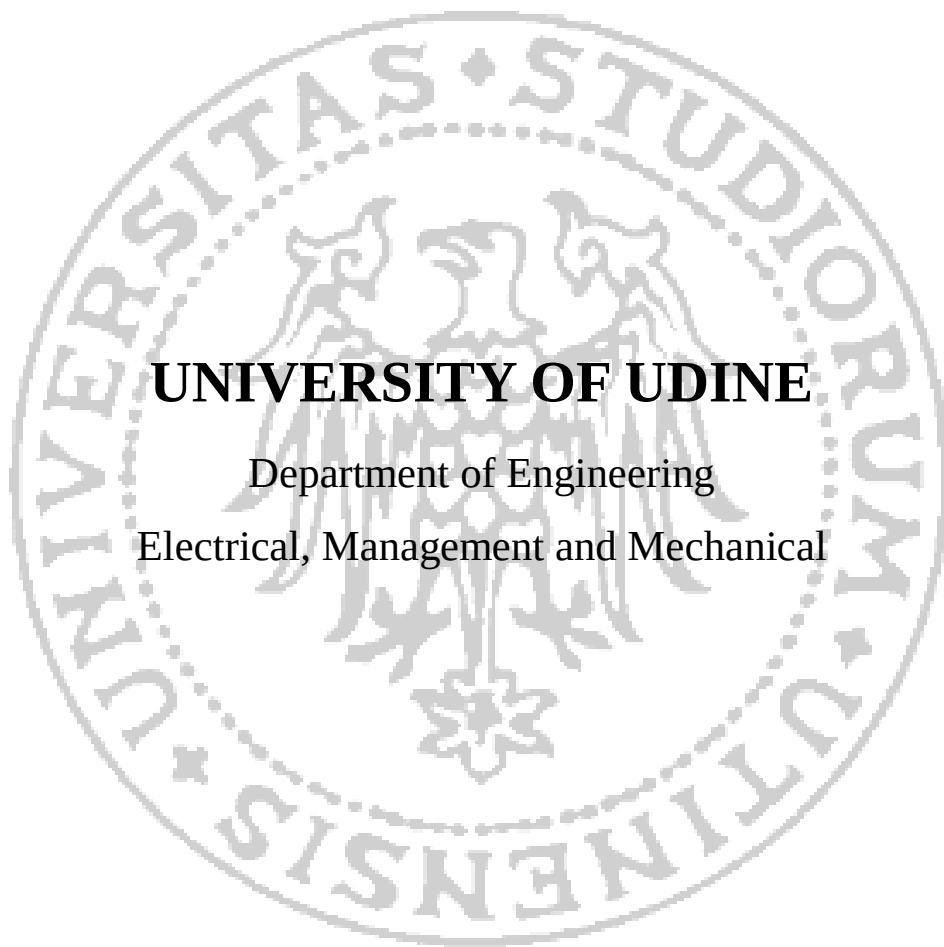


Air Cargo Challenge 2015

Stuttgart



UNIVERSITY OF UDINE

Department of Engineering

Electrical, Management and Mechanical

REPORT

Team 04

Index

1. Introduction	3
2. Project management	4
2.1. Financial budget	4
2.2. Time schedule	5
3. Aerodynamic design	6
3.1. Aircraft overall architecture	6
3.2. Propeller test	7
3.3. Airfoil	8
3.4. Stability analysis	10
4. Structural design	13
4.1. Flutter analysis	15
5. Electric and electronic components	18
5.1. Battery capacity	18
5.2. Servomotors size	18
6. Payload prediction	20
7. Outlook	20
References	22
Attachment – A: Drawing	
Attachment – B: Team Members and Guests Info	

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 reasonable chance to be able to participate in this kind of competition.

Despite this original handicap and some misadventures which happened during the development and testing of the first aircraft model, the team could finally participate in the 2013 SAE Aerodesign East competition, ranking fourth in the regular class.

This result was beyond expectations and so the team was motivated to go on in this kind of experience, which proved to be positive both from technical and human point of view.

The choice of a further competition was between a second time at SAE Aerodesign or a first time at Air Cargo Challenge. The second option has been chosen because the rules are significantly different from the previous one, thus configuring a fully new and challenging design and also because of reduced logistical problems and consequently of reduced budget.

The present team is officially composed by one professor (the pilot), two Phd students (who were also in the SAE Aerodesign 2013 team) and four master degree students. Two more students, who took their master degree in meanwhile, and the professor in charge have also helped in the development of the present project.

Being conscious of its own limits in aeronautical engineering, the team has adopted a prudent design philosophy and it has limited the design analyses to the essential issues of the project. In this perspective, a classical airplane architecture has been adopted and some typically aeronautical verifications, such as stability and flutter analyses, have been performed in a quite essential way.

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

2. Project management

2.1. Financial budget

The forecast for the required budget was initially based on the analysis of the previous experience in the SAE Aero Design competition, whose final cost turned out to be 20.000 Euros. One quarter of this budget came from sponsors and the rest was covered by the University of Udine. This total budget can be divided into two main parts: 6.000 Euros had been necessary to build up four airplanes (two prototypes and the two airplanes used in the competition) and 14.000 Euros were spent for the transfer of nine people (two professors and seven students) and two airplanes to the United States.

The site, where the Air Cargo Challenge 2015 will take place, is only few hundreds kilometres away from Udine and therefore the forecast for the logistic expenses can be greatly reduced.

The estimated budget for Air Cargo Challenge 2015 has been therefore fixed to about 10.000 Euros. Within this budget, the expenses to build up the airplanes are estimated to be about 6000 Euros: here again four airplanes will be constructed (one already made following the Air Cargo Challenge 2013 rules, a second one already finished to check the performances and to be used for the pilot's training, and the other two for the competition). The rest (4.000 Euros) should be enough to cover the team expenses during the competition. Up to now, this forecast seems to have been reasonable.

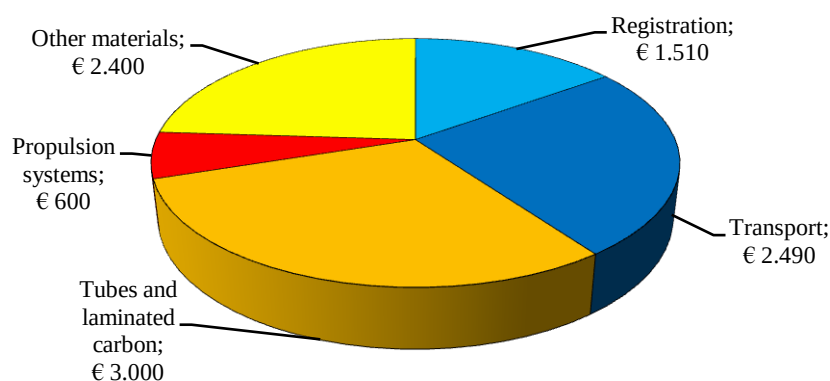


Figure 1 - *Distribution of total cost*

It has been clear from the early beginning that, because no restrictions concerning the allowed materials were fixed by the regulations, carbon fiber reinforced structures were the best solution. The team had no experience in using these materials and some of these structures could not be found directly on the market, like all other components. Thus, it was decided to design and construct the main carbon fiber structures in collaboration with an external

supplier, in order to be sure to obtain a reliable and professional product. This collaboration proved to be profitable at a cost of 600 Euros per airplane.

2.2 Time schedule

The project started in autumn 2013 it lasted a little bit less than two years. Since the team had no previous experience in using carbon fiber structures and electric motors for a competition, the first year was dedicated to acquire the necessary experience by designing, building up and testing a model aircraft following the ACC 2013 rules. Such an airplane was successfully tested in summer 2014. The project name of this first airplane was *FB1*.

As soon as the new ACC 2015 rules were published, the team started redesigning a new aircraft according to the new rules and modified the previous model in order to develop the second one. By the end of the year 2014 the new architecture was defined and the structural and aerodynamic design were completed. January and partly February 2015 were dedicated to define all detailed drawings and in March the construction has been completed. The new aircraft, following the ACC 2015 rules, was successfully tested by the end of March. The project name of this second airplane was *FB2*. The first half of May has been employed to produce the present report and to modify some details in order to obtain a lighter plane. The rest of the time before the competition will be used to build up the two airplanes in the final configuration and for the pilot's training.

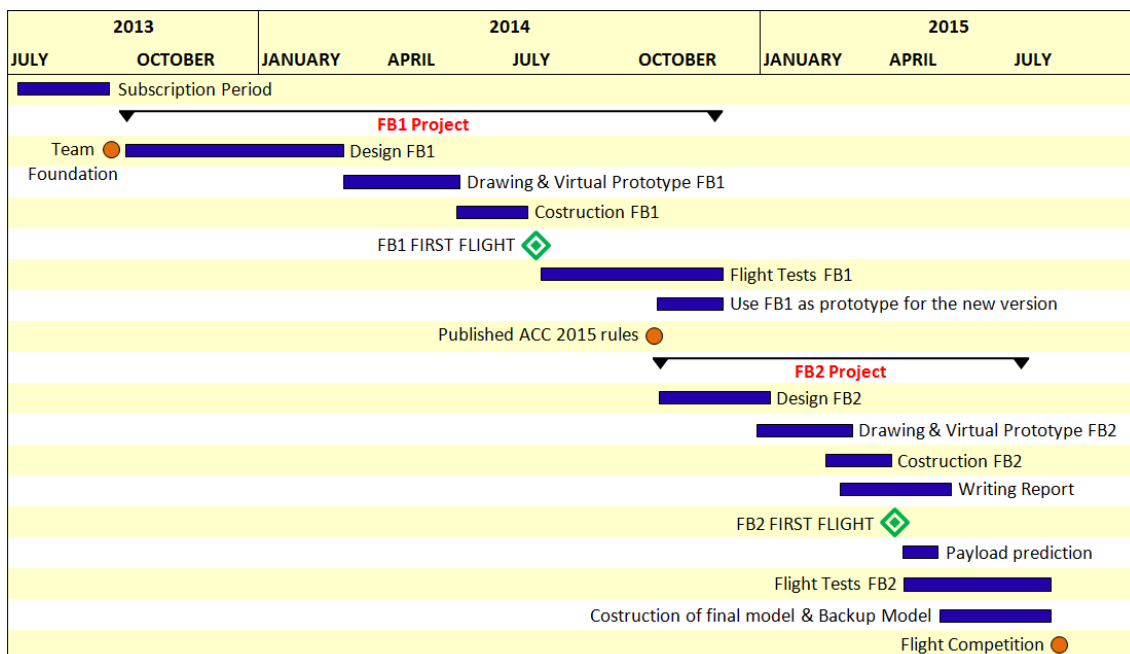


Figure 2 - Project timeline.

