

UNIVERSITA' DEGLI STUDI DI UDINE



TEAM #026

REGULAR CLASS

SAE AERODESIGN EAST - 2013

Appendix
2013 SAE AERO DESIGN

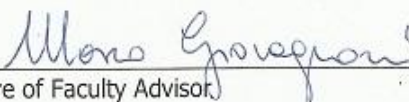
STATEMENT OF COMPLIANCE

Certification of Qualification

Team Name	AeroUD-MET	Team Number	026
School	UNIVERSITA' DEGLI STUDI DI UDINE		
Faculty Advisor	Dott. Ing. Marco Giovagnoni		
Faculty Advisor's Email	giovagnoni@uniud.it		

Statement of Compliance

As Faculty Advisor, I certify that the registered team members are enrolled in collegiate courses. This team has designed, constructed and/or modified the radio controlled airplane they will use for the SAE Aero Design 2013 competition, without direct assistance from professional engineers, R/C model experts or pilots, or related professionals.


Signature of Faculty Advisor

Team Captain Information:

Team Captain:	Kevin Altarac Marini
Captain's E-mail:	altaracmarini.kevin@spes.uniud.it
Captain's Phone:	+39-345-570-57-55

Note: A copy of this statement needs to be included in your Design Report as page 2 (see 6.1).

List of Contents

List of Figures.....	I
List of Tables.....	I
List of Symbols.....	II
Introduction.....	1
1. Configuration Selection Process.....	1
2. Aircraft Propulsion.....	2
3. Aerodynamic Design.....	6
4. Structural Design.....	13
5. Stability and Control.....	18
6. Estimated Performances.....	20
7. Conclusions.....	22
8. References.....	23
Appendix A - Payload Prediction Graph	
Appendix B – Aircraft plans	

List of Figures

Propeller efficiency from [1].....	4
Propeller velocities incidences.....	4
Optimum propeller pitch at varying flight velocity and engine speed, from [2].....	4
Test rig for propeller thrust measurements, left –engine mounted on the cantilever beam, center strain gauge, right – the engine placed in front of the open end wind tunne.....	5
Measured thrust of propellers with same diameter but different pitch.....	6
Measured thrust of commercial propellers.....	6
E423 (left), S1223 (center) and FX74CL5-140 (right) contours.....	9
Experimental and Xfoil C_l curves.....	10
Variation in C_l due to change in Reynolds Number.....	10
Lift distribution of the wing.....	11
Horizontal stabilizers performances.....	12
Destructive test for a foam specimen.....	14
Finite element model of the half-wing.....	17
Payload (left) and payload bay(right).....	20
payload prediction graph.....	21

List of Tables

FOM - Wing configuration options.....	2
FOM - Fuselage configuration options.....	2
FOM - Tail configuration options.....	2
FOM - Engine configuration .options.....	2
FOM – Selected aircraftconfiguration options.....	2
Airfoils Selection Process.....	9
Wing data.....	11
Static margin at varying payload.....	20
Take-off roll distance prediction.....	21

List of Symbols

Symbols

Units

Regular

AR	Aspect Ratio	dimensionless
b	Wingspan	mm
C_l	Airfoil coefficient of lift	dimensionless
C_d	Airfoil coefficient of drag	dimensionless
C_m	Airfoil moment coefficient	dimensionless
C_L	Airplane lift coefficient	dimensionless
C_D	Airplane drag coefficient	dimensionless
C_{ma}	mean aerodynamic chord	mm
e	Oswald's efficiency factor	dimensionless
g	Acceleration of gravity	m/s^2
i_h	Horizontal stabilizer incidence angle	degrees
l_f	Fuselage length	mm
l_T	distance between the 25% chords points of wing and horizontal stabilizer	m
l_V	distance between the 25% chords points of wing and vertical stabilizer	m
M	moment	Nm
L	Lift	N
S_{ref}	Wing planform reference area	m^2
S	surface/area	m^2
SM	Static margin	dimensionless
T	Thrust	kg
V_{LOF}	Lift-off velocity	m/s
V	Velocity	m/s
V_S	Stalling speed	m/s
W	Overall weight of the airplane	kg

Greek

α	Airplane angle of attack/flow incidence	degrees
α_{CL0}	Angle of attack at zero lift coefficient	degrees
α_{CLmax}	Angle of attack at maximum lift coefficient	degrees
ρ	Air density	kg/m^3
λ	Taper ratio	dimensionless

Subscript

W	wing
HT	horizontal tail
VT	vertical tail
ail	ailerons
cg	center of gravity
pn	neutral point

Introduction

The goal of the SAE Aero Design Competition is to design, build and fly a radio controlled aircraft to carry the maximum payload possible while adhering to certain design requirements. The most important requirements are the 200 ft. maximum takeoff distance, the cargo bay size requirement, and the ten foot minimum wingspan. The wingspan requirement adds a level of complexity to the design as it forces the designer to determine the optimum wingspan.

The design was broken into the following sections:

- Configuration/preliminary design
- Aircraft sizing
- Aerodynamics and structures
- Stability and control
- Weight and balance

The design philosophy of the team was to design an aircraft with good aerodynamic performance utilizing simple manufacturing techniques. This is due to the low amount of manufacturing experience and available time and money of the team. This philosophy played a large role in team design decisions. Much time was spent performing preliminary trade studies and sensitivity analyses to determine which design parameters have the greatest effects on aircraft performance. This was done to reduce the number of design variables to be considered, and to determine which parameters need to be closely monitored due to the large effect they have on aircraft performance.

1. Configuration selection process

To choose the conceptual design of the aircraft, the team determined the best performance assigning a Figure of Merit (FOM) to possible configurations, which scale from 1 to 5, being 5 the best result (Tabs 1-4). The team compared and analyzed each configuration and selected the total best result for each one.

Table 1: FOM- Wing configuration options

Wing	Performance	Manufacturing	Cost	Total
Monoplane	3.5	5	4.5	13
Biplane	4.5	3	2	9.5
Tandem	3	4	3.5	10.5
Canard	4	3.5	4	11.5

Table 2: FOM - Fuselage configuration options.

Fuselage	Performance	Manufacturing	Cost	Total
Conventional	3	5	4	12
Blended	3.5	3	4	10.5
Lifting	5	2.5	3.5	11

Table 3: FOM - Tail configuration options.

Tail	Performance	Manufacturing	Cost	Total
Conventional	4	4	5	13
V-tail	4.5	3	3.5	11
H-tail	4	4	4	12
Tailless	4	2	3	9

Table 4: FOM - Engine configuration options.

Engine configuration	Performance	Manufacturing	Cost	Total
Tractor	4	4	5	13
Pusher	3.5	2	3	8.5
Over the wing	5	1	2	8

Although the conventional monoplane wing configuration could have less wing area than a biplane, it was the best choice for eliminating complex airframe structures. The lifting body showed the best performance, but increases the overall empty weight of the airplane and the design and manufacturing time because of structural issues. Tractor engine configuration eliminates the necessity to rearrange tail booms and help the position of the center of gravity. The selected configuration is summarized in Tab. 5.

Table 5: Selected aircraft configuration

Wing	Fuselage	Tail	Engine Config
Monoplane	Conventional	Conventional	Tractor

2. Aircraft Propulsion

SAE AeroDesign main rules about aircraft propulsion are:

- the engine is the same for every team, precisely an OS61FX;
- the propeller must rotate at the same velocity.

The second one is a huge constrain for some reasons that will be explained latter in this section.

Since one of the principal target of the competition is to take-off in no more than 61 meters, it's obvious that accurate data about thrust are necessary to predict aircraft performances and give indication about the aerodynamic design (total drag must be lower than thrust).

Theoretical Background

During engine brake-in (performed with a Graupner 11x7) its rotational speed was measured by means of an optical tachometer, it turned out to be about 12000 rpm. This data can be used to understand what characteristics must have the propeller in order to maximize the thrust (e.i. the propeller efficiency). Propeller efficiency is a function of both its pitch and the dimensionless coefficient advance ratio, defined as:

$$J = V/nD$$

where V is the airspeed [m/s], n the revolutions per second [1/s] and D is the propeller diameter [m]. Since aircraft cruise speed is fixed at 13 m/s (see aerodynamics) and that the suggested propeller diameters for this engine are between 10 to 13 inches, the advanced ratio values are in the range $J=0.2-0.3$.

