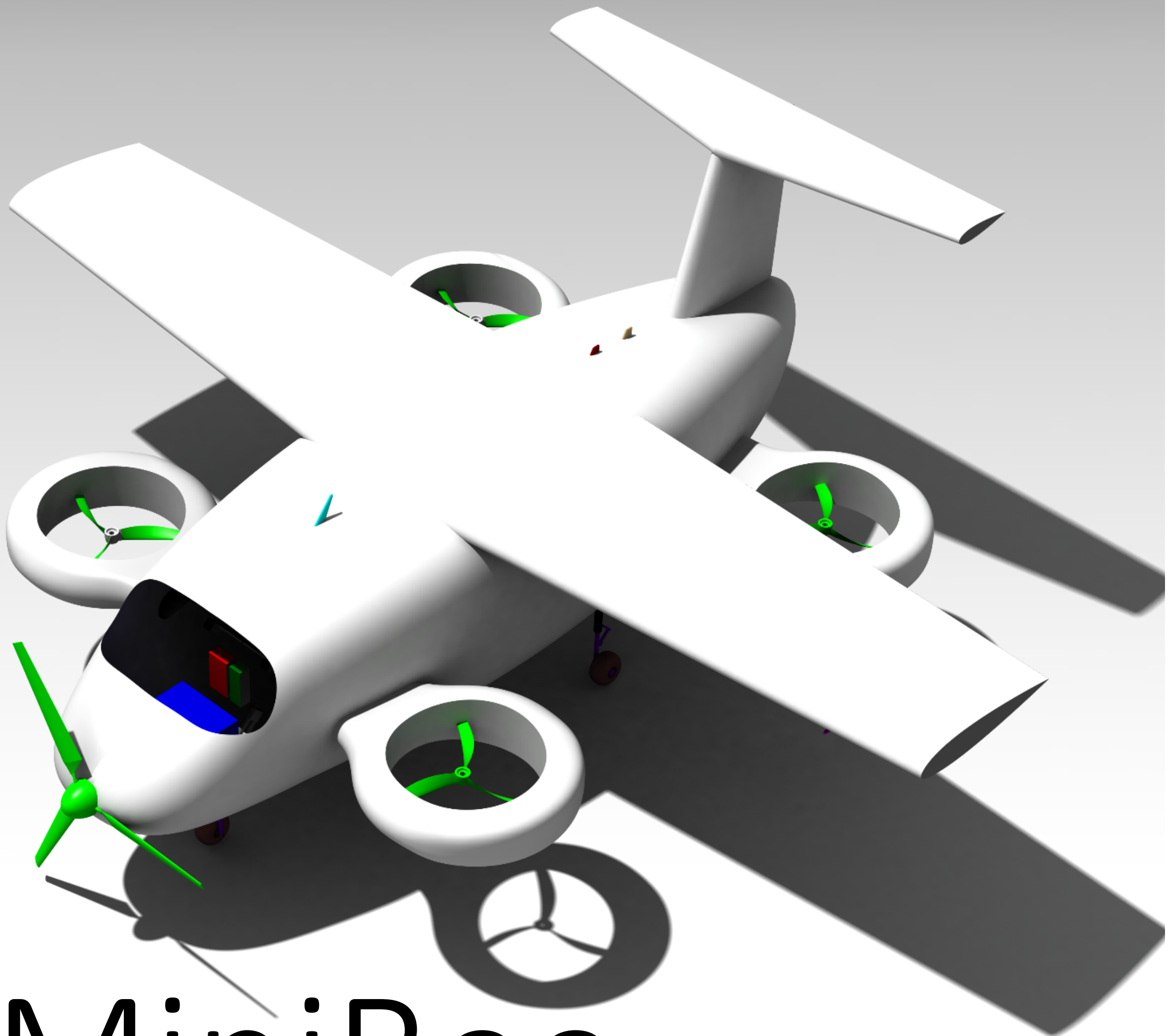

MiniBee Report

SUPMECA Team:
Angelica DI COLA
Claudia CONTE
Joao Pedro BADAN

Saint Ouen,
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1. INTRODUCTION

The MiniBee is a concept of aircraft that has the potential to revolutionize the way personal and low cost aviation works. It is a hybrid aircraft, that utilizes electric motors in conjunction with a thermic one to be able to perform in the gap of actuation between an airplane and a helicopter, presenting features as VTOL, which enables it to dispose high mobility as a helicopter, but also a high cruise speed as an airplane.

The development of an aircraft in general is composed by several procedures, and regularly is not a linear nor an intuitive process, it is composed by many levels of hierarchy and interactions inside and between those levels. As it is a complex task, it is appropriate to follow a procedure to facilitate the development process.

The following procedure was synthesized by BARROS in 2000, and it is an amalgam of the four main methodologies known in the project of aircrafts (TORENBECK 1981, RAYMER 1989, ROSKAN 1985, and VANDAELE 1962), applied to the project of lightweight subsonic aircraft.

As BARROS infers, the lightweight subsonic aircraft development can be broke in the following procedures:

- Specification and requirements
- Initial Study
- Initial Project
- Project
- Manufacturing
- Ground Tests
- Flight Tests

With some of them overlapping each other, as the development process evolves.

Each one of these procedures can be split into “Sub Procedures”, as shown below, and the development structure of MiniBee is going to follow this procedure. This report is going to cover all of the development already done on the MiniBee until the initial project phase. The procedures utilized are divided like the following:

- Specification and requirements
    - Finality of the airplane
    - Ambitioned performance
    - Missions/tasks
    - Objectives
    - Constraints and requirements
- Initial study
    - Comparative methods
        - Technical tables
        - Comparative table
    - Priorities list
    - Project delimitation
        - Determination of the external configuration
        - Study of the internal configuration
        - Study of internal ergonomics
        - Preliminary sizing
        - Propulsion determination
        - Materials and procedures
        - Equipment and infrastructure
- Initial project

- Initial sketch
- Wing sketch and stall information
- Definition of wing and empennage profiles
- Weight estimation
- External proportions
- Stability and control estimative
- Center of gravity tolerance
- Thermic motor alignment
- Modeling of the fuselage
- Refinement of the external geometry

2. SPECIFICATION AND REQUIREMENTS

2.1. Finality of the airplane

The finality of the MiniBee project is to build a low cost aircraft, with a high cruise speed, which presents high mobility, meaning that it's be able to land in the several conditions, which are normaly in the operational scope of the helicopters.

2.2. Ambitioned performance

- Cruise speed of 300+-50km/h: A high cruise speed is required for the aircraft to perform the intended tasks, and as well as a selling point.
- Service ceiling 6000m: Typical service ceiling for UML aircrafts.
- Between 600km and 800km of autonomy: A high autonomy is necessary for the aircraft to reach different branches of the market, and as well for it to be categorized as UML.
- Stall speed between 100Km/h and 150km/h: For comparison the stall speed of a Cessna 172 is about 80km/h. A high stall speed requires more power of the propulsion systems, but also enables the utilization of smaller wings.

2.3. Missions/tasks

The MiniBee aircraft, aims to be able to perform on various fields, ranging from air ambulance for emerging countries, to rescue missions and even low cost personal transportation. It seeks to be an affordable and accessible way to popularize air transportation in untapped fields.

2.4. Objectives

- VTOL Capabilities: The VTOL capability is needed to achieve the high mobility requirement. With a VTOL system, the aircraft will be able to take off and land, in virtually any of the fields inside of the helicopters and planes operational scopes. The VTOL system is intended to be created allying four electrical motors with the help of the thermal engine. When the aircraft reaches a safe altitude, it transitions to cruise flight. The VTOL system is critical in the project conception, and should be responsible for most of the advantages of the final product, but also for many possible compromises during the project phase.
- Hybrid configuration: The hybrid configuration is a result of the choice of electrical motors to power the VTOL System. To reduce the need of overweight batteries, and also the necessity to charge the aircraft into a power outlet, one possible solution is to use the thermal motor to power and charge the electrical engines in flight. The technology needed to implement this system can be inspired in already commercial packages, found in hybrid cars. The hydride configuration is, as the VTOL system, not usually found in airplanes, and in the same way, can be responsible for many

advantages and innovations in the final product, but may be also responsible for some compromises during the project phase.

2.5. Constraints and requirements

- Cost below 100,000€
- MTOW of 2000kg
- 4 or 5 seats
- Maximum dimension of 10m

3. PROJECT MANAGEMENT

The efficient project management represents one of the many keys to the success of a project. As an aeronautical project, MiniBee is no different, it has even a greater demand for efficiency because of that, the team decided to take care of the management carefully. Based on the procedures proposed by BARROS, which are a synthesis of the works of TORENBECK 1981, RAYMER 1989, ROSKAN 1985 and VANDAELE 1962, the four most known methodologies in the aeronautical field, the project can be divided as it follows:

- Specification and requirements
- Initial study
- Initial project
- Project
- Manufacturing
- Ground tests
- Flight tests

As the team hopes to advance only until the the "Initial project" phase, the management of work have been done only until this step of the project.

To simplify the comprehension of how the project management was performed, the creation process can be divided in three parts:

I) Identify, relate and create and hierarchy of the main project areas;
II) List the inputs and outputs of each of those areas
III) Relate I and II in a timeline

3.1. Identify the initial project areas

On the fist part (I), the main areas of the Initial project (and aeronautical projects in general) can be listed as:

- Concept estimations
- Aerodynamic calculations
- Performance calculations
- Force and Momentum Calculations
- Control and Stability calculations
- Structure Calculations
- Detailed Project of systems and subsystems

also to enable the parallel work between supmeca and other team, it’s possible to add the Detailed Project of **selected** systems and subsystems, as it follows:

- Fuel tanks
- Landing gear
- E-motors housing and support structure
- Thermal engine support structure
- Cockpit Subsystems

the relation between all of the areas listed above can be seen on the Figure 1:

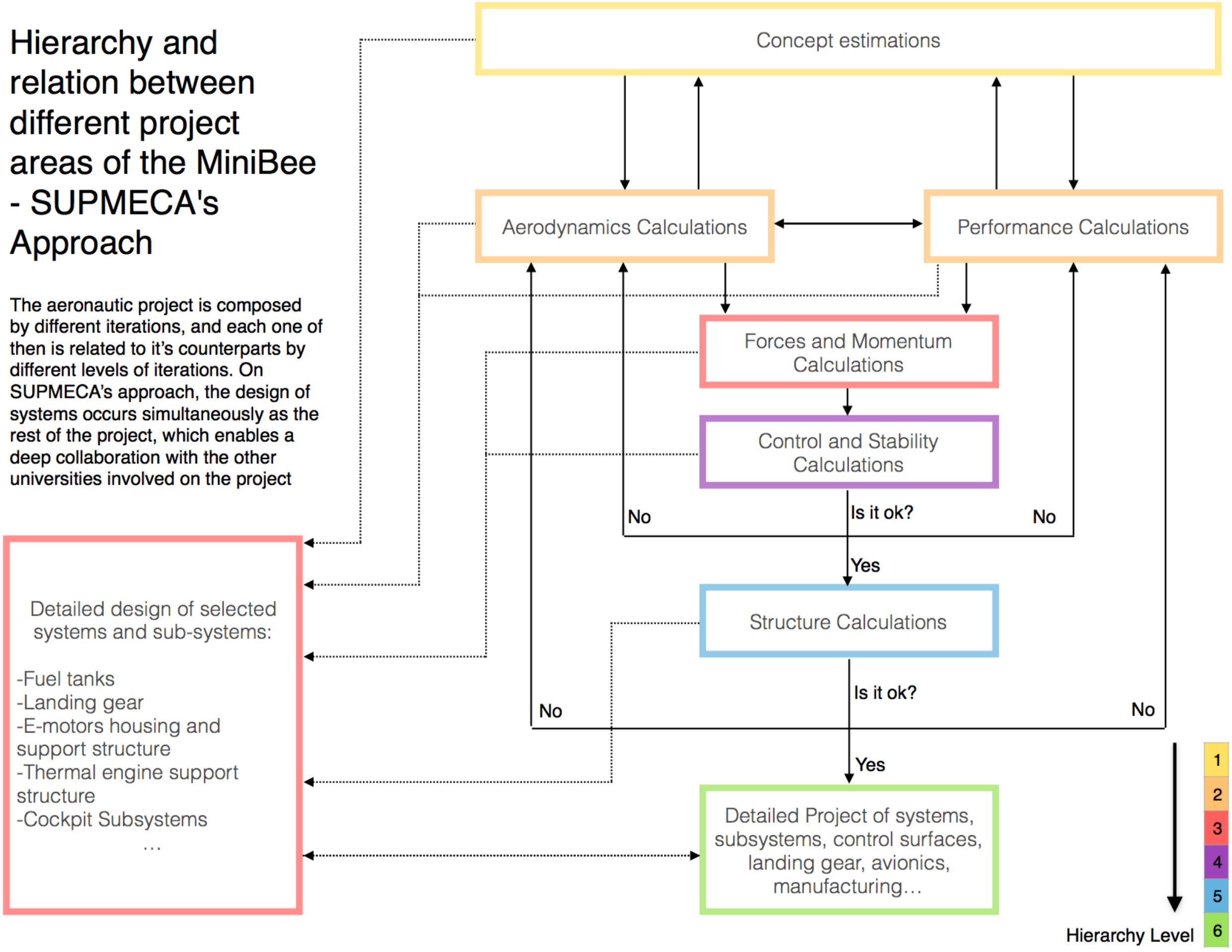


Figure 1 - Hierarchy level between diferent project areas

The black arrows represent how each area of the project is related to each other, for example, Concept and estimations doesn't have a direct link to the structure calculations, but it will indirectly influence it by the chunks that are placed between then. It is important to notice that the results also influent the outputs, meaning that the diagram needs lots of interactions to get more refined. To clarify the visualization, different hierarchy levels were colored differently. They can be seen on the right downside.

3.2. List of inputs and outputs

On the second part (II) of the project management, each chunk of the first part was analyzed individually to determine it's required inputs and outputs. This is going fetch the necessary data to generate the task list that will compose the activity board. Bellow on the Figure 2 thru Figure 11 is shown the result of that, for each of the areas listed above:

