



Team #09 AeroUD

Technical Report

Air cargo Challenge 2026 - Stuttgart

University of Udine



1st May 2026

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1 Introduction

The document here provided describes the main steps taken by the #09 AeroUD team to design, build and fly the model aircraft *SandoCAN* to compete in the 2026 Air Cargo Challenge (ACC 2026, Stuttgart).

AeroUD's history began in 2011 at the *University of Udine*, and this year marks its sixth participation in the Air Cargo Challenge, with several podium classifications to its name.

The current team was formed in October 2024 with a solid knowledge in composite materials and aerodynamic design, based on the experience of the previous teams whose members were available for an effective know-how transfer.

After the unsatisfying performance in the 2024 competition, the team spent the leap year preparing for the new regulations by experimenting with innovative methods and processes to close the gap with the winning team. In particular, the following research topics were pursued and successfully completed:

- Selection of the optimal sandwich material for the wing ribs and spar.
- Experimental characterization of the new wing spar.
- Manufacture and testing of new, lighter wing connectors.
- Design and testing of a retractable landing gear.
- Improvement of the aircraft performance simulation, including weight estimation.
- Accurate prediction of the aircraft aerodynamic drag using CFD simulations, also including the propeller effects.
- Update the telemetry system, incorporating a Pitot tube and an angle-of-attack sensor.

When the regulations were released, the team was fully prepared to rapidly design the aircraft, aiming to test the prototype in the first months of 2026. This schedule allowed sufficient time in the remaining months to optimize the competition aircraft.

The design and optimization process is described in detail in this report. First, the project and team management are outlined, followed by a comprehensive description of the aerodynamic and structural design. The development of the electronic systems, along with aircraft and flight strategy optimizations, is also presented. The report concludes with the payload prediction for the selected flight strategy and the complete aircraft drawings.

2 Project Management

This section describes the organizational structure adopted for the project and the approach used to manage the activities carried out by the team. It first details the group's composition, followed by an outline of the work distribution across the different functional areas.

The team is composed of students from the University of Udine. In this edition, participation was extended beyond engineering programs for the first time, allowing students from different academic backgrounds to contribute to the project. The core group includes seven Master's degree students and one Bachelor's degree student in Mechanical Engineering. The team is further complemented by one Bachelor's student in Electronic Engineering, one in Management Engineering, one in Industrial Engineering, and two students in Computer Science.

The overall coordination of the project is assigned to a team leader, who supervises the progress of the different activities and ensures communication between the various groups. Each technical section has its own division leader responsible for managing the work within the group. Coordination between the team leader and the division leaders is maintained through weekly meetings, while a meeting involving the entire team is organized approximately once per month to review progress and discuss the main project decisions.

Professor in charge is L. Casarsa, full professor in turbomachinery, who has a lifelong experience with model aircrafts and has supervised the AeroUD project at the University of Udine since 2011.

2.1 Team organization

Due to the multidisciplinary background of the members, the work was divided into four main sections. Each section focuses on a specific aspect of the aircraft development while maintaining close coordination with the others during the design and manufacturing phases.

The Aerodynamics section is responsible for the aerodynamic design of the aircraft. Its tasks include the selection of the wing airfoil and planform, as well as the definition of the fuselage and tail geometries.

The Structures section focuses on the design of the load-bearing components of the wing and other structural elements. This includes the structural layout, the selection of materials for composite lamination, the design of the landing gear, and the development of molds required for manufacturing the aerodynamic surfaces.

The Electronics section deals with the onboard systems of the aircraft. Its activities include the development and optimization of the telemetry system, the configuration of the propul-

Finally, the Marketing and Budget section is responsible for securing financial support and managing the resources available to the team. This group works on identifying potential sponsors and ensuring that the project remains within the financial constraints of the competition.

The work done during the leap year meant that the team had the possibility to rapidly design the aircraft, both aerodynamically and structurally. Delays were introduced in the process as the twin engine option was something completely new that the team had to test and simulate before the design phase.

- **Aug-Sep-Oct:** Aircraft design.
- **Nov-Dec-Jan:** Mold manufacturing and prototype construction.
- **Feb-Mar:** First competition aircraft manufacturing.
- **Apr-May:** Second competition aircraft manufacturing.

Air Cargo Challenge 2026		Aug		Sep		Oct		Nov		Dec		Jan		Feb		Mar		Apr		May		Jun																																					
Activity	Division	32	33	34	35	36	37	38	39	40	41	42	43	44	45	46	47	48	49	50	51	52	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27										
Aircraft design																																																											
Propeller testing	Electronics																																																										
Wing design	Aerodynamics																																																										
Fuselage design	Aerodynamics																																																										
Structural design	Structures																																																										
Tail design	Aerodynamics																																																										
Undercarriage design	Structures																																																										
Servosizing	Electronics																																																										
Prototype manufacturing																																																											
Mold manufacturing	Structures																																																										
Tail lamination	Team																																																										
Wing lamination	Team																																																										
Fuselage lamination	Team																																																										
Wing and fuselage assembly	Team																																																										
First model: SandoCan1																																																											
Second model: SandoCan2																																																											
Model optimization																																																											
Propeller validation	Team																																																										
ESC current regulation	Team																																																										
Battery choice	Team																																																										
Flight strategy	Team																																																										
Flight test																																																											

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2.3 Budget management and sponsors

The marketing and budget team had the opportunity to reach out to various local industries in search of sponsorship opportunities. The low visibility of the team made this task particularly challenging; however, with the support of the university, sufficient funding was secured to complete the project, as shown in Table 1.

Type	Name	Price (€)
Budget	ACC2024 leftovers	+2000
	University fund	+22000
	RALC Italia Srl	+2000
Materials	Carbon fiber	-3000
	PVC and Rohacell foam	0
	Resin and glues	-400
	MDF for molds	-200
	Aluminum Wing molds	-8500
Electronics	Receivers	-150
	Batteries	-600
	Motors	-150
	Propellers	-100
	Servo-motors	-500
	Electronic retractables	-200
Various	Participation fee	-1050
	Transport vans	-1400
	Accommodation and food	-3000
	Gasoline	-500
	Small costs	-2000
Total		+4250

Table 1: Budget management

The budget was more than enough to ensure the team could successfully complete the project, with some funds remaining for any unforeseen expenses.

3 Aerodynamic Design

This section is dedicated to the description of the methodologies used for the aerodynamic design of the aircraft.

Before delving into the details of the design, the aerodynamics team outlined some preliminary constraints:

- The aircraft must fit within a limited-size box, with total dimensions not exceeding 1500 mm.
- To increase wingspan while minimizing the occupied box volume, the wing is divided into multiple sections.
- The landing gear must be retractable to reduce drag.
- The aircraft must adopt a low-wing configuration to improve aerodynamic efficiency (taking advantage of ground effect) and to accommodate the landing gear.

A paradigm shift compared to previous ACC competitions was necessary. The overall aircraft design must not follow a sequential, step-by-step procedure (e.g., first selecting the configuration, then the propeller, then the airfoil, and finally the wing planform). Instead, as many design parameters as possible should be determined simultaneously, following a Multi-Domain Optimization (MDO) philosophy.

MDO refers to a meta-heuristic integrated design method, in which multiple disciplines such as aerodynamics, propulsion, structural mechanics, and performance constraints are considered concurrently. By optimizing across all relevant domains at once, MDO ensures that trade-offs between competing requirements (e.g., minimizing drag while maximizing structural efficiency) are balanced from the earliest stages of the design process, leading to a more globally optimal solution than traditional sequential methods.

3.1 Propeller thrust curve measurement

One of the biggest problems in the propeller selection for the 2024 aircraft was found to be the limited maximum speed (25 m/s) achievable in the wind tunnel installed at the University of Udine. Aiming for a podium finish, the team identified cruise speeds of about 30 m/s as a crucial requirement.

To give the aerodynamics team accurate propeller thrust curves, it was thus decided to use primarily the data provided by the organizers. Nevertheless in house tests were run in order to verify the given database and to also explore different propellers of renowned manufactures such as Aeronaut and APC.

During the propeller tests different air speeds ranging from 0 m/s up to 25 m/s were explored, simulating different flight conditions, extrapolating the data up to 30 m/s . In order to also test the twin motor configuration, propellers were tested at both 15 A and 30 A with a power supply at constant 11.5 V , simulating the average voltage of a 3-cell LIPO battery.

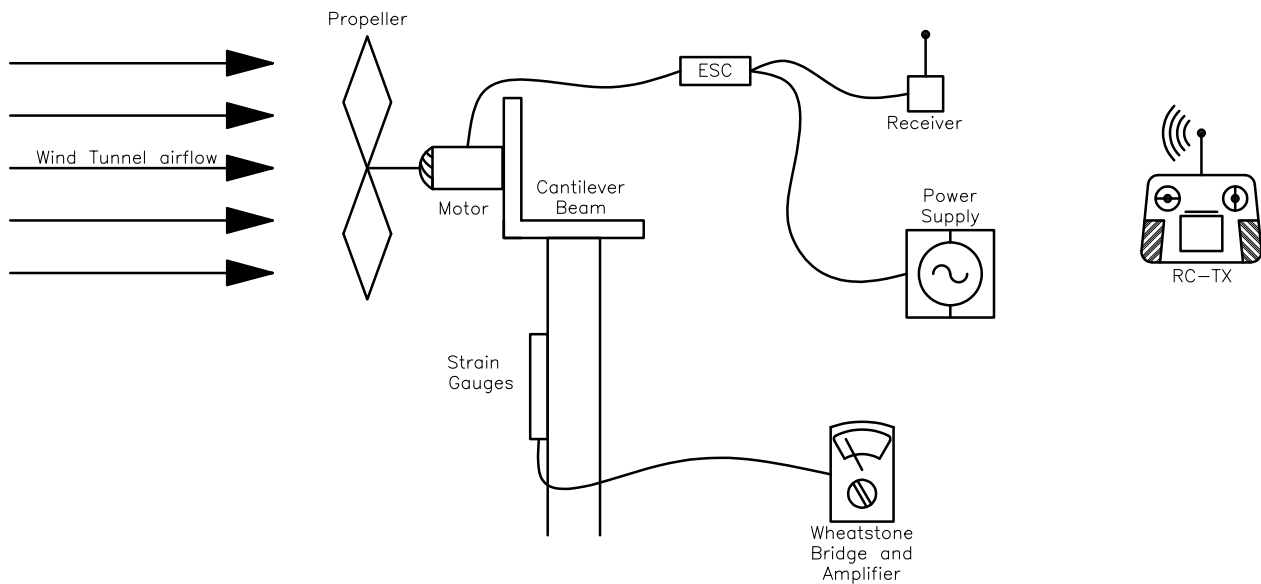


Figure 2: Simplified diagram of the propeller test setup

The downstream air speed was measured using a Pitot tube connected to a water column micro-manometer, while as shown in the schematic (Figure 2), the motor is installed at the free end of a cantilever beam. The thrust generated by the propeller causes the beam to deform, and the deformation is measured by 4 strain gauges in full bridge configuration. The setup was calibrated imposing a known force and interpolating the output, in order to properly estimate the strain gauge response.