As reported by Brandt [1] for a similar propeller (APC 11x8) (see Fig. 1) the maximum efficiency for J values in the range of above is reached at one third the actual rotational speed. Considering that the engine speed cannot be reduced without reducing consequently the available shaft power, the other way is to go for a lower pitch value. It can also be seen that for high advance ratios (i.e. high airspeeds) a propeller with coarse pitch will be more efficient, conversely at low advance ratios (i.e. low airspeeds and high rpm) a propeller with fine pitch is the best choice. This can be better understood by looking at Fig. 2 where a propeller blade cross section together with the pertinent velocity vectors are reported. If the situation depicted at the top of Fig. 2 (coarse pitch-low advance ratio) is considered as an example, it can be seen how the propeller blade sees a relative velocity with a very high incidence angle, which leads to propeller stall and consequent loose of thrust. The stall can be avoided by reducing the propeller pitch (see Fig. 2).

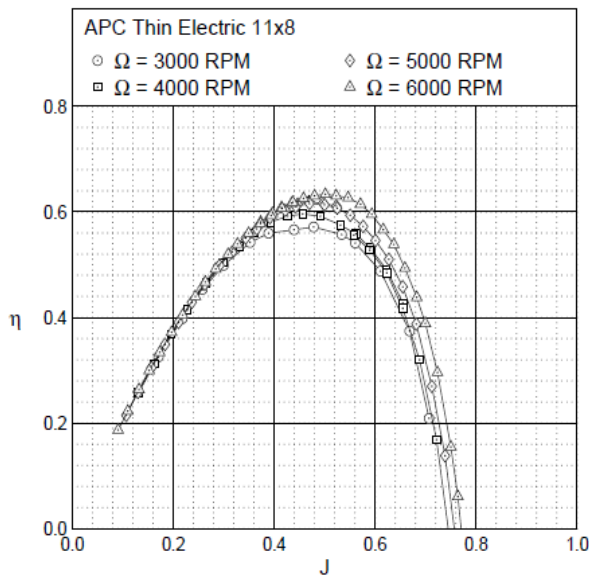


Figure 1: Propeller efficiency from [1]

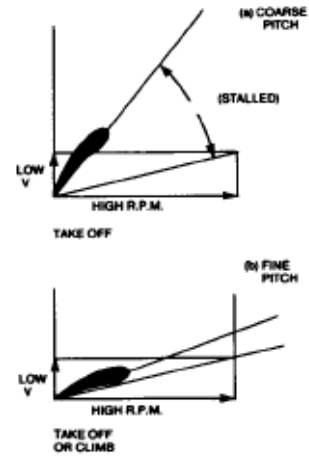


Figure 2: Propeller velocities

The propeller pitch was determined by the chart provided by Simmons [2] and reported in Fig. 3, while the diameter was calculated assuming that its tip velocity must be lower than the speed of sound:

$$D \leq \sqrt{\frac{\pi}{n} (a^2 - V^2)} \quad \text{where } a \text{ is the speed of sound.}$$

The best propeller configuration for n about 12000 rpm turned out to be 13 inches diameter and 3 inches pitch. Unfortunately, propellers with these dimensions are not available in the actual market. This justifies the propeller testing activity presented in the next sub-section.

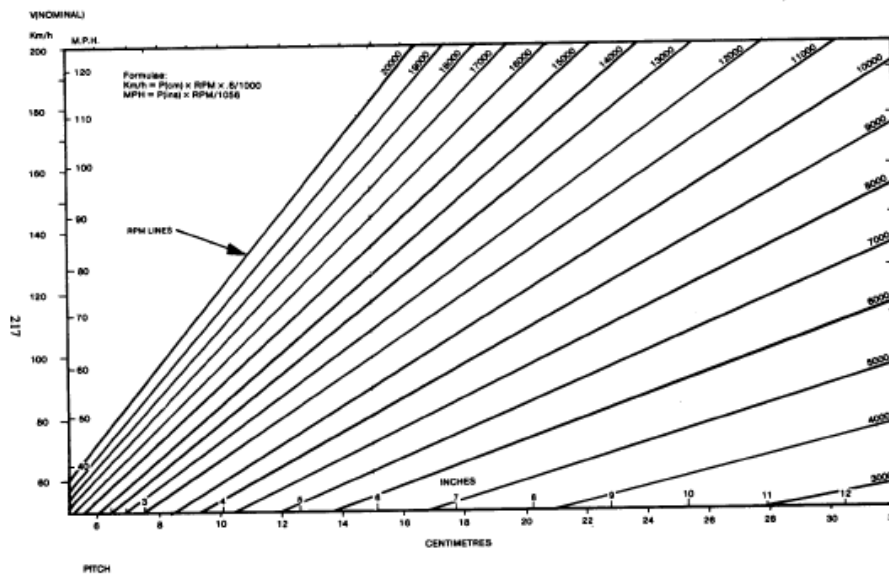


Figure 3: Optimum propeller pitch at varying flight velocity and engine speed, from [2]

Propeller Test

The aim of this experimental campaign was to gather data about thrust (both in static and flight conditions) of as much as possible commercial propellers, in order to be able to choose the best one.

Test rig

An engine support was designed, realized and mounted at the top of a vertical cantilever beam, Fig. 4. The small deformation of the cantilever was then associated to the engine/propeller thrust by means of a strain gauge placed near the bottom/clamped end of the beam, Fig. 4. For small deformation, the strain is proportional to the force applied at the free end, hence the thrust. With a simple calibration it was possible to calculate the constant of proportionality between the force and the electrical signal given as output from the amplifier linked to the strain gauge.

To perform the measurements at different airspeed (in order to simulate flight conditions) a small open-end wind tunnel was placed in front of the engine, Fig. 4. The wind tunnel is made of a counter rotating fan, driven by an inverter controlled electrical engine, and a pipe with 400 mm internal diameter. This simple rig was able to provide an almost uniform flow field in the whole section. This was checked by velocity measurements performed along the pipe diameter by means of a hotwire anemometer. These measurements were also used for the wind tunnel calibration (relationship between fan and air stream velocity).



Figure 4: Test rig for propeller thrust measurements, left –engine mounted on the cantilever beam, center- strain gauge, right – the engine placed in front of the open end wind tunnel.

Results

In Fig. 5 performances of three propellers of same diameter but different pitch are reported. It can be seen that, for the considered air stream velocities, the propeller with the higher pitch perform worst, as expected by the theoretical approach above described.

In Fig. 6, the results for the best propeller are provided. Many other propellers were tested before finding the best one but, for seek of clearness, only data of the best three are shown.

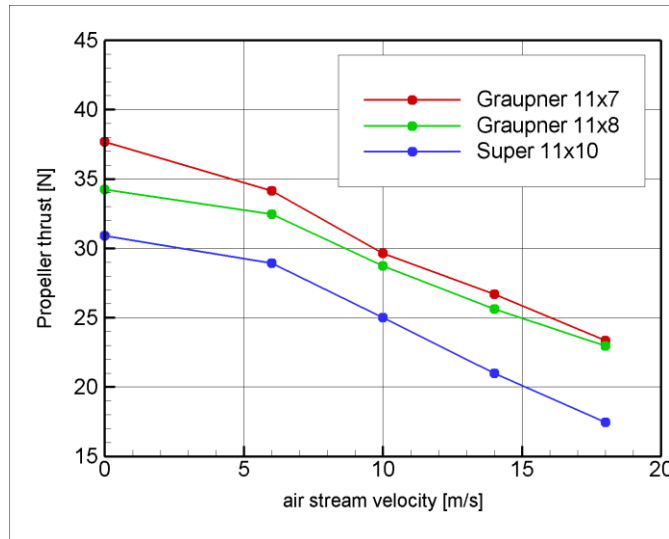


Figure 5: Measured thrust of propellers with same diameter but different pitch

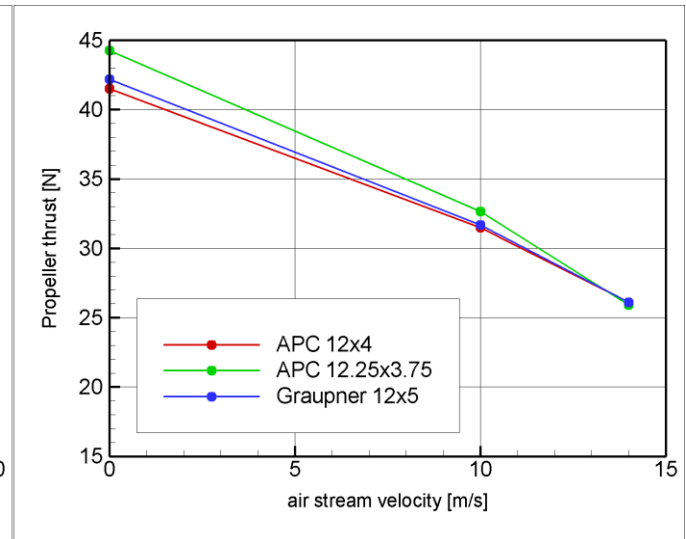


Figure 6: Measured thrust of commercial propellers.