3. Aerodynamic design

3.1. Aircraft overall architecture

In the perspective of adopting the simplest possible solutions, it was decided to use a simple monoplane architecture. This type of model aircraft has proven to be the best suited for this type of competition, as documented by the final rankings in both the SAE Aero Design and Air Cargo Challenge past competitions.

A second fundamental choice concerns the position of the payload with respect to the wing. Stability considerations would probably already suggest putting the payload in a lower position with respect to the wing, but another consideration in the same direction has been evaluated as a crucial one: if the load is put under the wing, the main carriage can be put directly under the payload. In this configuration, the whole load during the landing is directly transferred from the ground to the payload through the main carriage axle and no other structure of the airplane is severely involved. In this way, the structural configuration of the main carriage turns out to be very simple and the risk of failure during the landing for the other structural parts of the plane is greatly reduced.

The position of the motor in this type of plane is constrained by the need of a correct positioning of the centre of gravity. The motor and the battery, which must be close to each other in order to have the shortest possible connecting cables, are a non-negligible mass which can be usefully exploited to put the centre of gravity in the desired position. Thus, the motor position is at the airplane nose and the distance between the nose and the wing can be adapted to obtain the suited CG position.

Two options have been considered for the carriage: a tricycle and a bicycle configuration. A bicycle configuration would probably lead to a slightly lighter solution, but stability during take-off and landing could be significantly reduced. Therefore, the usual prudent design philosophy has led to a statically stable tricycle configuration.

Finally, an initial decision was taken for what concerns the structural components of the aircraft. As previously said, it was decided to develop the carbon fiber reinforced structures in collaboration with an external supplier in order to have a reliable and constant quality product. The selected supplier mainly produces windsurf masts and consequently the available solution consisted in straight carbon reinforced tubes with circular or elliptical sections. The circular section tube has been chosen. Once taken this decision, the resulting main structures of the plane are straightforward: two straight tubular structures for fuselage

and wing spar (forming a simple cross) and straight tubular structures for spars of the tail surfaces.

3.2. Propeller test

Some early thrust measurements had already been carried out in order to choose the best propeller for the first airplane, which was built up following the ACC 2013 rules. These measurements were carried out with the apparatus shown in Figure 3.

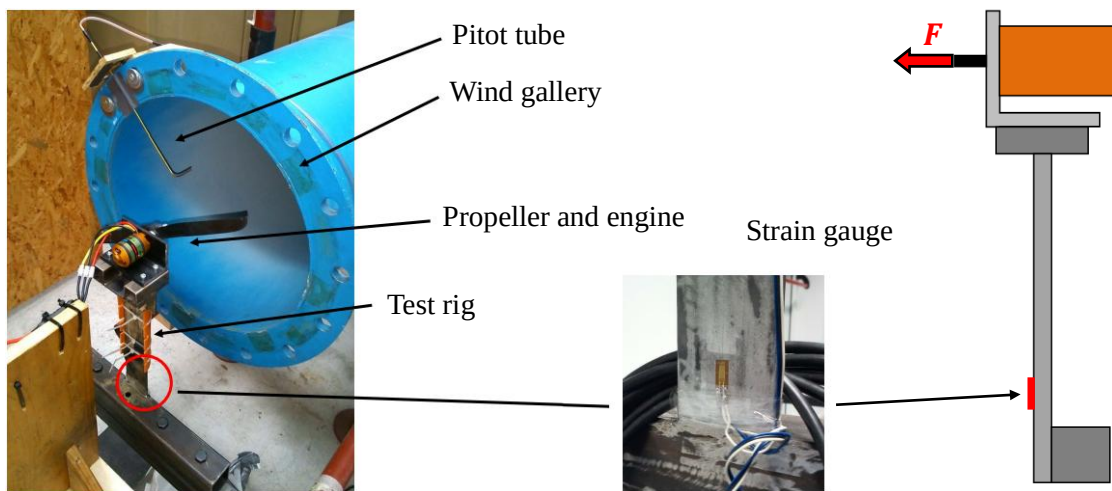


Figure3 - Thrust measurement apparatus.

Thrust measurements have been repeated with the APC 13x7 propeller, imposed by the regulations, and it was soon clear that the main problem with this propeller was the current absorbed by the motor.

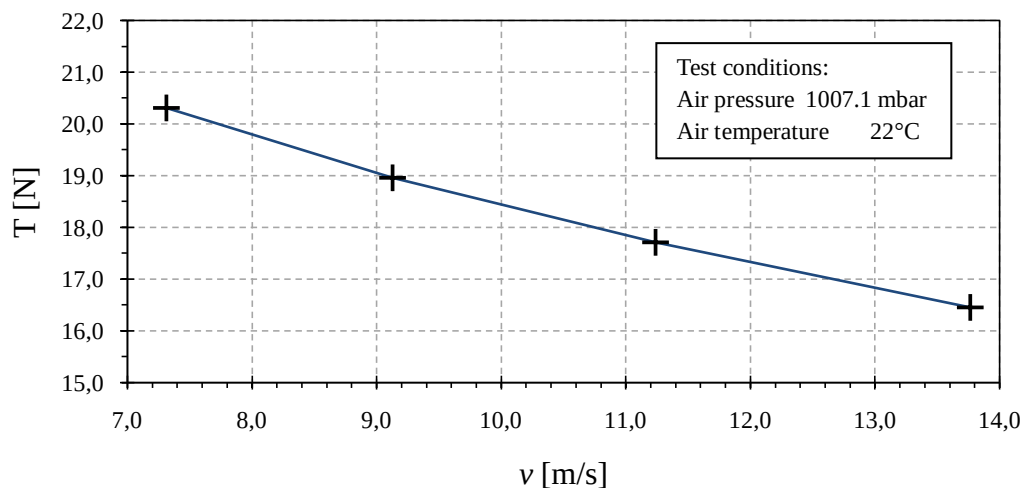


Figure 4 - Propeller 13x7 experimental results: thrust vs. air speed.

A reasonably realistic design thrust, compatible with the motor working temperature, can be estimated at about 17.4 N with an air speed of 12 m/s. Drag estimates had proven to be unreliable and therefore the design air speed was chosen looking at the flying tests of the first prototype according the ACC 2013 rules.

3.3. Airfoil

The first decision to take was the choice of the wing airfoil from the *UIUC Airfoils Coordinate Database*[1]. Two high lift airfoils have been preliminarily considered: the FX74CL5-140 and the Selig 1223, both shown in Figure 5.

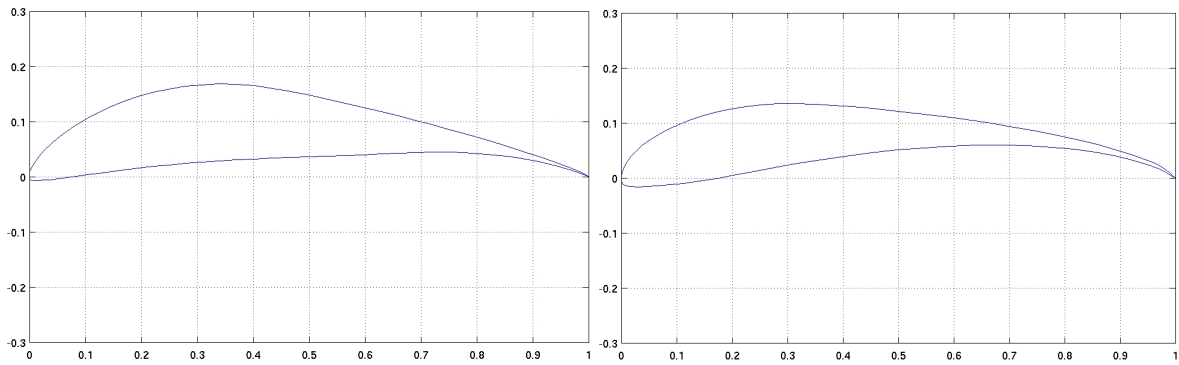


Figure 5 -High lift FX74CL5-140 (on the left) and S1223 airfoils (on the right).

The performances of these two airfoils have been comparatively estimated with the XFLR5 code (based on XFOIL's theory) at a Reynolds number of 200.000 and thanks to the data that we have read in the *Summary of Low-Speed Airfoil Data* [3] and *High-Lift Low Reynolds Number Airfoil Design* [4]. The results show that both airfoils have a quasi-identical lift versus angle-of-attack behaviour, while the C_m coefficient of the S1223 is about 10% larger than that of the FX74CL5-140 airfoil. On the other hand, the S1223 airfoil (red line in Figure 5) should be significantly more efficient than the FX74CL5-140 (blue line in Figure 5), if it is employed in the range from -2° up to 5° for the angle of attack.