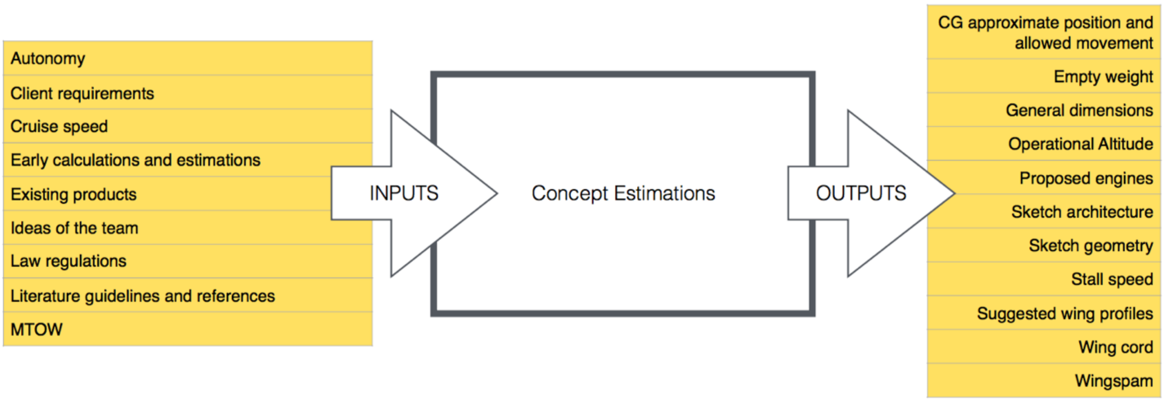


Figure 2 - Concept estimation IO diagram

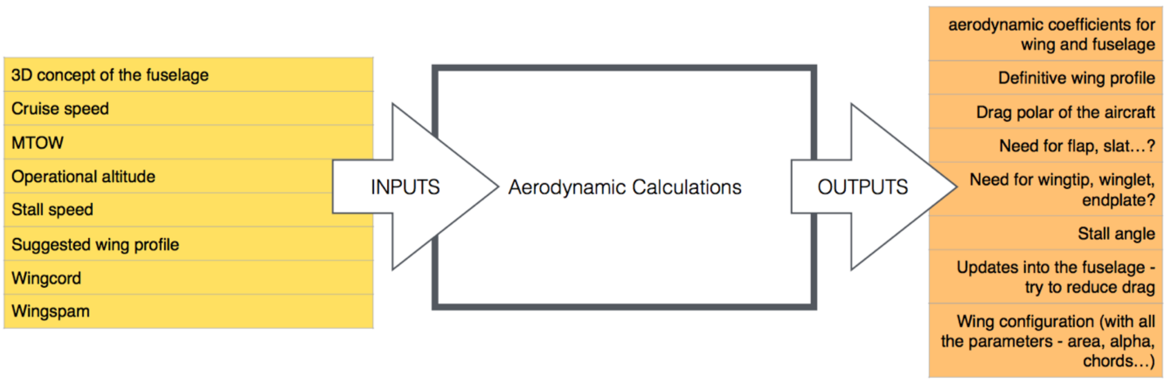


Figure 3 Aerodynamic calculations IO diagram

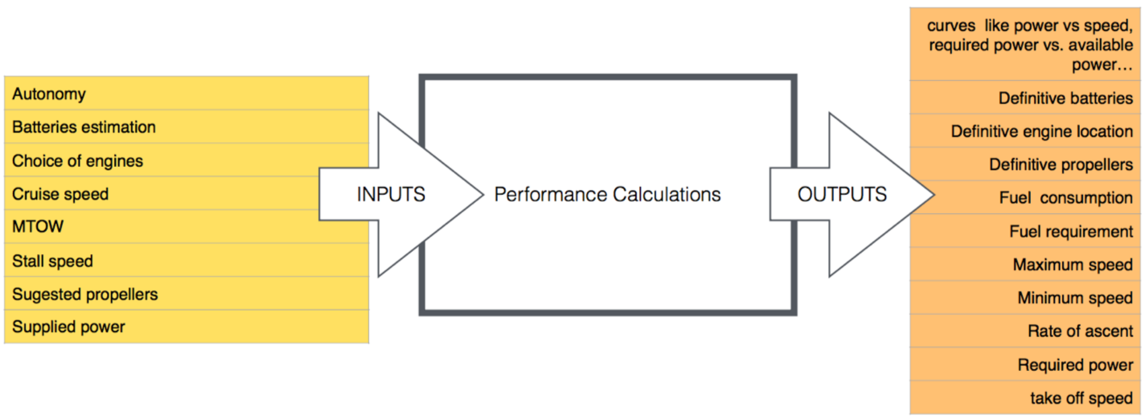


Figure 4 - Performance Calculations IO diagram

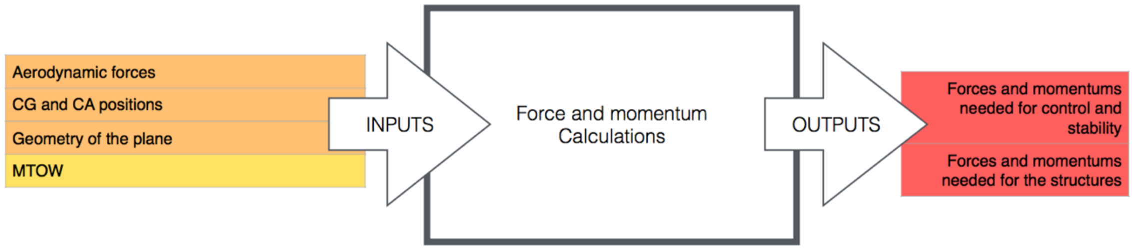


Figure 5 - Force and momentum calculations IO diagram

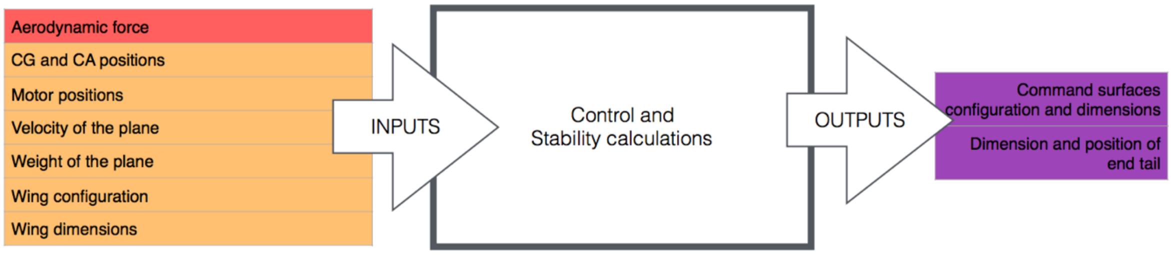


Figure 6 - Control and stability calculations IO diagram

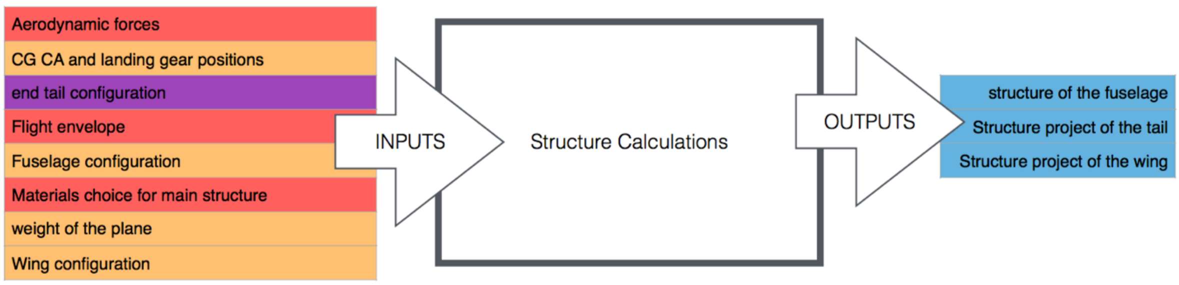


Figure 7 - Structure calculations IO diagram

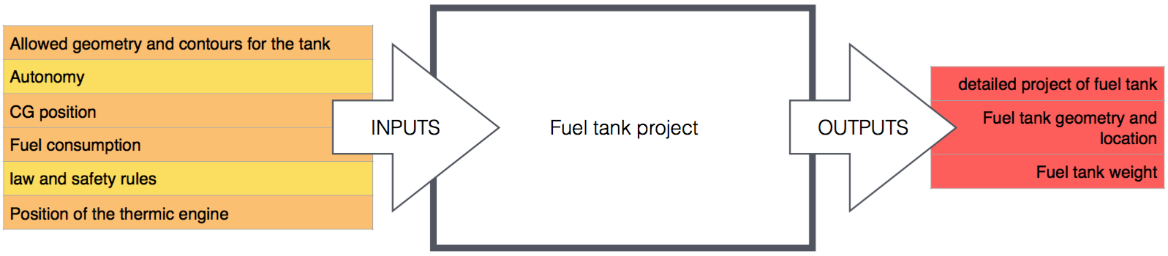


Figure 8 - Fuel tank project IO diagram

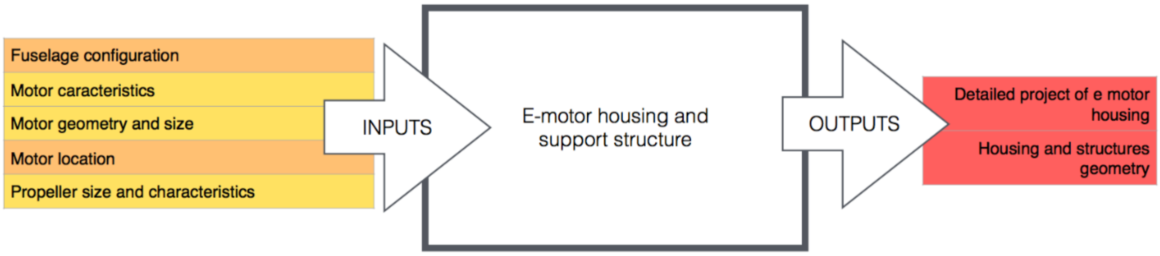


Figure 9 - E-Motor Housing and support structure IO diagram

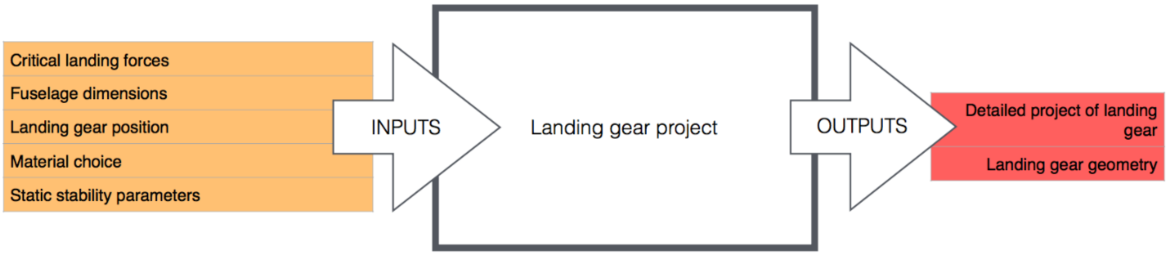


Figure 10 - Landing Gear project IO diagram

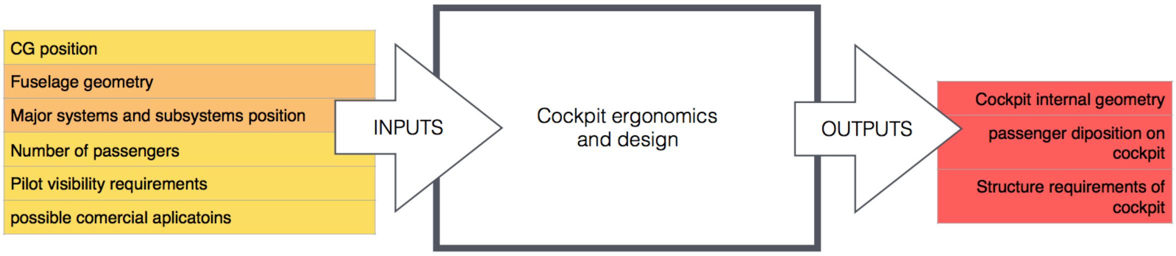


Figure 11 - Cockpit ergonomics and design IO diagram

3.3. Activity Board

To create the activity board for the project, presented in a timeline, was only a matter of addressing the proposed tasks (listed on part II) obeying the hierarchy level, and after suit then to each university working on the project, and choose an appropriate time to complete each task. In that way, after the completion of a task by a university, others can use the results to develop further tasks. The activity board in a timeline is shown on the next page:

Tasks\Deadline	Completed? (Y/N)	11/Dec/2015	18/Dec 2015	25/Dec/2015	1/Jan/2016	8/Jan/2016	15/Jan/2016	22/Jan/2016	29/Jan/2016	5/Feb/2016	12/Feb/2016
3D concept of the fuselage											
Autonomy											
Batteries estimation											
CG approximate position and allowed movement		Supméca	Supméca								
CG position		Supméca	Supméca								
Choice of engines											
Client requirements											
Cruise speed											
Early calculations and estimations											
Empty weight											
Existing products											
General dimensions											
Ideas of the team											
Law regulations											
Literature guidelines and references											
Motor caracteristics											
Motor geometry and size											
MTOW											
Number of passengers											
Operational Altitude											
Pilot visibility requirements		Estaca	Estaca								
possible comercial aplicatoins											
Propeller size and characteristics											
Proposed engines											
Sketch architecture											
Sketch geometry											
Stall speed											
Sugested propellers		Supméca	Supméca								
Suggested wing profile											
Supplied power											
Wing cord											
Wingspam											
aerodynamic coefficients for wing and fuselage		Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	
Allowed geometry and contours for the fuel tank											
CG and CA positions		Supméca	Supméca	Supméca	Supméca	Supméca					
Critical landing forces		Supméca									
curves like power vs speed, required power vs. available power…								Central	Central	Central	
Definitive batteries		Estaca	Estaca								
Definitive engine location		Estaca	Estaca								
Definitive propellers		Estaca	Estaca								
Definitive wing profile											
Drag polar of the aircraft		Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	
Fuel consumption		Polito									
Fuselage configuration		Supméca	Supméca								
Geometry of the plane										Supméca	
Landing gear position			Estaca								
Major systems and subsystems position		Polito									
Material choice for the fuselage		Estaca									
Maximum speed				Central	Central	Central					
Minimum speed				Central	Central	Central					
e-Motor location		Estaca									
Need for flap, slat…?						Supméca	Supméca				
Need for wingtip, winglet, endplate?		Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca			
Position of the thermic engine		Estaca									
Rate of ascent		Central	Central								
Required power		Central	Central								
Stall angle						Supméca	Supméca				
Static stability parameters		Estaca									
take off speed				Central	Central	Central	Central				
Updates into the fuselage - try to reduce drag							Supméca	Supméca	Supméca		
Velocity of the plane				Central	Central	Central					
Weight of the plane		Supméca	Supméca								
Wing configuration		Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	
Wing configuration (with all the parameters - area, alpha, chords…)		Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	
Wing dimensions		Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	Supméca	
Aerodynamic forces									Supméca	Supméca	
Cockpit internal geometry				Estaca	Estaca	Estaca					
Detailed project of e motor housing		Supméca				Supméca	Supméca	Supméca	Supméca	Supméca	
detailed project of fuel tank				Polito	Polito	Polito	Polito				
Detailed project of landing gear			Supméca	Supméca	Supméca	Supméca					
Flight envelope								Central	Central	Central	
Forces and momentums needed for control and stability								Central	Central	Central	
Forces and momentums needed for the structures calculations								Central	Central	Central	
Fuel tank geometry and location			Polito								
Fuel tank weight				Polito	Polito	Polito	Polito				
e-Housing and structures geometry		Supméca									
Landing gear geometry			Supméca	Supméca	Supméca	Supméca					
Materials choice for main structure							Estaca	Estaca	Estaca		
passenger diposition on cockpit				Estaca	Estaca	Estaca					
Structure requirements of cockpit							Estaca	Estaca	Estaca		
Command surfaces configuration and dimensions											
Dimension and position of end tail											
end tail configuration											
structure of the fuselage											
Structure project of the tail											
Structure project of the wing											

The colors on this side of the table obey the hierarchy code

The colors on this side of the table are to clarify visualization only, and dont obey any special order

4. INITIAL STUDY

4.1. Comparative methods

4.1.1. Technical information of Cessna 172, R44 and AW119

The Cessna 172, R44 and AW119 are three aircrafts which represents well the scope of comercial aircraft that MiniBee looks for to be in the gap. They are, in each respectively categories, in the same class of aircraft of the MiniBee, so they can be used to comparison porposes, and also to donate initial estimations for the MiniBee project.

First of all, **Cessna 172 Skyhawk** is a four-seat, single-engine, high wing, fixed-wing aircraft made by the Cessna Aircraft Company. Measured by its longevity and popularity, the Cessna 172 is the most successful aircraft in history. The three views of this airplane are shown in the Figure 12:

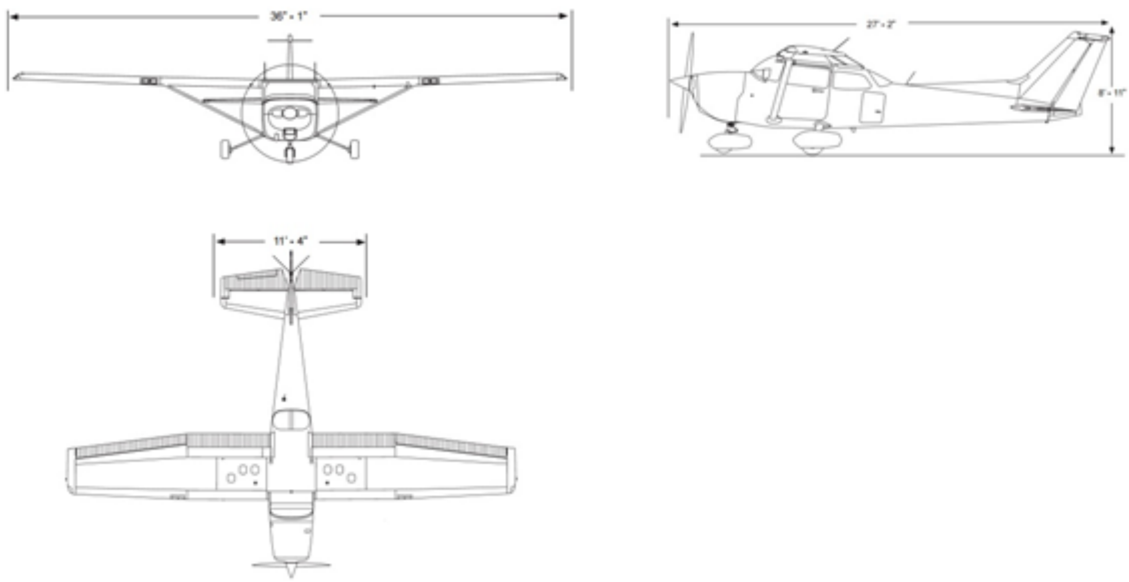


Figure 12 - Three views of the Cessna 172

Main characteristics of Cessna 172:

- **Crew:** one
- **Capacity:** three passengers
- **Length:** 27 ft 2 in (8.28 m)
- **Wingspan:** 36 ft 1 in (11.00 m)
- **Height:** 8 ft 11 in (2.72 m)
- **Wing area:** 174 sq ft (16.2 m2)
- **Airfoil:** modified NACA 2412
- **Empty weight:** 1,691 lb (767 kg)
- **Gross weight:** 2,450 lb (1,111 kg)
- **Fuel capacity:** 56 US gallons (212 litres)
- **Powerplant:** 1 × Lycoming IO-360-L2A four cylinder, horizontally opposed aircraft engine, 160 hp (120 kW)

The **Robinson R44** is a four-seat light helicopter produced by the Robinson Helicopter Company since 1992. The three views of this helicopter are shown in the Figure 13 :

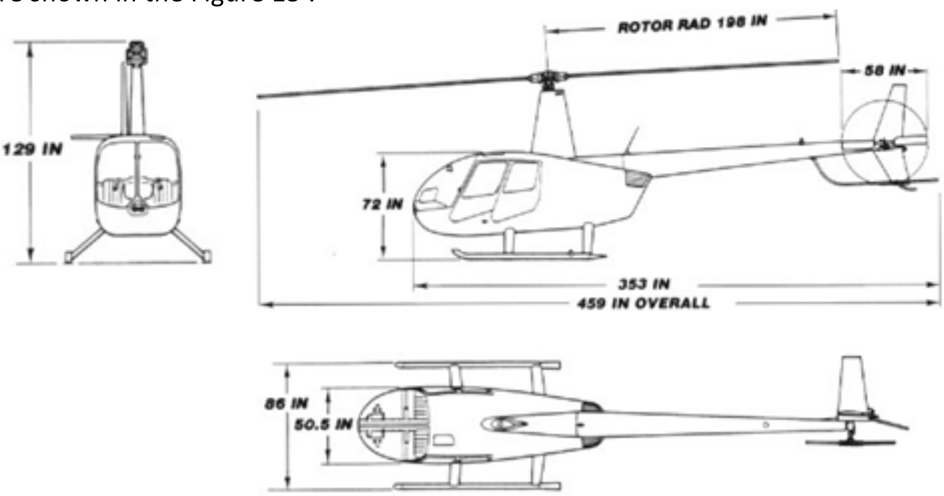


Figure 13 - Three Views of the Robinson R44

Main characteristics of R44:

- **Crew:** one or two pilots
- **Capacity:** four, including pilot
- **Payload:** 748 lb (340 kg)
- **Length:** 38 ft 3 in (8.96 m)
- **Rotor diameter:** 33 ft (10.1 m)
- **Tail rotor diameter:** 4 ft 10 in (1.5 m)
- **Height:** 10 ft 9 in (3.3 m)
- **Empty weight:** 1,450 lb (657.7 kg)
- **Loaded weight:** 2,500 lb (1,134 kg)
- **Powerplant:** 1 × Lycoming IO-540-AE1A5 6 cylinder, flat engine with fuel injection, 245 bhp (183 kW)

The **AgustaWestland AW119 Koala** is an eight-seat utility helicopter powered by a single turboshaft engine produced for the civil market. The three views of this helicopter are shown in the Figure 14:

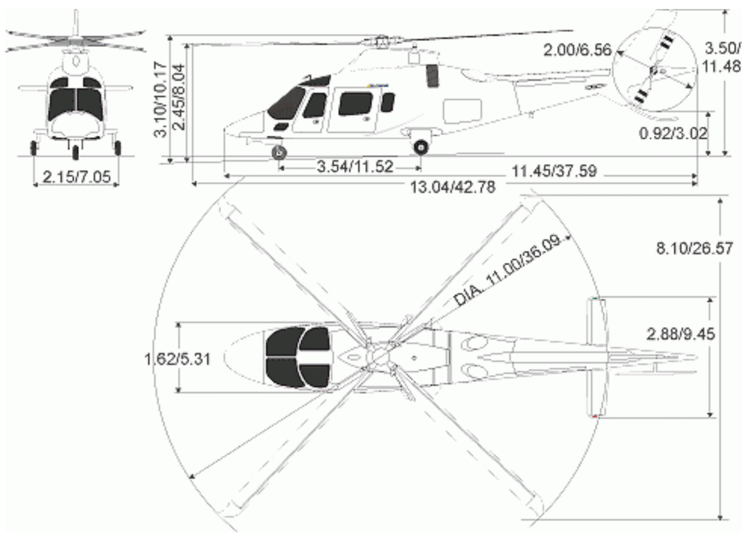


Figure 14 - Three views of the AW119

Main characteristics of AW119:

- **Crew:** 1 pilot
- **Capacity:** 6-7 passengers
- **Length:** 13.01 m (42 ft 8 in)
- **Rotor diameter:** 10.83 m (35 ft 6 in)
- **Height:** 3.77 m (12 ft 4 in)
- **Disc area:** 92.1 m^2 (991 ft^2)
- **Empty weight:** 1,430 kg (3,152 lb)
- **Loaded weight:** kg (lb)
- **Max. takeoff weight:** 2,720 kg (6283 lb)
- **Powerplant:** 1 × Pratt & Whitney Canada PT6B-37A turboshaft, 747 kW (1,002 hp)

4.1.2. Comparative table between Cessna 172, R44, AW119 and MiniBee

With the comparison table, it is possible to make estimations, as well as validations, about the minibee. As the three choosen aircrafts are in the boundaries of minibee's operational scope, the final values of minibee's properties should be next to it's comparisons.