The data is subsequently post processed using piece-wise linear interpolation, because of its robustness and relative simplicity. The data is finally used as input for the aerodynamic optimization to select the propeller and the wing geometry simultaneously (Section 3.3).

3.2 Considerations on the twin motor configuration

Following the regulatory change allowing also a twin-propeller configuration, the team conducted a series of preliminary considerations before proceeding with the design phase.

To assess the potential advantages and disadvantages of the twin-propeller layout, several preliminary geometries were analyzed through CFD simulations. These simulations were performed using the commercial software STAR-CCM+, which was made available through

our University.

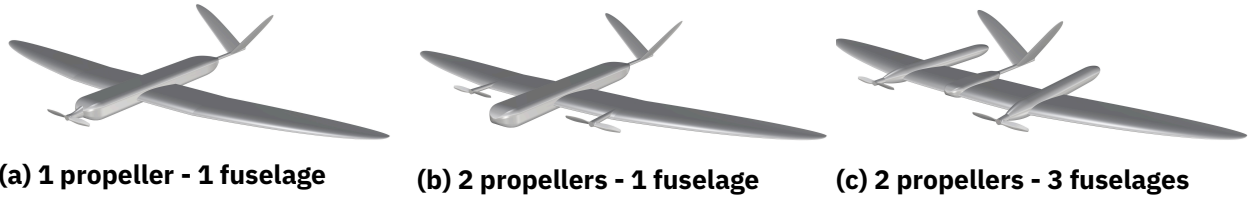


Figure 3: Tested configurations

In order to capture the propeller–wing interaction more accurately, a rotating domain was implemented in the computational setup. This approach allowed the simulations to account for the turbulence and flow disturbances generated by the propellers, which inherently influence the aerodynamic performance of the wing.

Lift and drag values were therefore compared across the analyzed geometries. The results, reported in Table 2, clearly indicate that the twin-engine configuration leads to an increase in drag and a reduction in lift.

	Lift difference [%]	Drag difference [%]
1 propeller - 1 fuselage	0	0
2 propellers - 1 fuselage	-3.9%	+10%
2 propellers - 3 fuselages	-3.7%	+6.5%

Table 2: Lift and drag comparisons of the wing in different geometric configurations

Nevertheless, given the higher thrust predicted for the twin-engine configurations, this option was not discarded. These effects were instead taken into account during the combined propeller–wing design process discussed in Section 3.3.

3.3 Propeller choice and wing design

A key component of the aerodynamic design is the definition of the wing geometry. In accordance with the MDO philosophy, the team moved away from traditional sequential design methods and developed a dedicated Python-based optimization tool. The methodology and structure of this code are described in detail in the following sections.

3.3.1 Wing sizing process overview

Figure 4 illustrates the workflow of the simulation script, whose objective is to evaluate, with the highest possible fidelity, the performance of a given aircraft geometry defined by the parameter set $A = \{p, a, c, N_S, d\}$.

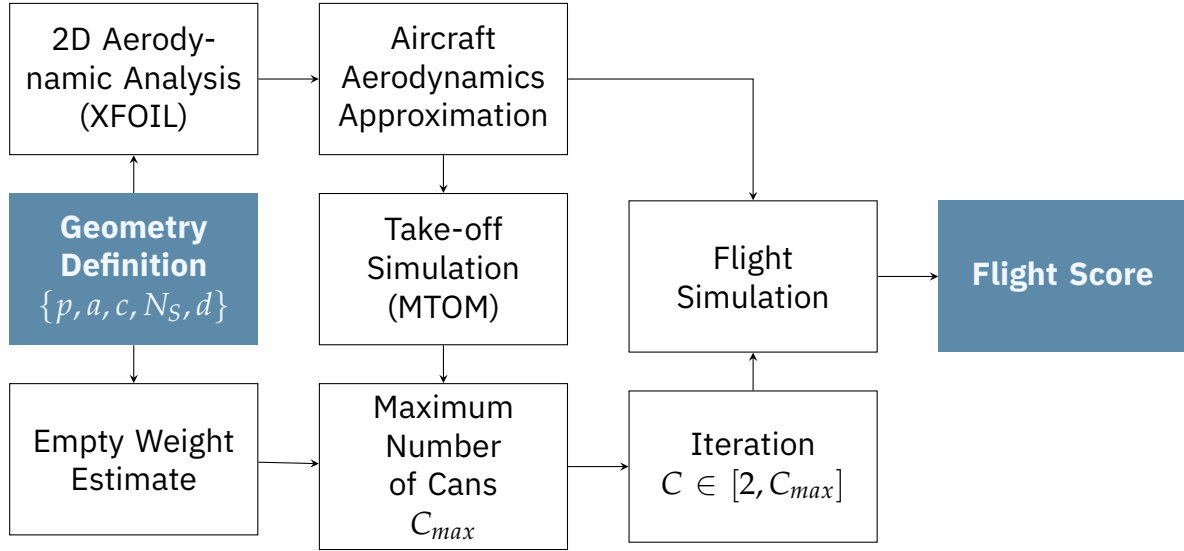


Figure 4: Wing optimization process

Geometry definition:

The code iterates through various possible configurations:

- **Propellers** $\{p\}$: with thrust curves given by the organization committee (Section 3.1);
- **Airfoils** $\{a\}$: selected from an online database, filtered according to the following criteria: $0 < \text{camber} < 6\%$, $7\% < \text{thickness} < 14\%$;
- **Chord lengths** $\{c\}$: in the range $200 \text{ mm} < c < 400 \text{ mm}$;
- **Number of sections** $\{N_S\}$: making the wing dismountable to fit into the transport box;
- **Double motor option** $\{d\}$: $[yes, no]$.

Each iteration starts with the definition of the geometrical configuration $A = \{p, a, c, N_S, d\}$. The wingspan is determined by the dimensions of the transport box, (B_1, B_2, B_3) , satisfying:

$$B_1 + B_2 + B_3 = 1500 \text{ mm}.$$

Here, B_1 accounts for the total wing sections and the fuselage thicknesses, B_2 corresponds to the segment length l_S and B_3 represents the chord length c , such that:

$$B_1 = N_S \cdot T_a \cdot c + F_T, \quad B_2 = l_S, \quad B_3 = c,$$

where F_T is the fuselage thickness, T_a is the airfoil percentage thickness, and $b = l_S \cdot N_S$ defines the wingspan.

Assuming an elliptical wing planform to reduce lift-induced drag, the corresponding wing

surface area and aspect ratio are calculated as follows:

$$S = \frac{\pi}{4}bc, \quad \Lambda = \frac{b^2}{S} = \frac{4b}{\pi c}.$$

This formulation allows the code to systematically generate feasible wing geometries that fit within transport constraints while maintaining aerodynamic consistency.

2D aerodynamic analysis:

To reduce computational cost, for each airfoil, a 2D aerodynamic map of the form $C_l, C_d = f(Re, \alpha)$ is pre-computed over a series of Reynolds numbers and angles of attack, specifically: $Re \in [10k, 2M]$ and $\alpha \in [-15^\circ, 25^\circ]$.

For each (Re, α) pair, an XFOIL analysis is performed for both the unflapped and flapped configurations, using a 15° downward flap at 70% hinge position and a critical amplification factor of $N_{crit} = 7$.

If convergence is not met in the XFOIL simulation, it is repeated using *Neuralfoil*¹. While this approach is slightly less accurate and slower, has the advantage of producing C^∞ -continuous results due to the properties of the neural network on which it is based.

Once the map is generated, any (Re_i, α_i) within the defined domain can be obtained via linear interpolation using the two nearest points in both Reynolds number and angle of attack.

Aircraft aerodynamics approximation:

Because of the three-dimensional effects of the finite aircraft which are not captured by the 2D XFOIL airfoil analysis, an approximation is adopted to simulate the entire aircraft. Prandtl's finite wing theory is applied, considering that additional components such as fuselage and tails do not generate lift but contribute to drag:

$$C_L = \frac{C_l}{1 + \frac{2}{\Lambda e}} k_{wb}$$
$$C_D = C_d + \frac{C_L^2}{\pi \Lambda e} + C_{D,tails} + C_{D,fuse} + C_{D,else}$$

where the lower case letters (l, d) stand for the airfoil lift and drag coefficients, while the upper case letters (L, D) stand for the total aircraft coefficients.

- **Lift-induced drag:** $\frac{C_L^2}{\pi \Lambda e}$, where e is Oswald efficiency factor (1 for elliptical wings);
- **Tail drag:** $C_{D,tails} = f(v)$, interpolated as a function of aircraft velocity based on a geometry similar to that of ACC2024, using XFLR5;

¹Python module developed by Peter D. Sharpe at MIT: Neuralfoil employs a neural network trained on tens of thousands of XFOIL airfoil simulations to predict aerodynamic characteristics of the input airfoils.

- **Fuselage drag:** $C_{D,fuse} = \frac{k_f}{S} (2\pi D_H^2 + 4\pi D_H l) \left(1 + 1.5 \left(\frac{D_H}{l} \right)^{1.5} + 7 \left(\frac{D_H}{l} \right)^3 \right) \frac{0.074}{Re_l^{0.2}}$, where k_f is determined via CFD analysis assuming a hypothetical fuselage geometry, D_H , l are respectively the hydraulic diameter and the length of the fuselage;
- **Additional drag:** $C_{D,else} = \frac{k_e}{Re_c^{0.2}}$, which accounts for all losses not previously considered (e.g. wing–fuselage interactions or disturbed flow caused by the propeller).

The empirical coefficients k_{wb} and k_e were obtained from flight tests on LEObeta, the AeroUD ACC2024 aircraft.

Take-off run simulation:

The team considered a 40 m take-off mandatory due to its associated bonus.

To simulate the take-off run, the dynamic equation along the x -axis is imposed:

$$m\ddot{x} = T(\dot{x}) - D(\dot{x}, \alpha = 0) - \mu (gm - L(\dot{x}, \alpha = 0))$$

where the lift and drag forces are defined as $L = \frac{1}{2}\rho v^2 C_L S$ and $D = \frac{1}{2}\rho v^2 C_D S$, respectively.

The thrust T is interpolated as a function of $\dot{x}$, and μ is the rolling drag coefficient, obtained empirically from flight tests on the LEObeta (ACC 2024) aircraft on a concrete runway.

The maximum take-off mass (MTOM) is determined iteratively as the largest mass that satisfies the condition:

$$L(\dot{x}_{40}, C_L = 0.9C_{L,max}) > 1.1 \cdot MTOM \cdot g$$

where $\dot{x}_{40}$ is the aircraft speed at the end of the runway.

Empty weight estimate:

Aircraft weight plays a crucial role in determining the optimal configuration. Aerodynamically, a high aspect ratio wing combined with a slim airfoil is known to improve maximum speed but also increase weight due to the required additional structural strength.