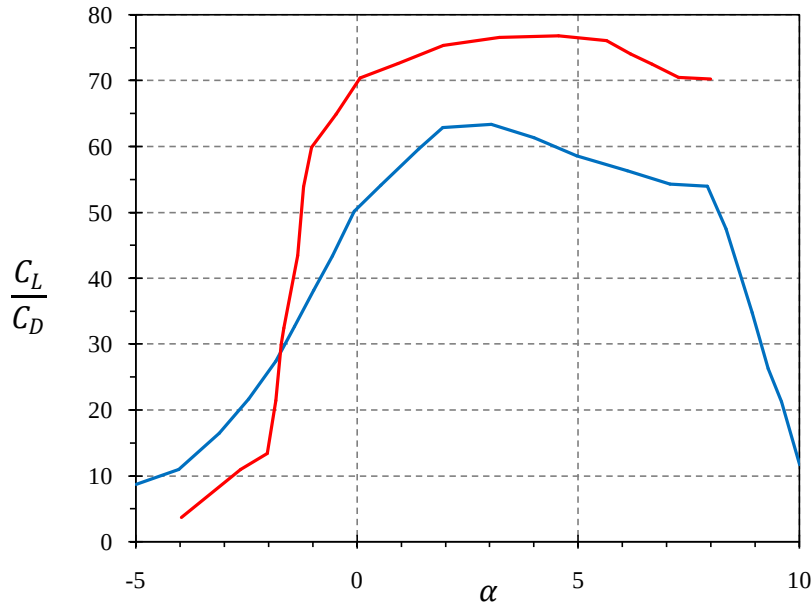


Figure 6 - C_L/C_D for the S1223 (red line) and for the FX74CL5-140 (blue line).

Efficiency has been considered more important than the possible problems, which might arise due to the 10% increase in the pitching moment and therefore the Selig 1223 airfoil was finally selected.

The second important decision concerns the area of the surface and the shape of the plant of the wing. Considering an estimated total weight of about 14 kg, an air speed of 12 m/s and the lift coefficient of the selected airfoil (reduced by the 3D effect of the finite length wing), it turns out that the area of the wing surface should be about 1.2 m². In these flying conditions and with this wing area, the angle of attack is about 3.5°, which corresponds more or less to the mid span of the range where the efficiency of the S1223 airfoil should be best. Based on these considerations, the wing area was fixed at 1.2 m². A flat wing with no dihedral has been chosen because of its structural simplicity resulting from the coupling of the straight carbon fiber tubes to build up the spar.

Once established the area, the shape of the plant has been determined, taking into account the geometrical constraints fixed in the regulations and with the aim of approximating an elliptical shape to reduce the drag. Because of the dimensions of the transportation box and of the 2.5x2.5 m square constraint, it was soon clear that the wing should have been divided into three parts. In order to approximate as far as possible the elliptical shape with a simple polygon, the central part was designed as the largest possible rectangle, which could be easily

put into the transportation box, while the remaining two parts have been designed as two symmetric trapezoids. The resulting plant is depicted in Figure 7.

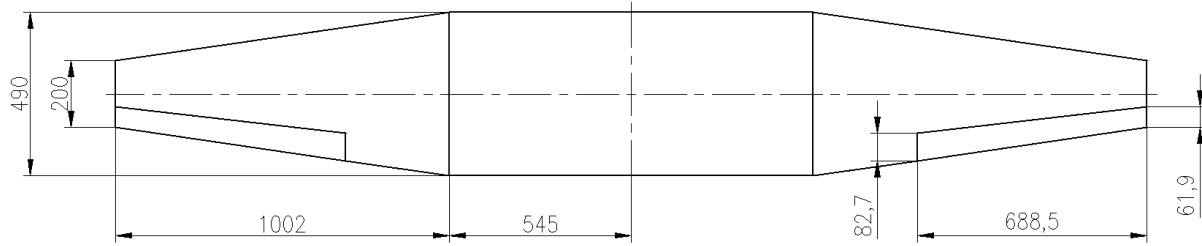


Figure 7 - Wing plant.

The symmetric naca009 airfoil has been chosen for both the vertical and horizontal stabilizer. The decision of using a symmetric airfoil for the elevator was taken in the perspective to have an airplane with a nearly zero lift on the horizontal stabilizer when flying in nominal conditions, despite the considerable pitching moment given by the wing with S1223 airfoil. The area of the elevator surface has been determined by static longitudinal stability considerations, which will be described in more details in the following stability analysis section. The result was an area of 0.265 m^2 for the horizontal stabilizer. Here again, in order to approximate the elliptical plant with a reasonably simple polygonal while respecting the $2.5 \times 2.5 \text{ m}$ square constraint, the plant of the horizontal stabilizer has been defined as a symmetrical hexagon with a wingspan of 1 m, a root chord of 0.365 m and a chord of 0.175 m at the tips.

3.4. Stability analysis

The stability analysis of the airplane has been limited to longitudinal static stability.

First, the positions of the two aerodynamic centres of both the wing and the horizontal stabilizer have been determined taking into account of the shapes of the corresponding plants. Once both aerodynamic centres determined, the body axes have been defined in the following manner: the origin of the body axes (y, z) is positioned in the aerodynamic centre of the wing, the y -axis lies along the line connecting the two centres and is oriented towards the tail, while the orthogonal z -axis is oriented upwards. Only one x -axis of the wind axes needs to be defined: the origin of the wind axes coincides with the origin of the body axes and the x -axis is oriented parallel to the air speed vector.

The angle of attack α_A of the whole aircraft has been defined as the angle of attack of the line

connecting the two aerodynamic centres. In other words, the angle of attack α_A of the aircraft is the angle between the y-axis and the x-axis. The α_A angle should be always quite small in normal flying conditions and thus the transformation between body axes and wind axes has been linearized to the simple form $x = y + \alpha_A z$.

The y-coordinate of the neutral point of the aircraft has been determined by the classical formula:

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

where

y_{pn} is the distance of the neutral point from the aerodynamic centre of the wing;

b is the distance between the two aerodynamic centres;

A_W, A_T are the areas of the wing and horizontal stabilizer surfaces;

$C_{L,\alpha}^W, C_{L,\alpha}^T$ are the derivatives of the lift coefficients for the wing and for the elevator;

k_ϵ is the derivative $k_\epsilon = \frac{d\epsilon}{d\alpha_A}$ of the angle ϵ which represents the downwards deflection of the air stream at the leading edge of the horizontal stabilizer.

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 and 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. It must be said that this correspondence was satisfactory, but not particularly precise.

The static margin is defined as $m_s = \frac{x_{pn} - x_{cg}}{c_w}$ where c_w is the mean aerodynamic cord of the wing and x_{pn} and x_{cg} are the distances in wind coordinates of the neutral point and of the center of gravity from the wing aerodynamic centre.

Using the linearized transformation: $x_{pn} - x_{cg} = y_{pn} - y_{cg} - \alpha_A z_{cg}$ one obtains for the static margin:

$$m_s = \frac{y_{pn} - y_{cg}}{c_w} - \alpha_A \frac{z_{cg}}{c_w}$$

The static margin depends on the angle of attack. As a matter of fact, if the distance of the centre of gravity from the line connecting the two aerodynamic centres is not negligible, a variation in the angle of attack produces a back and forth displacement of the centre of gravity, parallel to the air speed vector. In the chosen aircraft configuration, where the payload is much lower than the wing and thus $z_{cg} < 0$, increasing the angle of attack gives larger static margins.

Straight flight conditions and longitudinal static stability have been analysed by means of a Matlab code written to this purpose. As an example, the following paragraph reports the program output for an air speed of 12 m/s.

```
Speed = 12m/s
Total mass = 11.8 kg
Angle of attack of the line passing through the wing a.c. and the hor. stab. a.c. = -5.5 °
Angle of attack of the fuselage = -3.5 °
Angle of incidence of the wing = 5.0 °
Angle of attack of the wing = 1.5 °
Wing lift = 118 N
Wing drag = 9 N
Distance between LE and centre of pressure of the wing = 0.252 m
Angle of incidence of the horizontal stabilizer = 2.2 °
Angle ε (deflection of the air stream) = 0.6 °
Angle of attack of the horizontal stabilizer = -1.9 °
Elevator lift = -2.8 N
Elevator drag = 0.2 N
Static margin = 10.7 %
```

The area of the horizontal stabilizer surface has been chosen as a compromise between geometrical constraints, reasonable static margin in all loading conditions and minimum horizontal stabilizer lift with large payloads. The centre of gravity of the aircraft without payload is a little bit more towards the nose than the cargo bay. In this way, the static margin in empty flight is about 20%, while it reduces regularly to about 10% with large payloads. Should the centre of gravity of the plane be placed at the same horizontal coordinate than the

centre of the cargo bay, than the static margin at intermediate payloads would become too small and might become dangerous in a possible dive, due to increased speed and reduced angle of attack.

Table 1 - *Static margin values in different loading conditions.*

Payload [kg]	Speed [m/s]	Static margin %	Wing angle of attack [deg]	Horizontal stabilizer lift [N]
0	9	19.6	-3.9	-4.4
1	10	15.4	-3.8	-4.8
2	10	13.4	-2.2	-4.0
3	11	11.6	-2.5	-4.5
4	12	10.2	-2.9	-5.2
5	12	10.0	-1.8	-4.5
6	12	10.0	-0.7	-3.9
7	12	10.3	0.4	-3.3
8	12	10.7	1.5	-2.8
9	12	11.2	2.6	-2.3

4. Structural design

As previously said, all main structural parts are realized by means of straight carbon fiber reinforced tubes. The structural design process has been developed in the following steps:

- experimental investigation of the modules and the ultimate strengths of the tubes
- static analysis of the wing spar in terms of stress and deformation
- dynamic analysis of the motor supporting structure to exclude possible resonances
- flutter analysis of the wing and of the tail surfaces (separately described in the next section)

The Young's modulus has been determined by suspending one end of the tubes to a wire and by measuring the natural bending frequencies with an impact test. Analytical solutions are available for the natural frequencies in bending of uniform beams in free-free boundary conditions and, if one uses a long wire, the real boundary condition is very close to the ideal free one.