	MINIBEE PLANE	*CESSNA 172*	*R44*	*AW 119*
	Hybrid aircraft	Fixed-wing aircraft	Rotary-wing aircraft	Rotary-wing aircraft
General characteristics				
Capacity	three passengers	three passengers	three passenger	six or seven passengers
Length [m]	-	8.28	8.96	13.01
Wingspan [m]	-	11		
Rotor diameter [m]	-		10.1	10.83
Wing area [m2]	-	16.2		
MTOW [kg]	2000	1111	1134	2720
Performance				
Cruise speed [km/h]	300+-50	226	200	220
Range [km]	600/800	1289	560	991

Table 1 - Comparison between MiniBee, Cessna 172, R44 and AW119

4.2. Priorities list

The priorities list, as the name infers is a tool that is useful to define priorities based on the client needs. It contains two rows, the first one which is the flexibility column, which represents how flexible the implementation of certain characteristic is to the project. It varies from 0 to 4, which 0 means not flexible and 4 means flexible. The second column is called criticity, and it represents how critical is the implementation of certain feature to the project.

EFO: Exigences functionelles			
The MiniBee has to be *HYBRID*		0	H
The MiniBee can be equipped with limited seats	4 \ 5	1	L
The MiniBee can have a determinate PAYLOAD VOLUME		3	M
The MiniBee must use use a thermical engine		0	L
The MiniBee has to be *LOW-COST* machine	< 100,000€	0	H
EPE: Exigences de performance			
The MiniBee has *HIGH RANGE*	600 to 800 km	0	M
The MiniBee can have *ELEVATED CRUISE SPEED*	300 ± 50 km/h	1	H
EIN: Exigences d'interfaces			
The MiniBee is provided with PASSENGER ENTRANCE		0	L
The MiniBee need an interface for REFUELING		3	L
The MiniBee need a CHARGING PORT		3	L
The MiniBee need a COMMUNICATION SYSTEM		1	L
The MiniBee need a NAVIGATION SYSTEM		1	L
The MiniBee is provided with car-like pilot interface		1	H
EOP: Exigences opérationelle			
The MiniBee must *Take-Off Vertical*		0	M
The MiniBee must have ELECTRIC CONTROL AND STABILITY		0	H
The MiniBee must consider EMERGENCY LANDING		2	H
The MiniBee can have MAX DIMENSION LIMIT		2	M
The MiniBee can have *ELEVATED CRUISE SPEED*	300 ± 50 km/h	1	H
The MiniBee is provided with a LANDING GEAR		3	L
The MiniBee can have *LIMITED OPERATIONAL COST*		2	M
C: Contraintes enviromentales			
The MiniBee must consider TEMPERATURE LIMIT		2	L
The MiniBee must consider WEATHER LIMIT		2	M
The MiniBee is provided with ANTI/DE-ICEING SYS		3	M
The MiniBee is designed for low *FUEL CONSUMPTION*		2	M
The MiniBee is designed for low *EMISSIONS*		3	M
The MiniBee can be composed by *RECYCLABLE* parts		3	M
EV: Exigences de verification et validation			
The MiniBee must consider ACCELERATION LIMIT		1	M
The MiniBee must consider NOISE LIMIT		1	M

The MiniBee has to be conform with *SAFETY GUIDELINE*		0	H
The MiniBee can have *Maximum Take-Off Weight*	MTOW < 2000 kg	0	H
EMA: Exigences de maintenance			
The MiniBee can be equipped with *INTERCHANGEABLE PIECES*		3	M
The MiniBee can have high *ACCESSIBILITY*		3	M
The MiniBee is designed for *CHEAP MAINTENANCE*		3	H
Flexibilité: 0=not modificable; 4=free			
Criticité: L=low; M=medium; H=high			

Table 2 - Priorities list table

4.3. Project delimitation

4.3.1. Determination of the external configuration

Initialy three configurations for the MiniBee were proposed by the team. A “X” based configuration, as shown on Figure 15, a “H” based configuration as shown on Figure 16, and a usual Airplane configuration as shown on Figure 17.

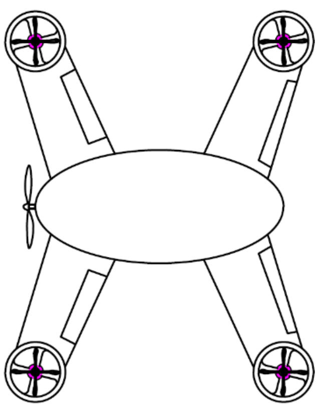


Figure 15 - X configuration

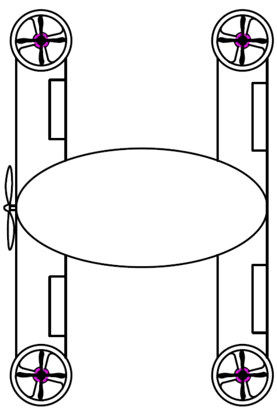


Figure 16 - H configuration

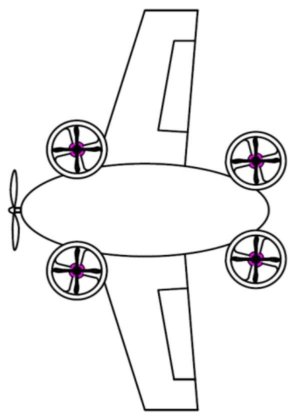


Figure 17 - Airplane configuration

Each one of the proposed configurations has it’s avantages and disadvantages, for instance, the X and H configurations present a fairly more reasonable control of the quadcopter system, while the Airplane configuration presents a lower weight (due to less structure requided), and a far more efficient general aerodynamic. To choose the most appropriate configuration, a choice matrix, was created and is shown below.

	"A"	"X"	"H"	
ROTORS LOCATION	2	5	5	1=DIFFICULT TO CONTROL/LOWER STABILITY
SMALL CONTROL SURFACES	4	2	2	5=EASY TO CONTROL/HIGH STABILITY
SWEEP ANGLE	4	1	4	
WING LOCATION	4	1	2	1=NOT "CLEAN" AERODYNAMICS
ROTORS LOCATION	3	1	1	5="CLEAN" AERODYNAMICS
SWEEP ANGLE	5	2	5	1=HEAVY STRESSED STRUCTURE
ROTORS LOCATION	4	2	2	5=LIGHT STRUCTURE
SUM	26	14	21	

Table 3 - Choise Matrix for the configuration

The rotors close to the fuselage A represent a stability decrease and control difficulties; the forward swept angle X gives yaw instability; the small wing on X and H doesn't allow effective control surfaces.The wings at the same height on X and H can interact with each other and rotors located at the wing-tip waste bearing surface on X and H and also makes the wing structure more haevy and stressed. The forward sweep angle reduces the divergence speed X.

Considering the results from the choise matrix, and the explanation, the A – Airplane configuration is considered the optimal configuration, and by that it is the choosen one.

4.3.2. Propulsion determination

The Thermal engine is needed to ensure hybrid capabilities, and also as the core of the power provider of the aircraft. It has to be able to provide enough power to the propellers, to ensure cruise flight, as well as provide power to a generator, to charge the batteries of the electric system during flight. To search for thermic engines, the following criterias were considered as the most Important:

- Reliability: represents the ability of a system or component to function under stated conditions for a specified period of time. This characteristic is very important in the aeronautical field because a failure can compromise people safety. Components reliability is usually inversely proportional to the components complexity.
- Robustness: is the ability of a structure to withstand events like fire, explosions, impact or the consequences of human error, without being damaged to an extent disproportionate to the original cause. Also in this case this characteristic is very important because it can affect the safety of the aircraft.
- Smooth rotation of the shaft : with cylinder engine we don't have a perfect continuity in the rotation of the shaft because motion is the result of the succession of the strokes. In wankel rotatory engine there are no cylinder, this engine uses an eccentric rotary design to convert pressure into rotating motion.
- No thermic shock cooling : in a diesel engine during the compression, the temperature of about 550 °C is reached. After during the induction stroke the cylinder is again filled with clean air at low temperature so there is a big difference in temperature. In the rotatory engine we don't have this problem because cold air and exhaust gas are always in the same part of the engine.

- Temperature gradient : for the same reason of the point above in the Wankel rotatory engine we have a big temperature gradient between the two part of the motor.
- Low purchase cost : we want to project a low cost aircraft so this point is very important. We need a good but low cost engine.
- High power to weight ratio : is a calculation commonly applied to engines and mobile power sources to enable the comparison of one unit or design to another. Power-to-weight ratio is a measurement of actual performance of any engine or power source.
- Compact size : the size of the engine is very important for the aeronautical field because it affects the frontal area of the aircraft.
- Simple construction : this characteristic is linked to reliability and maintainability, a less complex engine is more easy to repair.
- Low cost fuel : a low cost fuel reduces the cost of the aircraft mission.

The caracterystics of two different engines were analysed:

- SMA SR305-230E is an air/oil-cooled, horizontally opposed, four-cylinder, four-stroke, diesel piston aircraft engine as shown on the Figure 20 bellow:

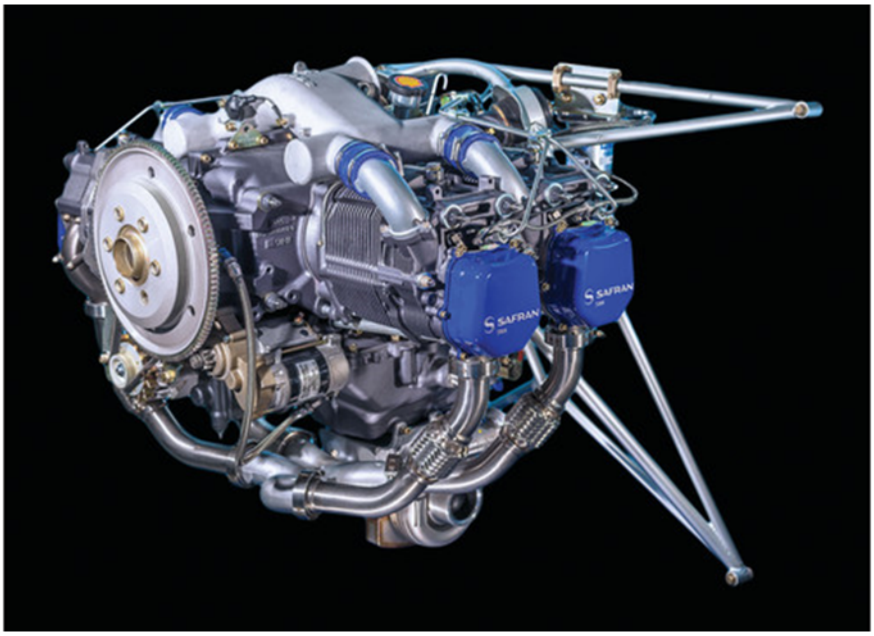


Figure 18 - SMA SR305-230E Engine

- MISTRAL G300 is a Wankel rotary engine being developed by the Mistral Engine Company for use in light aircraft and helicopters shown on the Figure 19 bellow:

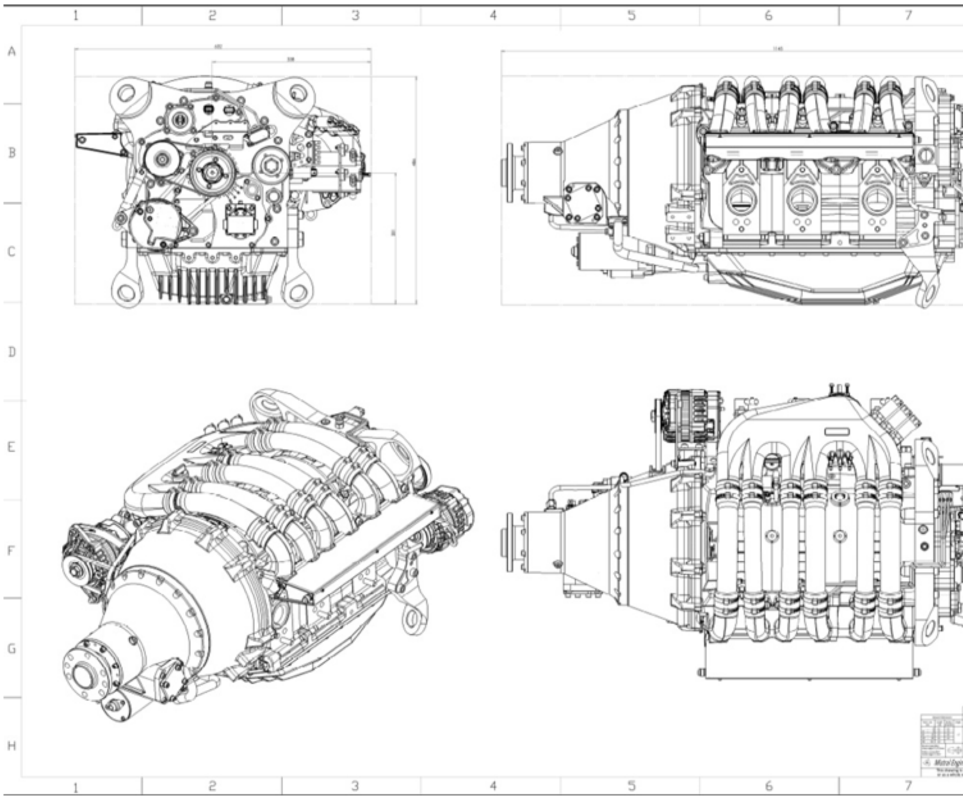


Figure 19 - MISTRAL G300 Wankel Engine

The table below show the difference between the two engine:
(a scale of quality is used : 3->good 1->bad)

Characteristic	SMA SR305-230[E]		MISTRAL G300	
	Data	N	Data	N
Reliability		2		2
Robustness		2		3
Smooth rotation of the shaft		1		3
No thermic shock cooling		1		3
Low purchase cost	75,000 $	2	50,000€	3
High power to weight ratio	1.1(=230hp/206Kg) hp/kg	1	1.7(=300hp/177kg) hp/kg	3
Compact size	834*931*784 [mm]	1	632*1145*486 [mm]	3
Simple construction		1		3
Temperature gradient		3		1
Low cost fuel	Diesel 8.06 $/gallone	3	10LL avgas 12.68 $/gallone	2
TOT		17		26

Table 4 - Comparison between the two choosen engines

The biggest problem with the diesel engine is the bigger dimension and the lower power to weight ratio. These are caused by the not very good optimization of the engine form and also by the high pressure reached during compression that requires thicker casing.

The wankel rotary engine is the one which meets most of these criteria. It has no major internal parts exposed to reversing stresses, meaning that it can present an extended durability in comparison to other types of engines. In theory the life of the major parts in the engine such as the e-shaft and rotor is unlimited. There are no exhaust valves to melt or burn, meaning that it provides more safety for aeronautical applications. The two rotor engine is downright tiny compared to an air cooled aircraft piston engine and also the power to weight ratio of the engine is excellent. The disadvantages of the Wankel engine can be listed as:

- Low Thermic efficiency
- High fuel consumption
- Exhaust gases may contain more carbon Monoxide
- High temperature gradient inside of the motor, causing thermic stress.

Also there were durability issues in the past, due to the way the engine works, but new materials have allowed today these problems to be extinguished.

How a Wankel rotatory engine works:
https://www.youtube.com/watch?v=6BCgl2uumlI

The most important choice criteria for MiniBee concept are High "power to weight ratio" and compact size , to minimize weight and volume. Ability to run low cost fuel and Low purchase cost to make the project inexpensive.

The choice is focused on Mistral G-300, a Wankel rotary engine that uses three rotating combustion chambers and provides 300hp (maximum take off power), a liquid-cooling system and full electronic control.

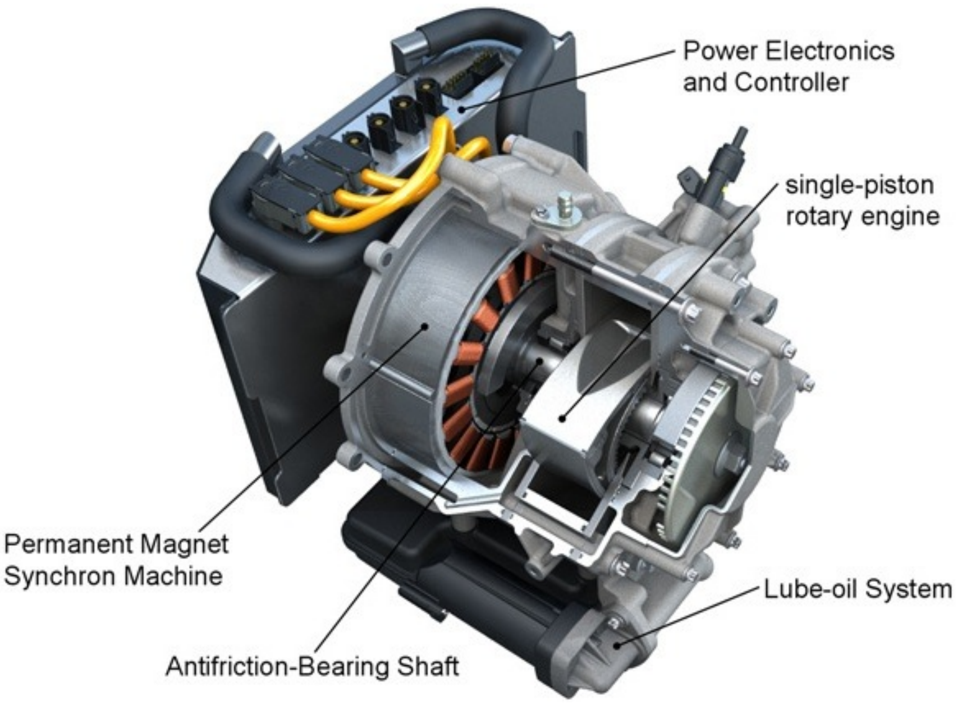


Figure 20 - Cut view of a Mistral G-300 power generator

Power		300 hp (224kW)	Maximum take off power
Weight		177 kg	With accessories
Dimesions	Lenght	672 mm	With accessories
	Width	540 mm	With accessories
	Height	520 mm	With accessories
Price		50.000 $	

Table 5 - Characteristics of Mistral engine

A preliminary power requirement estimation in cruise flight shows that:

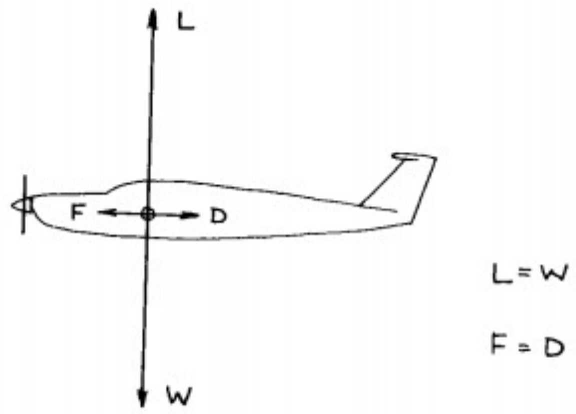


Figure 21 - General forces diagram of an airplane

$$F = Tpropeller = D = 1/2\,\rho\, Vcruise^2\, S\, Cd$$

Equation 1- Drag definition

$$P = 1/\eta\, T * Vcruise$$

Equation 2 - Power definiton

F	Vertical force		
Tpropeller	Traction propeller force		
D	Drag		
ρ	Air density	1,21 kg/m^3	
Vcruise	Cruise speed	300 km/h	Project goal
S	Surface	17 m^2	Previously explained
η	Propeller efficiency	0,8	
C_D	Drag coefficient	0,02	Referred to minimum Clcruise [8]
P	Needed power	150 kW	TO ESTIMATE

Table 6 - Synthesis of equations 1 and 2

It is required to consider that wings are not the only source of drag and the thermal engine has to charge the batteries during the cruise, therefore it is reasonable to assume a minimum power of 200 kW. It corresponds at the power request of same category's aircraft.