The total mass of the aircraft is estimated as:

$$M_{tot} = M_{wing} + M_{fuse} + M_{motors} + M_{payload}$$

with the individual components computed as follows:

- **Wing mass:** $M_{wing} = k_1 \frac{m\Lambda}{t} + k_2 S + k_3 (N_S - 1) + k_4$, where the term $\frac{m\Lambda}{t}$ accounts for the wing spar weight, and $k_3 (N_S - 1)$ represents the mass of the wing connectors;
- **Fuselage mass:** $M_{fuse} = k_5 M_{tot} + k_6$, reflecting that the fuselage must be stronger (and therefore heavier) as the total mass increases;
- **Motor weight:** $M_{motors} = k_7 N_{motors}$, penalizing the twin-engine configuration;

- **Payload weight:** $M_{\text{payload}} = 0.35 \text{ g/cans} \cdot N_{\text{Cans}}$.

All coefficients were obtained from aircrafts that participated in previous editions, in particular PH-2 and LEObeta, corresponding to the ACC2022 and ACC2024 competitions, respectively.

Maximum number of Cans:

The maximum number of liftable cans must be determined iteratively. Specifically, $N_{\text{cans,max}}$ is defined as the largest number of cans for which the total mass satisfies:

$$M_{\text{tot}}(N_{\text{cans,max}}) < \text{MTOM}$$

Flight Simulation Iteration

At this stage, the flight can finally be simulated. An iterative procedure is used: for each number of cans $N_{\text{cans}} \in [2, N_{\text{cans,max}}]$, the flight is simulated and a corresponding score is computed. This approach ensures that configurations with a partially filled payload are also evaluated, as they may yield a higher overall score.

The dynamic equations of motion considering just the horizontal and vertical displacements degrees of freedom are:

$$\begin{cases} m\ddot{x} = T(v) \cos \alpha - L(v, \theta) \sin \alpha - D(v, \theta) \cos \alpha, \\ m\ddot{z} = T(v) \sin \alpha + L(v, \theta) \cos \alpha - D(v, \theta) \sin \alpha - mg, \end{cases}$$

where: $v = \sqrt{\dot{x}^2 + \dot{z}^2}$; α is the ground angle of attack; $\gamma = \tan^{-1}(\frac{\dot{z}}{\dot{x}})$ is the velocity vector angle; and $\theta = \alpha - \gamma$ is the angle between the wing chord and the velocity vector.

The optimal flight strategy is defined as follows:

- Full throttle is applied throughout the entire flight;
- $\alpha = \alpha_r$, chosen such that $C_L(\alpha_r) = 0.9C_{L,\text{max}}$, is maintained up to 60 s;
- $\alpha = \alpha_s$, chosen such that $z(t = 180 \text{ s}) = z_{\text{lim,inf}}$, ensures that the maximum potential energy is converted in kinetic energy.

The flight is integrated using an explicit Euler method. At each instant $t \in [0, 180 \text{ s}]$, the state

$$\text{state}(t) = \{x, \dot{x}, \ddot{x}, z, \dot{z}, \ddot{z}\}$$

is fully known, allowing straightforward computation of total covered distance during the speed section (D_{SF}).

Flight score:

Finally the flight score can be computed as:

$$SCORE = N_{cans} \cdot D_{SF}^2$$

For each different airfoil the best score is computed such that the optimal aircraft $A_{opt} = \{p, a, c, N_s, d\}$ is found.

3.3.2 Final aircraft geometry

From the optimization process described above, the best-performing aircraft was identified (Table 3 and Figure 5).

Airfoil	AG24
Propeller	14x8
Twin propeller (d)	NO
Number of sections (N_s)	3
Root chord (c)	340 mm
Wing span (b)	2.85 m
MTOM	6.2 kg
Empty weight	2.6 kg
Number of cans	10
Speed flight (D_{SF})	3.138 km

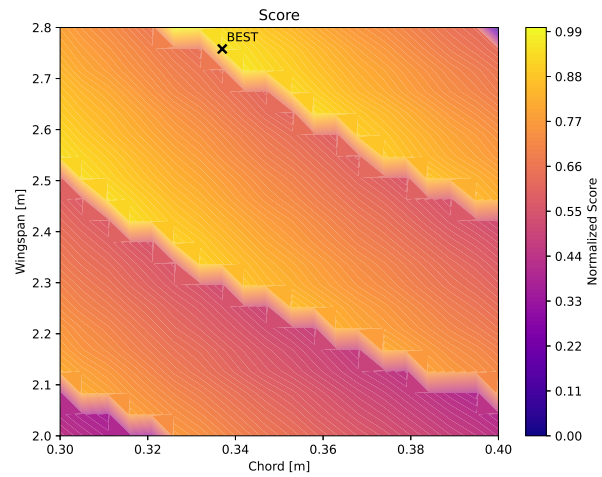


Table 3: Optimization results

Figure 5: Optimization results for AG24-optS

The selected airfoil was further optimized using *xoptfoil* with the following objectives:

- **Minimum thickness:** $th_{min} = 10\%$, ensuring the retractable undercarriage would fit inside the wing;
- **Lift maximization at low speed:** C_l maximized at $Re = 220k$, taking into account a 70% hinge flap with the positive angle treated as an optimization variable;
- **Drag minimization at high speed:** C_d minimized at $Re = 550k$, considering negative flap deflections at the same 70% hinge, with the angle again optimized.

And the obtained airfoil is shown in Figure 6.



Figure 6: AG24-optS

Finally, an elliptical wing planform was designed, with the fine tuning of the dimensions

obtained from the optimization tool for the specific final configuration (Figure 7). This was done to approximate an elliptical lift distribution as closely as possible in accordance with the Schrenk Criterion, while ensuring a tight fit in the transportation box.

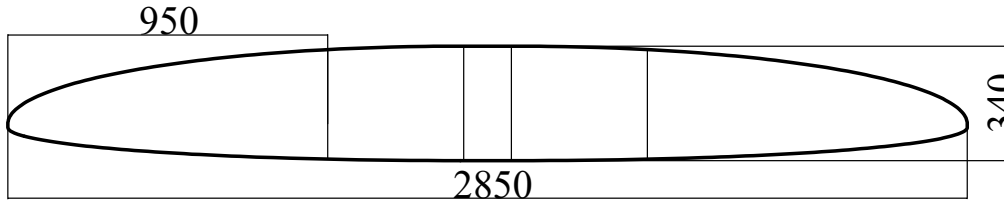


Figure 7: Wing planform

3.4 Stability analysis and tail design

The aircraft adopts a full-flying V-tail configuration, continuing the team tradition, due to its good maneuverability, aerodynamic efficiency, and manufacturability. The design and stability analysis were carried out using the software *XFLR5*.

A symmetrical *NACA-0008* airfoil was selected for its low drag and sufficient thickness to house servos and spars. The tail pitch angle was set to 0° to minimize drag.

A conventional T-tail was first sized using literature correlations and later converted into the final V-tail configuration. Stability requirements were imposed using the horizontal (V_H) and vertical (V_V) tail volume coefficients and the spiral stability parameter (B) (MIT Unified Engineering, 2006):

$$V_H = \frac{S_H l_H}{S_W MAC} = 0.4 \quad V_V = \frac{S_V l_V}{S b} = 0.25 \quad B = \frac{l_V \gamma_e}{b C_L} = 4$$

where: the tail arm $l_H = l_V = L_{TB} - x_{CG} + 0.25c_T$ and L_{TB} is the tail boom length. The tail chord c_T was fixed at $c_T = 162 \text{ mm}$ to be as small as possible to minimize drag while still large enough to house the servo.

Coefficient values were based on the successful *PH-2* aircraft (ACC2022), an elliptical planform was adopted to reduce induced drag without increasing manufacturing complexity and a static margin of $SM = 10\%$ was imposed:

$$SM = \frac{x_{NP} - x_{CG}}{MAC} = 10\%$$

The center of mass was computed considering component weights and positions. The wing motor distance X_w and the tailplane dimensions (b_h and c_H) have to be computed.

The horizontal tail sizing was obtained through the following iterative procedure:

Step 1: First try L_{TB} .

Step 2: Compute b_H from $S_H = \frac{\pi}{2} b_H c_H = \frac{V_H S MAC}{l_H}$.

Step 3: Compute X_{CG} given the aircraft mass distribution.

Step 4: Compute x_{NP} using XFLR5.

Step 5: Compute $x_{CG} = x_{NP} - MAC \cdot SM$.

Step 6: Modify wing positioning X_w to match the desired CG position, L_{TB} changes, repeat 2-6 until convergence.

Step 7: Modify b_H so that $c_M = 0$ at $AOA = 0$, x_{NP} changes, repeat 2-7 until convergence.

The resulting horizontal tail parameters are reported in Table 4.

The vertical tail is then sized using the corresponding volume coefficient:

$$S_V = \frac{S b V_V}{l_V} \quad b_V = \frac{S_V}{c_T \frac{\pi}{4}}$$

Results are reported in Table 4.

Due to the low-wing configuration, additional lateral stability was ensured by introducing tip dihedral. Using literature correlations, the equivalent dihedral angle was computed as

$$\gamma_e = \frac{b C_L B}{l_V}$$

Assuming an elliptical lift distribution:

$$\gamma_e = \frac{\int_0^1 \gamma(\eta) \sqrt{1-\eta^2} d\eta}{\int_0^1 \sqrt{1-\eta^2} d\eta} = 3 \int_0^1 \gamma(\eta) \sqrt{1-\eta^2} d\eta$$

The resulting values are summarized in Table 4.

Finally, the V-tail geometry was derived by equating its horizontal and vertical projections to those of the T-tail, as successfully done by the team in previous models:

$$V \sin^2(\delta/2) = \frac{S_H}{2}, \quad V \cos^2(\delta/2) = \frac{S_V}{2}$$

where δ is the V-tail internal angle, V is the V-tail surface. Assuming the same chord $c_T = 162 \text{ mm}$ and an elliptical planform, the final V-tail parameters are shown in Table 4.