The shear modulus has been determined with a similar impact test: two equal, known and large rotatory inertias has been fixed to both ends of the tubes, which were suspended to a wire at the mid span. The natural torsional frequencies allow one to easily determine the G modulus.

The ultimate strength of the tubes has been measured in simply supported boundary conditions by loading them at the mid span.

The static analysis of the wing spar has immediately shown that the major problem was the compression stress due to the bending moment, which caused buckling in the wall of the tubes, if the wall thickness was too small. Therefore, after some attempts, it was decided to use a five-layer wall for all the tubes of the wing spar, resulting in a wall thickness of 1.35 mm. The structure of the layers, going from the inner to the outer surface of the tube, is the following:

1 x HS150g @90°	
2 x HM300g @ 0°	for bending
1 x HS150g @ 0°	for bending
1 x Biax300g @ ±45°	for torsion

With this type of wall, an outer diameter of 24.7 mm was needed at the mid span of the wing to insure an appropriate safety factor against failure due to bending moment. The spar of the central piece of the wing is therefore a constant section tube with a 24.7 mm outer diameter and it is 1090 mm long.

The spars of the two lateral pieces of the wing are realized keeping in mind that if one uses tubes with decreasing diameters, the easiest (and even lightest) solution is to fit the smaller tube directly into the bigger one. With this criterion, the spars are composed by three tubes, each 333 mm long and with decreasing outer diameters of 21.7 mm, 18.7 mm and 15.7 mm at the tip. These diameters are certainly excessive from the point of view of the ultimate stress in bending, but they were found to be the simplest and lightest possible solution. However, they increase the wing flexural and torsional stiffnesses and this helps against excessive tip displacements/rotations and flutter.

Considering a lift of 150 N with the spar positioned at 38% of the root cord and considering the centre of pressures at 45% of the cord, the computed tip displacement was 69 mm and the corresponding rotation was -1.5° ($E = 82 \text{ GPa}$ and $G = 8 \text{ GPa}$).

The fuselage is composed by an anterior tube, which carries the motor, the electronic speed

controller and the batteries, and by the tail boom. The thinnest possible wall suggested by the supplier was 0.76 mm thick and this was preliminarily chosen to realize both the anterior tube and the tail boom.

No significant static loads (causing possible buckling problems) are acting on the fuselage structure, while there could be in principle resonance problems in the anterior tube, due to the running motor, and flutter problems in the tail boom, due to lack of stiffness.

Analysing the natural frequencies of the anterior tube (with an outer diameter of 29.5 mm and a wall thickness of 0.76 mm) it was soon clear that no resonance problems had to be feared, because, considering it as a cantilever, its fundamental natural frequency was far below 1000 rpm. As a matter of fact, no vibration problems have been experienced on the anterior tube at any running speed of the motor.

The second step was to verify if the tail boom, composed of two pieces of tube (with an outer diameter of 29.5 mm on the wing side and 27.5 mm on the tail side and the 0.76 mm wall thickness) would be stiff enough to avoid flutter problems of the tail. Fortunately, the flutter analyses of the vertical and horizontal stabilizer have shown that the tail boom was stiff enough to avoid flutter (see next section for the details).

In conclusion, all the three tubes which compose the fuselage have a wall thickness of 0.76 mm with the following layer structure, going from the inner to the outer surface of the tube:

1 x HS150g @90°

1 x HM300g @ 0° for bending

1 x Biax300g @ ±45° for torsion

4.1. Flutter analysis

Flutter analysis has been carried out starting from the formula:

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

where

v is the air speed;

k_{θ} is the torsional stiffness of the wing;

k_u is the translational stiffness of the wing;

I_g is the mass moment of inertia with respect to the center of mass of the wing;

m is the sum of the mass of the wing and of the surrounding air cylinder;

x_g is the distance of the center of mass from the leading edge;

x_{ct} is the distance of the center of twist from the leading edge;

$C_{L,\alpha}$ is the derivative of the lift coefficient expressed in N/rad;

A is the area of the wing surface;

ρ is the air density;

c is the wing cord.

This formula can be obtained by considering a 2D problem, i.e. an infinitely long and rigid wing suspended at the centre of twist by a translational and a torsional spring. It results from the application of the Routh-Hurwitz stability criterion to a 2d.o.f.system, as shown in the famous book by J.P. Den Hartog “Mechanical vibrations” [2].

Some simplifications have been immediately introduced. Noting that $c/4 < x_g < c/2$ and thus $(x_g - c/4) < c/4$ and evaluating that $\frac{I_g}{m} \cong \frac{1}{2} \frac{c^2}{12}$, the previous formula has been approximated by

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

The problem is now the application of this formula to the half-wing, to the elevator and to the vertical stabilizer, to evaluate the risk of flutter phenomena. In fact, all these three structures behave essentially as cantilevers, with additional flexibilities for the vertical and horizontal stabilizer due to the tail boom: something quite different from a rigid wing on a flexible suspension. The key problem is the definition of appropriate values for the stiffnesses k_θ and k_u : if one employs the stiffnesses at the tip of the cantilever, they are evidently overestimated. Thus, a criterion must be found to define reasonable values for these stiffnesses.

The adopted criterion is based on the following energy equivalence: the translational (or torsional) stiffness is chosen so that a concentrated load (or concentrated moment) equal to the resultant of the applied distributed load (or the resultant of the distributed moment) introduces in the suspension spring k_u (or k_θ) the same strain energy as that introduced into the cantilever by the distributed load itself (or distributed moment itself).

This criterion has firstly been applied to the flutter analysis of the wing spar, where the suspension stiffnesses depend only on the cantilever stiffnesses. Furthermore, in this case the

centre of twist is in the centre of the tube and the centre of gravity should be not so far away from it. Thus, in the case of the wing spar the term $(x_g - x_{ct})^2$ can be neglected with no significant loss. In this way the minimum flutter speed for the half-wing is found to be 27 m/s, if the spar is realized as previously reported in the structural analysis section. This value has been considered satisfactory and thus the structural design has been confirmed by this flutter analysis.

The flutter analysis of the tail surfaces needs some additional considerations. As a matter of fact, the determination of the stiffnesses to be introduced in the flutter verification formula in the case of the vertical and horizontal stabilizer must take into account for the stiffnesses of the surfaces themselves, but also for the stiffnesses of the tail boom: they are stiffnesses put in series. The energy equivalence criterion is used to define the equivalent stiffnesses of the half-stabilizer and of the vertical stabilizer, considered as cantilever beams. Once these determined, they must be put in series with the stiffnesses of the tail boom, considered as a cantilever beam too. Another important difference with the previous case of the half-wing is that here the center of twist is near the mid span of the tail boom, both for the vertical stabilizer and horizontal stabilizer. Therefore, the positions of the two centres of twist must be determined and the term $(x_g - x_{ct})^2$ should not be neglected in the flutter analysis of the tail surfaces.

If the spars of the vertical and horizontal stabilizer are constant section tubes with 10 mm outer diameter (the largest allowed by the airfoil thickness) and with the same wall like that one used for the fuselage (0.76 mm thick), the minimum flutter speeds are 24 m/s for the stabilizer and 21 m/s for the vertical stabilizer. These results, in particular the last one, were not considered sure enough, especially in case of a dive.

It was noted that the most significant improvements could be obtained by increasing the torsional stiffness of the spars. A new 4-layer structure with a theoretical thickness of 1.07 mm has been defined as follows (going from the inner to the outer surface of the tube):

1 x Biax300g @ $\pm 45^\circ$	for torsion
1 x HM300g @ 0°	for bending
1 x HS150g @ 0°	for bending
1 x Biax300g @ $\pm 45^\circ$	for torsion

With an outer diameter of 10 mm and this new wall structure the minimum flutter speeds are 30 m/s for the horizontal stabilizer and 26 m/s for the vertical stabilizer. These values were

considered acceptable, this was the actual solution for the vertical stabilizer and horizontal stabilizer spars.

5. Electric and electronic components

5.1. Battery capacity

It was decided to use Lithium based battery (LiPo type) to supply the airplane engine. To select battery size, it was calculated the minimum capacity required. The total time of an average flight is 4 minutes, formed by:

- take-off time, assumed about 1 minute considering several attempts;
- time window for gaining some altitude, 30 seconds imposed by regulation;
- 2 minutes to fly the greatest number of legs;
- 30 seconds for landing.

Assuming to operate the motor at max throttle for all the flight time and considering an average current of 42 A, the minimum battery capacity must be at least 2800 mAh, according to the following equation:

$$C = 1000 I \frac{\Delta t}{60}$$

where

C battery capacity [mAh];

I current [A] ;

Δt total flight time [min].

The engine will be supplied by a battery having a capacity greater than that specified, compatible with the commercial availability and reliability.