At last, it seems appropriated to consider tractor propeller installation because it improve the aircraft stability and allow smaller control surfaces [9].

TRACTOR INSTALLATION	PUSHER INSTALLATION	WING INSTALLATION
Heavy engine up front		
shorten forebody		
smaller tail area		
improved stability		
indisturbated cooling air for the propeller	disturbated air for the propeller	
disturbater air for the wing	indisturbated air for the wing	
	reduced aircraft friction drag	
	reduced cabin noise	
	improved pilot's outside vision	
	required longer landing gear	
	damage by the rocks thrown up by the wheels	
		normally for the multi engines
		reduce fusolage drag
		introduce controllabily problem
		increase tail and rubber size
		longer landing gear

4.3.3. Electric Motors

Sizing of propellers used for the VTOL is one of the most difficult part of the project. However a simple estimation can be find through the momentum theory or disk actuator theory [10]: according to this theory the rotor is modeled as an infinitely thin disc with an infinite number of blades that induce a constant pressure jump over the disk area and along the axis of rotation.

The power required (considering 4 propellers) in the ideal case, for each propeller, is:

$$P = T^{(3/2)}/(2*\sqrt{\rho * A})$$

Equation 3 - Power estimation for the propeller

P	Power needed	TO ESTIMATE
T	Propeller traction	
ρ	Air density	1,22 kg/m^3
A	Area = $r_{propeller}{}^2*\pi$	TO ESTIMATE

Table 7 - Synthesis of equation 3

Considering that $T = Weight = \frac{1}{4}9{,}81*2000\,N$ for each propeller, the needed power can be choose in function of the most convenient diameter:

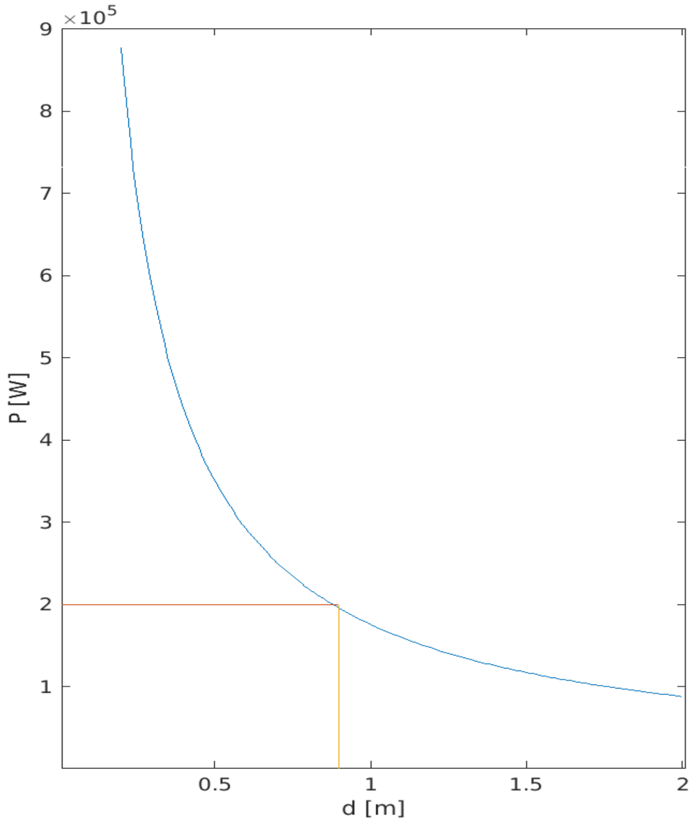


Figure 22 - Power vs rotor diameter diagram

During this first analysis, it seems appropriate to choose a diameter of 0.9 m for each propeller, which corresponds to a required power of approximately 200.000 W.

Below there are some comparisons between few proposed models:

		Yuneec Power Drive 40 [11]	Siemens Concept [12]	Emrax 268 [13]
Dimensions	Length	163 mm		91 mm
	Width	240 mm		268 mm
	Height	240 mm		268 mm
Weight		19 kg	50 kg	20,3 kg
Power		40 kW at 2400 rpm	260 kW	80 kW at 4000 rpm
Power to weight ratio		***2,1 kW/kg***	***5,2 kW/kg***	***4 kW/kg***

Table 8 - Comparison between the 3 proposed e-motors

The table shows matrix comparison between the 3 motors:

	Yuneec Power Drive 40	**Siemens Concept**	**Emrax 268**
Production	4	2	4
Dimensions	1	5	4
Weight	2	5	4
Power to Weight Ratio	2	5	4
Feasibility *	1	5	3

	10	22	19
1= inappropriate			
5= preferable			

Table 9 - Choose matrix of the e-motors

It is clear that the best choice is the Siemens Concept, but it is not yet available; so we opted for Emrax 268 (although we need more units).

5. INITIAL PROJECT

5.1. Initial sketch

Based on the previously choosen concept "A", a initial proposition for the MiniBee was proposen by the sketch bellow (Figure 23). Note that this is only a general preliminarly concept, and it probably is going to be deeply changed as the development process evolves.

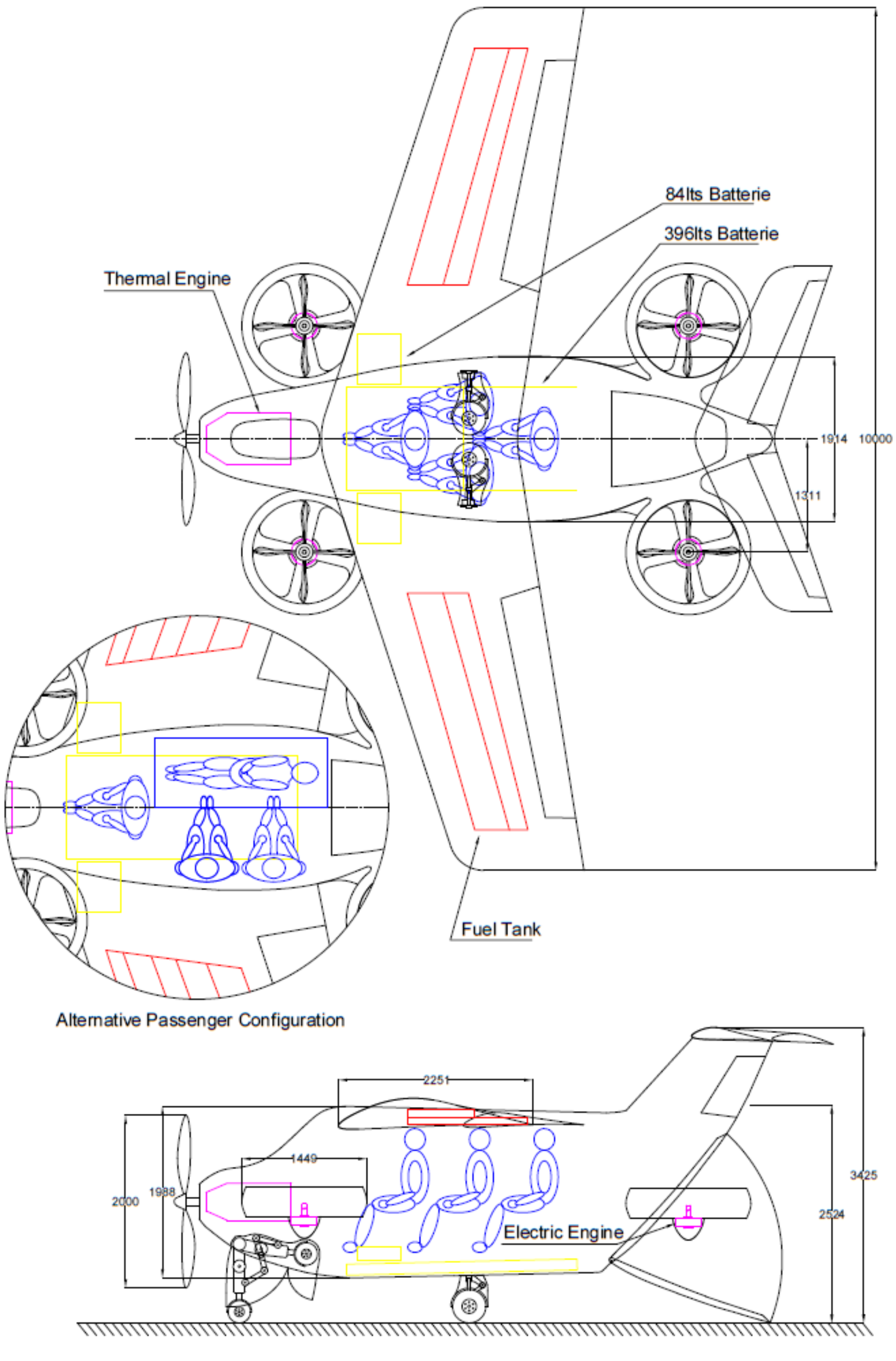


Figure 23 - MiniBee initial concept

5.2. Wing sketch and stall information

In order to estimate the wing size, a simple lift-definition application [2] can be used

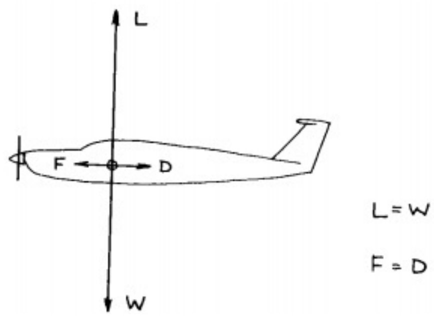


Figure 24 - Cruise equilibrium

$$L = MTOW = \frac{1}{2} * \rho * Vstall^2 * Clmax * S$$

Equation 4 - Lift definition

L	Lift	-	-
MTOW	Maximum take off Weight	2000kg	Constrain specified for 4 passengers MiniBee version
ρ	Air density	1,29 kg/m^3	Sea level value- Take off Value
Vstall	Stall speed	130 km/h (36,1 m/s)	Accepted value for the same category aircraft's [1]
Clmax	Maximum lift coefficient	1,6	Accepted value for the same category aircraft's [1]
S	Wing surface	15 m^2	TO ESTIMATE

Table 10 - Synthesis of equation 4

The results show that a 24 m^2 wing surface guarantees enough vertical force during the cruise. It is clear that without wings, the rotors can provide lift but MiniBee can't reach high cruise speed: the aim is to increase stall speed in order to reduce the wing surface, respecting the maximum rotational speed of the propellers.

In fact, if we assume 130km/h (36 m/s) stall speed, the necessary surface is only 15 m^2: this can be a better solution considering that one of the most important MiniBee characteristics is the ability to take off and land everywhere.

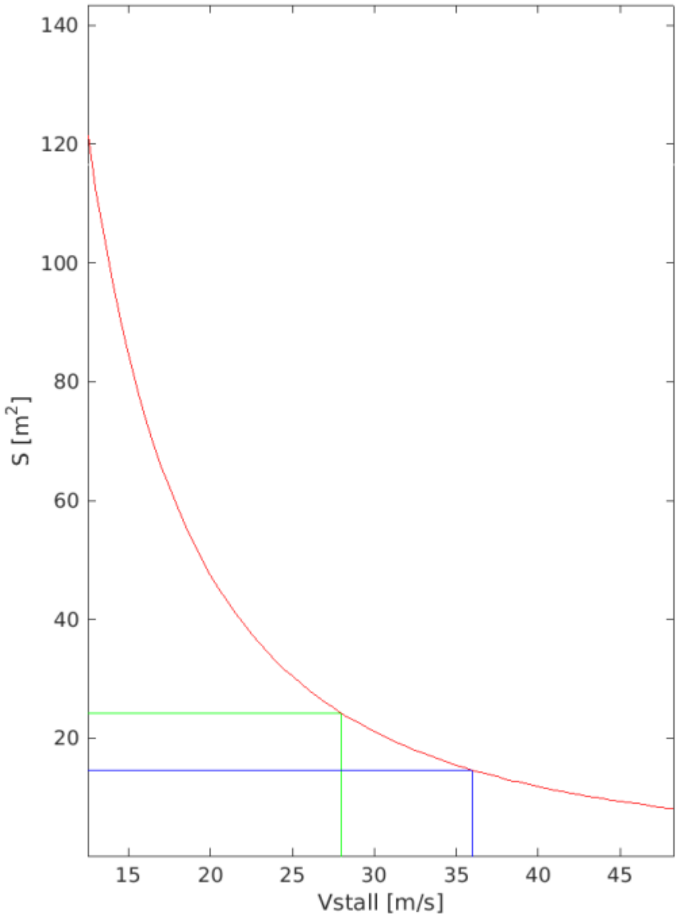


Figure 25 - Wings surface estimation

5.3. Definition of Airfoil and Geometry Selection

Any section of the wing cut by a plane parallel to the aircraft symmetrical plane is called **Airfoil**: It usually looks like a positive cambered section that the thicker part is in front of the airfoil.
An airfoil-shaped body moved through the air will vary the static pressure on the top surface and on the bottom surface of the airfoil.
A typical airfoil section is shown in figure, where several geometric parameters are illustrated.

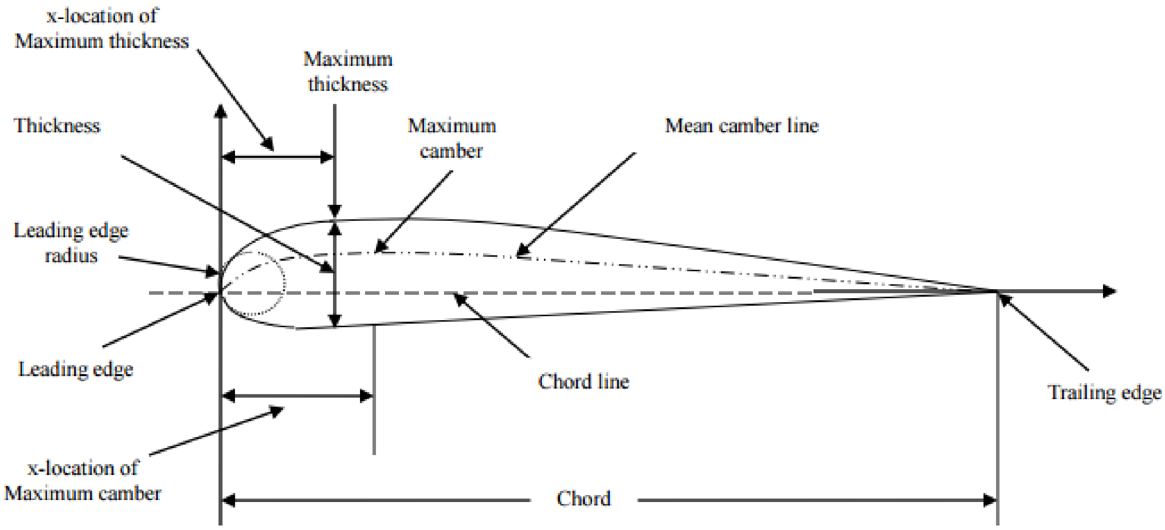


Figure 26 – Typical Airfoil Section

The camber of airfoil is usually positive: the upper surface static pressure in less than ambient pressure, while the lower surface static pressure is higher than ambient pressure. This is due higher airspeed at upper surface and lower speed at lower surface of the airfoil. As the airfoil angle of attack increases, the pressure difference between upper and lower surfaces will be higher.

In the process of wing airfoil selection, we do not look at airfoil geometry only, or its pressure distribution. Instead, we examine the airfoil operational outputs that are more informative to satisfy design requirements.

There are several graphs that illustrate the characteristics of each airfoil when compared to other airfoils in the wing airfoil selection process. Thus, we evaluate the performance and characteristics of an airfoil by looking at the following graphs.

1- The variations of lift coefficient versus angle of attack

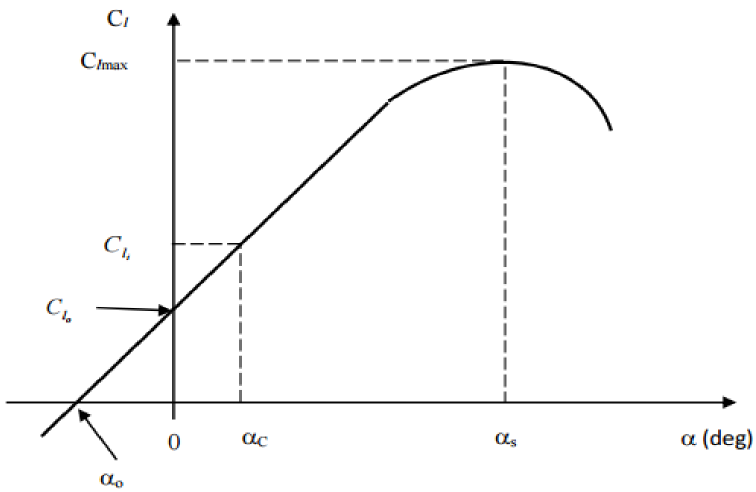


Figure 27 - Lift coefficient versus angle of attack

a. The stall angle (αs) is the angle of attack at which the airfoil stalls; i.e. the lift coefficient will no longer increase with increasing angle of attack.

The stall angle is directly related to the flight safety, since the aircraft will lose the balance of forces in a cruising flight. It the stall is not controlled properly; the aircraft may enter a spin and eventually crash. In general, the higher the stall angle, the safer is the aircraft, thus a high stall angle is sought in airfoil selection.

b. The maximum lift coefficient (Clmax) is the maximum capacity of an airfoil to produce non-dimensional lift; i.e. the capacity of an aircraft to lift a load (i.e. aircraft weight). The maximum lift coefficient is usually occurs at the stall angle.

The stall speed (Vs) is inversely a function of maximum lift coefficient, thus the higher Clmax leads in the lower Vs. Thus the higher Clmax results in a safer flight. Therefore, the higher maximum lift coefficient is desired in an airfoil selection process.

c. The zero lift angle of attack (αo) is the airfoil angle of attack at which the lift coefficient is zero. The design objective is to have a higher αo (more negative), since it leaves the capacity to have more lift at zero angle of attack.

This is essential for a cruising flight, since the fuselage center line is aimed to be level (i.e. zero fuselage angle of attack) for variety of flight reasons such as comfort of passengers.

d. The ideal lift coefficient (Cli) is the lift coefficient at which the drag coefficient does not vary significantly with the slight variations of angle of attack. The ideal lift coefficient is usually corresponding to the minimum drag coefficient.

This is very critical in airfoil selection, since the lower drag coefficient means the lower flightcost. Thus, the design objective is to cruise at flight situation such that the cruise lift coefficient is as close as possible to the ideal lift coefficient.

e. The lift coefficient at zero angle of attack (Cl0) is the lift coefficient when angle of attack is zero. From design point of view, the more Cl0 is the better, since it implies we can produce a positive lift even at zero angle of attack.

f. The lift curve slope (Cl α) is another important performance feature of an airfoil. The lift curve slope is the slope of variation of lift coefficient with respect to the change in the angle of attack, and its unit is 1/deg or 1/rad. Since the main function of an airfoil is to produce lift, the higher the slope, the better the airfoil.

g. Another airfoil characteristic is the shape of the lift curve at and beyond the stall angle of attack (stall behavior).