The payload tray center is positioned at the aircraft centers of mass. Given that the tray is designed for 11 cans and is thus not symmetric, two battery positions (one for odd number

SM[%]	$x_{NP}[mm]$	$x_{CG}[mm]$	$x_W[mm]$	$c_T[mm]$	$b_T[mm]$	$V[m^2]$	$\delta_T[^\circ]$	$\gamma[^\circ]$
10	169	130	608	162	469	0.06	106	3.5

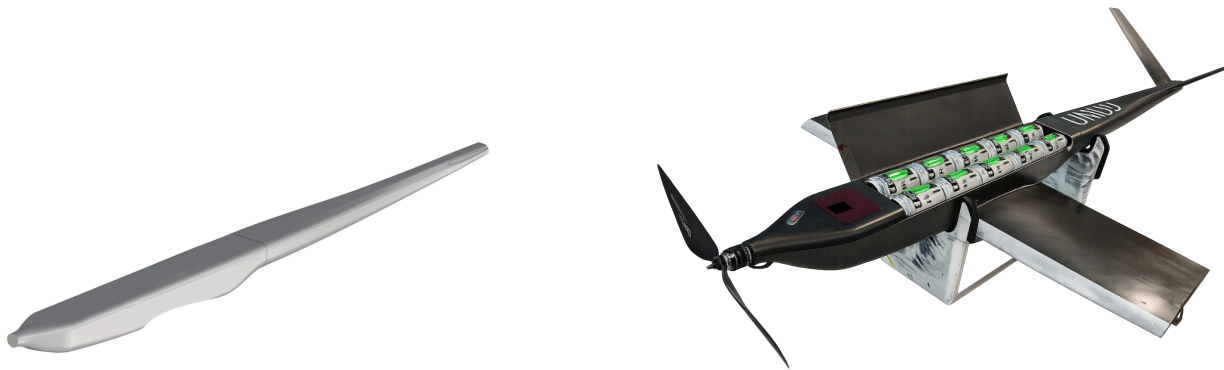
Table 4: Stability analysis results

of cans one for even) are found so that the total center of gravity stays in the same position, ensuring a 10% static margin for each configuration.

3.5 Fuselage design

The design of the aircraft fuselage is a crucial aspect that directly affects performance and efficiency. The main goal was a design that could accommodate the regulatory payload while minimizing the wet surface and the presence of sharp edges. This approach aims to improve the aircraft's overall aerodynamics, reducing drag and improving flight efficiency.

The fuselage, like the rest of the aerodynamic surfaces, was modeled using Rhinoceros 7. The cross-section was chosen based on the cargo bay to be transported (Figure 8a) where the cans are arranged longitudinally in two parallel rows to minimize frontal area and increase aerodynamic efficiency (Figure 8b).



(a) Fuselage CAD model

(b) Assembled fuselage, with cans

Figure 8: Fuselage final design

The fuselage was designed in two parts of equal length (same length as the wing section) so that it could be disassembled and placed into the transport box. The front part, designed to house the cargo bay, has an almost rectangular shape with filleted edges and draft angles to facilitate extraction from the molds.

Given the low-wing configuration, the lower section of the fuselage features a shaped area to house the wing.

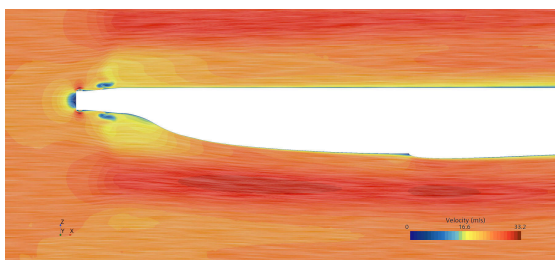
The nose was modeled to contain the ESC, the batteries and the telemetry box, featuring a circular cross-section at the front to accommodate the motor, transitioning to a semi-

rectangular shape in order to minimize drag. The tail end was specifically designed to have two flat surfaces shaped like the tail airfoil, serving as the insertion point for the tail assembly.

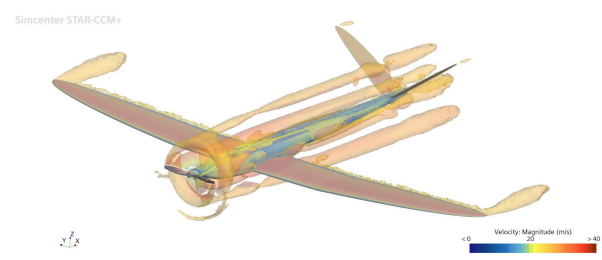
The fuselage was designed so that the center of gravity of both the cargo bay and the aircraft coincide, leaving some margin to accommodate potential displacements due to small errors in weight prediction.

Regarding the shape of the front section, several options with different fairing shapes between the nose and the wing's leading edge were evaluated to determine the most efficient one. To this end, CFD simulations of the entire aircraft were performed by means of Star-CCM+, leading to the configuration shown in the Figure 9a.

The simulations, which were also conducted by introducing a rotating domain to account for the propeller's effect on the entire structure, allowed for the evaluation of the aircraft's aerodynamic behavior (Figure 9b).



(a) Fuselage flow field



(b) Turbulence around the aircraft

Figure 9: CFD simulations for the fuselage design

4 Structural Design

This section presents the internal structure of the aircraft and the key design choices that guided the final configuration.

Due to the complexity of the structural design of composite components, the team relied on previous competition experience and conducted an experimental campaign to improve the wing spar sizing.

4.1 Wing structure

The internal wing structure is composed of a wing spar (made of a hand laminated CFRP sandwich as web and UD tapes as flanges) and ribs (made of a lightweight CFRP sandwich). To correctly size the wing spar, an experimental campaign was launched, with two distinct phases. The first with the objective to identify the best material for the spar and the second

to find a relation between the number of tapes used and the wing deflection under load.

In the first phase, in order to find the sandwich for the spar web with the best compromise between weight and resistance, different cores and reinforcement materials were explored, also accounting for their orientation. Tensile and bending tests were conducted for each configuration using the universal testing machine available at the University of Udine (Figure 10).

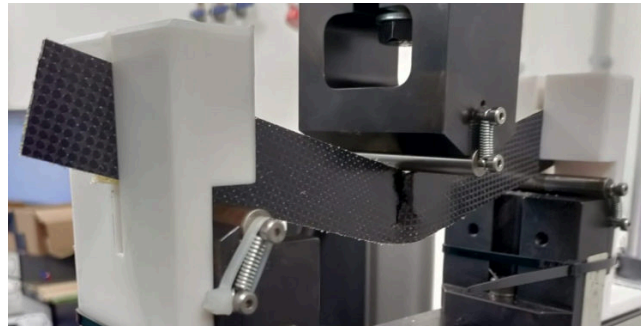


Figure 10: Bending test for one of the specimens

For the core materials, balsa wood — nowadays widely used in the design of composite structures such as wind turbine blades — and PVC expanded foam were selected, both in 2 mm sheets. For the reinforcement, both carbon fiber (40 g/m² and 80 g/m²) and glass fiber (80 g/m²) were investigated in 0–90° and ±45° orientations.

The results, shown in Figure 11, were processed using beam theory to obtain comparable quantities across different configurations. With the goal of identifying the best compromise between structural strength and weight, balsa wood clearly outperformed PVC, while carbon fiber outperformed glass fiber. Moreover, a 40 g/m² carbon fiber reinforcement at ±45° orientation proved sufficient for the application.

In the second phase, in order to characterize the selected configuration, representative prototypes of the spar were manufactured by gluing a web made of ±45° carbon fiber-reinforced balsa (2 mm) as selected in the first phase and with flanges made of unidirectional layered carbon tapes (*Textreme* 38 g/m² IM), with each specimen consisting of an increasing number of tapes.

The specimens were produced using a hand lay-up technique with the different part bonded together using structural glue (*Elan-Tech* ADH 891.892). The specimens were then tested under three-point bending with varying configurations and load levels (Figure 12). The results were again processed using beam theory to determine the equivalent bending stiffness:

$$\{EI\}_{eq} = \frac{PL^3}{48f}$$

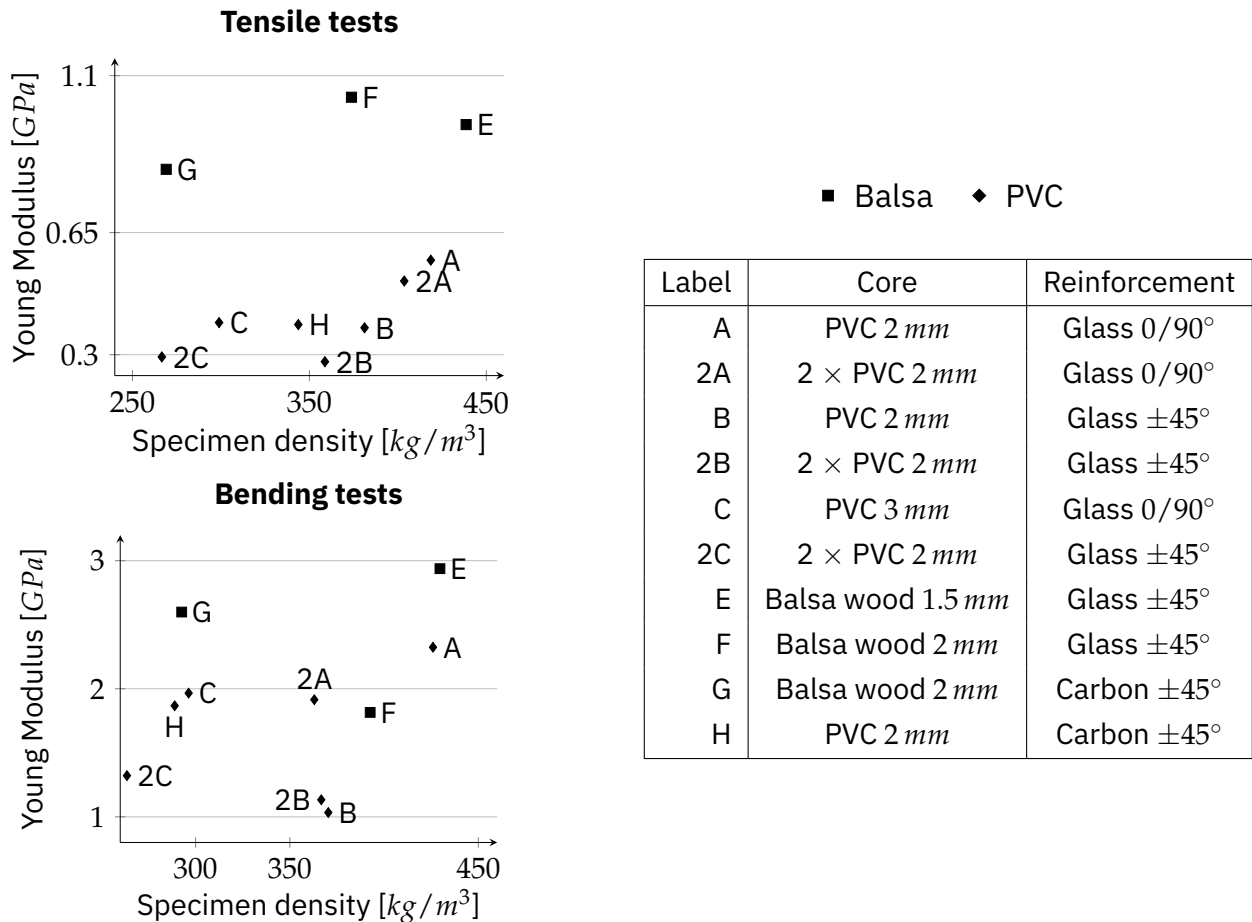


Figure 11: Results of the preliminary experimental tests. Note that, in the bending tests, indentation effects occurred, leading to an overestimation of the Young's modulus.