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

5.2. Servomotors size

The formula used to calculate the torque is as follows:

$$M = \frac{1}{4} \rho v^2 c^2 L \sin(\alpha) \frac{\tan(\alpha)}{\tan(\beta)}$$

where

- c control surface average chord [m] ;
- L control surface length [m] ;
- v speed [m/s];
- ρ air density in standard conditions [kg/m^3];
- α max control surface deflection [degrees], we assumed 35°;
- β max servo deflection [degrees], we assumed 60°;

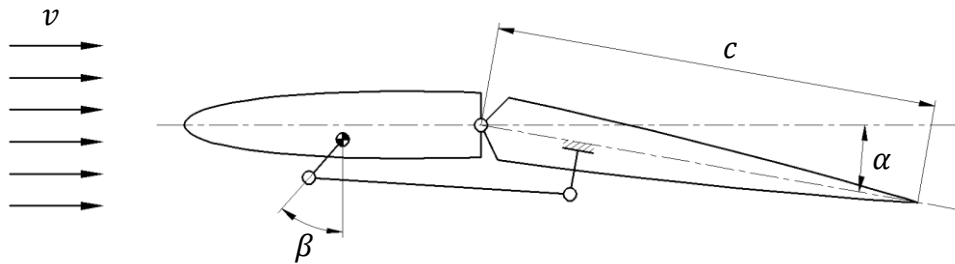


Figure 8 - Sketch of the parameters considered for the servo torque calculation.

Obtained with the following assumptions:

1. 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.
2. Control linkages have zero offset at hinge line and are perpendicular to horns at neutral.
3. The pushrods are longer than the servo and control horns.
4. Control mechanisms are frictionless and surfaces are mass-balanced.

The assumptions should generally yield conservative predicted torques (except the last assumption).

Table 2 - Servomotors torque required.

Control surface	Average chord [m]	Length [m]	Torque [kg·cm]
Aileron	0,072	0,686	0,57
Rudder	0,100	0,443	0,71
Elevator	0,087	1,000	1,21

We have chosen to use the servomotor HS-5665MH (Hitec) with 8.8 kg·cm of torque to the rudder and elevator, while for the ailerons the servomotor HS-5125MG (Hitec) with 3.0

kg·cm of torque. All the four servos installed in the airplane posses a torque far above that necessary.

6. Payload prediction

Payload prediction chart is based on data from test flights and propeller thrust measurements. 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 CL , 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:

$$\text{Predicted payload [kg]} = 13.05 \cdot \text{Air density} \left[\frac{\text{kg}}{\text{m}^3} \right] - 7.89$$

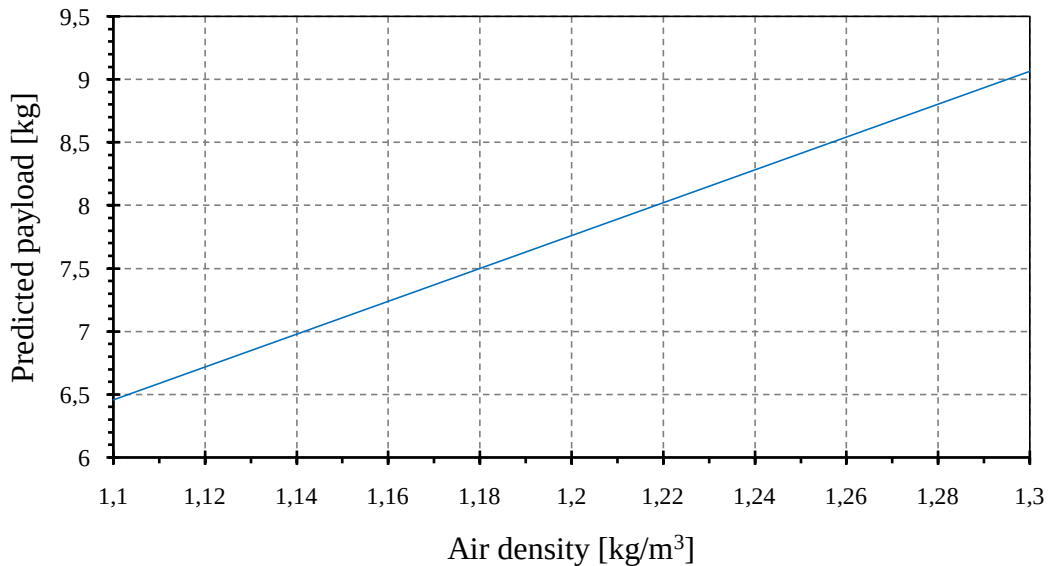


Figure 9 - Predicted payload over air density.

7. Outlook

Currently the team built an airplane model that respects the rules imposed by ACC2015 competition. This model is used to perform competition simulations, to evaluate the maximum number of legs and the maximum payload transportable. Indeed, based on the scoring system, a tradeoff has to be found between maximum payload and flight speed, to allow for the highest possible number of flown legs.

At present, the dedicated flight test shown the best compromise to be adopted during the completion and that will be further validated with the planes for the competitions that are currently under construction.

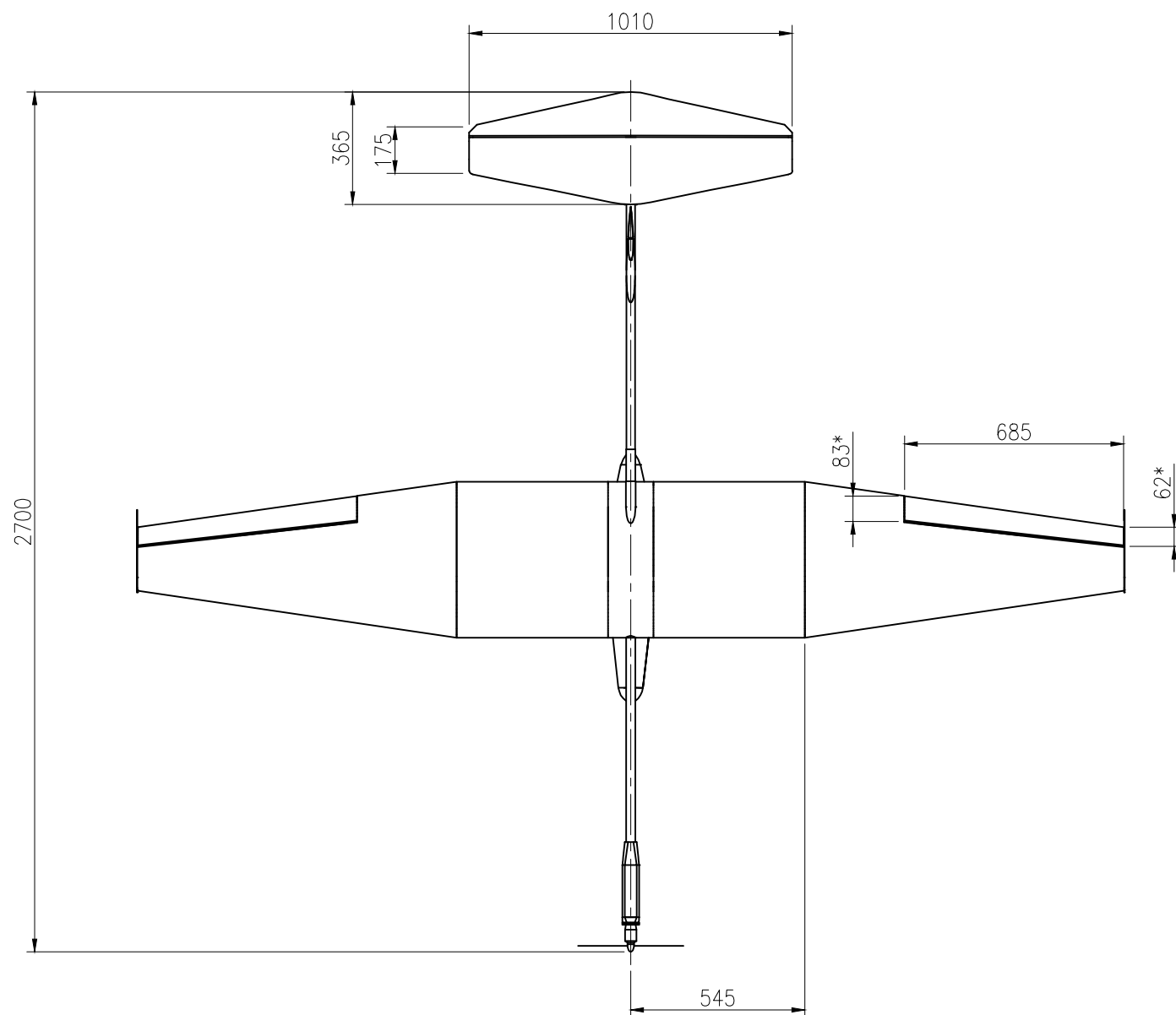
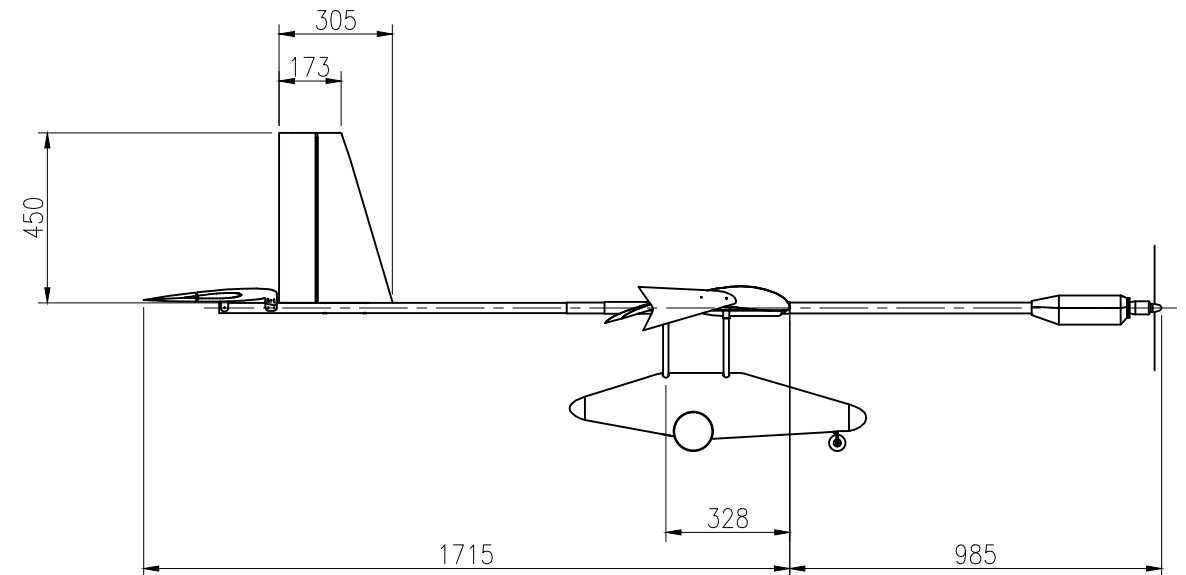
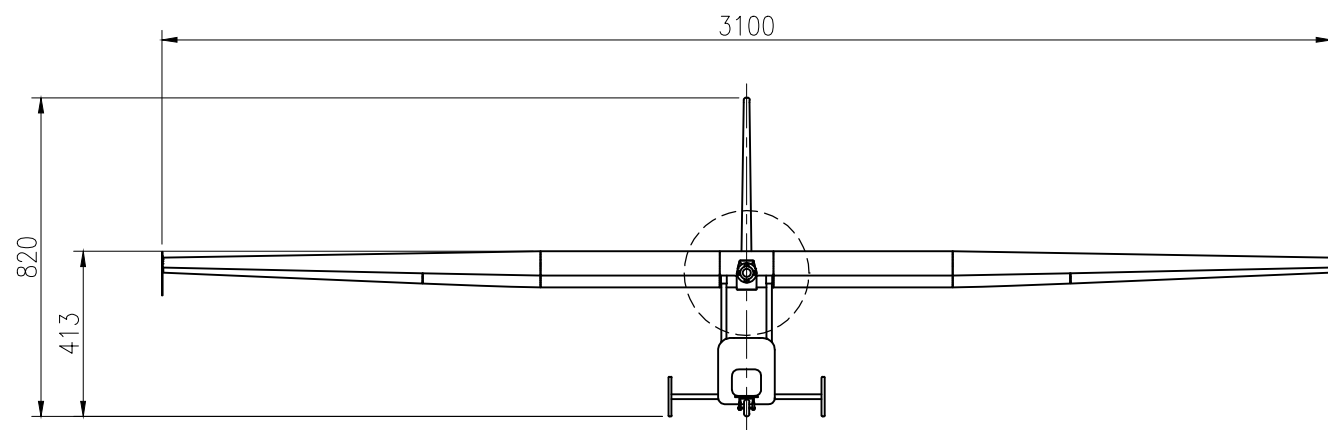
Indeed, the team started the construction of other two models, which will be used during the competition. During the construction of these new models, we are operating a series of improvements, in order to minimizing as much as possible the weight of the aircraft, ensuring the same resistance and safety.