An airfoil with a gentle drop in lift after the stall, rather than an abrupt or sharp rapid lift loss, leads to a safer stall from which the pilot can more easily recover

.

2- The variations of pitching moment coefficient versus angle of attack

The slope of this graph is usually negative and it is in the region of negative Cm for typical range angle of attacks. The negative slope is desirable, since it stabilizes the flight, if the angle of attack is disturbed by a gust. The negative Cm is sometimes referred to as nose-down pitching moment. This is due to its negative direction about y-axis which means the aircraft nose will be pitched down by such moment.

3- The variations of drag coefficient versus lift coefficient

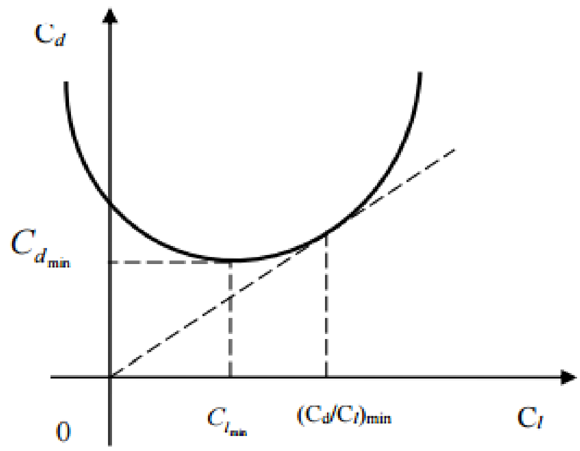


Figure 28 - Drag coefficient versus lift coefficient

The lowest point of this graph is called minimum drag coefficient (C dmin).
The corresponding lift coefficient to the minimum drag coefficient is called Cl min. As the drag is directly related to the cost of flight, the C dmin is of great importance in airfoil design or airfoil selection. A line drawn through the origin and tangent to the graph locates a point that denotes to the minimum slope. This point is also of great importance, since it indicates the flight situation that maximum Cl-to-Cd ratio is generated, since (Cd/Cl)min = (Cl/Cd)max.
This is an important output of an airfoil, and it is referred to as the maximum lift-to-drag ratio. In addition of requirement of lowest dmin C , the highest (Cl/Cd)max is also desired.

5.3.1. Airfoil Families

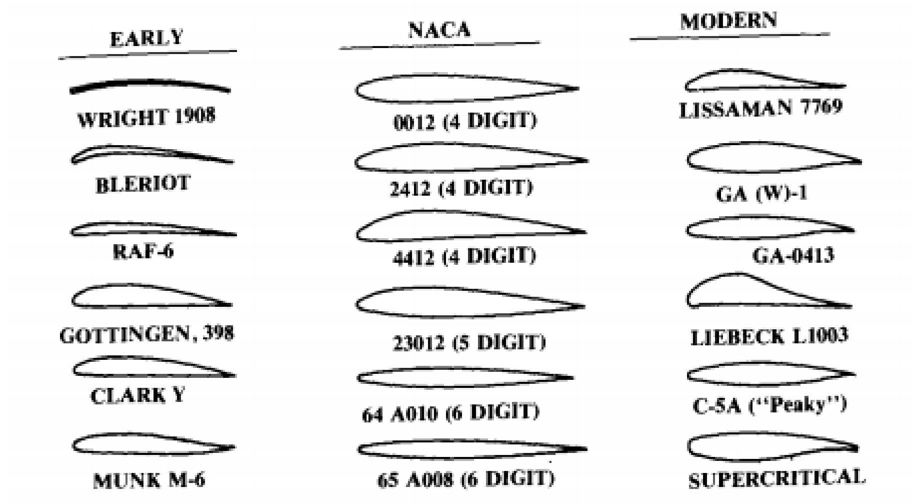


Figure 29 - Airfoil Families

Three following groups of NACA airfoils are more interesting:

a. Four-digit NACA airfoils

The four-digit NACA airfoil sections are the oldest and simplest NACA airfoils to generate. The camber of a four-digit airfoil has made up of two parabolas. One parabola generates the camber geometry from the leading edge to the maximum camber, and another parabola produces the camber shape from the maximum camber to the trailing edge. In a Four-digit NACA airfoil, the first digit indicates the maximum camber in percent chord. The second digit indicates the position of maximum camber in tenths of chord length. The last two digits represent the maximum thickness-to-chord ratio. Although these airfoils are easy to produce, but they generate high drag compared with new airfoils.

b. Five-digit NACA airfoils

The camber of a five-digit airfoil section has made up of one parabola and one straight line. The parabola generates the camber geometry from the leading edge to the maximum camber, and then a straight line connects the end point the parabola to the trailing edge. In a five-digit NACA airfoil section; the first digit represents the 2/3 of ideal lift coefficient in tenths. It is an approximate representation of maximum camber in percent chord. The second digit indicates the position of maximum camber in 200 of chord length. The last two digits represent the maximum thickness-to-chord ratio.
The NACA five-digit airfoils were developed to allow shifting the position of maximum camber forward for greater maximum lift.

c. 6-series NACA airfoils

The four- and five-digit airfoil sections where designed simply by using parabola and line.

They were not supposed to satisfy major aerodynamic design requirements, such as laminar flow and no flow separation. When it became clear that the four- and five-digit airfoils have not been carefully designed, NACA researchers begin the investigation to develop new series of airfoils that have been driven by design requirements.

On the other hands, newly designed faster aircraft require more efficient airfoil sections. The six series airfoils were designed to maintain laminar flow over a large part of the chord, thus they maintain lower Cdmin compared with four and five digit airfoils.

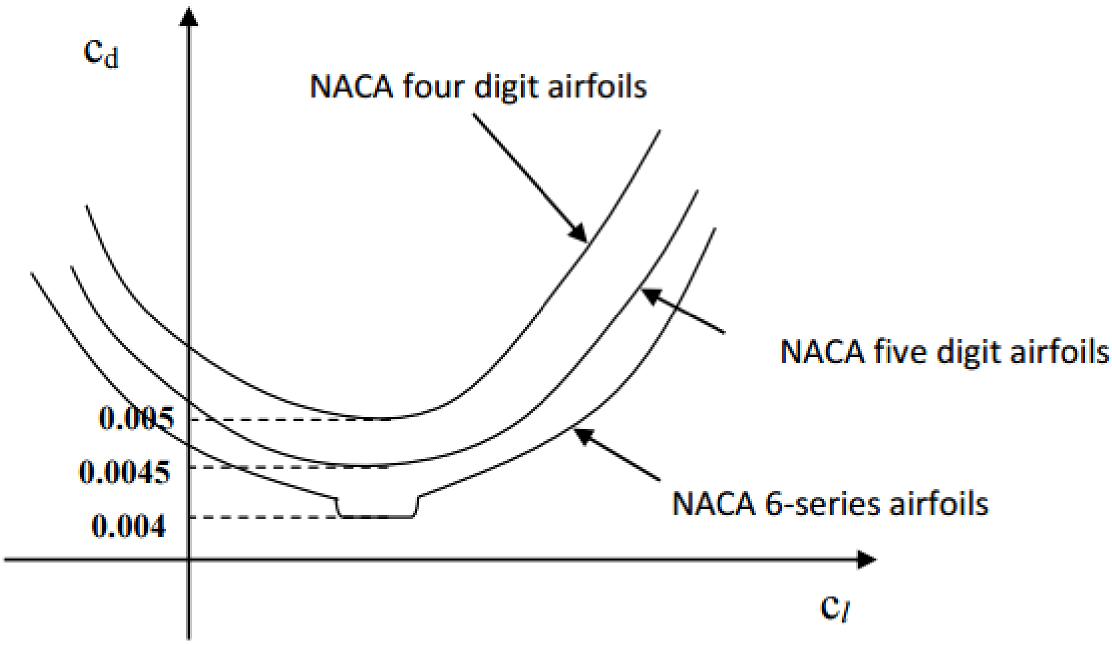


Figure 30 - Airfoil families comparison

5.3.2 Airfoil Selection Criteria

Selection of an airfoil for a wing begins with the clear statement of the flight requirements. For the Minibee Plane, the main requirements are:

- The airfoil with the highest maximum lift coefficient (Cl max) that means the lowest stall speed;
- The airfoil with the proper ideal or design lift coefficient (Cld or Cli);
- The airfoil with the lowest minimum drag coefficient (Cdmin) that means the highest maximum speed;
- The airfoil with the highest lift-to-drag ratio ((Cl/Cd)max) that means the highest endurance;

We must also consider other requirements such as airworthiness, structural, manufacturability, and cost requirements; we can neglect the stall quality and the lowest Cmo because Minibee can use the electrical motors for control and stability.

5.3.3 Airfoil Selection

I. **Calculate the aircraft ideal cruise lift coefficient**

No	Airfoil section	$C_{l\max}$ at $Re=3\times10^6$	α_s (deg)	C_{mo}	$(C_l/C_d)_{max}$	C_{li}	C_{dmin}	$(t/c)_{max}$
1	0009	1.25	13	0	112	0	0.0052	9%
2	4412	1.5	13	-0.09	125	0.4	0.006	12%
3	2415	1.4	14	-0.05	122	0.3	0.0065	15%
4	23012	1.6	16	-0.013	120	0.3	0.006	12%
5	23015	1.5	15	-0.008	118	0.1	0.0063	15%
6	63_1-212	1.55	14	-0.004	100	0.2	0.0045	12%
7	63_2-015	1.4	14	0	101	0	0.005	15%
8	63_3-218	1.3	14	-0.03	103	0.2	0.005	18%
9	64-210	1.4	12	-0.042	97	0.2	0.004	10%
10	65_4-221	1.1	16	-0.025	120	0.2	0.0048	21%

Figure 31- Airfoil Caratheristics

In a cruising flight, the aircraft weight is equal to the lift force so:

$$Lift = Weight = \frac{1}{2} * \rho * Vc^2 * C_{Lc} * S$$

Equation 5 - Lift definition

where Vc is the aircraft cruise speed, Xis the air density at cruising altitude, and S is the wing platform area. Considering that:

- Vc= 320km/h = 90 m/s
- S =17 m^2
- ❒ = 0,8 kg/m^3

It can be obtained $C_{lc} = 0{,}35$.

II. **Calculate the wing ideal cruise lift coefficient**

Basically, the wing is solely responsible for the generation of the lift. However, other aircraft components also contribute to the total lift; negatively, or positively; sometimes, as much as 20 percent.
Thus the relation between aircraft cruise lift coefficient and wing cruise lift coefficient is a function of aircraft configuration. The contribution of fuselage, tail and other components will determine the wing contribution to aircraft lift coefficient.
In the preliminary design phase the following approximate relationship can be used:

$$C_{Lci} = C_{Lc}/0{,}95 = 0{,}368$$

III. **Instead the aircraft maximum lift coefficient (C_{lmax} =1,6) is estimated on accepted value for the same category aircraft's**

IV. **Identify airfoil section alternatives that deliver the desired Cli and Clmax**

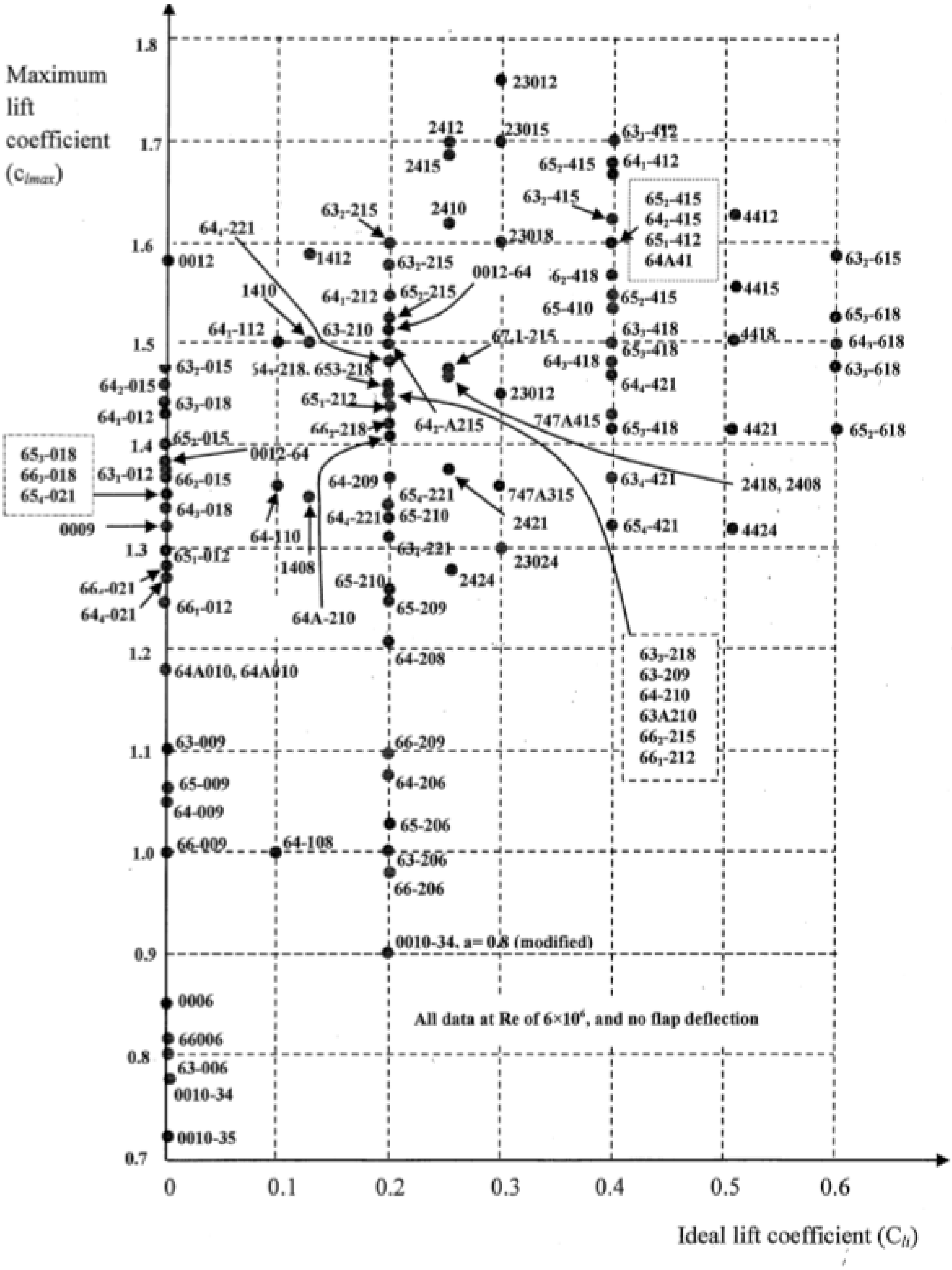


Figure 32 - Maximum Lift Coefficient vs Ideal lift coefficient

The figure shows a collection of Cli and Clmax for several NACA airfoil sections in just one graph. The horizontal axis represents the airfoil ideal lift coefficient while the vertical axis the airfoil maximum lift coefficient. Every black circle represents one NACA airfoil section.

Thus, we need to look for NACA airfoil sections that yield an ideal lift coefficient of 0.36 and a maximum lift coefficient of 1,6.

Referring to figure we find the following airfoils whose characteristics match with our design requirements:

NACA	Clid	Clmax	Cdmin	(Cl/Cd)max	Cm	Stall Quality
23018	0,3	1,6	0,007	114,3	-0,009	Sharp
65_2-415	0,4	1,6	0,004	133,3	-0,057	Docile
64_2-415	0,4	1,6	0,005	135	-0,068	Docile
65_1-412	0,4	1,6	0,004	145,5	-0,072	Moderate

Table 11 - Comparison between different profiles

Using a comparison table that incorporates the weighted design requirements, we obtain:

Requirement		NACA 23018	NACA 65_2-415	NACA 64_2-415	NACA 65_1-412
Highest maximum lift coefficient	30%	4	5	5	5
proper ideal lift coefficient	25%	4	4	4	4
Lower minimum drag coefficient	25%	3	5	4	5
highest lift-to-drag ratio	10%	2	3	3	4
lowest Cmo	5%	2	3	3	4
Stall quality	5%	2	4	4	3
	1	3,35	4,4	4,15	4,5

Table 12 - Matrix of choice for different profiles

1= inappropriate
5= preferable

The aircraft performance (stall speed, endurance and maximum speed) are of greatest important design requirement so the airfoil section NACA 65_1-412, NACA 65_2-415, and NACA 64_2-415 are the best respectively. We can neglect the stall quality and the lowest Cmo because Minibee can use the electrical motors for control and stability.

5.4. Propelers estimation

To fly, planes exploit the principle of action and reaction, i.e. they move on pushing behind the air. There are many ways to exploit this principle, one of these is using a propeller.
Before the advent of jet propulsion propellers were the only means to enable aircraft to exist. Nowadays the propellers are mainly used on small size airplanes flying at relatively low speeds for commercial aviation. The propellers in fact present problems at high speed, in contrast with turbojets, because the tip of the blades tends to easily reach Mach 1, the blades must therefore be stalled and loses its efficiency.

For a small aircraft like the one in this project the use of the propeller is more than justified because the required powers is low and speeds in the game are well below Mach 1.
Moreover, as already seen in the section on the engine, the internal combustion rotary engine coupled with propeller is the best solution in terms of cost and simplicity.

The propeller can be seen as a screw that goes into the air. Imagining that the propeller screw in a solid body, the distance covered (in the direction parallel to the axis of the propeller) after a full turn is called the geometric pitch. The space that a propeller performs in the air is rather lower, because of the compressibility of this one, and takes the name of real pitch. The difference between these two size takes the name of propeller slip. This factor describes the goodness of the propeller, a good propeller has a very little slip.

The efficiency of a propeller is determined by the equation 6:

$$\eta = \frac{\text{propulsive power out}}{\text{shaft power in}} = \frac{\text{thrust} \cdot \text{axial speed}}{\text{resistance torque} \cdot \text{rotational speed}}.$$

Equation 6 - Propeller Eficiency

The profile of a blade is very similar to the profiles used by the wings and is subject to the same forces as shown on figure 33:

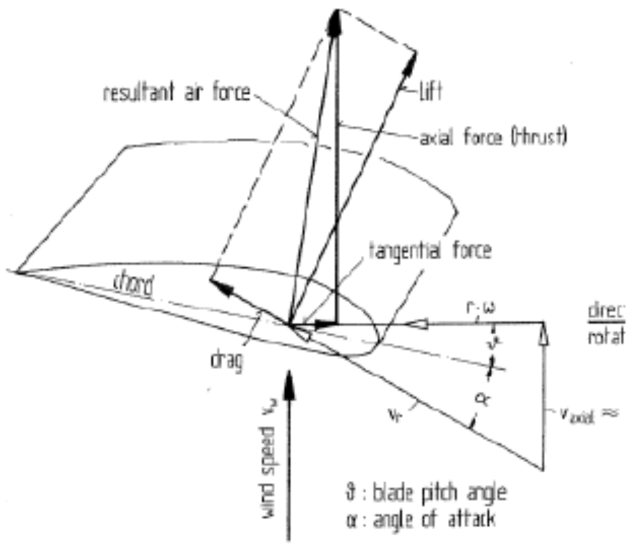


Figure 33 - Forces diagram of the Propeller

-Drag: parallel to the chord;
-Lift: perpendicular to the chord.