3. Aerodynamic Design

Aircraft Sizing

Wing

The wing of the AeroUD-MET T2 was decided to be made as a constant chord, tapered outer and it has a total wingspan of 2.8 meters. This derives from rough magnitude drag analysis between a constant chord wing and a constant chord with tapered outer which showed a 10% less overall drag. Although, this was not the most aerodynamically efficient configuration, it showed up to be easier to manufacture than an elliptical wing. Also, this wingspan was necessary for obtaining the calculated amount of lift required without lowering the aspect ratio below 6 for performance issues.

In order to obtain further information on wing basic dimensions, such as the surface area and average chord length, the following equations were used:

$$AR = \frac{b^2}{S_W} \text{ where } S_W = \frac{b^2}{AR} = \frac{2.8^2}{6.12} \text{ thus } S_W = \mathbf{1.28 \, m^2}$$

$$S_W = b * c_{ma} \text{ where } c_{ma} = \frac{S_W}{b} \text{ thus } c_{ma} = \mathbf{0.457 \, m}$$

Discussion of airfoils and expected performance is shown later.

Horizontal stabilizer

Empirical relations which relate the tail area and its location respect to wing area, chord and span, based on historical data and some reference numbers were used to size the horizontal stabilizer. With this in mind, the following equation is used to obtain the volume ratio with the previous data obtained from the wing and the values of S_T and l_T which are subjected to the following relations:

$$V_{HT} = \frac{S_W \cdot c_{ma}}{S_{HT} \cdot l_T} \text{ where } S_W = 1.28 \, m^2, c_{ma} = 0.457 \, m, \frac{S_{HT}}{S_W} \leq 0.2, \frac{l_T}{c_{ma}} \geq 0.22$$

Setting the values $S_{HT} = 0.2 \, m^2$ and $l_T = 1.10 \, m$ which complied with their respective relations, the volume ratio resulted in:

$$\mathbf{V_{HT} = 2.6}$$

Since large scale transportation aircraft have a reference value of 1 for this volume ratio [3], the team decided to maintain a great margin for control power. As for the wing, more information is available in the aerodynamic discussion.

Vertical stabilizer

For the vertical stabilizer volume ratio calculation, the process is similar and uses the same values of wing area and chord previously obtained.

$$V_{VT} = \frac{S_{VT} \cdot l_V}{S_W \cdot b} \text{ where } S_W = 1.28 \, m^2 \text{ and } b = 2.8 \, m$$

Knowing that the value of V_{VT} must remain within the range of $\mathbf{0.02 < V_{VT} < 0.09}$ [3] and setting the values $S_{VT} = 0.1 \, m^2$ and $l_V = 1 \, m$ the following is attained:

$$\mathbf{V_{VT} = 0.0279}$$

Ailerons

The first approach used to size the ailerons, ensuring their effectiveness for medium roll rates standing within the $30^\circ/\text{sec}$, was done with Pajno's aileron chord to wing chord ratio [4] and double checked with Raymer's traditional aileron vs. percent chord [5]

The ailerons were designed for the outer tapered part of the wing which span is 0.6 m , with 20% of the local wing chord. This results in an aileron with mean chord equal to $C_{ail} = 0.08\text{ m}$ and $S_{ail} = 0.096\text{ m}^2$. These values are in the recommended range for an effective aileron design, [5].

Aerodynamics

For the aerodynamics analysis, the focus was pointed towards the purpose of the aircraft: lifting the most payload possible. Aiming at that goal gave the pivot points to start a research of airfoils with better performance and make the trade-offs required in order to design an airplane that meets the restrictions imposed by the competition.

Wing

Taking the wing configuration selected during the aircraft sizing process and with the wing area set the first approximation of the necessary coefficient of lift for velocities over 10 m/s . Moreover, computed Reynolds numbers tuned out to be low and it was clear that the choices available were either searching for a high cambered low Reynolds airfoil with the minimum drag possible or study a flapped configuration selecting a less cambered airfoil. This latter possibility was not considered due to the lack of wind tunnel data at low Reynolds numbers for non NACA airfoils, difficult manufacturing, increased weight by the cause of structural and equipment arrangement and an increased drag to be overcome with the available thrust in take-off and landing configuration.

Searching for a high cambered airfoils, the UIUC database [6] was selected since it makes available not only the airfoils coordinates, but also a series of papers with low Reynolds wind tunnel data [7] that were used to validate and tune computer based simulations.

The airfoils selected for a first comparison were the Eppler 423, the FX74CL5-140 and the Selig 1223, as shown in Fig.7.

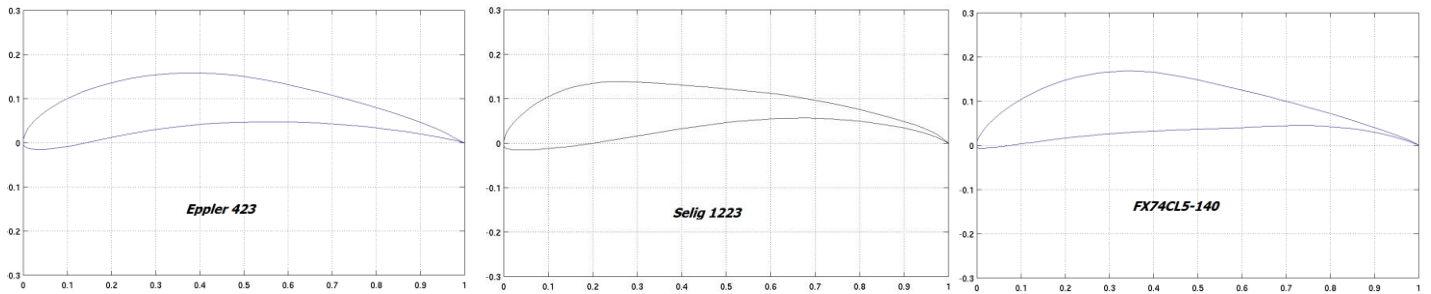


Figure 7: E423 (left), S1223 (center) and FX74CL5-140 (right) contours.

Four categories were studied : lift, drag, moment and ease of construction. Each one of them is presented in the following figure of merit (Tab. 6 with an importance weight from one to four, being the last the greater).

Table 6: Airfoils Selection Process

	Airfoils		
FOM	Eppler 423	FX74CL5-140	Selig 1223
Lift	3	3	4
Drag	2	3	4
Momentum	2	3	1
Construction	3	2	1
OVERALL	10	11	10
RANK	2	1	2

The three of them have similar performance but the Eppler was excluded because of overall performance compared to the FX74CL5-140 and so the Selig 1223 for a high moment coefficient and awkward trailing edge geometry to manufacture.

The next step was to determine the real performance of the FX74CL5-140 working in a finite wing, so the wind tunnel data at low Reynolds to tune the simulations was obtained from [8]. Mark Drela's XFOIL code was used with special attention to airfoils coordinate distribution points and also to the smoothness of the contour for the best numerical accuracy.

Computed results for $Re = 0.3 \cdot 10^6$ revealed to be near the wind tunnel data with a 140 point airfoil coordinates setting, Fig. 8.