References

- [1] UIUC Applied Aerodynamics Group. *UIUC Airfoils Coordinate Database*. University of Illinois in Urbana Champaign. www.ae.illinois.edu/m-selig/ads/coord_database.html
- [2] J.P. Den Hartog. *Mechanical vibrations*. Dover publication, New York.
- [3] Michael S.Selig, James J. Guglielmo, Andy P.Broeren and Philippe Giguère. *Summary of Low-Speed Airfoil Data*. University of Illinois in Urbana Champaign.
- [4] Michael S.Selig and James J. Guglielmo. *High-Lift Low Reynolds Number Airfoil Design*. University of Illinois in Urbana Champaign.

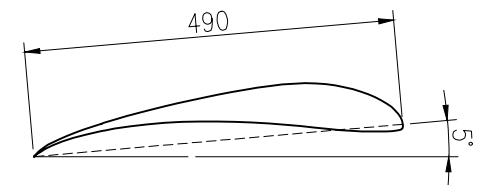
ATTACHMENT – A

Drawings

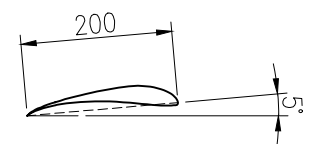


Airfoil – SCALE 1:10

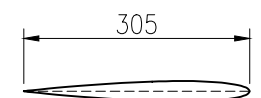
Wing root airfoil
(S1223)



Wing tip airfoil
(S1223)

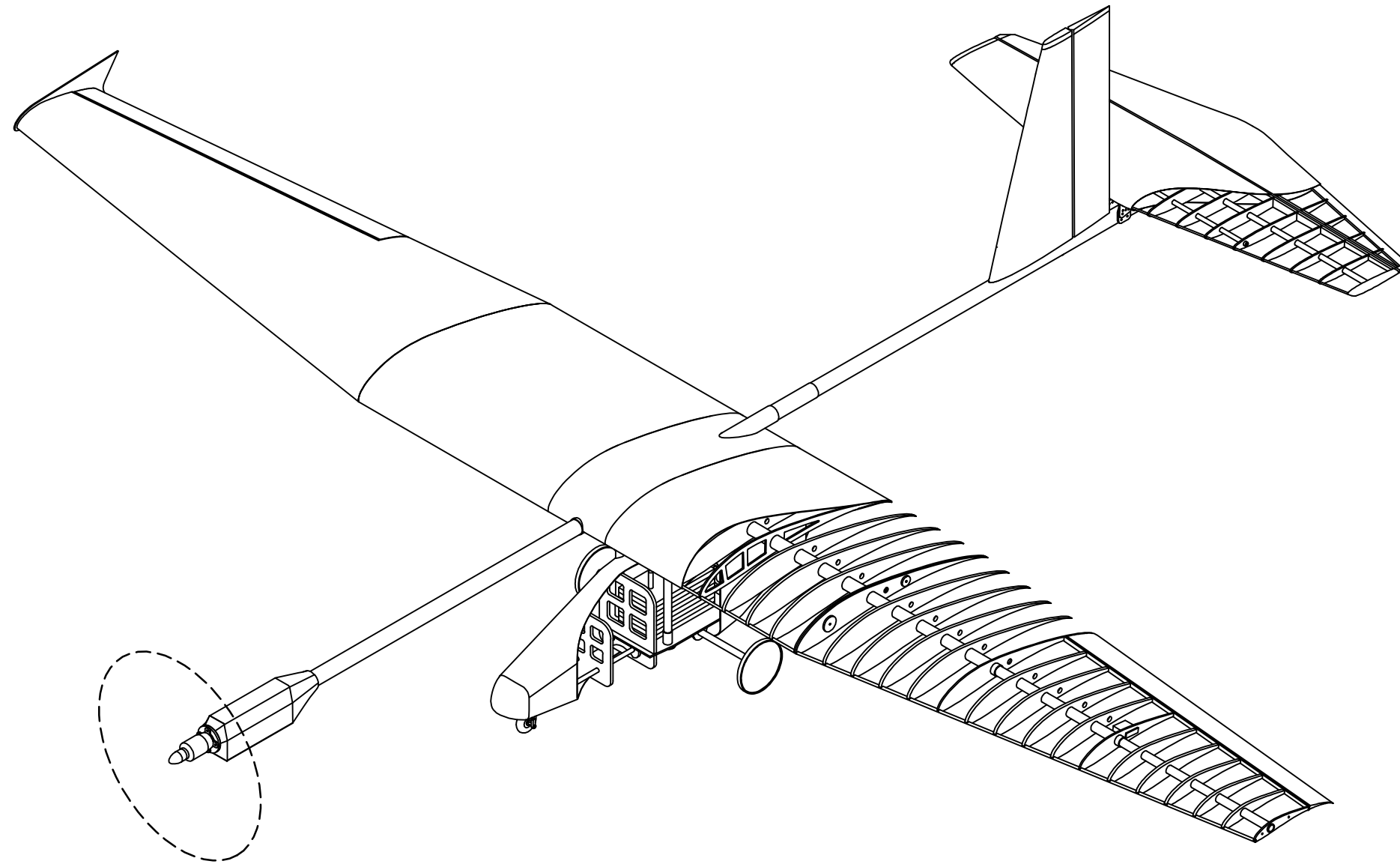


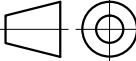
Rudder root airfoil
(naca009)