A simple linear relationship between equivalent bending stiffness and reinforcement was observed:

$$\{EI\}_{eq} = \alpha \cdot N \quad (1)$$

where α is the coefficient of the interpolating line, obtained experimentally, N is the number of unidirectional carbon tapes in the flanges (Figure 13).

Based on these results, a design criterion for spar sizing was established starting from a maximum allowable deflection.

Using the simply supported beam equation under a concentrated load, with the required reinforcement N directly determined, enabling a fast and repeatable sizing procedure:

$$f = \frac{PL^3}{48\{EI\}_{eq}} \quad (2)$$

The spar is manufactured as an I-beam structure through manual lamination followed by

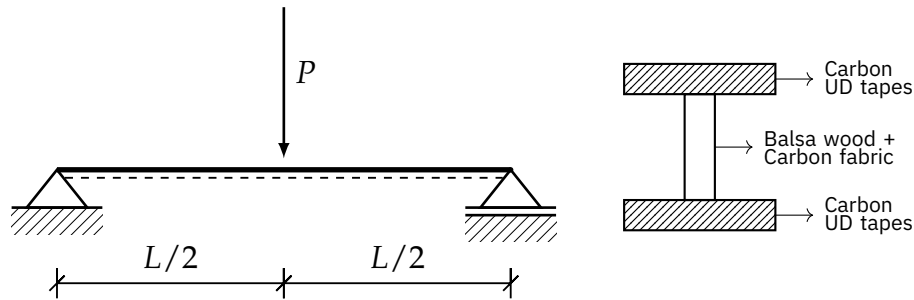


Figure 12: Three point bending test for the beam specimens

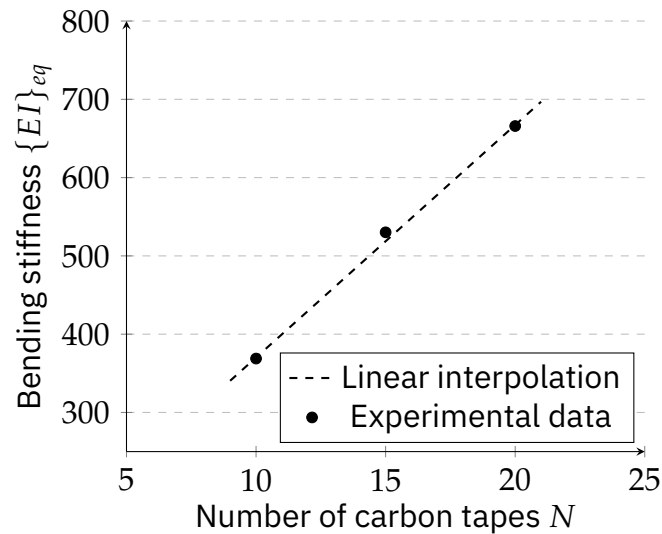


Figure 13: Equivalent bending stiffness as a function of reinforcement quantity

bonding:

- flanges made of 12 UD carbon tapes, laminated directly into the wing skin;
- web made of 2 mm balsa core with $40 \text{ g/m}^2 \pm 45^\circ$ carbon fiber fabric.

Wing ribs are produced using the same material of the spar web, a 2 mm balsa core reinforced using $40 \text{ g/m}^2 \pm 45^\circ$ carbon fiber fabric.

The wing skin is a sandwich structure designed to withstand torsional loads while preserving the aerodynamic shape. It consists of a composite sandwich made of non-crimp carbon fabric (*CarboWeave* 40 g/m^2 , $\pm 45^\circ$ IMS) on both the internal and external sides and *Rohacell* IG-F31 foam (1 mm) as the core.

Due to the undercarriage being mounted in the wing root, reinforcement made of triaxial 60 g/m^2 carbon fiber and 36 g/m^2 kevlar fabric are placed on both sides of the sandwich to ensure sufficient rigidity.

As determined by aerodynamic optimization, the wing is divided into three sections: one central section mounted beneath the fuselage (the root), and two outer sections (the tips).

These sections are connected using joiners manufactured in-house (Figure 14), preserving the designed dihedral angle, withstanding bending moments and shear forces acting on the wing.

To meet these requirements, a hand layup lamination process was adopted using CNC-milled aluminum molds (Figure 14).

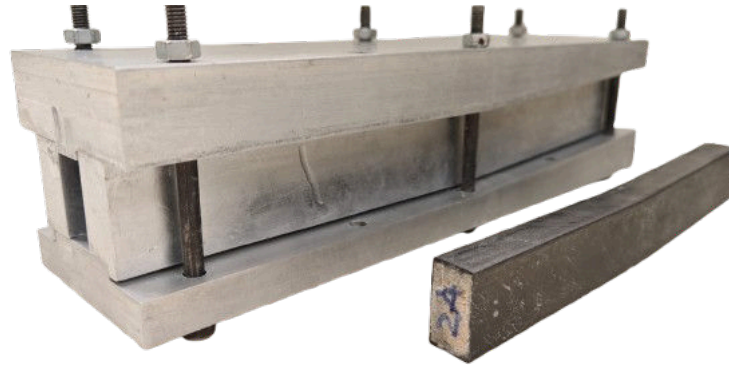


Figure 14: Wing connector and its aluminum mold

Load tests, performed on the joiners, allowed for material selection and determination of the required reinforcement. The chosen configuration is a sandwich structure. The core is made of hand-milled PVC (*Divinycell 80*), while the primary reinforcement consists of 12 carbon rovings (*DowAksa 24k 1600tex*) on each side. A carbon fiber layer wraps the structure to ensure shear strength and full consolidation of the connector. The pockets are made of a carbon fiber braided sleeve (*Toroyaca $\phi 35$ 45° 200 tex 3k*).

4.2 Fuselage structure

The wing is directly attached to the fuselage using 3 M5 nylon screws to ensure rigid coupling with particular interest on torsional stiffness.

The fuselage integrates the payload bay, engine mount, telemetry box and all aircraft electronics. The main duties for the fuselage are allowing rapid payload loading and withstanding inertial loads during flight. It consists of two sections connected by a coupled flange made of 6 M4 nylon screws with alignment pins made from 3 mm carbon fiber rods.

Each section is manufactured via vacuum bagging, consisting of triaxial non-crimp carbon fiber fabric (*CarboWeave 60g/m², 0° ± 45°*) and 2 mm *Rohacell IG-F31* sandwich panels as a single composite part, resulting in improved aerodynamic efficiency and a self-supporting structure, reducing the need for internal frames and minimizing weight.

The payload is loaded from above supported by a tray made from CNC milled polystyrene (30 kg/m³), providing lightweight support.

4.3 Landing gear

In order to reduce aerodynamic drag, the adoption of retractable landing gear was proposed.

Integrating the mechanism within the limited space inside the wing proved challenging. Different landing gear systems were tested. A commercial gear purchased on *Amazon* proved to be structurally robust, lightweight, and easy to integrate into the wing. The iron hinge that was originally included with the mechanism was replaced with an aluminum hinge produced using the SLM machine at the LAMA laboratory at the University of Udine. This change was made to reduce the mechanism's weight, making each mechanism 15 g lighter than the original (Figure 15).

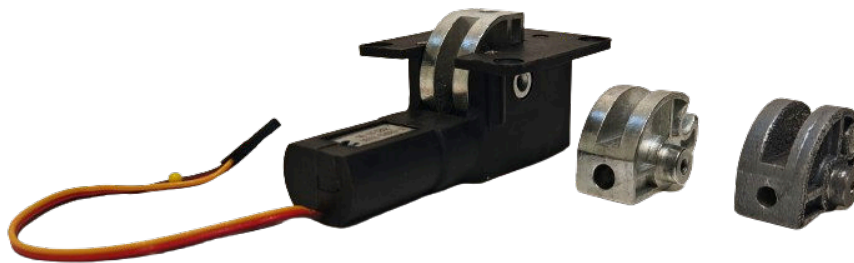


Figure 15: Retractable mechanism, comparison between iron and aluminum hinges

4.4 Construction process

Vacuum bagging hand lay-up technique was selected for the manufacturing of all composite components.

Different molds were used:

- wings: outsourced aluminum molds for superior surface finishing;
- fuselage and tail surfaces: cheap in-house manufactured MDF molds.

Molds are machined from MDF panels using a 3-axis CNC router. This required the team to learn using CAD/CAM technologies for complex geometries. After machining, molds are painted and polished using 1000 grit sandpaper to achieve a low surface roughness. The molds are finally treated with 5 hands of silicone-free release agent.

A high-quality epoxy resin was selected as matrix material, after the lay-up phase, components are cured under vacuum at temperatures above 24°C for at least 24 hours.

During final assembly, careful alignment and interference checks are performed to ensure correct geometry and structural integrity, while molds are closed using structural glue.

5 Propulsion System Optimization

This section is dedicated to describing the propulsion system and its optimization process that guided the final configuration.

5.1 Current regulation system

Among the variables that contribute to the competition's score calculations, the current limit is a key parameter: exceeding 30 A results in a big penalty on the final score. For this reason, the team found essential to identify a solution that allows for precise, stable and reliable current regulation.

In the 2024 edition an external current regulator was implemented. For the 2026 season, an ESC with integrated current limit capabilities was chosen in order to simplify the system's architecture and improve responsiveness. Several ESCs were evaluated, and the Ibex 45 LV ESC was selected due to its compatibility with the MAV Sense development framework, enabling firmware-level customization through direct collaboration with the MAV Sense development team to meet the specific requirements outlined below.

The team started an experimental campaign leveraging the test rig already described in Section 3.1, with the objective of fine tuning the ESC current regulation. The data was acquired through both the Unisens-E sensor (the one used in the competition), a 70 A 10 kHz amperometric clamp and the internal telemetry of the ESC, which provided among others measures for current, voltage and temperature through a proprietary interface. The data collected was subsequently processed using a MATLAB script and analyzed through comparative graphical representations and analytical calculations of the current penalties across the different configurations.

During the first tests, it was observed that although the current limit was set to 30 A on the base firmware, the measurements recorded by the Unisens-E showed an abnormal behavior of the current, with both a progressive decrease over time and an anomalous ramp (Figure 16).