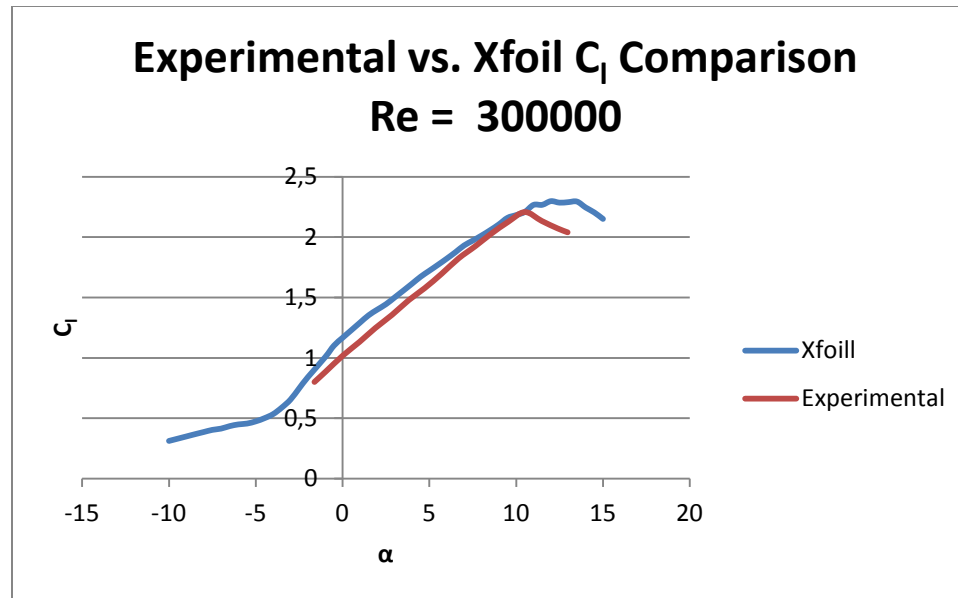


Figure 8. Experimental and Xfoil C_l curves.

In addition, small variations in C_l values according to [7] were verified (Fig. 9) and polars ranging from $0.3E6$ to $0.7E6$ were used to predict wing's performance using another code called XFLR5.

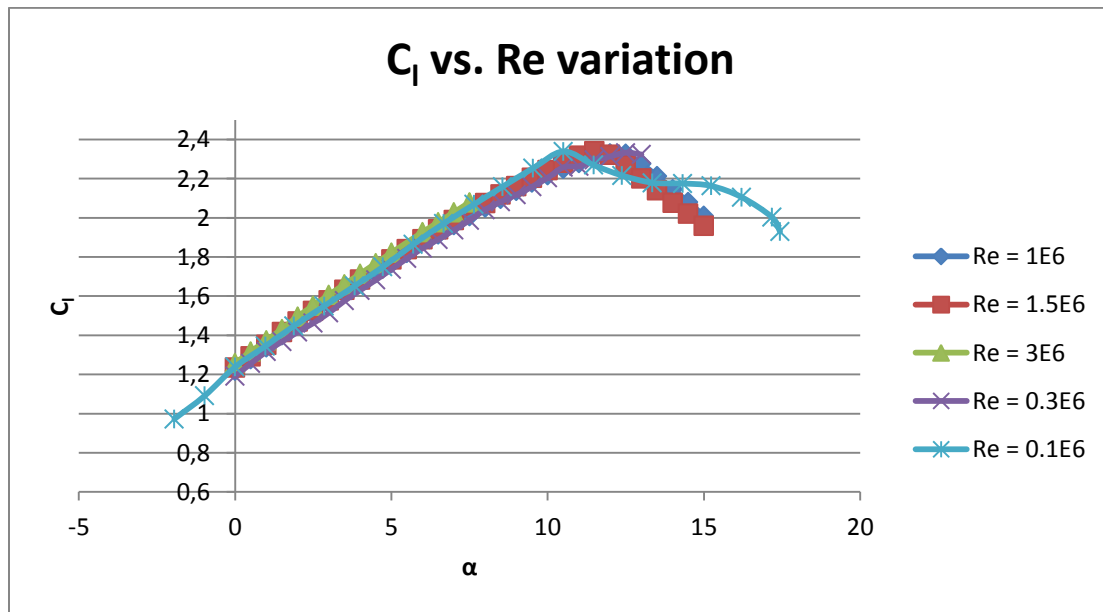


Figure 9: Variation in C_l due to change in Reynolds Number.

Using the Lifting Line Theory method to approximate the performance of the designed wing, the main results for take-off velocity, maximum velocity and lift distribution are shown in Tab. 7 and Fig. 10.

Table 7: Wing data.

Velocity / Coefficient	10 m/s	13 m/s
C_l min	0.785	0.785
C_l max	1.781	1.781
C_d min	0.0689	0.0668
C_d max	0.1367	0.1811
C_m min	-0.17	-0.18
C_m max	-0.22	-0.22

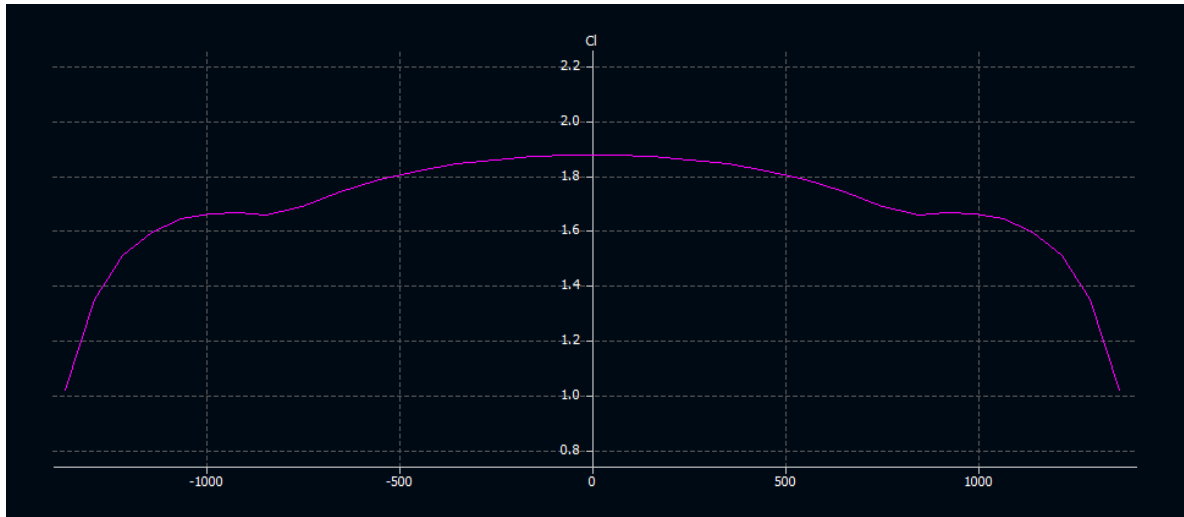


Figure 10: Lift distribution of the wing.

A -3 degree washout optimization was included after seeing the first results in order to mitigate tip stall at high angles of attack.

Horizontal Stabilizer

The horizontal stabilizer is a fundamental part of the airplane because of stability and control purposes, for this reason careful attention was paid to its design. Once area required and distance from the quarter chord of the wing were established (see aircraft sizing), two planforms with two different airfoils were conceived and tested strictly following the design guidelines found in [3] concerning aspect ratio, minimum control moments and scissors curve directives needed for critical phases of flight, specially the take-off rotation.

The first one being a rectangular planform flapped stabilizer with a NACA 0015 symmetrical airfoil and the last a tapered full flying stabilizer incorporating the Eppler 205 airfoil inverted to create the need down force.

Same process described previously to tune airfoils simulations and analyze wing performance was employed. The next plots show a comparison between the performances of both arrangements.