The resultant of these forces can be decomposed, in the case of a helix, in:
-Thrust: force parallel to the axis of the propeller
-Strength on the plane of rotation: which must be overcome by the torque

To better understand the operation of propeller we can use the actuator disc theory (ADT).
This is a theory of mechanical origin whose purpose is to study the exchange of momentum through a propeller.
The assumptions of this theory are:
-ideal disc: the propeller is imagined as an infinitely thin disk with an infinite number of blades, the drag of this one is zero and it is permeable;
-there is a pressure drop between the two surfaces of the disk;

-The flow is irrotational;
-the downstream speed of the disk is constant.
Through these considerations, we have that:

$$u = \sqrt{\frac{T}{A}}\sqrt{\frac{1}{2\rho}}$$

Equation 7 - Ideal velocity equation

Where:

- T=thrust
- A=disk area
- p=air density
- u=ideal velocity

We can then consider the definition of power:

$$W_{id} = Tu = 2\rho A u^3$$

Equation 8 - Ideal power definition

Where:

- Wid= ideal power

And at the end derive the relationship between the area of the thrust disk and the ideal power:

$$Wi = \sqrt{\left(\frac{T^3}{A \cdot 2 \cdot p}\right)}$$

Equation 9 - Derivate of the ideal power

This report is very useful for a first dimensioning in disk size and can be observed graphically in the figure 34:

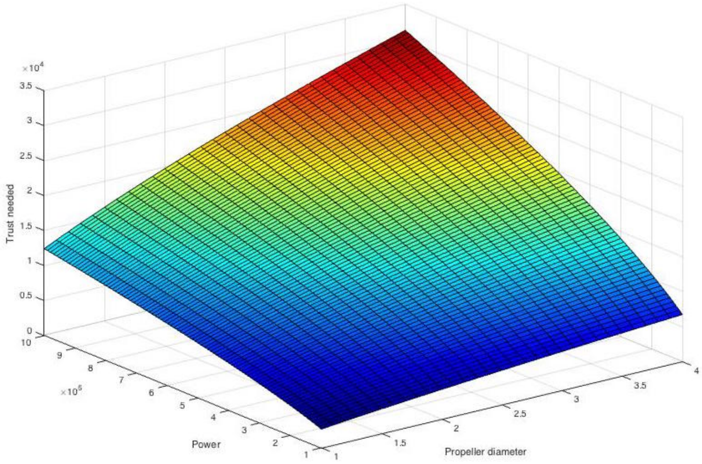


Figure 34 - Power vs Diameter vs Trust

It must remember that this theory does not take account of the blade profile, or their strength or their number.

It is still a good approximation for the calculation for the dimensioning of the propeller in the early stages of design. Therefore, we will use to select plausible values of thrust, disc area and ideal power then to analyze in more detail by means of simulation programs.

The selected data is therefore:

- wid = 220 kW
- T = 7666 N
- D = 2.2 m

Account must be taken of the fact that the data so obtained are valid at a fixed point, i.e. it is not considered the forward speed of the aircraft.

After approximate data are obtained, we can use a program to calculate better data for the propeller.

The program chosen for this task is "JavaProp", an application that can be easily downloaded from the web (website: http://www.mh-aerotools.de/airfoils/javaprop.htm) based on the theory of the optimum propeller, as developed by Betz, Prandtl, Glauert.

The parameters you enter are relatively few:

- number of blades;
- speed axial flow;
- propellerdiameter;
- used profiles;
- available power;
- density.

Let's look at how the choice of these initial parameters is made:

Number of blades: the number of blades affect, only lightly, propeller efficiency. Tests are made with 3 an 4 blades without changing other characteristics. It has been found that passing from 3 to 4 blades efficiency is increased from 86.7% to 87.3%, this increment can be considered negligible and for this reason, preferring the simplicity of construction, it was decided to use a rotor with 3 blades.

Flow axial velocity (v): This value coincides in our case with the cruise speed of the aircraft.

Propeller diameter (D): In this section, we use the estimated diameter through the actuator disc Theory as already explained above.

Profiles used for blade: the choice of the profiles is limited to those in the program, after some simple consideration we opted for the following profiles:

-root (r / R = 0.00): MH 112 16.2%

-Middle (r / R = 00:33 - r / R = 0.66): MH 114 13%

-Tip (r / R = 1.00): MH 116 9.8%

Available power (P): the available power correspond to that which can be delivered by the engine. Having chosen a G300 Mistral the available power is 220 kW.

Density (p): we are projecting an aircraft propeller so the density is that of air, for this reason, it is set equal to 1.22 kg / m ^ 3.

After entering this data we start the calculation of the ideal profile, the results are provided in the form of graphics and data. The analysis was carried out considering the power value given (it is set by the selected engine) and leaving the thrust and torque as variables.

The first data we get there are provided in the form of a table:

v/(nD)	1.011
v/(ΩR)	322
Efficiency η	86.473%
Thrust T	2,281.02 N
Power P	220 kW
Torque Q	933.69 Nm
β at 75%R	24.6°
Ct	0.0567
Cp	0.0663
Cs	1.7395
Pitch H	2.37 m

Table 13 - Propeller properties

Where:

n = revolution par second (1/s)

Ω = angular speed (1/s)

η = propeller efficiency

Ct = thrust coefficient

Cp = power coefficient

Cs = speed-power coefficient

β = blade angle

H = pitch heigh

The data in the table above were calculated in the design point, i.e. at the point where it is expected that the plane works for most of the time. These data, therefore, are only partially representative of the entire operation of the aircraft, during the flight, in fact, this will pass through various regimes.

The second thing that the program provides us is the geometric shape along the blade length. These data are provided in table form:

r/R	c/R	β	H/D	r	c	H	t	Airfoil
[-]	[-]	[°]	[-]	[mm]	[mm]	[mm]	[mm]	[-]
0.0000	Spinner	-	-	-	-	-	-	-
0.0500	Spinner	-	-	-	-	-	-	-
0.1000	Spinner	-	-	-	-	-	-	-
0.1500	0.0343	66.4	1.1	165.0	37.7	2369.4	6.2	interpolated
0.2000	0.0506	59.7	1.1	220.0	55.6	2369.4	8.8	interpolated
0.2500	0.0640	53.9	1.1	275.0	70.4	2369.4	10.8	interpolated
0.3000	0.0737	48.8	1.1	330.0	81.1	2369.4	12.0	interpolated
0.3500	0.0797	44.4	1.1	385.0	87.7	2369.4	12.7	MH 114 13%, Re=500,000
0.4000	0.0827	40.6	1.1	440.0	90.9	2369.4	13.2	interpolated
0.4500	0.0833	37.3	1.1	495.0	91.6	2369.4	13.3	interpolated
0.5000	0.0822	34.4	1.1	550.0	90.4	2369.4	13.1	interpolated
0.5500	0.0797	31.9	1.1	605.0	87.7	2369.4	12.7	interpolated
0.6000	0.0763	29.7	1.1	660.0	84.0	2369.4	12.2	interpolated
0.6500	0.0722	27.8	1.1	715.0	79.4	2369.4	11.5	MH 114 13%, Re=500,000
0.7000	0.0707	26.1	1.1	770.0	77.7	2369.4	11.0	interpolated
0.7500	0.0701	24.6	1.1	825.0	77.1	2369.4	10.5	interpolated
0.8000	0.0685	23.2	1.1	880.0	75.4	2369.4	9.8	interpolated
0.8500	0.0653	22.0	1.1	935.0	71.9	2369.4	8.9	interpolated
0.9000	0.0592	20.9	1.1	990.0	65.1	2369.4	7.7	interpolated
0.9500	0.0470	19.8	1.1	1045.0	51.7	2369.4	5.9	interpolated
1.0000	0.0024	18.9	1.1	1100.0	2.6	2369.4	0.3	MH 116 9.8%, Re=500,000, M=0.5

Table 14 - Geometric shape

Where:

- r = radius station
- c = chord length
- Airfoil = the airfoil at each station
- t = thickness

It is supplied also a view of the blade in 2D and a graph showing the trend of the ratio Pitch / Diameter along the blade as shown in figure 35:

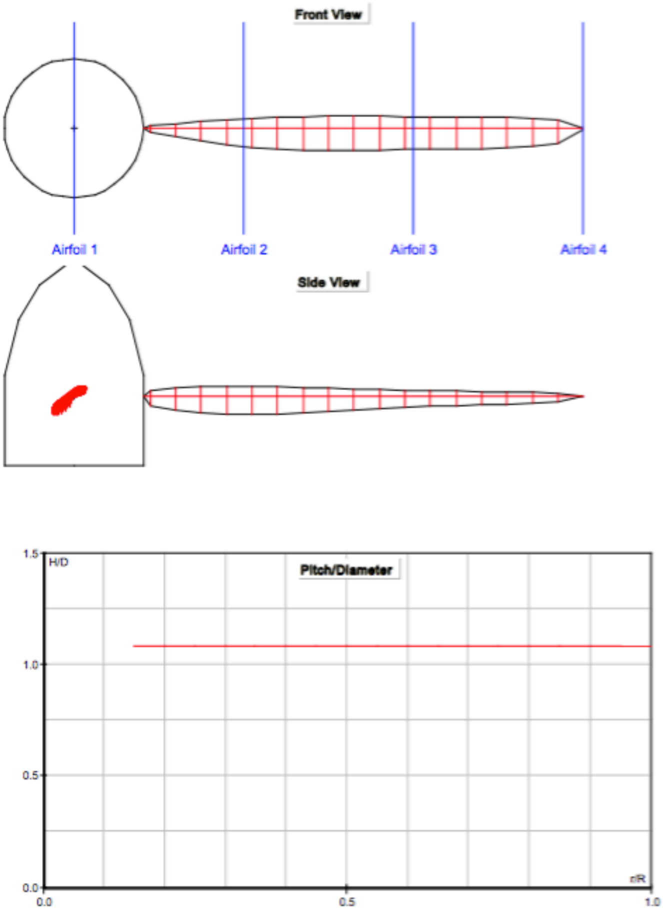


Figure 35 - Blade geometry

The program also allows you to export the file format .igs of the blade itself figure 36, which makes it easily unpresentable already in 3DEXPERIENCE.

Figure 36 - Blade output on 3DExperience

As already said, the data above represent the propeller in its operating point. It is possible to carry out indicative analyses of the propeller behavior outside the conditions sets for the calculation. The analyzes were carried out considering the number of engine revolutions constant and leaving the other parameters vary. Excluding the phases of takeoff and landing this represents a trend fairly common in aircraft, therefore, even if the analyzes remain theoretical, allow to better understand the trend of the variables during operation.

v/(nD)	v/(ΩR)	Ct	Cp	Cs	η	η*	stalled	v	rpm	Power	Thrust	Torque
[-]	[-]	[-]	[-]	[-]	[%]	[%]	[%]	[m/s]	[1/min]	[kW]	[kN]	[kNm]
0.000	0.000	0.110247	0.079624	0.000052	0.00	0.01	15.00 !	0.00	2250	264.221	4.434	1.121
0.100	0.032	0.108033	0.079617	0.165882	13.57	31.11	39.00 !	8.25	2250	264.198	4.345	1.121
0.200	0.064	0.109684	0.078579	0.332635	27.92	51.54	24.00 !	16.50	2250	260.754	4.412	1.107
0.300	0.095	0.109920	0.079638	0.497619	41.41	65.13	15.00	24.75	2250	264.268	4.421	1.122
0.400	0.127	0.108536	0.081947	0.659710	52.98	74.37	8.00	33.00	2250	271.931	4.366	1.154
0.500	0.159	0.105583	0.084256	0.820068	62.66	80.83	0.00	41.25	2250	279.592	4.247	1.187
0.600	0.191	0.101024	0.085958	0.980154	70.52	85.47	0.00	49.50	2250	285.238	4.063	1.211
0.700	0.223	0.094392	0.086285	1.142643	76.58	88.96	0.00	57.75	2250	286.325	3.797	1.215
0.800	0.255	0.083844	0.082916	1.316321	80.89	91.81	0.00	66.00	2250	275.146	3.372	1.168
0.900	0.286	0.071175	0.076216	1.506029	84.05	94.08	0.00	74.25	2250	252.912	2.863	1.073
1.000	0.318	0.058150	0.067392	1.715055	86.29	95.82	0.00	82.50	2250	223.631	2.339	0.949
1.100	0.350	0.044811	0.056336	1.955398	87.50	97.18	0.00	90.75	2250	186.944	1.802	0.793
1.200	0.382	0.031179	0.042946	2.252152	87.12	98.26	0.00	99.00	2250	142.509	1.254	0.605
1.220	0.388	0.028416	0.039975	2.322748	86.72	98.45	0.00	100.65	2250	132.651	1.143	0.563
1.240	0.395	0.025646	0.036911	2.398782	86.16	98.63	0.00	102.30	2250	122.483	1.032	0.520
1.260	0.401	0.022863	0.033746	2.481565	85.36	98.81	0.00	103.95	2250	111.982	0.920	0.475
1.280	0.407	0.020068	0.030480	2.572809	84.27	98.98	0.00	105.60	2250	101.143	0.807	0.429
1.300	0.414	0.017264	0.027116	2.674839	82.77	99.14	0.00	107.25	2250	89.981	0.694	0.382
1.320	0.420	0.01444	0.023651	2.79128	80.6	99.2	0.00	108.9	225	78.483	0.58	0.333

		9		0	4	9		0	0		1	
1.340	0.427	0.011624	0.020084	2.927751	77.55	99.44	0.00	110.55	2250	66.646	0.468	0.283
1.360	0.433	0.008769	0.016389	3.094759	72.76	99.59	0.00	112.20	2250	54.385	0.353	0.231
1.380	0.439	0.005921	0.012615	3.309016	64.77	99.73	0.00	113.85	2250	41.862	0.238	0.178
1.400	0.446	0.003059	0.008732	3.613329	49.04	99.86	0.00	115.50	2250	28.975	0.123	0.123
1.420	0.452	0.000199	0.004762	4.137413	5.95	99.99	0.00	117.15	2250	15.802	0.008	0.067
1.440	0.458	-0.002731	0.000601	6.347160	0.00	99.99	0.00	118.80	2250	1.995	-0.110	0.008
1.460	0.465	-0.005632	-0.003612	4.495851	0.30	50.87	0.00	120.45	2250	-11.985	-0.227	-0.051

Table 15 - Project Analysis

Where:
η* = maximum possible efficiency
stalled = relative disc area swept by stalled airfoils

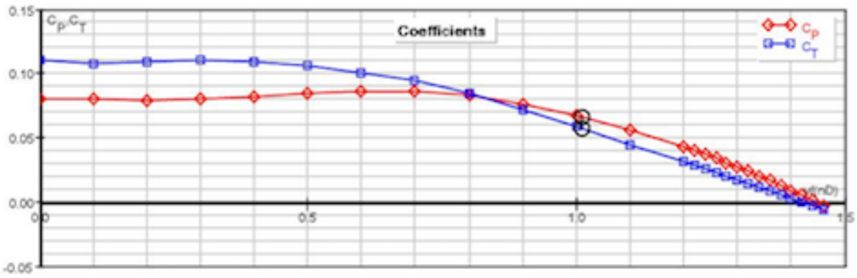


Figure 37 - Cp vs Cy

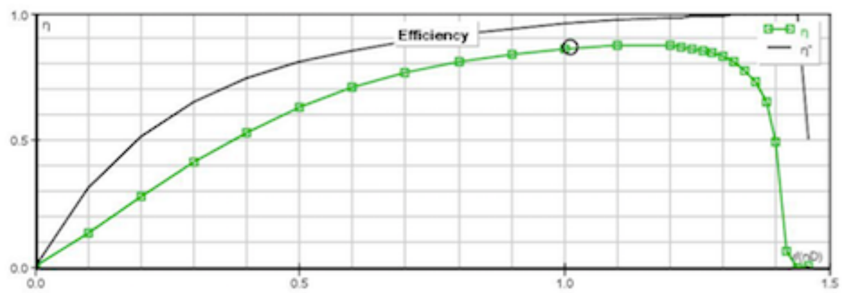


Figure 38 - Efficiency graph

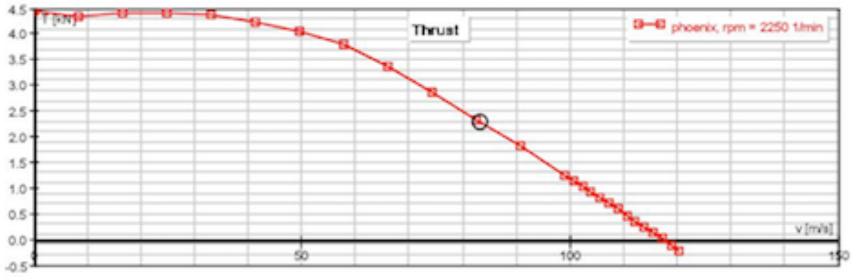


Figure 39 – Trust vs speed

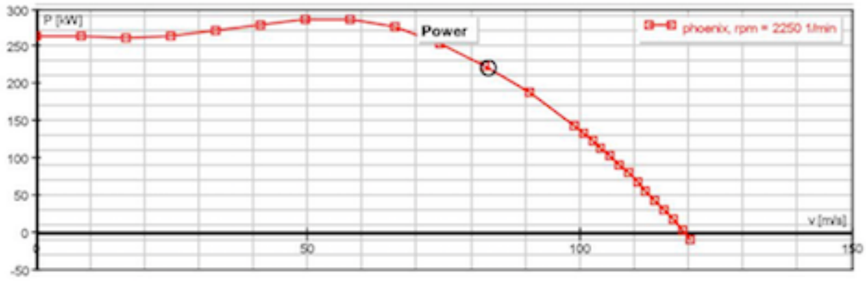


Figure 40 - Power vs speed

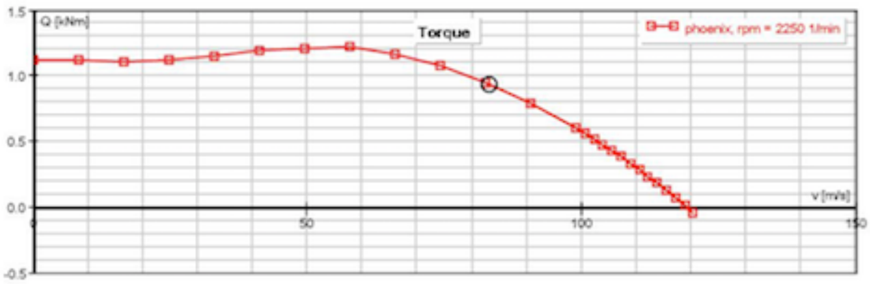


Figure 41 - Torque vs Speed

From the graphs above it is possible to see the trend of the power (P) and of the torque (Q) reflect that of the Cp. As is to be expected the Thurst (T) decreases as the speed increases, this means that the plane can move forward only up to a set speed. The collapse of efficiency is due to the fact that at some point the jump in speed between the upstream and the downstream section of the disk becomes zero, in simple words the propeller is not able to accelerate the flow over a certain speed. A blade so designed show a good behaviour in the operating point and its characteristics do not degrade suddenly outside of it. Thanks to program used it is possible to have an approximate vision of the flow velocity through the propeller. As shown in figure 42, before the propeller there is a quite uniform velocity, in contrast after it, it is possible to see the gradient of velocity given by the propeller itself.