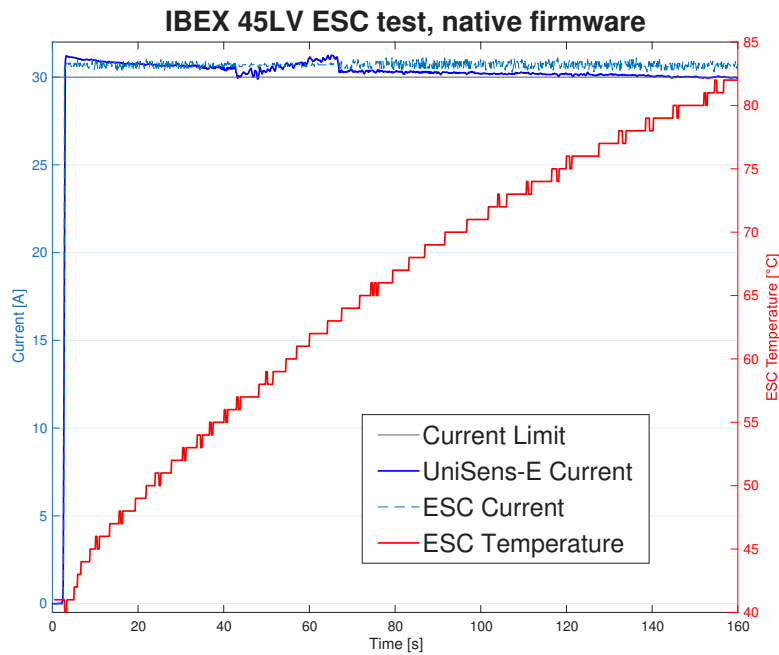
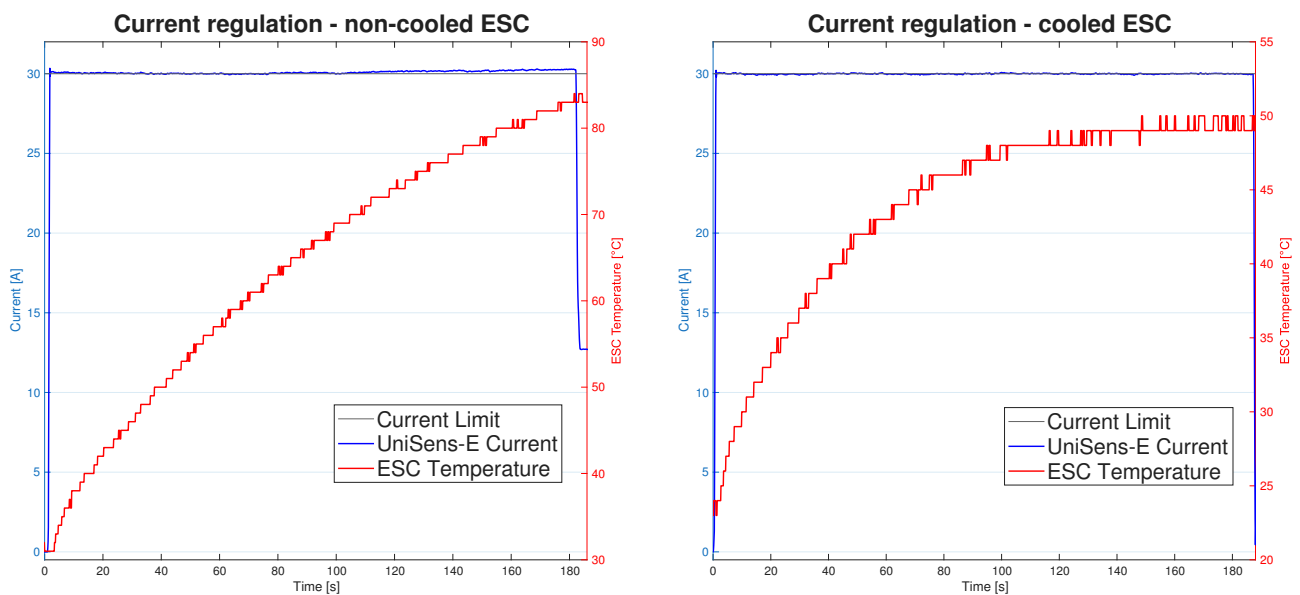


Figure 16: Out of the box test for the Ibex 45 LV ESC

Through an in-depth analysis of the data, the electronics team identified the temperature rise as the cause of the current drop, which was not accounted for by the base firmware in the ESC. To address this problem, a temperature correction was introduced. Still, as shown in Figure 17a, a small error is measured with the temperature rise. Therefore, cooling fins, soldered to the ESC, are introduced, ensuring that the temperature does not exceed 50°, enabling stable regulation with a current penalty of just 0.3% (Figure 17b).



(a) Current regulation - non cooled ESC

(b) Current regulation - cooled ESC

Figure 17: Non-cooled vs cooled ESC current regulation

Using the updated firmware version and the cooling fins, a marked improvement was observed with a system that could stably regulate the current at the desired 30 A.

Finally, the firmware was updated to implement an in-flight current limitation set point via one of the radio transmitter potentiometers. This allows the current limit to be fine-tuned to the official Unisense-E that will be used during the competition, and enables the limit to be increased during flight if necessary (e.g. to reduce take-off length, gain altitude, or increase speed).

5.2 Battery pack selection

Once the current regulation system was ready, the electronics team shifted its focus to the choice of the battery pack. Ground tests were performed using the same test rig (Section 3.1) with the objective of finding the battery pack with the best ratio between weight and voltage drop.

The measurements were conducted over a 180 s time interval, corresponding to the competition flight time, while maintaining a constant flow velocity of about 20 m/s keeping the current limited to 30 A and measuring the voltage drop via the same Unisens-E used for the ESC optimization.

Batteries from different renowned manufacturers were tested, between them also a parallel of two smaller 1800 mAh Full Power batteries exhibiting the lowest voltage drop between the tested packs. Among batteries with comparable capacities, the 3000 mAh Xtron unit showed a more contained voltage drop compared to the other alternatives tested (Figure 18).

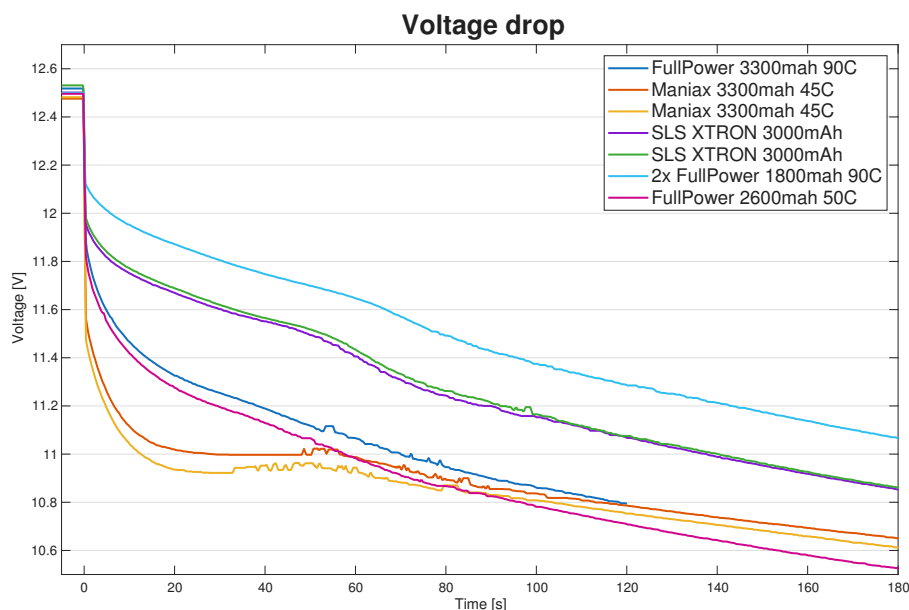


Figure 18: Voltage drops for the tested batteries

Test flights dedicated to the choice of the final pack will be conducted in order to assess the impact of weight and voltage drop on the final score to select the most suitable solution.

6 Control surfaces design and optimization

The team identified the control surfaces as a key factor in improving aircraft performance. This section describes their design and test stages.

From the wing aerodynamic optimization (Section 3.3.1) both positive and negative flaps were proposed (Figure 19), the former option to increase take off mass, the latter to reduce aerodynamic drag.

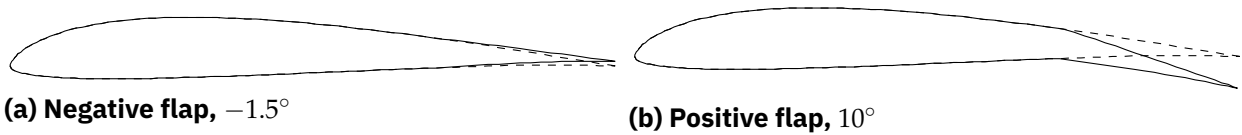


Figure 19: Positive and negative flap geometries, base profile in dashed line

Both flaps and ailerons have equivalent push and pull -rod driven kinematics (Figure 20) which is relatively simple and fits entirely inside the wing. The sizing process is presented in detail for the ailerons, given their larger size.



Figure 20: Flap and ailerons kinematics

To initially size the servo, the following geometric relation is used:

$$\tau = n \frac{1}{4} \rho v^2 S_F c_F \sin \varphi \frac{\tan \varphi}{\tan \beta}$$

where $\rho = 1.23 \text{ kg/m}^3$ is the air density, $v = 35 \text{ m/s}$ is the peak speed, $S_F = 0.06 \text{ m}^2$ is the aileron surface, $c_F = 0.075 \text{ m}$ is the mean flap chord, φ is the flap angle and β is the servo actuation angle. In the worst case, which is the fully activated surface, $\varphi = 40^\circ$ and $\beta = 65^\circ$. The obtained torque with a safety factor $n = 2$ is:

$$\tau = 0.831 \text{ Nm} = 8.48 \text{ Kg} \cdot \text{cm}$$

The *KST X10*—previously used for ACC2024—was thus selected and evaluated, providing

a torque of $8.9 \text{ kg} \cdot \text{cm}$ @ 7.0 V at a weight of 28 g , proving to be sufficiently strong. To reduce weight, the lighter *KST X10 Mini PRO* was also tested, delivering $6.9 \text{ kg} \cdot \text{cm}$ @ 7.0 V at 20 g which was likewise found to be adequate and was therefore selected due to its lighter weight.

Initially, a flap–aileron configuration was adopted (Figure 21), primarily to increase the maximum take-off mass. Flight tests were conducted to compare this setup with a flaperon configuration, revealing that the additional weight introduced by the flaps outweighed the benefits in take-off mass.

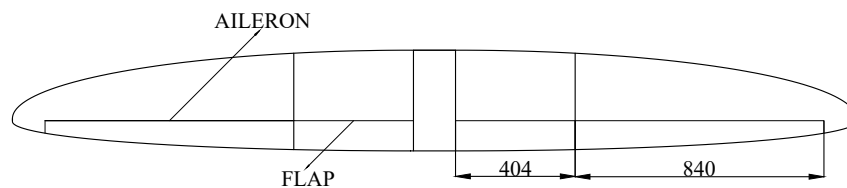


Figure 21: Wing initial flap configuration

Finally, the effectiveness of the negative flaperon configuration was evaluated during the speed segment of the flight, showing a clear increase in mean speed, as illustrated in Figure 22.