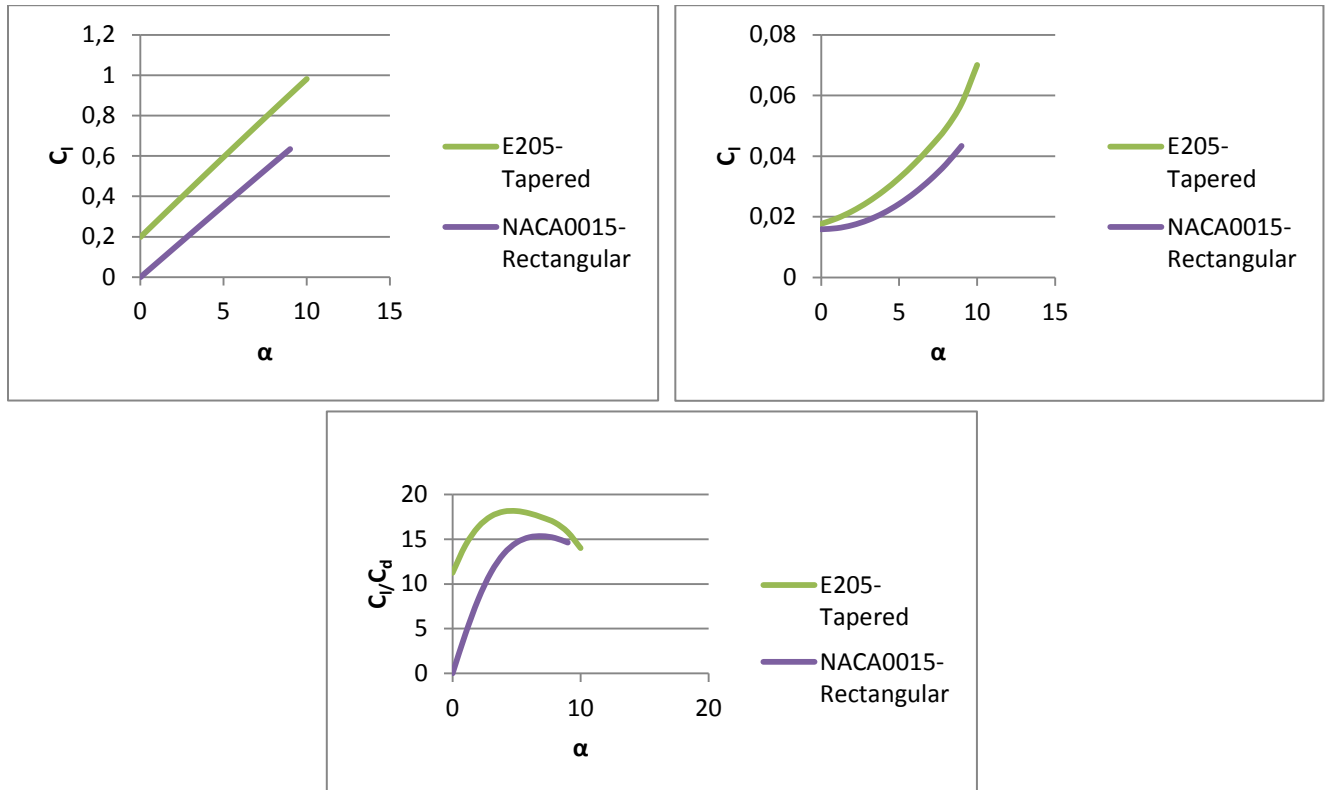


Figure 11: Horizontal stabilizers performances.

Longitudinal static stability is studied later in the section 5 of this report and the procedure explained, combined with a Matlab® code developed on purpose, allowed to have the first trim data of the airplane that was successfully proven during test flights.

The tapered stabilizer was chosen as a consequence of better efficiency (See C_l/C_d plot in Fig. 11) that was clearly noticeable on better control during the rotation phase and increased maximum velocity by 1.3 m/s measured with a Pitot probe connected to a telemetry system.

Drag

Drag forces on an airplane can be classified in two main categories: induced drag and parasite drag.

Induced drag is a byproduct of lift while parasite drag is the result of the interaction of the aircraft in general with the environment or any other drag generated which is not the induced drag. Also, the parasite drag can be classified in three types: *Form Drag*, that is produced by any structure which disrupts the flow of air and can be reduced as the aircraft is made more aerodynamic, *Skin Friction*

Drag, which is produced by the friction between the aircraft's surface and the air, and *Interference Drag*, which is produced when two currents of air flowing through the aircraft interact and have different directions. This occasionally occurs where the wing and the fuselage meet. Anderson [9] equations were used to estimate the Induced and Parasite Drag coefficient. The results are:

Induced Drag Coefficient

Wing: 0.0689

Parasite Drag Coefficients

Wing: 0.00181

Wing Interference: 0.000542

Fuselage: 0.00985

Horizontal Stabilizer: 0.000703

Vertical Stabilizer: 0.000272

Nose Landing Gear: 0.000272

Main Landing Gear: 0.0005

Engine Shape: 0.0128

Engine Temperature :0.015

Total Aircraft Drag Coefficient

$C_D = 0.11$

4. Structural Design

Material Testing

Materials chosen by the AeroUD-MET Team to build the airplane are traditional and widely used. The team decided to perform simple destructive and non-destructive tests in order to measure the mechanical properties and increase the level of confidence in structural calculations.

Since the wing and stabilizers use a sandwich type structure composed of a core of high density foam and a resin-epoxy covering on the upper and lower surfaces, testing was focused on those materials.

First the ultimate stress and Young's modulus of the foam have been measured and then the mechanical properties of the covering have been determined by testing sandwich panel specimens, once the foam properties were known.

For both the foam and the sandwich two kind of tests were performed on beam-like specimens: a natural frequency measurement in free-free boundary conditions to determine the Young's modulus and an ultimate strength test in simply supported boundary conditions to determine the ultimate stress.

Non-destructive Tests

The specimens were hanged by a long thread to get as near as possible to free-free boundary conditions in bending, for which analytical solutions exist for the natural frequencies. Each specimen was hit at the mid-span and the output was recorded by a light piezo-accelerometer fixed at the lower end of the specimen to obtain the value of the fundamental natural frequency.

The following Young's moduli were found:

- Young's modulus for the foam = $8.37 \times 10^6 \text{ N/m}^2$
- Young's modulus for a 1 mm thick covering = $1.69 \times 10^{10} \text{ N/m}^2$

Destructive Tests

Bending tests in simply supported boundary conditions were performed to obtain either the foam and the sandwich compound (foam-resin-obeche) ultimate strength. The specimens were loaded filling up a tank of water attached at the midspan between the two supports until they cracked (Figure 12).

By measuring the amount of water inside the tank, the ultimate stress σ can be obtained for the foam and the covering (considered as a 1 mm thick membrane). The following results were found:

- Ultimate stress for the foam $\sigma = 0.20 \text{ N/mm}^2$
- Ultimate stress for a 1 mm thick covering $\sigma = 17 \text{ N/mm}^2$



Figure 12: Destructive test for a foam specimen

Wing Spar Design

The wing structure is supported by 2 aluminum wing spars and this configuration came up to allow the separation into two half-wings for transportation issues e to be able to load and unload the payload easily from above the aircraft.

The wing spars have been subjected to the following verifications:

Stress 2) Deformation 3) Lateral buckling

The maximum design load for the couple of spars was set to 200N, corresponding to the most severe straight flight condition. Then considering the maximum roll angle at 45° , an amplification coefficient of 1.5 is applied to take into account the load increment imposed during the maneuver. Under these hypotheses, the maximum bending moment at the center of the wing , and therefore at the center of the spars, is 105 Nm and the maximum load is 300N.

Since two wing spars are used and the payload is arranged over them, determining the maximum percentage of load that needs to be supported by every single wing spar is key. Considering the position of the center of pressures over the wing chord in a full payload flight condition is the first aim point for that purpose. Such position was evaluated at 38% of the wing chord and therefore the rear wing spar can carry up to 68% of the total load.

Therefore the maximum bending moment for a single wing spar is 71 Nm (105×0.68) while maximum load is 204 N (300×0.68). These values were used for stress, deformation and lateral buckling tests.

Stress verification is straight forward and it is by far the less demanding between the three. With the selected wing spar cross-section, the safety factor is 3.7 for yield and even 5.0 for ultimate strength.