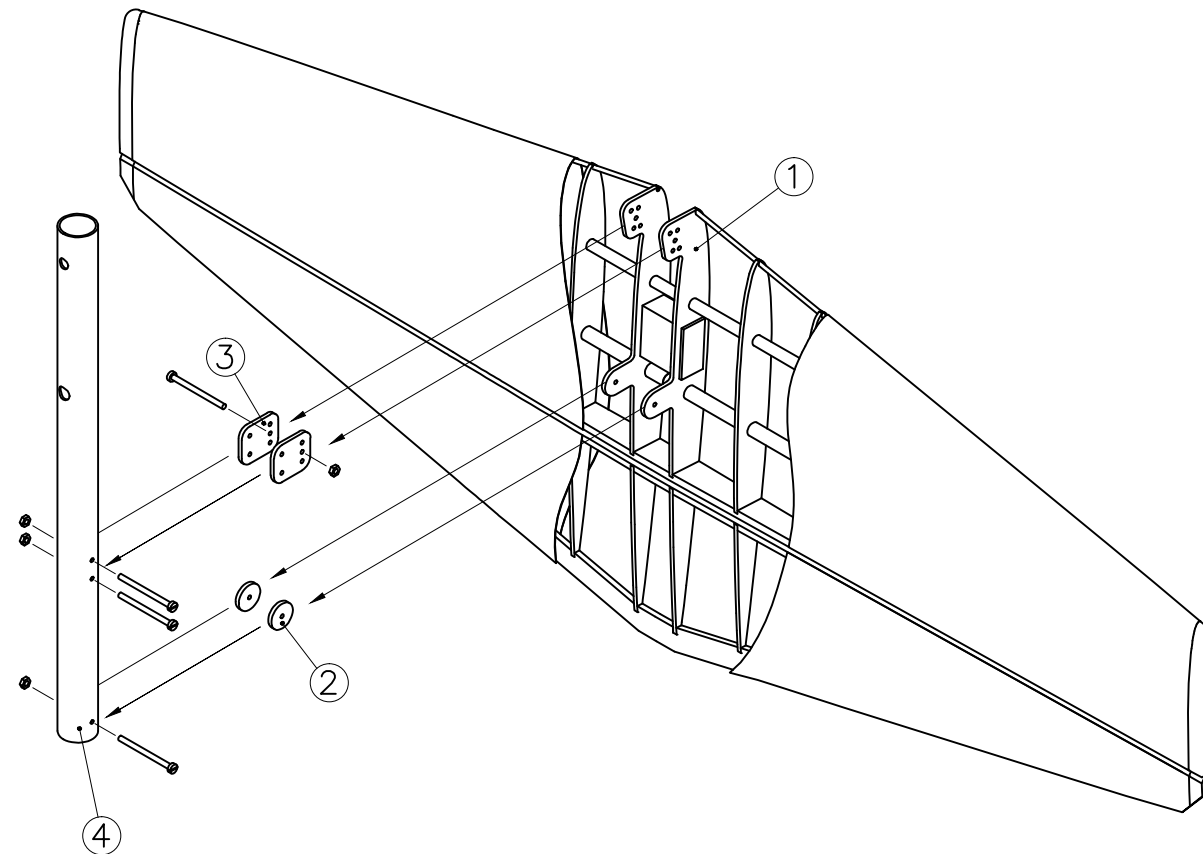
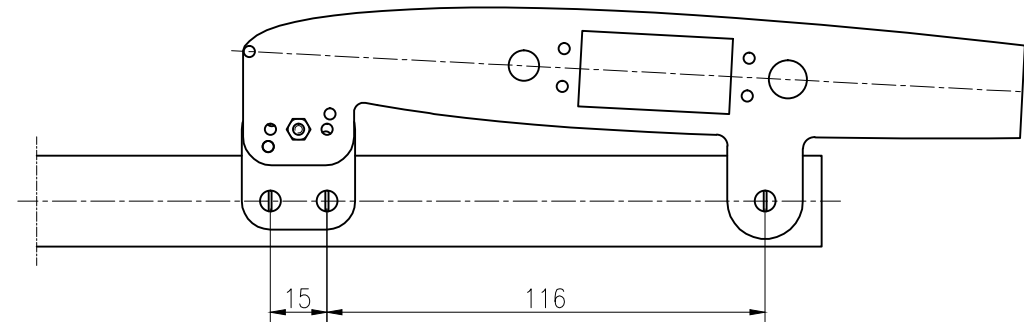
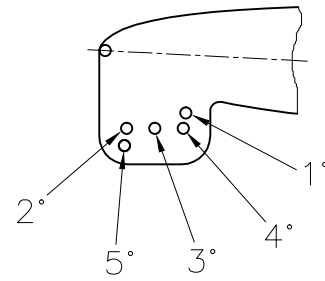
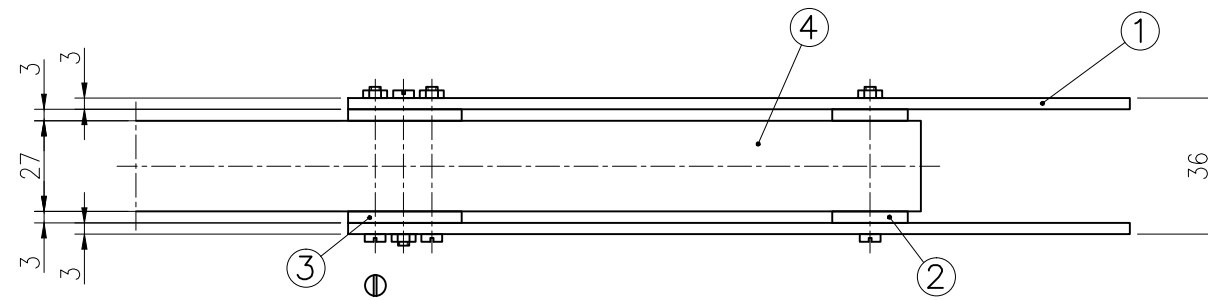
Surface	Area	Surface	Area
Horizontal stabilizer	0.2650 m ²	Rudder	0.0450 m ²
Elevator	0.0947 m ²	Wing	1.2241 m ²
Vertical stabilizer	0.1076 m ²	Ailerons	0.0998 m ²
TEAM'S UNIVERSITY University of Udine		TEAM'S NAME AeroUD-MET	
TEAM'S UNIVERSITY University of Udine		TEAM'S NUMBER TEAM #04	
TITLE 3-VIEW DRAWING		DATE 04/05/2015	
DESCRIPTION - front view at the top left corner - top view below - starboard view on the right		SCALE 1:20	

(*) Length measured on the wing chord

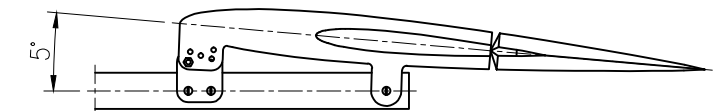
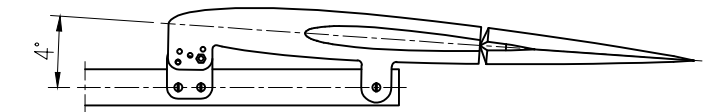
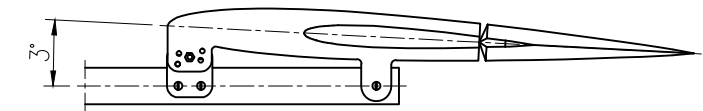
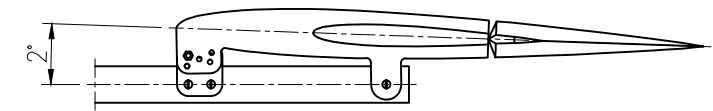
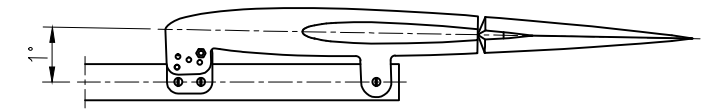


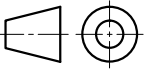
TEAM'S UNIVERSITY University of Udine	TEAM'S NAME AeroUD-MET	TEAM'S NUMBR TEAM #04	
	TITLE ISOMETRIC PERSPECTIVE VIEW		DATE 04/05/2015
	DESCRIPTION Isometric view with some glimpses of the internal structure		SCALE 1:10 