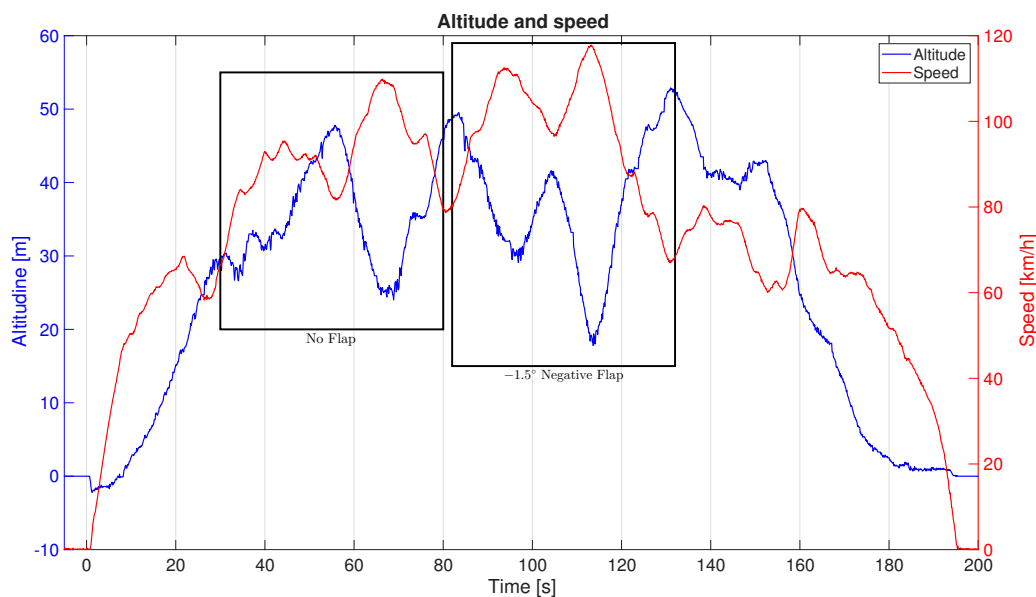


Figure 22: Negative flaperon tests

7 Payload Prediction

To define a payload prediction, the team launched a flight test campaign to find both the aircraft limits and the optimal strategy. To simplify the matter, given the scoring scheme as-

signed to payload prediction which heavily penalizes any overestimates, the team decided to just consider the worst case scenario – no wind at take off – to predict the strategy.

On a mild winter day (approximately 15°C) the maximum take off mass at 40 m was measured with the prototype at $MTOM_r = 6.3\text{ kg}$, with an air density of approximately $\rho = 1.23\text{ kg/m}^3$. It was thus possible to calibrate the code explained in detail in Section 3.3.1 for the take-off simulation, to extrapolate a simple linear equation for the maximum take off mass at different air densities, also accounting for a 5% safety factor:

$$MTOM = 2.17\rho + 3.61$$

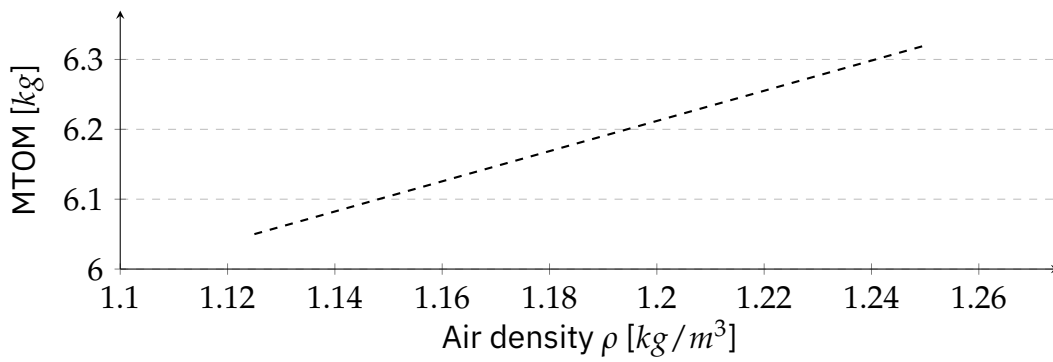


Figure 23: Payload prediction chart

Expecting an ambient temperature between 18°C and 30°C , corresponding to an air density range of $\rho = 1.16\text{--}1.21\text{ kg/m}^3$, and considering an aircraft empty weight of $E_W = 2.6\text{ kg}$, the predicted payload for our competition aircraft is 10 cans.

8 Outlook

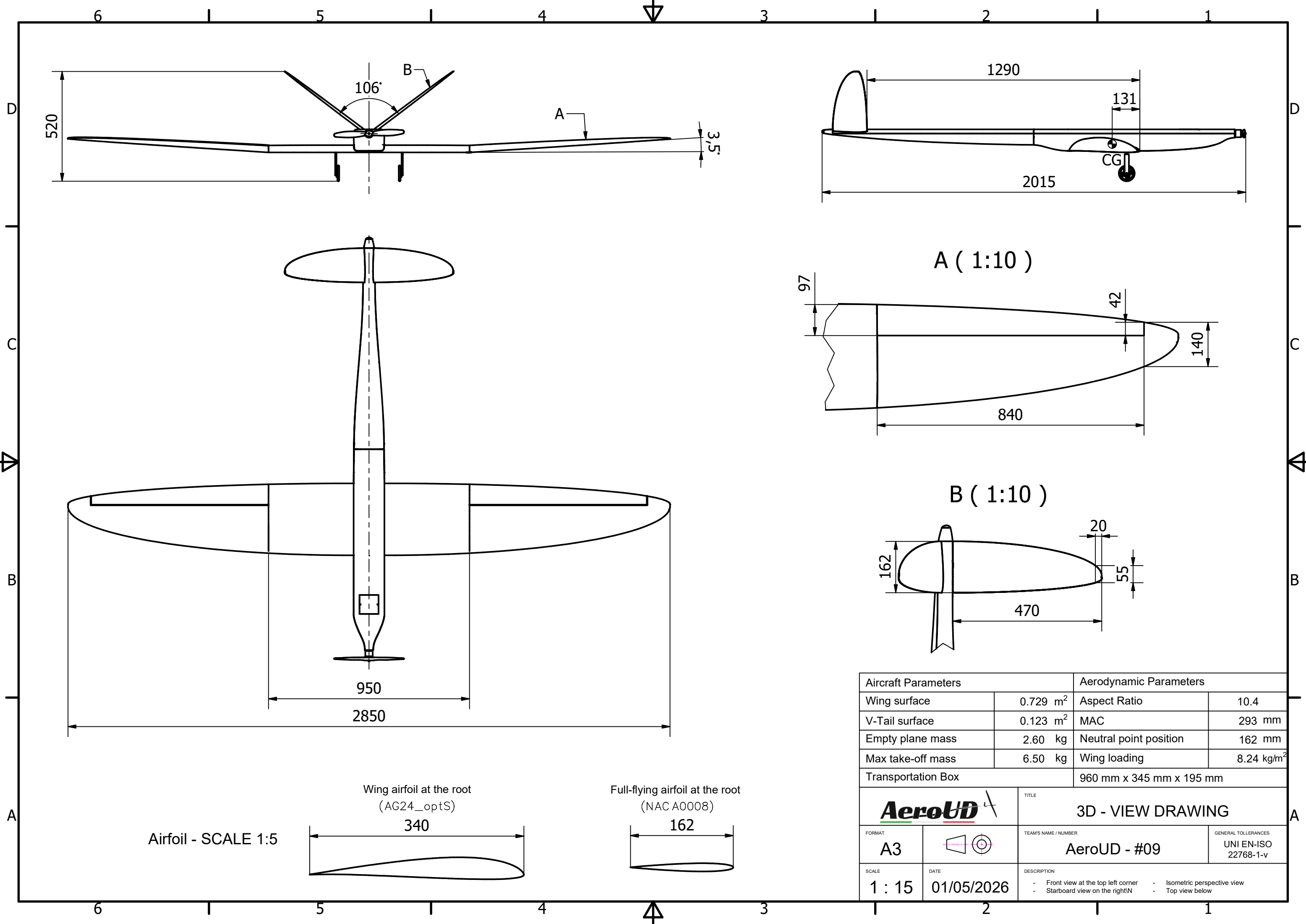
At the time of submitting this report, the team has already built and tested the first competition aircraft, and the parts for the second are being laminated. The second aircraft is expected to be completed in about 15 days. Once the second aircraft is ready, the team will focus on further structural optimization, selecting the battery pack and optimizing the flight strategy, with almost 50 flights already scheduled. During the design and construction phase, the team faced many difficulties, primarily due to the fact that almost the entire team was new to the aeronautical world.



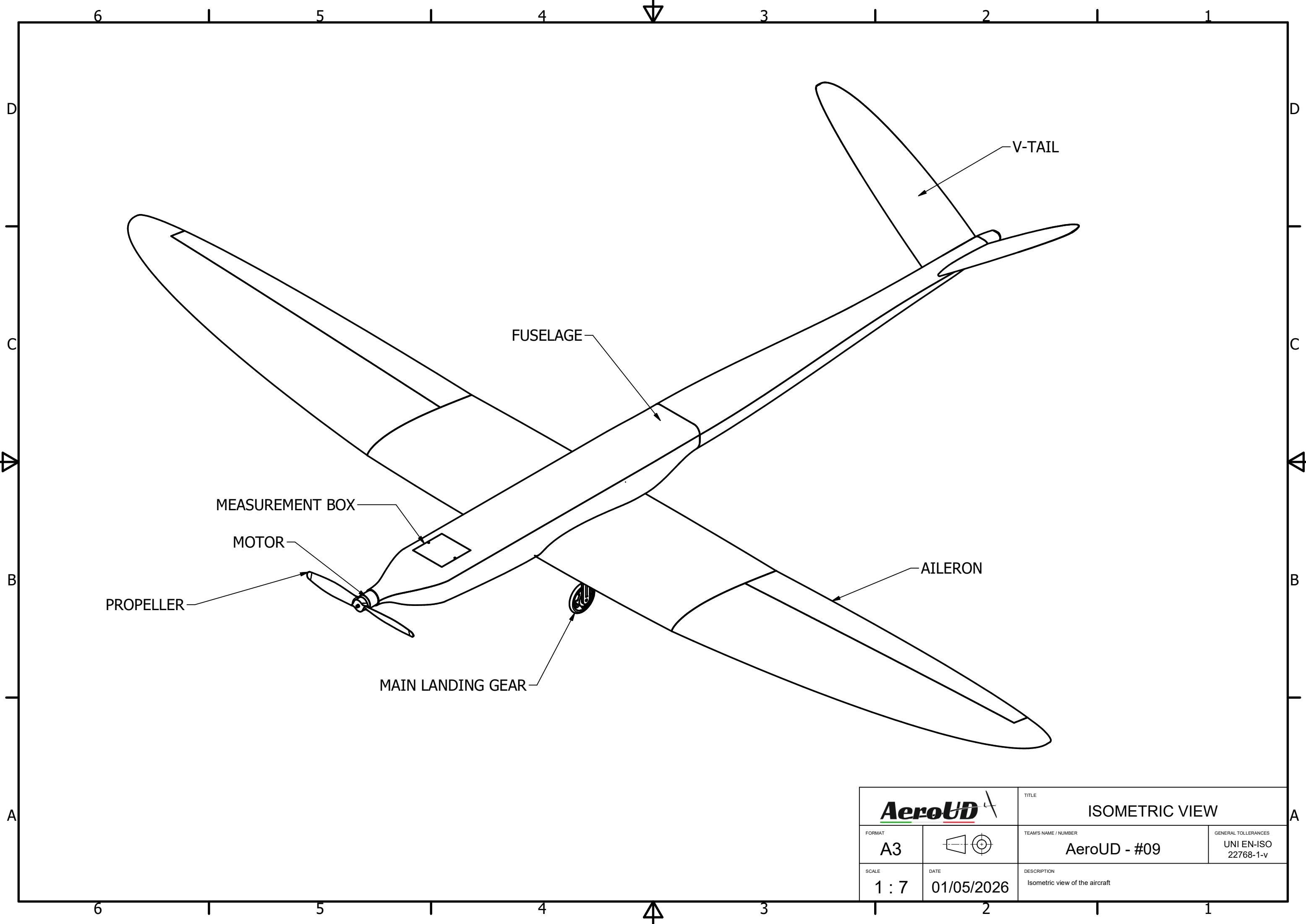
Figure 24: SandoCan, 2026 AeroUD aircraft