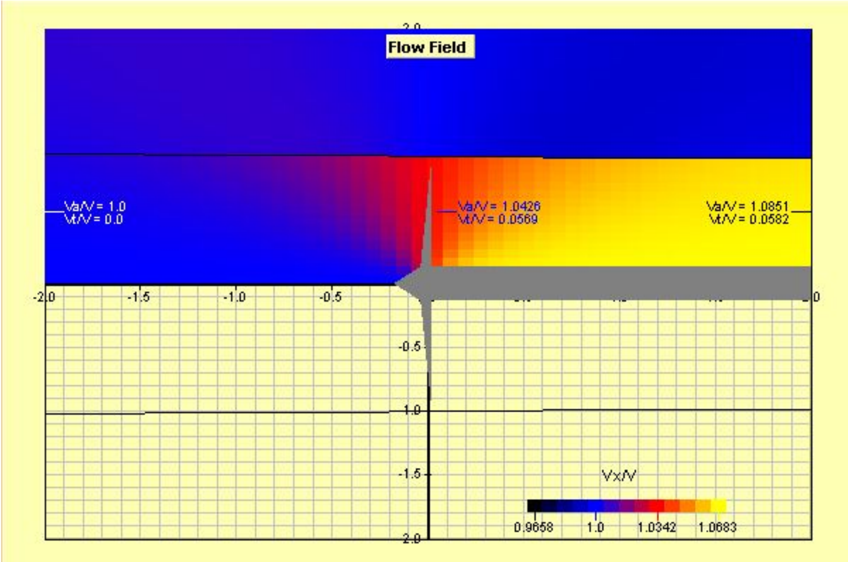


Figure 42 - Propeller aerodynamic analysis

There are basically three ways in which a blade can be mounted to the hub:

- **fixed pitch**: in this configuration the blade, once mounted to the hub, has no possibility to rotate around its own axis. This configuration is used in small planes devoid of electronic stuff for reasons of simplicity and to reduce the load on the pilot who otherwise would be forced to manually change the pitch of the blades.
- **variable pitch on the ground:** this configuration, as the previous one, does not allow the modification of the pitch during the phase of flight, but it allow to set the pitch when the aircraft is on the ground according to the flight conditions. The possibility of variation of the pitch causes larger hub and more complex construction and structure.
- **Variable pitch:** also in this case we have a more complex hub, but this is compensated by the possibility of orienting the blades also depending on the phases of flight. This configuration is very used when the aircraft mounts already on board electronic stuff. The pitch variability also allows greater maneuverability on the ground and reduction of landing space thanks to the possible reversal flow.

Given the characteristics of the airplane it is decided to opt for a variable pitch so as to ensure the necessary handling and to increase efficiency in all phases of flight.

5.5. Weight estimation

MiniBee must meet narrow objectives (range, endurance, cruise speed) and it is important to predict, for a given mission specification, the take off gross weight (Wto), the empty weight (We) and the mission fuel weight (Wf). It will be only a simple estimation based on relationships between existing types of airplanes [Jan Roskam fonte], but it's very important in order to begin the detailed design of aircraft's components.

It is significant to underline that this procedure has been developed for "traditional" airplane configuration, i.e. not for VTOL airplanes or helicopters; nevertheless it can be adapted considering that during the cruise flight the Minibee acts as traditional aircraft and paying attention during take off and landing phases.

A convenient way to break down Wto is:

$$\mathrm{Wto} = \mathrm{Woe} + \mathrm{Wf} + \mathrm{Wpl}$$

Equation 10 - Weight breakdown

Where:

- Wto is the take off gross weight and it is a constrain which means fixed Wto=2500kg;
- Wpl is the payload weight; it is normally specified in the profile mission and it consist of passengers, baggage and crew members.
- Wf is the mission fuel weight;
- Woe is the operating weight empty.

5.5.1. Determination of mission payload weight

The payload weight can be easily determined for each mission profile considering an average weight of 80kg per person and 15kg of baggage for short to medium distance flights. As the Minibee can also be used as ambulance, a reasonable assumption is to consider 100kg of medical equipments.
Since there are 4 passengers in Minibee's configuration and the crew consists of one person:

$$Wpl = (80 + 15) * 5 + 100 = 575kg$$

5.5.2. Determination of mission fuel weight

Mission fuel weight, Wf, can be written as:

$$Wf = Wf_{used} + Wf_{res}$$

Where

- Wfused is the fuel actually used during the mission
- Wfres are the fuel reserves required for the mission and it can be estimated as 5% of Wfused.

To determine the Wfused the so-called *Fuel Fraction Method* will be used: the Minibee mission, as can be seen in figure 43, is broken down into different mission phase and the fuel used during each phase is found from a simple calculation or estimated on the basis of experience. The fuel fraction for each phase is defined as the ratio of final weight to start weight. It is necessary to study both VTOL and conventional TOL, to have more accurate valuation.

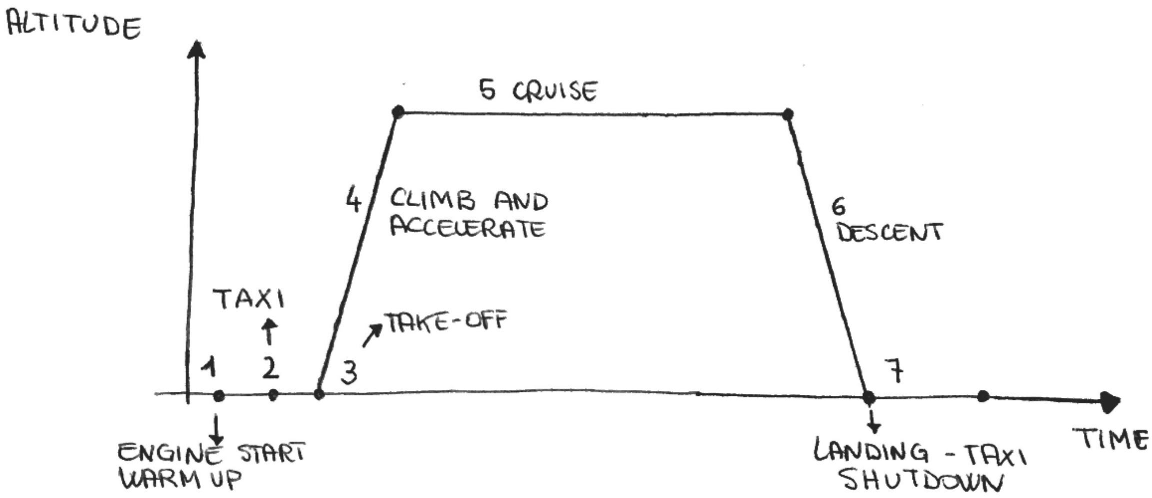


Figure 43 - Mission profile

- Phase 1 : Engine start and warm-up.
- Phase 2 : Taxi
- Phase 3 : VTOL
- Phase 3a : Conventional take off
- Phase 4: Climb to cruise altitude and accelerate to cruise speed
- Phase 5: Cruise
- Phase 6: Descent
- Phase 7: Landing – Taxi and shut-down

For each phase, the Fuel fraction can be determined following the table –figure 44, by Roskam. The suggested fuel-fractions are based on experience, on judgement or on existing airplanes.

PHASE 1		**FUEL FRACTION**	**PHASE 4**		**FUEL FRACTION**
Begin Weight	Wto	W1/Wto= 0.995	Begin Weight	W3	W4/W3= 0.992
End Weight	W1		End Weight	W4	
PHASE 2		**FUEL FRACTION**	**PHASE 5**		**FUEL FRACTION**
Begin Weight	W1	W2/W1= 0.997	Begin Weight	W4	W5/W4= 0.93 **
End Weight	W2		End Weight	W5	
PHASE 3		**FUEL FRACTION**	**PHASE 6**		**FUEL FRACTION**
Begin Weight	W2	W3/W2= 1	Begin Weight	W5	W6/W5= 0.993
End Weight	W3		End Weight	W6	
PHASE 3a		**FUEL FRACTION**	**PHASE 7**		**FUEL FRACTION**
Begin Weight	W2	W3a/W2= 0.998	Begin Weight	W6	W7/W6= 0.993
End Weight	W3a		End Weight	W7	

Table 16 - Fuel fraction

** The ratio W5/W4 can be estimated from Breguet's range equation, which can be written:

$$Rcr = 375\left(\frac{np}{cp}\right)\left(\frac{L}{D}\right)\ln\frac{W4}{W5}$$

Where:

Rcr [miles]	Cruise Range	435	mission costrain
np	Propeller efficency	0,8	
cp [lbs/hp/hr]	Propeller specific consumption	0,5	
L/D	Lift to Drag Ratio	10	

Table 2.2 Suggested Values For L/D, c_j, η_p, And For c_p For Several Mission Phases

Mission Phase No. (See Fig.2.1)	Cruise (5) L/D	Cruise c_j lbs/lbs/hr	Cruise c_p lbs/hp/hr	Cruise η_p	Loiter (6) L/D	Loiter c_j lbs/lbs/hr	Loiter c_p lbs/hp/hr	Loiter η_p
Airplane Type								
1. Homebuilt	8-10*		0.6-0.8	0.7	10-12		0.5-0.7	0.6
2. Single Engine	8-10		0.5-0.7	0.8	10-12		0.5-0.7	0.7
3. Twin Engine	8-10		0.5-0.7	0.82	9-11		0.5-0.7	0.72
4. Agricultural	5-7		0.5-0.7	0.82	8-10		0.5-0.7	0.72
5. Business Jets	10-12	0.5-0.9			12-14	0.4-0.6		
6. Regional TBP's	11-13		0.4-0.6	0.85	14-16		0.5-0.7	0.77
7. Transport Jets	13-15	0.5-0.9			14-18	0.4-0.6		
8. Military Trainers	8-10	0.5-1.0	0.4-0.6	0.82	10-14	0.4-0.6	0.5-0.7	0.77
9. Fighters	4-7	0.6-1.4	0.5-0.7	0.82	6-9	0.6-0.8	0.5-0.7	0.77
10. Mil.Patrol, Bomb, Transport	13-15	0.5-0.9	0.4-0.7	0.82	14-18	0.4-0.6	0.5-0.7	0.77
11. Flying Boats, Amphibious, Float Airplanes	10-12	0.5-0.9	0.5-0.7	0.82	13-15	0.4-0.6	0.5-0.7	0.77
12. Supersonic Cruise	4-6	0.7-1.5			7-9	0.6-0.8		

Notes:
1. The numbers in this table represent ranges based on existing engines.
2. There is no substitute for common sense! If and when actual data are available, these should be used.
3. A good estimate for L/D can be made with the drag polar method of Sub-section 3.4.1.

* Homebuilts with smooth exteriors and/or high wing loadings can have L/D values which are considerably higher.

Figure 44 - Fuel Fraction

Now it is possible to calculate the mission fuel fraction, Mff from:

$$Mff = \frac{W1}{Wto} \prod_{i=1}^{i=6} \frac{W\ i+1}{W\ i}$$

Equation 11 - Mission Fuel Fraction

The fuel used during the mission can be found from:

$$Wfused = (1 - Mff)Wto$$

And the value for mission fuel weight can finally be determined from equation 11:

$$Wf = (1 - Mff)Wto + Wfres$$

Equation 12 - Fuel Weight

According that, the results for VTOL and conventional take off and landing are summarized in table 17.

VTOL		CONVENTIONAL TOL	
Mff	0,9024255	Mff	0,900620645
Wf [kg]	256,133073	Wf [kg]	260,8708072

Table 17 - Fuel Weight

5.5.3. Determination of operating empty weight

Considering the equation 10 and the achieved results, the operating empty weight can be obtained:

Wto [kg]	take off gross weight	2500
Wf [kg]	fuel weight	260
Wpl [kg]	Payload weight	575
Woe [kg]	*Operating Empty weight*	*1665*

This evaluation is coherent with the weight tendency of single engine propeller airplane, as can be noticed in the figure 45.
The trend line was established with regression analysis and it shows the state-of-the-art of airplane design: although the source is not updated with the latest data, it can be reasonably accepted in this preliminary analysis.
Finally, can be noticed that it is desirable to have as small a value for We but in this case the batteries weight and the rotors weight have a preponderant role in structure weight, therefore in empty operating weight.

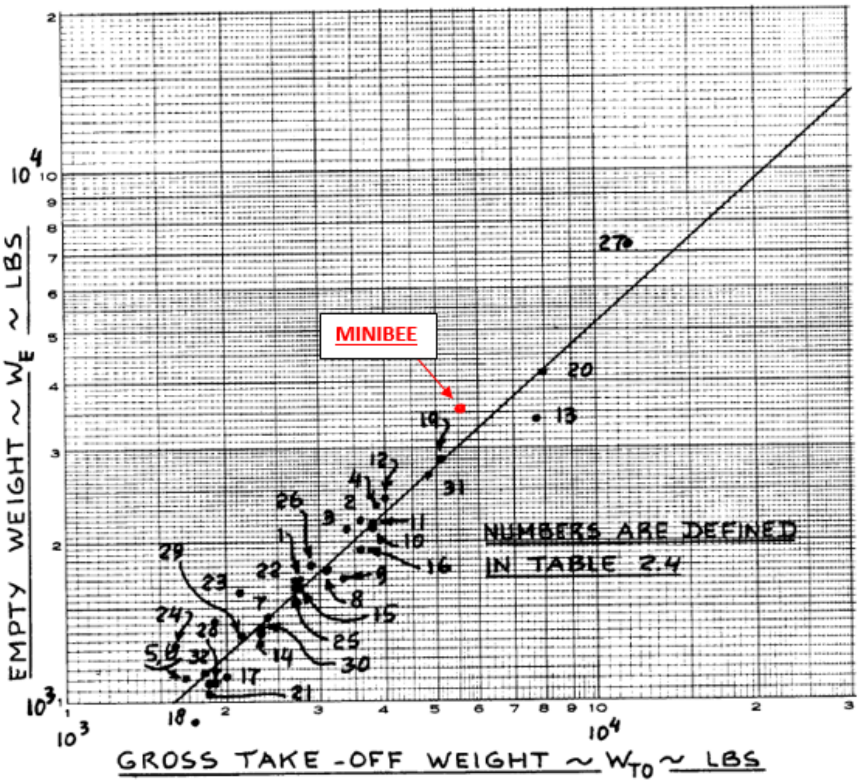


Figure 45 - Single Engine Propeller Weight Tendency

Table 2.4 Weight Data for Single Engine Propeller Driven Airplanes

No.	Type	Gross Take-off Weight, W_{TO} (lbs)	Empty Weight, W_E (lbs)	Maximum Landing Weight, W_{Land} (lbs)	Max. Internal Fuel Weight, W_{MIF} (lbs)
	BEECH				
1	Sierra 200	2,750	1,694	2,750	335
2	Bonanza A36	3,600	2,195	3,600	434
3	Bonanza V35B	3,400	2,106	3,400	434
4	Turbo Bonanza	3,850	2,338	3,850	599
5	Skipper 77	1,675	1,100	1,675	170
	CESSNA				
6	152	1,670	1,112	1,670	229
7	Skyhawk II	2,400	1,427	2,400	252
8	Skylane RG	3,100	1,757	3,100	517
9	Skywagon 185	3,350	1,700	3,350	517
10	Stationair 8	3,800	2,123	3,800	358
11	Centurion II	3,800	2,153	3,800	511
12	Centurion Press.	4,000	2,426	3,800	511
13	Caravan 208 (TBP)	7,750	3,385	7,000	2,194
	PIPER				
14	Warrior II	2,325	1,348	2,325	282
15	Arrow IV	2,750	1,637	2,750	452
16	Saratoga	3,600	1,935	3,600	628
17	Tripacer PA22	2,000	1,110	2,000	211
18	Super Cub PA18-150	1,750	930	1,750	211
	DeHAVILLAND				
19	DHC-2 Beaver (land)	5,100	2,850	5,100	556
20	DHC-3 Otter (land)	8,000	4,168	8,000	1,286
	SOCATA				
21	Rallye 125	1,852	1,125	1,852	149
22	Diplomate ST-10	2,690	1,594	2,690	310

Figure 46 - Weight Data from Roskam Part I

Table 2.4 (Cont'd) Weight Data for Single Engine Propeller Driven Airplanes

No.	Type	Gross Take-off Weight, W_{TO} (lbs)	Empty Weight, W_E (lbs)	Maximum Landing Weight, W_{Land} (lbs)	Max.* Internal Fuel Weight, W_{MIF} (lbs)
	ZLIN				
23	142	2,138	1,609	2,138	194
24	Z50L	1,587	1,256	1,587	93
	MOONEY				
25	201(M20J)	2,740	1,640	2,740	376
26	231 Turbo(M20K)	2,900	1,800	2,900	462
27	Antonov AN-2	11,574	7,275	N.A.	1,984
28	Beagle B.121-2 Pup	1,900	1,090	1,900	169
29	Partenavia P66C	2,183	1,322	2,183	251
30	Fuji FA-200	2,335	1,366	2,335	317
31	Pilatus PC-6(TBP)	4,850	2,685	4,850	832
32	Varga 2150A Kachina	1,817	1,125	1,817	205

Figure 47 - Weight Data from Roskam Part I

5.5.4. Airplane components weight

It is possible to express the weights of major airplane components as a simple fraction of gross take off weight, Wto: it will be sufficient to use average fraction values obtained from a number of airplanes with mission not too much different from the mission of Minibee.
It is very important to observe whether or not:

- An airplane is pressurized
- The landing gear is mounted on the fuselage or on the wing
- The engine are mounted on the wing or on the fuselage
- The primary structure is made of aluminum or of composites materials

Considering that the Minibee is a not-pressurized airplane with landing gear mounted on the fuselage, engine mounted both on the wing and on the fuselage, structure made of composites materials, it is not easy to determinate a class of existing airplanes to fit in; so this estimation needs to be only a starting point of recursive process (see figure 48) that leads to determinate Center of Gravity position and finally stability and control parameters.

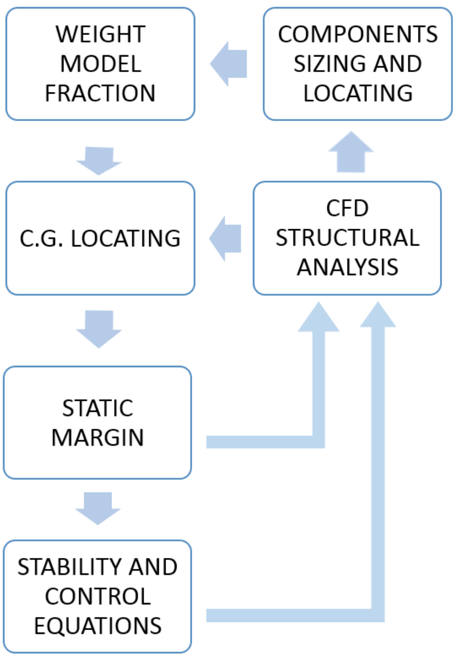


Figure 48 - Iterative process

5.5.5. Selection of comparable airplanes

Specifically, the following airplanes seem to be appropriate for the comparison:

- The **Cessna** 210J, that is a six-seat, high-performance, retractable-gear, single-engine, high-wing general aviation aircraft.

Figure 49 - Cessna 210J

- The Cessna 310, that is an American six-seat, low-wing, twin-engined monoplane.

Figure 50 - Cessna 310

- The Beechcraft 95 Travel Air was a twin-engine development of the Beechcraft Bonanza.

Figure 51 - Beechcraft 95

- The Cessna 182 Skylane is an American four-seat, single-engined light airplane.