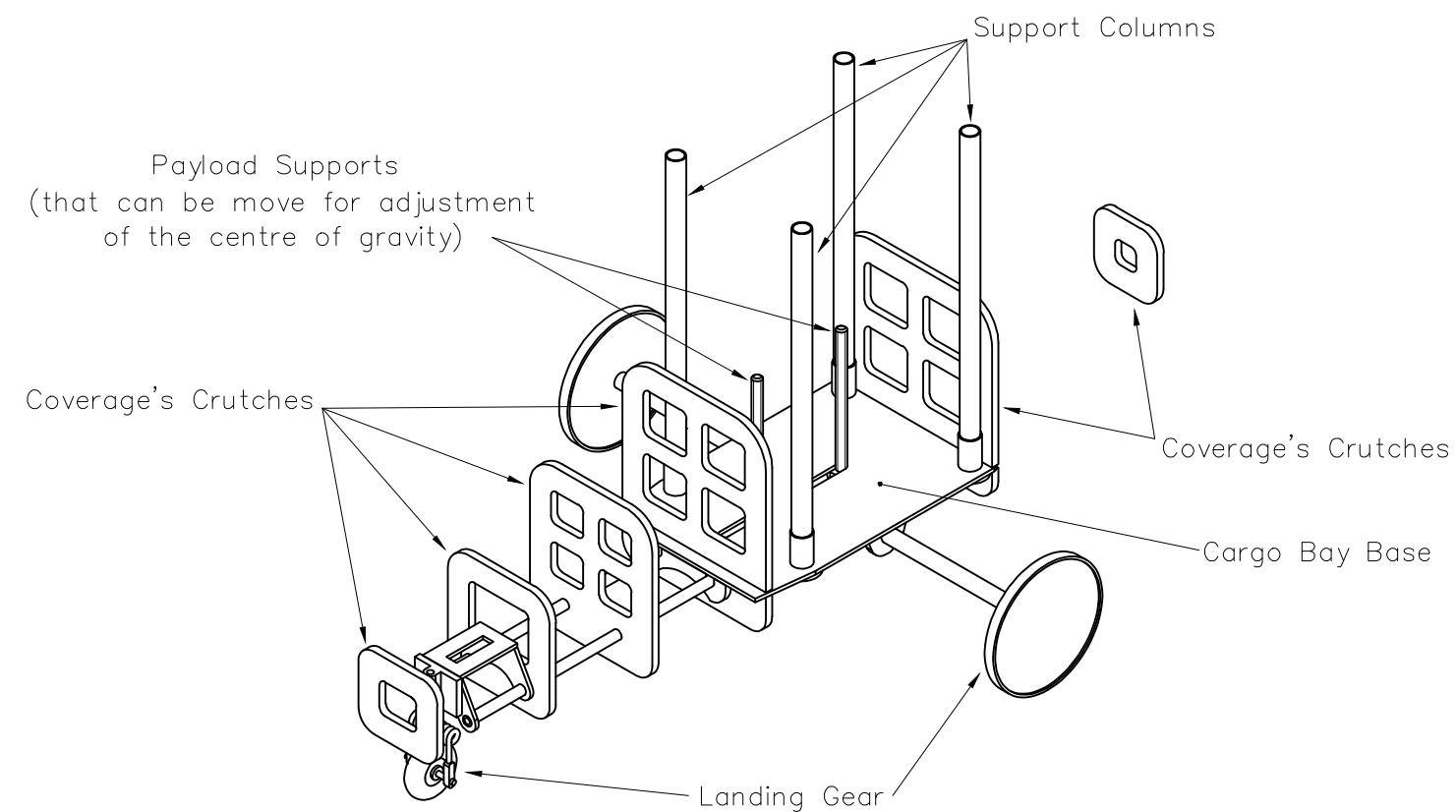
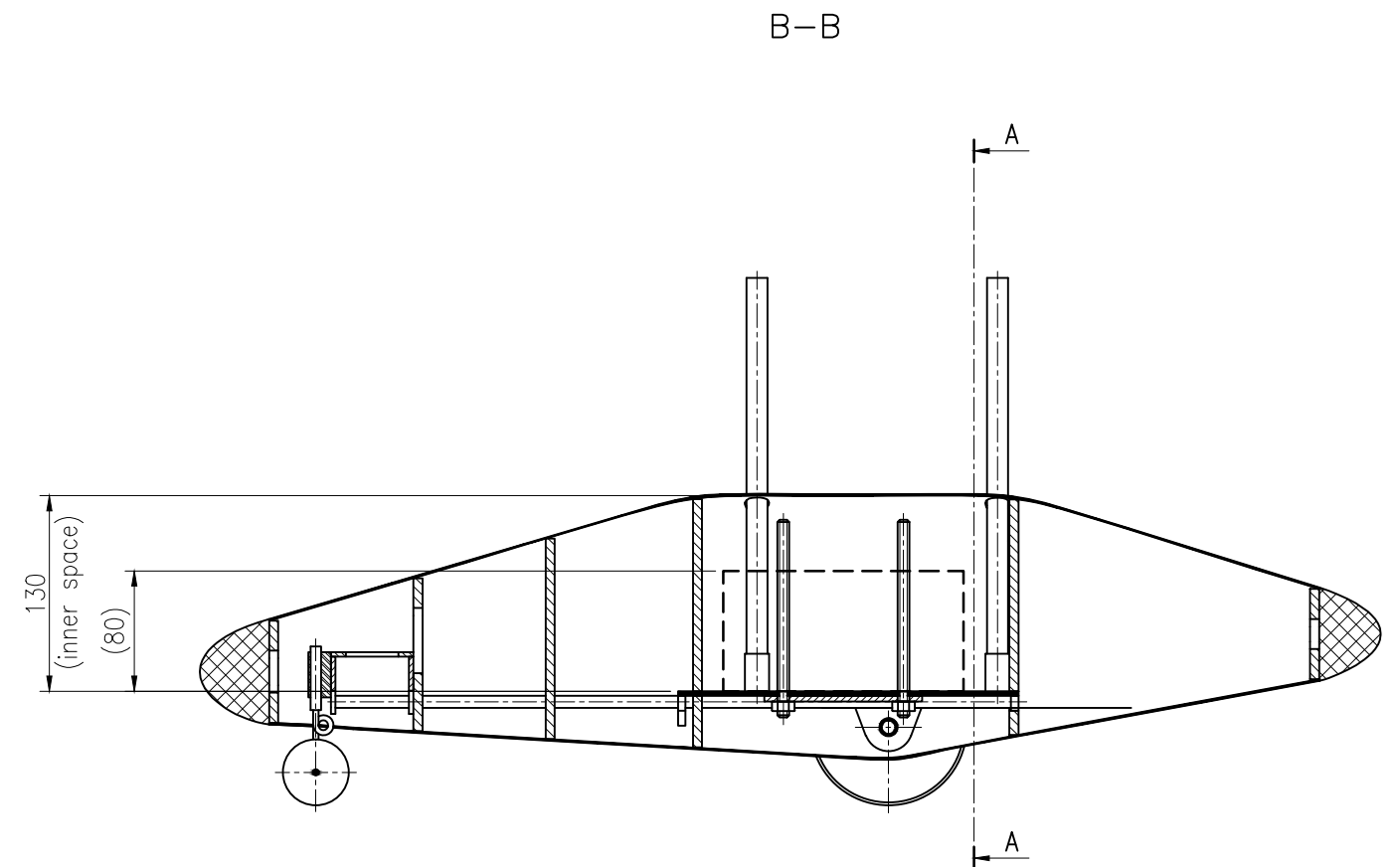
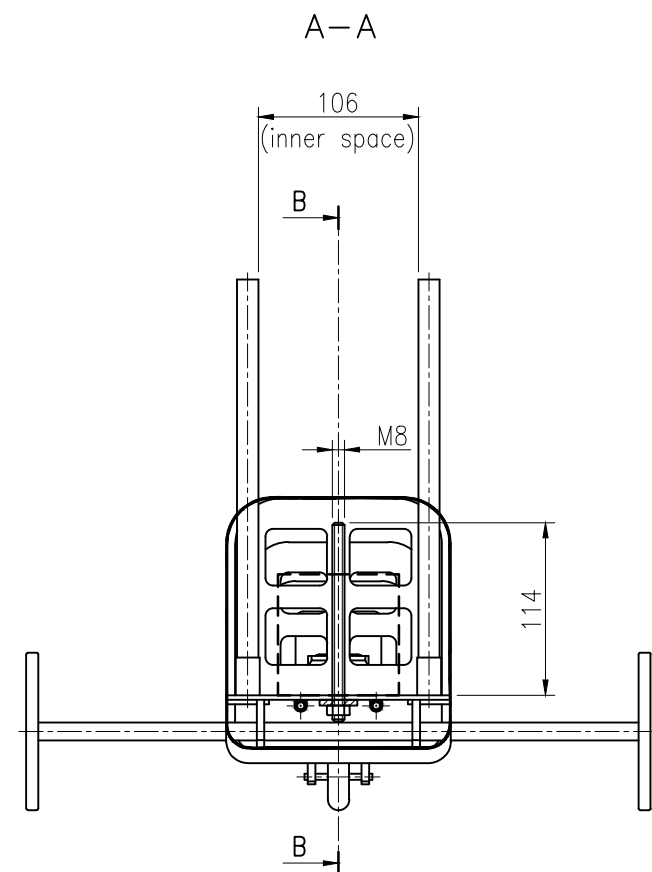
SCALE 1:2



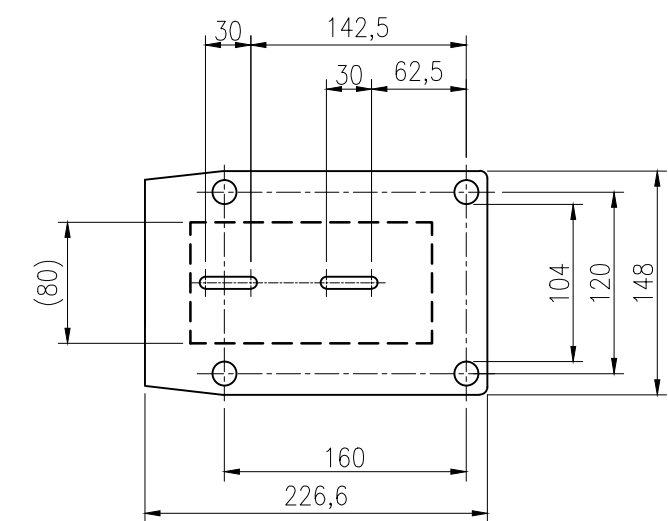
Possible settings – from 1° to 5°



TEAM'S UNIVERSITY University of Udine	TEAM'S NAME AeroUD-MET	TEAM'S NUMBR TEAM #04
	TITLE ISOMETRIC PERSPECTIVE VIEW	
	DESCRIPTION Attack angle regulation system for the horizontal stabilizer	
DATE 04/05/2015	SCALE 1:5	



Isometric View of the Assembly (hidden coverage)



Detail of Cargo Bay Base

TEAM'S UNIVERSITY University of Udine	TEAM'S NAME AeroUD-MET	TEAM'S NUMBR TEAM #04
	TITLE CARGO BAY	DATE 04/05/2015
	DESCRIPTION Cargo bay location and principal dimensions On dashed line: orthogonal volume of 160x80x80mm for technical verifications	SCALE 1:5
		

ATTACHMENT – B

Team Members and Guests Info



TEAM 04: Participants

NAME	SURNAME	GENDERS	DATE OF BIRTH	T-SHIRT SIZE	
Andrea	Lorenzon	Male	09/11/1991	S	Team Leader
Fabio	Pagnacco	Male	06/12/1989	M	Team Member
Leonardo	Rigo	Male	29/10/1991	M	Team Member
Luca	Furlani	Male	02/07/1989	L	Team Member
Luca	Gasparin	Male	17/06/1990	S	Team Member
Roberto	Costantini	Male	20/11/1991	M	Team Member
Luca	Casarsa	Male	18/12/1974	M	Pilot
Francesco	Fabbro	Male	31/07/1989	M	Guest
Marco	Giovagnoni	Male	21/12/1954	XXL	Guest
Stefano	Da Re	Male	20/05/1988	XXL	Guest

Team Leader
Andrea Lorenzon

Pilot
Luca Casarsa