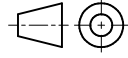
It is thanks to the passion of each team member, which is deeply rooted in the aeronautical history of our territory, that the project has been made possible once again this year. It has been a great challenge for all of us, but we are also very grateful to our university for giving us the opportunity to pursue our passions and compete against universities from all around the world on such a prestigious stage.

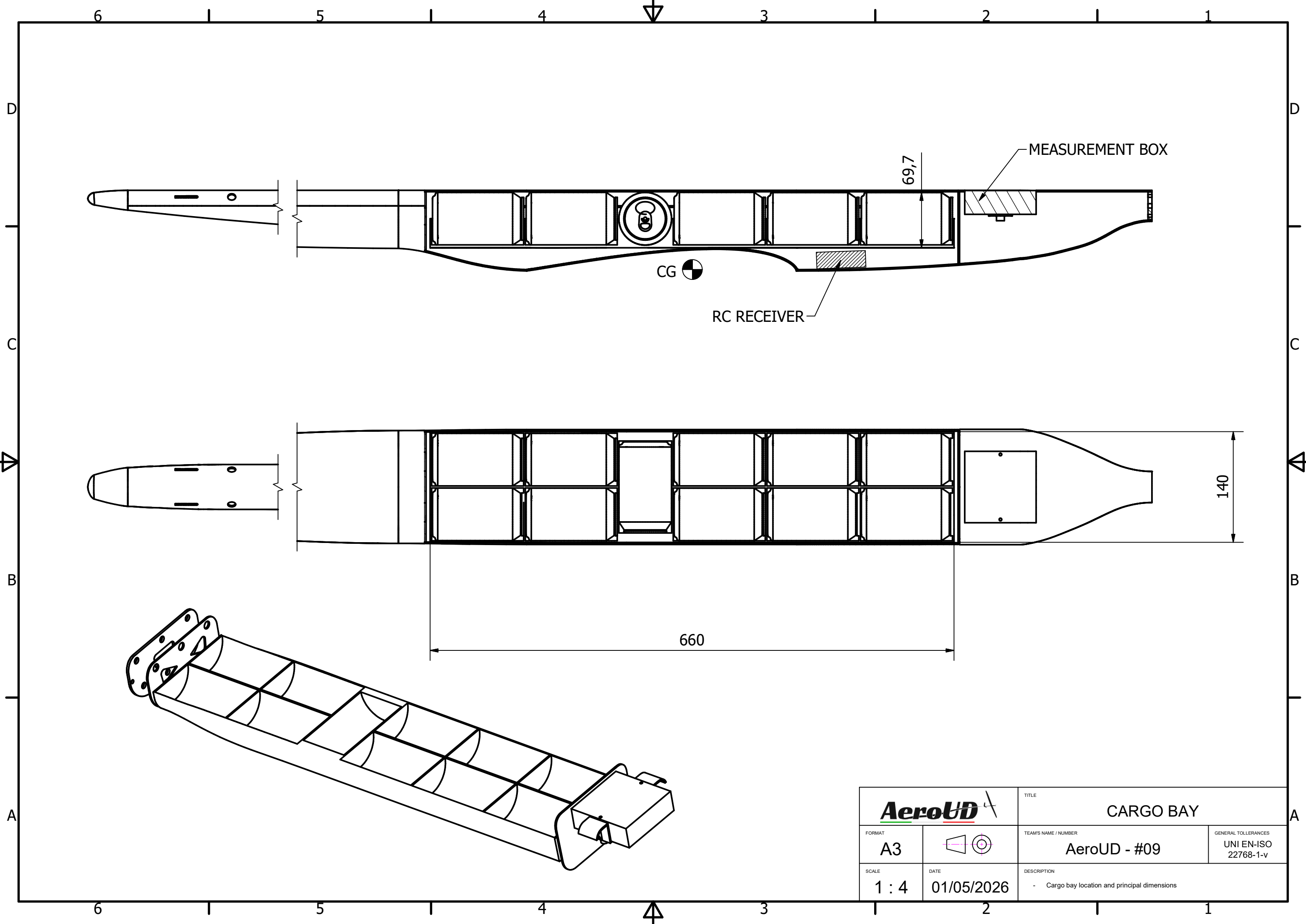
The 2026 team - AeroUD



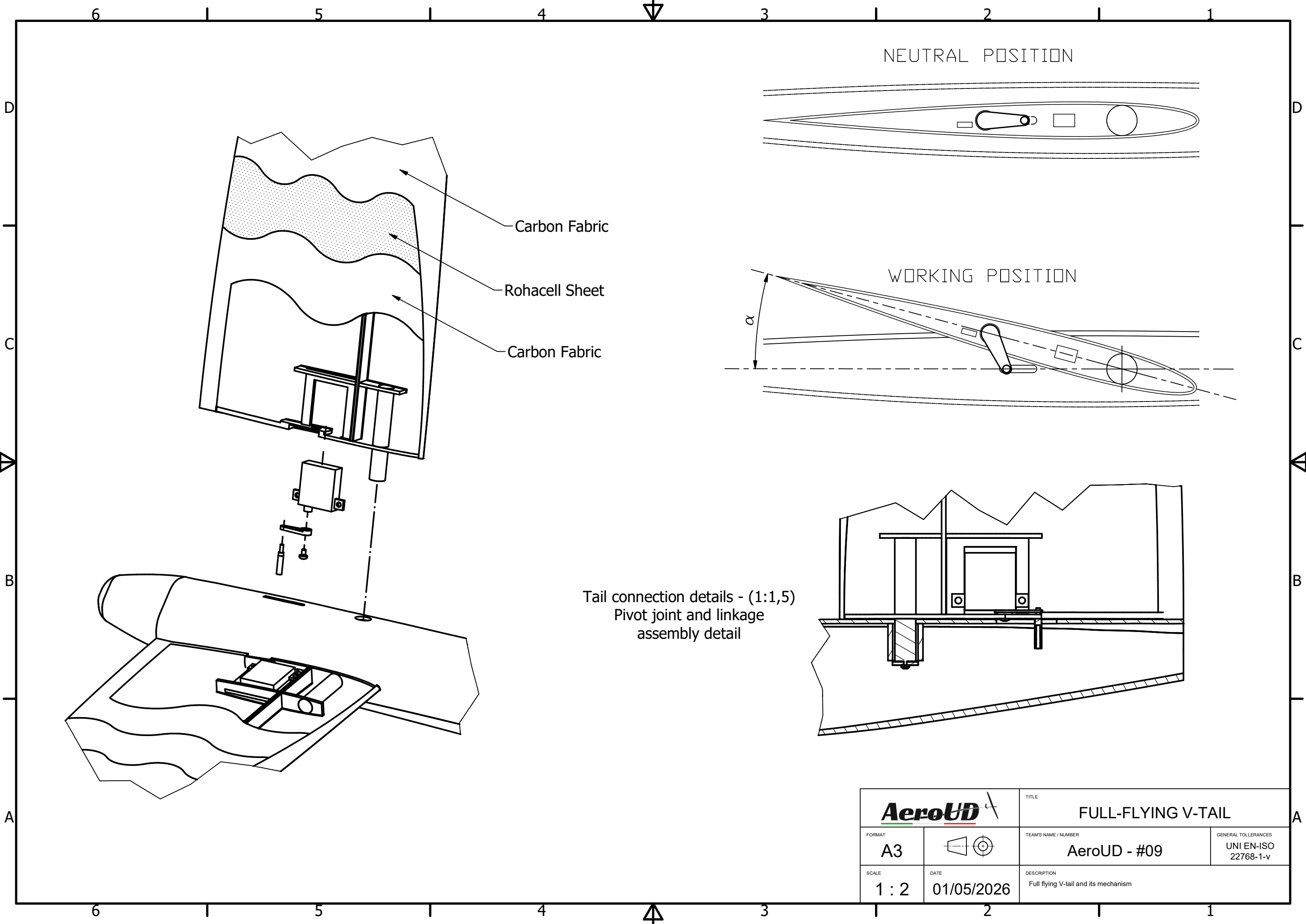
Aircraft Parameters		Aerodynamic Parameters	
Wing surface	0.729 m ²	Aspect Ratio	10.4
V-Tail surface	0.123 m ²	MAC	293 mm
Empty plane mass	2.60 kg	Neutral point position	162 mm
Max take-off mass	6.50 kg	Wing loading	8.24 kg/m ²
Transportation Box		960 mm x 345 mm x 195 mm	
AeroUD		TITLE 3D - VIEW DRAWING	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - #09	GENERAL TOLLERANCES UNI EN-ISO 22768-1-v
SCALE 1 : 15	DATE 01/05/2026	DESCRIPTION - Front view at the top left corner - Starboard view on the rightN - Isometric perspective view - Top view below	



		TITLE ISOMETRIC VIEW	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - #09	GENERAL TOLERANCES UNI EN-ISO 22768-1-V
SCALE 1 : 7	DATE 01/05/2026	DESCRIPTION Isometric view of the aircraft	



		TITLE CARGO BAY	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - #09	GENERAL TOLLERANCES UNI EN-ISO 22768-1-v
SCALE 1 : 4	DATE 01/05/2026	DESCRIPTION - Cargo bay location and principal dimensions	



AeroUD		TITLE FULL-FLYING V-TAIL	
FORMAT A3		TEAM'S NAME / NUMBER AeroUD - #09	GENERAL TOLERANCES UNI EN-ISO 22768-1-V
SCALE 1 : 2	DATE 01/05/2026	DESCRIPTION Full flying V-tail and its mechanism	