Deformation test is done by calculating the displacement of the wingtip as a consequence of the wing spar deformation. With this objective in mind, a half-spar was considered as a cantilever beam and loaded at the free tip. The ratio between the displacement of the tip and the length of the cantilever times the length of the half wing gives the total displacement of the wing tip in a full payload configuration due to spar deflection. The result is a 18 mm displacement of the wing tip due to spar deformation.

The most critical verification concerns lateral buckling due to torsion. For this task, it was decided to employ the analytical solution for a simply supported beam with rectangular cross-section and loaded in the mid-span. The critical load under these conditions is given by:

$$P_{cr} = 16.94 \frac{\sqrt{B_y C}}{LS^2} \quad \text{where } LS \text{ is the wing spars length.}$$

Considering a section of height h (effective height of the spar) and width a (core width), the previous formula has the term $B_y = EJ_y$, where J_y corresponds to the rectangular cross-section minimum moment of inertia $J_y = \frac{1}{12} ha^3$ and $C = \frac{Gha^3}{\beta}$ where G is the shear modulus. The β coefficient depends on the h to a ratio and its value is 3.2.

All in all, the buckling safety factor of the wing spar remains in 1.7 and the core width couldn't be less than 3.5 mm, being those the main reasons that restricted lighten.

Wing Spar Box

For the spar box resistance verification, it was assumed that shear stresses transmitted from the wing covering to the box were uniformly distributed over the box length and also that the shear stress distribution all over the height at the box ends could be considered constant as well.

With these assumptions, the verification process becomes very easy because the walls of the spar box are in constant shear stress conditions.

Constant shear stress is $\tau = \frac{M}{2slh}$, where h is the height of the spar box wall, l is its length and s its width.

This stress must be compared with the ultimate shear stress in the plane of the wall made of birch that is assumed equal to 1.6 MPa.

Wing sandwich structure

The half-wing structure has been verified by means of a finite element model. Figure 13 shows a picture of the model. The half-wing model has been simplified into a rectangular plant half-wing, loaded using the pressure results from a 2D Xfoil analysis.

The model is built up with brick elements modeling the foam and shell elements modeling the covering.

The main difficulty is to take into account the pressure loads acting on the areas of the holes. To this

purpose the holes are modeled with brick elements with a very small Young modulus, so that the pressure loads can be applied but these elements do not contribute to the structural stiffness.

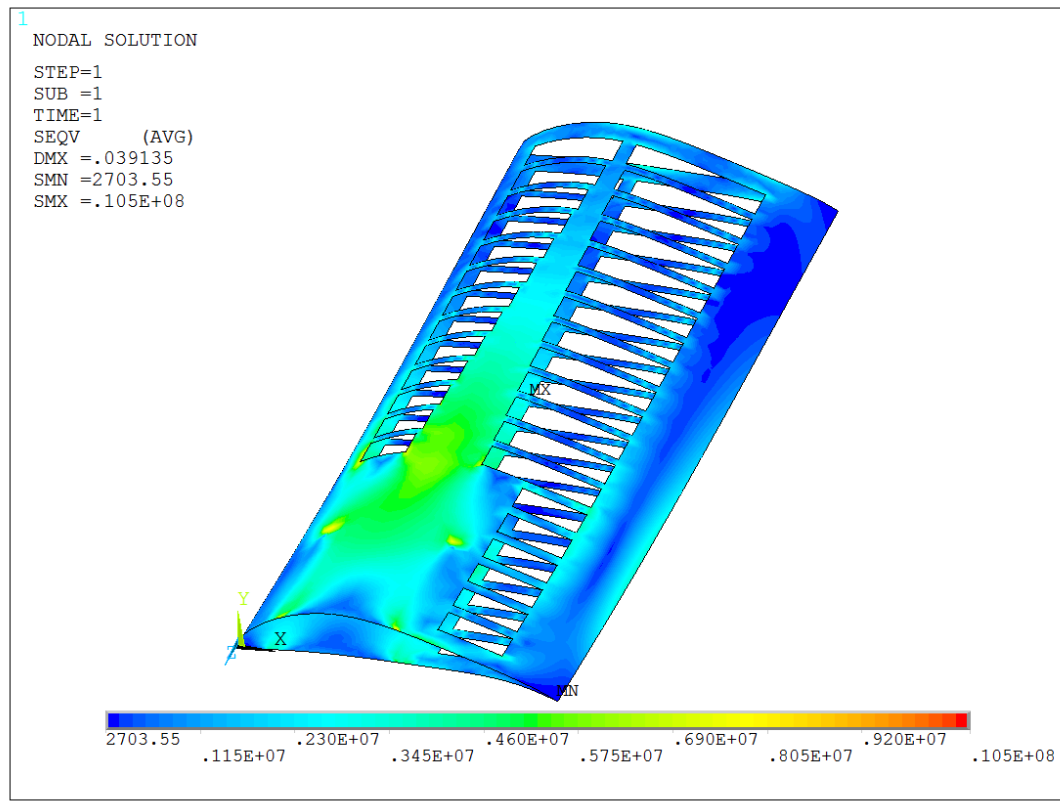


Figure 13: Finite element model of the half-wing

Carriage

The carriage structure has been realized by means of an aluminum plate and a cable. For the structural verification the aluminum plate has been modeled with beam elements and the cable as a rigid truss. The maximum load has been determined considering a vertical impact speed of 3 feet/s and equating the corresponding kinetic energy to the elastic energy of the impact. The vertical stiffness of the carriage is 72500 N/m , corresponding to a 15 mm maximum deflection in the landing impact , and the dynamic amplification factor for the impact force is 5.6 (The maximum impact force is 5.6 times the force due to gravity). The maximum impact load on a single wheel is therefore 560 N and the maximum cable tension 610 N.

5. Stability and Control

Longitudinal Static Stability

The y,z body axes are defined in the following manner: the origin is in the aerodynamic center of the wing, the y-axis is oriented towards the aerodynamic center of the horizontal stabilizer and the z-axis (perpendicular to y) is oriented upwards.

The x,t wind axes are defined in the following manner: the origin is in the aerodynamic center of the wing, the x-axis oriented along the air stream velocity vector and the t-axis perpendicular to x and oriented upwards.

The “airplane angle of incidence” is defined as the angle α_A between the y-axis and the x-axis.

Angle α_A will be always considered small in order to have $\cos(\alpha_A) \simeq 1$ and $\sin(\alpha_A) \simeq \alpha_A$, thus allowing to always use the linear transformation $x = y + \alpha_A z$. The static equilibrium equation for the moment “M” of the aerodynamic forces with respect to the neutral point can be therefore written as

$$M_{pn} = M_o^T - L_T(b_{ac} - y_{pn}) + M_o^W + L_W y_{pn}$$

where b_{ac} is the distance between the two aerodynamic centers (of the wing and of the horizontal stabilizer) and y_{pn} is the distance between the aerodynamic center of the wing and the neutral point. The coordinate y_{pn} of the neutral point can be found by imposing that the moment due to aerodynamic forces be constant with respect to the angle of incidence of the airplane, that is $\frac{dM_{pn}}{d\alpha_A} = 0$. It follows that:

$$y_{pn} = \frac{(1 - k_\epsilon) C_{L,\alpha}^{HT} S_{HT}}{(1 - k_\epsilon) C_{L,\alpha}^{HT} S_{HT} + C_{L,\alpha}^W S_W} b_{ac}$$

where k_ϵ is the derivative of the downwash angle with respect to the angle of attack of the wing, and $C_{L,\alpha}^{HT}, C_{L,\alpha}^W$ are the derivatives of wing and horizontal tail lift coefficients with respect to the angle of incidence α_A .

At this point, the coordinates $(y_{pn}, 0)$ of the neutral point in the body axes are known.

Once defined the maximum take-off weight of the airplane, the climb angle and the velocity, the next step is to determine the incidence angle of the airplane α_A and the stagger angle for the horizontal stabilizer in rectilinear flight conditions. To do so, two static equations are used: the first one being the rotational equilibrium equation with respect to the center of gravity and the last being the translation equilibrium equation in the direction perpendicular to velocity. These are respectively:

$$M_{pn} - (L_W + L_{HT})(x_{pn} - x_{cg}) = 0 \quad L_W + L_{HT} = mg \cos(\gamma) \text{ where } m \text{ is the airplane mass.}$$

These two equations, assuming small values for α_A , form a linear system (in which the coefficients depend on the weight, the climb angle γ and the velocity) that allows the simultaneous solution for α_A and the stagger angle of the horizontal stabilizer in straight flight.

Once the incidence angle of the airplane is known, it is possible to compute the corresponding static margin.

The static margin is defined as $m_s = \frac{x_{pn} - x_{cg}}{c_{ma}}$ and using the following relation:

$$x_{pn} - x_{cg} = y_{pn} - y_{cg} - \alpha_A z_{cg},$$

$$\text{the static margin becomes: } m_s = \frac{y_{pn} - y_{cg}}{c_W} - \alpha_A \frac{z_{cg}}{c_W}.$$

The static margin therefore depends upon the incidence angle of the airplane. In particular, if the center of gravity is lower than the neutral point ($z_{cg} < 0$), which is the situation of our airplane, the static margin increases with increasing angles of incidence.

Payload design

The support assembly and the plates of the payload have been designed in accordance with the following criteria:

- The payload must be loaded and secured to the airframe in one minute; thus, the payload must be loaded from upside into the aircraft and it must be secured by means of screws.
- The payload must be directly attached to the wing structure to avoid swinging during the flight; thus, the support assembly is directly fastened to the two wing spars by means of four sliders which can be blocked by means of screws.

- The static margin of the aircraft must be kept as near as possible to a theoretical constant target value in all loading conditions; thus, the plates are designed to insure a backwards shift of the center of gravity for larger loads.

It must be possible to trim the position of the center of gravity after the flight tests and according to the pilot's indications; thus, each plate can be rotated by 180 degrees and this allows to shift the center of gravity of each plate in the most suitable position.

Table 8 reports as an example some results of the computations for the static margin, the angle of attack of the wing and the stagger angle of the stabilizer, for 13 m/s speed and zero climb angle.

Table 8: Static margin at varying payload

Payload	Weight (kg)	Static margin	Speed (m/s)	Wing attack	Stabilizer stagger
No payload	6.0	12.8%	13	-3.3°	9.0°
4 plates	11.7	11.3%	13	2.4°	7.2°
6 plates	14.2	11.1%	13	4.9°	6.3°
7 plates	15.6	11.3%	13	6.3°	5.7°
8 plates	16.3	11.0%	13	7.0°	5.6°
9 plates	17.0	10.8%	13	7.7°	5.5°

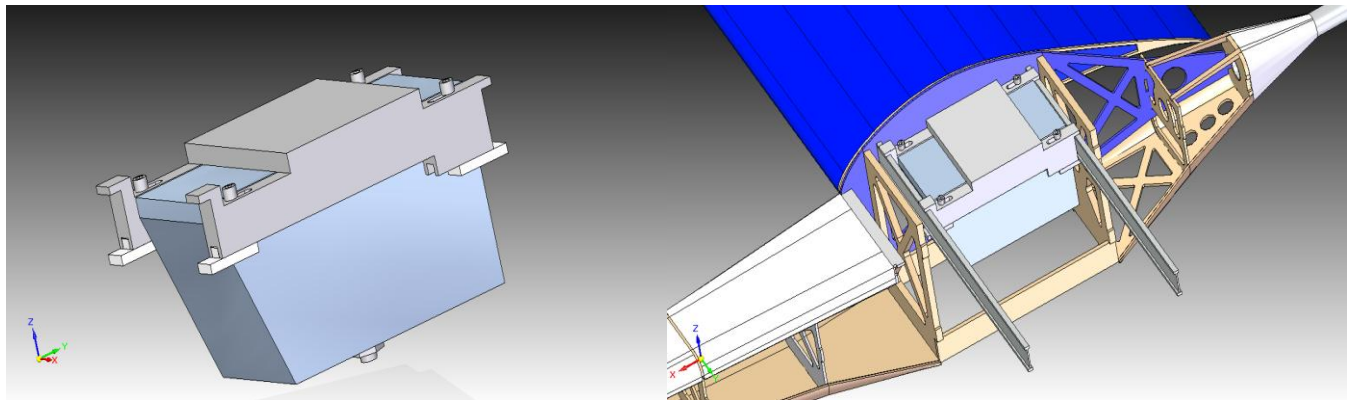


Figure 14: Payload (left) and payload bay (right)

6. Estimated performances

Pay load prediction curve as a function of the density altitude is reported in Fig. 15. The relationship is expressed by the following equations:

$$\text{Payload [N]} = -0.0134 \text{ Den. alt. [m]} + 89.333 \quad \text{Payload [lb]} = -0.000918 \text{ Den. alt. [feet]} + 20.082$$

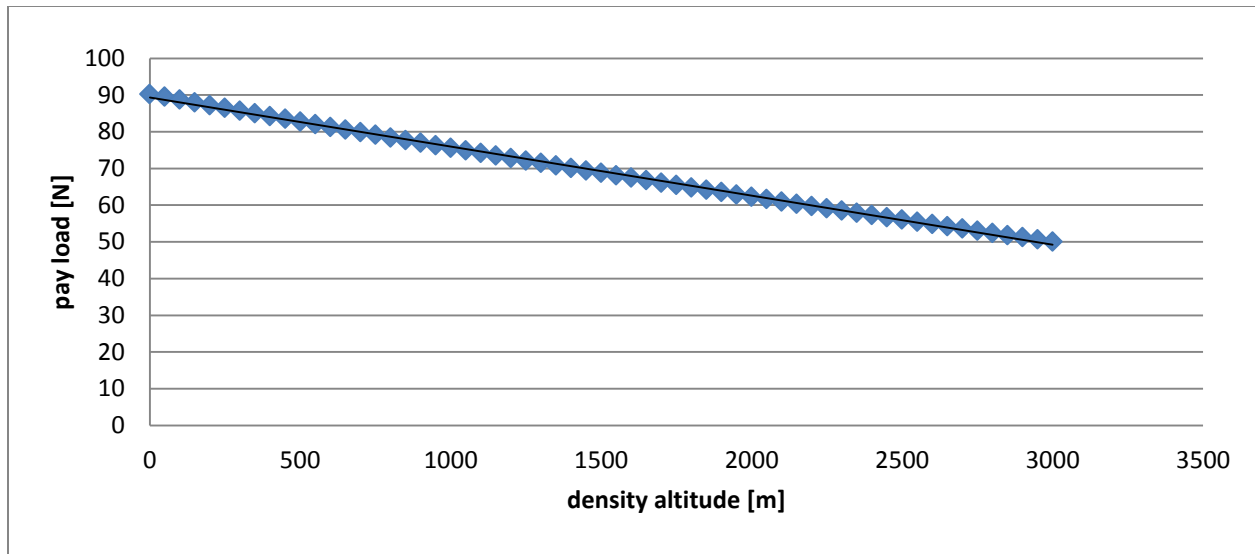


Figure 15: payload prediction graph.

The estimate of the take-off roll distance has been done by numerically integrating the differential equation of motion of the take-off transient. The thrust has been interpolated from the measured data for the selected propeller. The drag coefficient for the whole airplane was set to 0.11 (the angle of attack of the wing during take-off roll is -3 degrees). The friction coefficient for the wheels is set to 0.03.

The following are the results of the simulations:

Table 9: Take-off roll distance prediction

Payload	Weight [kg]	Stall speed [m/s]	Rotation speed [m/s]	Take-off roll distance [m]	Take-off roll duration [s]
No payload	6.0	6.6	7.3	5	1.3
4 plates	11.7	9.3	10.2	23	4.0
6 plates	14.2	10.2	11.2	37	5.8
7 plates	15.6	10.7	11.8	47	7.0
8 plates	16.3	11.0	12.0	53	7.6
9 plates	17.0	11.2	12.3	60	8.4

Another coefficient of performance in the ratio between maximum thrust and maximum take-off-weight

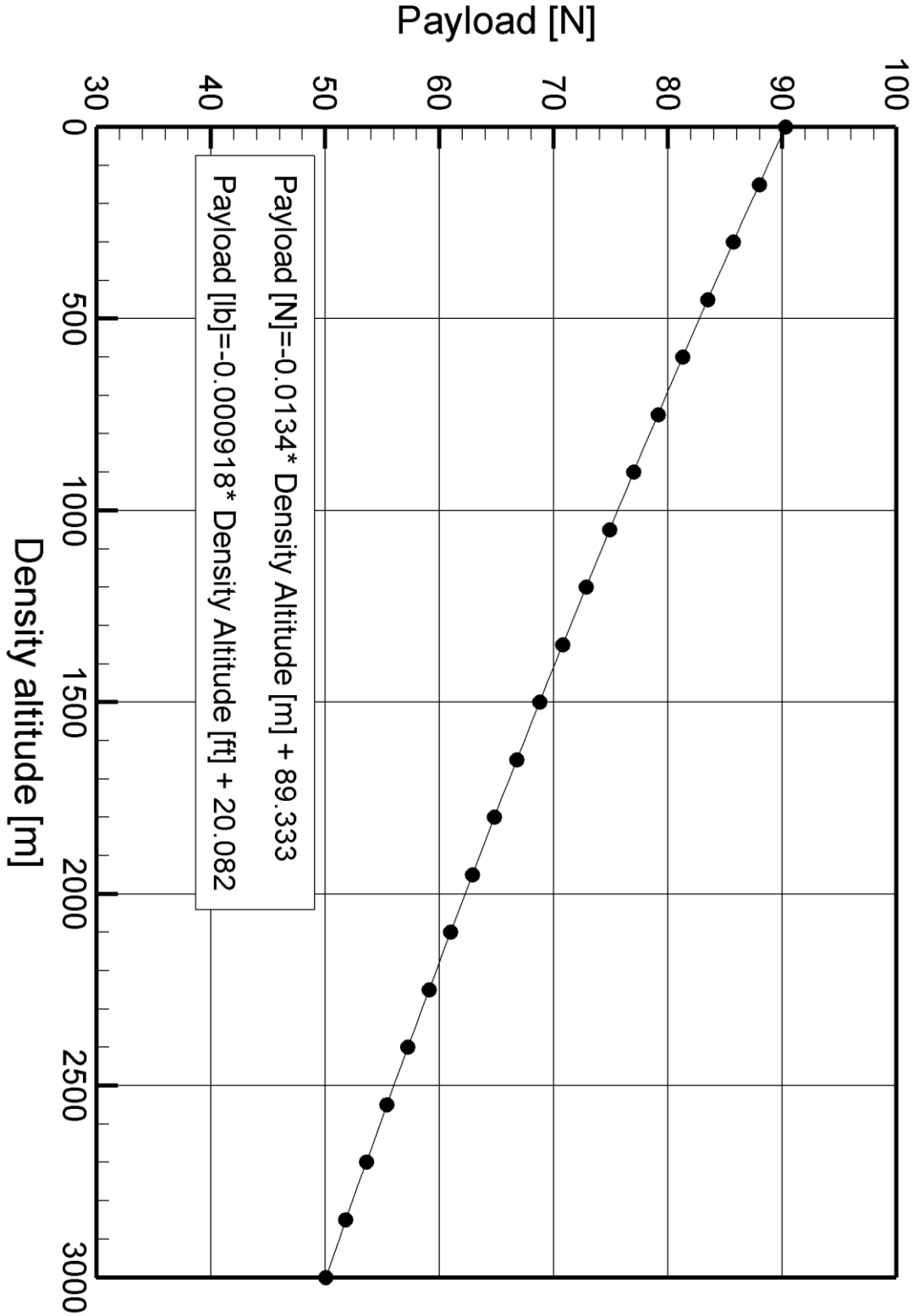
(MTOW). In the present case, it results $\frac{T_{max}}{MTOW} = 0.266$ which is actually close to the value 0.25 that characterizes cargo airplanes.

7. Conclusions

The AeroUD-MET team through the process of designing, building, testing and optimizing its first ever prototype has achieved a robust aircraft that meets all of the requirements set by the SAE Aero Design rules committee. The T2 demonstrated that is capable of carrying 98.1 N of payload under sea level conditions, which is an acceptable performance based on information from previous competitions. The 5.71 meters of maximum allowed sum of dimensions and power available had to be analyzed carefully in order to make the best trade-offs for wing efficiency and stability of the aircraft, while restriction on building and payload materials added a challenging design and test processes. All in all, engineering judgment must be employed to design a successful aircraft and continuous development of previous models leads to better overall results.

8. References

- [1] Brandt, J. B. and Selig, M. S. *Propeller Performance Data at Low Reynolds Numbers*, AIAA Paper 2011-1255, January 2011
- [2] Simmons, M. *Model Aircraft Aerodynamics*, Chapter 14, pagg. 217-218.
- [3] Kroo, Ilan. *Aircraft Design: Synthesis and Analysis*. University of Stanford, California.
adg.stanford.edu/aa241/AircraftDesign.html
- [4] Pajno, V. *Sailplane Design*. Macchione Editore. Varese, 2006.
- [5] Raymer, Daniel P. *Aircraft Design: A Conceptual Approach*. American Institute of Aeronautics, Inc. Washington, 2002.
- [6] UIUC Applied Aerodynamycs Group. *UIUC Airfoils Coordinate Database*. University of Illinois in Urbana Champaign. www.ae.illinois.edu/m-selig/ads/coord_database.html
- [7] Selig, Donovan, Fraser. *Airfoils at Low Speeds*. UIUC, 1989.
- [8] Wortmann, F.X., *The Quest For High Lift*. AIAA Paper N. 74-1018. 1974.
- [9] Anderson, John. *Aircraft Performance and Design*. McGraw-Hill. New York, 1998.



Technical Drawing of the T2 Aircraft

The drawing includes three views of the aircraft: a top view, a side view, and a rear view. Dimensions are provided in inches and feet.

Top View Dimensions:

- Wingspan: 110.63
- Wing Root Chord: 63.15
- Wing Tip Chord: 40.71
- Wing Root Chord (inner): 9.85
- Wing Root Chord (outer): 5.91
- Wing Root Chord (total): 11.81
- Wing Root Chord (total): 19.69
- Wing Root Chord (total): 23.62
- Wing Root Chord (total): 3.94
- Wing Root Chord (total): 2.36

Side View Dimensions:

- Wing Root Chord: 31.87
- Wing Root Chord (inner): 5.32
- Wing Root Chord (outer): 2.15
- Wing Root Chord (total): 3.94
- Wing Root Chord (total): 2.76
- Wing Root Chord (total): 76.68

Rear View Dimensions:

- Wing Root Chord: 21.22

Table of Specifications:

UNIT	SCALE	DATE
UNIVUD	1:16.5	06/02/2013
T2 PLANS		
AEROD-MET #026		
UNIVERSITA' DEGLI STUDI DI UDINE		
EMPTY WEIGHT [lb]	13.3	
WINGSPAN [in]	110.63	
LENGTH [in]	76.68	
HEIGHT [in]	31.87	
WING ROOT CHORD [in]	19.69	
WING TIP CHORD [in]	11.81	
TOTAL WING AREA [ft ²]	13.78	
HORIZONTAL STABILIZER SPAN [in]	40.71	
CENTER OF GRAVITY (from wing leading edge) CG (x,y) = (-5.32,-2.15)		
WING AIRFOIL	FX740LS-140	
HORIZONTAL STABILIZER AIRFOIL	EPPLER 205	
RUDDER AIRFOIL	NACA 0012	

UNIU.D		SCALE 1:16.5	
TITLE		DATE	
T2 PLANS		06/02/2013	
SUB AEROD-MET #026 UNIVERSITA' DEGLI STUDI DI UDINE			
EMPTY WEIGHT	[lb]	13.3	
WINGSPAN	[m]	110.63	
LENGTH	[m]	76.68	
HEIGHT	[m]	31.87	
WING ROOT CHORD	[m]	19.69	
WING TIP CHORD	[m]	11.81	
TOTAL WING AREA	[ft ²]	13.78	
HORIZONTAL STABILIZER SPAN	[m]	40.71	
CENTER OF GRAVITY (from wing leading edge)	CG (X _y) = (-5.32, -2.15)		
WING AIRFOIL		F740J5-140	
HORIZONTAL STABILIZER AIRFOIL		EPPLER 205	
RUDDER AIRFOIL		NACA 0012	