Figure 52 - Cessna 182

5.5.6. Weight fractions

TYPE	CESSNA 182	CESSNA 210J	BEECH 95	CESSNA 310C
MTOW [KG]	1202	1542	1814	2190
PAYLOAD [KG]	324,32	210,47	332,5	537,96
FUEL [KG]	176,9	314,31	305	277,6
EMPTY WEIGHT [KG]	700,78	891	1176,5	1374,44
FUSELAGE [KG]	181,44	180,07	125,2	144,7
WING [KG]	106,6	151,95	207,75	205,48
TAIL [KG]	28,12	39	36,86	53,52
POWER PLANT [KG]	247,21	263,54	396	566,99
LAND GEAR [KG]	59,87	86,64	98,88	119,3
FIX EQUIP [KG]	77,54	169,8	437,01	284,45

Table 18 - Weight Data

TYPE	CESSNA 182	CESSNA 210J	BEECH 95	CESSNA 310C	AVERAGE
MTOW [KG]	1202	1542	1814	2190	
EMPTY WEIGHT/WTO	0,583	0,578	0,649	0,628	0,609
FUSELAGE/WTO	0,151	0,117	0,069	0,066	0,101
WING/WTO	0,089	0,099	0,115	0,094	0,099
TAIL/WTO	0,023	0,025	0,020	0,024	0,023
POWER PLANT/WTO	0,206	0,171	0,218	0,259	0,213
LAND GEAR/WTO	0,050	0,056	0,055	0,054	0,054
FIX EQUIP/WTO	0,065	0,110	0,241	0,130	0,136

Table 19 - Weight Fraction

5.5.7. Components weights

Overall it is important to note that the empty weight ratio which follow from the preliminary sizing is 1665/2500 = 0.653. This is not so close to the average value of 0.609 in the tabulation, but it is due to the hybrid nature of Minibee and to the presence of rotors and batteries: we can assume it as a reasonable adjustment, but is necessary to distribute this difference over all items, in proportion to their individual weight.

MTOW [KG]	AVERAGE	Weigh	Adjustment due to round-off errors		Adjustment due to hybrid nature	
EMPTY WEIGHT/WTO	0,609	1566,26	-43,14	1523,12	141,88	1665,00
FUSELAGE/WTO	0,101	251,76	-6,93	244,83	22,81	267,63
WING/WTO	0,099	247,24	-6,81	240,43	22,40	262,82
TAIL/WTO	0,023	58,40	-1,61	56,79	5,29	62,08
POWER PLANT/WTO	0,213	533,61	-14,70	518,91	48,34	567,25
LAND GEAR/WTO	0,054	134,36	-3,70	130,66	12,17	142,83
FIX EQUIP/WTO	0,136	340,89	-9,39	331,50	30,88	362,38

Table 20 - Minibee Weight Fraction

5.6. Center of gravity tolerance

In order to estimate the CG position and the CG excursion, it is required to estimate the empennages moment arms and dimension. This estimation need to be verified and eventually corrected after the equilibrium and stability studies: it means CG position is fundamental step of recursive process that links components weights and dynamic stability of Minibee.

5.6.1. Empennage sizing and disposition

At the beginning, relating to existing airplanes is the most accepted way to proceed. From the general arrangement drawing - figure 53, can be remarked:

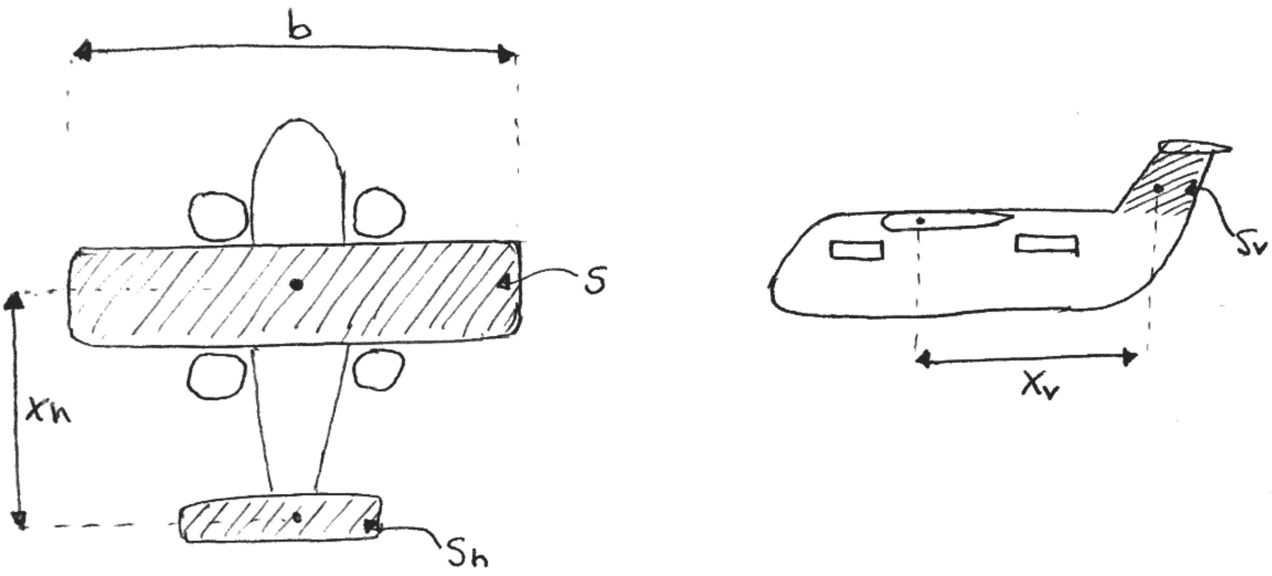


Figure 53 - General arrangement drawing

Where:

- Wing surface, S= 18m2
- Mean cord, c = 1.875m
- Wingspan, b= 10m
- Horizontal tail moment arm, Xh= 4.5m
- Vertical tail moment arm, Xv= 4.5m
- Horizontal tail surface, Sh
- Vertical Tail surface, Sv

From this data, the volume coefficient quantities can be defined as follow:

$$\bar{V}h = \frac{Xh * Sh}{S * \bar{c}}$$

Equation 13 - Horizontail tail volume coefficient

$$\bar{V}v = \frac{Xv * Sv}{S * b}$$

Equation 14 - Vertical tail volume coefficient

Looking at the table in the following page, accepted value for the volume coefficients are:

- Vh = 0.70
- Vv = 0.05

Type	Wing Area S ft^2	Wing Span b ft	Vert. Tail Area S_v ft^2	S_r/S_v	x_v ft	$\bar{V}_v$	Rudder Chord root/tip $fr.c_v$	S_a/S	Ail. Span Loc. in/out fr.b/2	Ail. Chord in/out $fr.c_w$
CESSNA Skywagon 207	174	35.8	16.0	0.44	18.0	0.046	.46/.46	0.10	.61/.94	.25/.22
Cardinal RG	174	35.5	17.4	0.37	13.5	0.038	.35/.43	0.11	.65/.97	.38/.37
Skylane RG	174	35.8	18.6	0.37	15.8	0.047	.41/.42	0.11	.47/.96	.17/.24
PIPER Cherokee Lance	175	32.8	13.8	0.31	15.3	0.037	.26/.50	0.064	.56/.88	0.20
Warrior	170	35.0	11.5	0.36	13.2	0.026	.29/.52	0.078	.48/.96	.27/.24
Turbo Saratoga SP	178	36.2	15.9	0.29	15.2	0.038	.23/.58	0.057	.52/.84	0.19
Bellanca Skyrocket	183	35.0	18.1	0.33	13.2	0.037	.28/.40	0.076	.60/1.0	.25/.22
Grumman Tiger	140	31.5	8.4	0.43	12.6	0.024	.36/.46	0.055	.56/.92	0.24
Rockwell Commander	152	32.8	17.0	0.28	11.4	0.039	.30/.46	0.072	.64/.97	.27/.36
Trago Mills SAH-1	120	30.7	17.1	0.40	18.6	0.086	.35/.54	0.080	.58/.97	.25/.29
Scottish Aviation Bullfinch	129	33.8	22.7	0.39	11.9	0.062	.35/.56	0.073	.61/.95	.23/.30

Figure 54 - Single Engine propeller data from Roskam

Type	Wing Area S ft^2	Wing mgc $\bar{c}$ ft	Wing Airfoil root/tip NACA*	Hor. Tail Area S_h ft^2	S_e/S_h	x_h ft	$\bar{V}_h$	Elevator Chord root/tip $fr.c_h$
CESSNA Skywagon 207	174	4.55	2412	44.9	0.45	16.2	0.92	.48/.47
Cardinal RG	174	4.79	64A215/64A412	35.0	1.00	14.3	0.60	stabilator
Skylane RG	174	4.52	2412	38.8	0.41	14.3	0.71	.47/.39
PIPER Cherokee Lance	175	5.25	65,415	34.6	1.00	16.1	0.61	stabilator
Warrior	170	4.44	65,415	26.5	1.00	13.5	0.48	stabilator
Turbo Saratoga SP	178	4.71	NA	36.2	1.00	16.2	0.70	stabilator
Bellanca Skyrocket	183	5.30	63,215	42.6	0.38	13.8	0.61	.36/.42
Grumman Tiger	140	4.44	NA	37.6	0.28	12.6	0.76	0.39
Rockwell Commander	152	4.58	63415	31.2	0.34	10.9	0.49	.33/.44
Trago Mills SAH-1	120	3.94	2413.6	22.0	0.46	17.8	0.83	0.46
Scottish Aviation Bullfinch	129	3.97	63,615	27.5	0.58	11.9	0.63	0.45

* Unless otherwise indicated.

Figure 55 - Single Engine propeller data from Roskam

Hence:

$$Sh = \frac{\bar{V}h * S * c}{Xh} = 5.25\ m^2$$

$$Sv = \frac{\bar{V}v * S * b}{Xv} = 2\ m^2$$

Last step for empennage sizing is to decide on the planform geometry. This involves the following choices:

- Sweep angle
- Dihedral angle
- Airfoil
- Incidence angle
- Aspect ratio

Referring to Roskam's books, typical accepted value are:

	Horizontal tail	Vertical Tail
Sweep angle	7°	30°
Dihedral Angle	0°	90°
Airlfoil	NACA 0009	NACA 0018
Incidence Angle	variable	0°
Aspect ratio	5,2	1,7

5.6.2. Component weight breakdown and centers of gravity

The center of gravity of Minibee can be found from the CGs of all components and our purpose is to determine if it is correctly positioned for different loading scenarios. The most important airplane components can be identified as:

- Passengers + baggage
- Crew
- Fuel
- Wing group
- Fuselage group
- Empennage group
- Power plant group
- Landing gear group
- Fixed equipment group

And for each component the CG has to be individuate from simple geometrical and symmetrical consideration, as can be seen in figure 56 and 57.

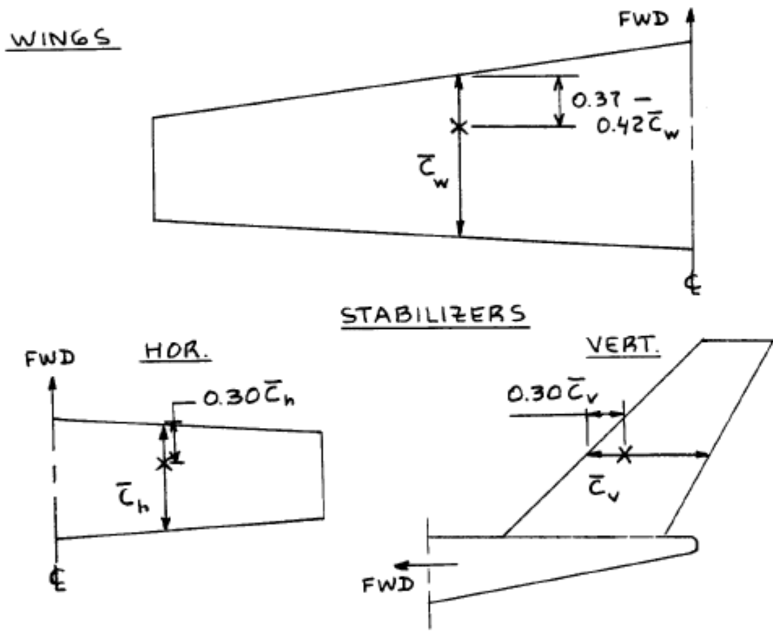


Figure 56 - CG component locating

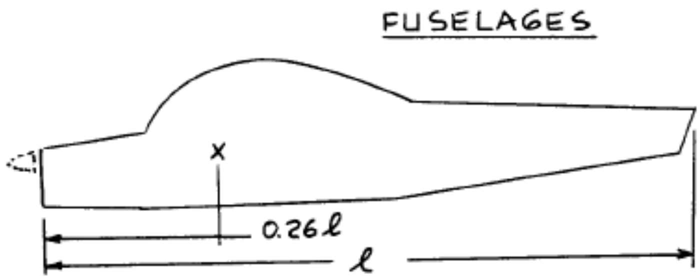


Figure 57 - CG component locating

The figure 58 shows a preliminary drawing: the sideview is sufficient because the Minibee is symmetrical.

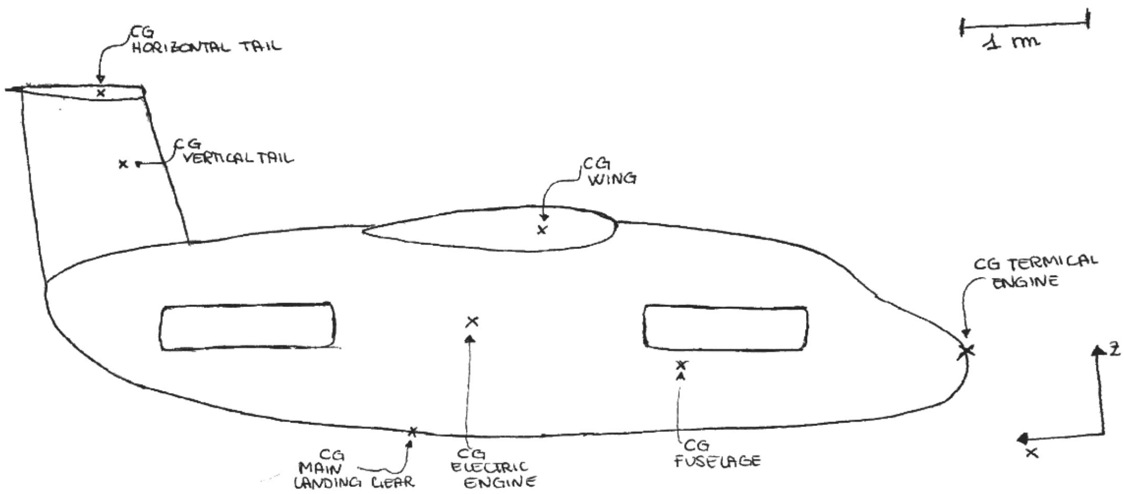


Figure 58 - Preliminary CG locating

With the appropriate coordinates of each components, referring to the zero neutral point, airplane CG can be found from the equation:

$$X_{CG} = \sum_i \frac{Xi * Wi}{Wto}$$

Equation 15 - CG definition

Hence:

COMPONENT	WEIGHT	Xi	WEIGHT x Xi
	kg	m	kg*m
Wing	262,82	4,96	1303,5872
Vertical tail	18,2	8,4	152,88
Hor tail	43,8	8,68	380,184
Fuselage	267,63	3,17	848,3871
Main Landing gear	142,83	6,3	899,829
Electric Engine	243,6	5,14	1252,104
Termical Engine	177	1,335	236,295
*Fixed equip **	*510*	*7*	*3570*
*Fuel **	*260*	*4,96*	*1289,6*
Crew	115	3,05	350,75
Passengers	460	5,15	2369

Table 21 - Components Centres of Gravity

* The position of fixed equipments and fuel can be used to define the CG position according to stability and control requirements

$$X_{CG} = 5.06\ m$$

These CG locations must be calculated for all loading scenarios; they depend in particular on the mission of the airplane and typically can be summarized as:

- Empty Weight
- Empty Weight + crew
- Empty Weight + crew + fuel
- Empty Weight + crew + fuel + payload = take-off weight

Therefore it is possible to construct a weight-CG excursion diagram (figure 59) identifying the loading sequences as well as the critical weights.

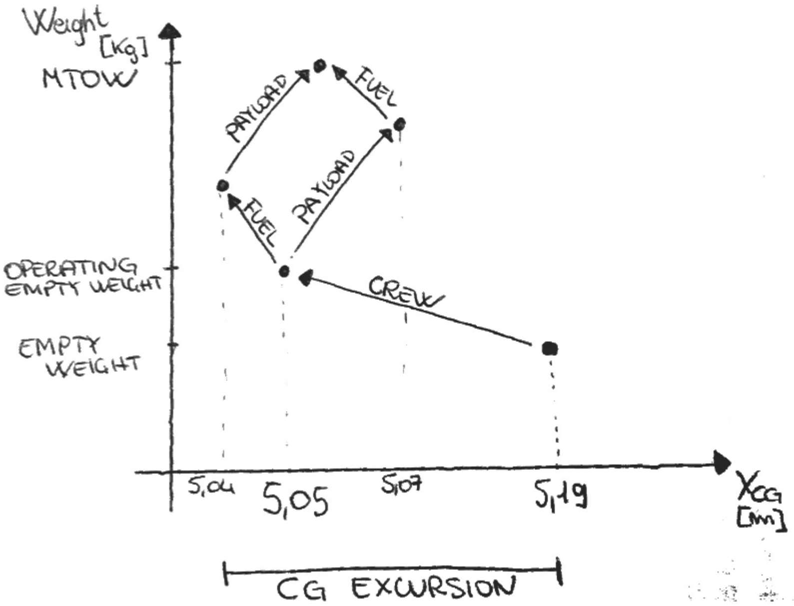


Figure 59 - CG Excursion

5.7. Stability and control analysis

Stability analysis is one of the most crucial part in aircraft design, however there are not all the necessary elements to complete it:

- aerodynamic study of the lift surfaces, in order to obtain wing (tail) lift distribution on wingspan to determinate both vertical forces and momentum.
- sizing and locating of flaps, elevator, spoilers, ailerons
- structural analysis

5.7.1. Longitudinal static stability.

Although, it is necessary to understand if the preliminary estimation of center of gravity and horizontal (vertical) tail leads to a stable static configuration or not; in detail the longitudinal static stability, that refers to the aircraft's stability in the pitching plane, is significantly influenced by the distance (moment arm or lever arm) between the center of gravity and the aerodynamic center, as in figure 60.

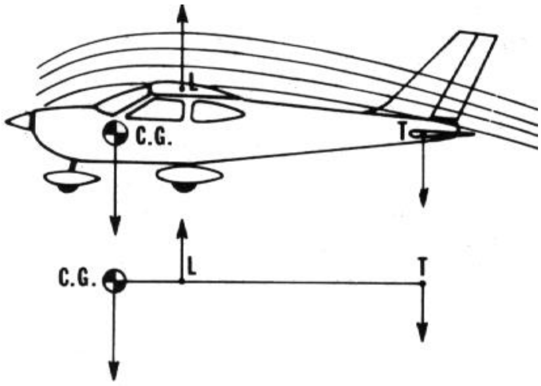


Figure 60 - Forces arm in Longitudinal Stability

If an aircraft is longitudinally stable, a small increase in angle of attack will cause the pitching moment on the aircraft to change so that the angle of attack decreases. Similarly, a small decrease in angle of attack will cause the pitching moment to change so that the angle of attack increases.

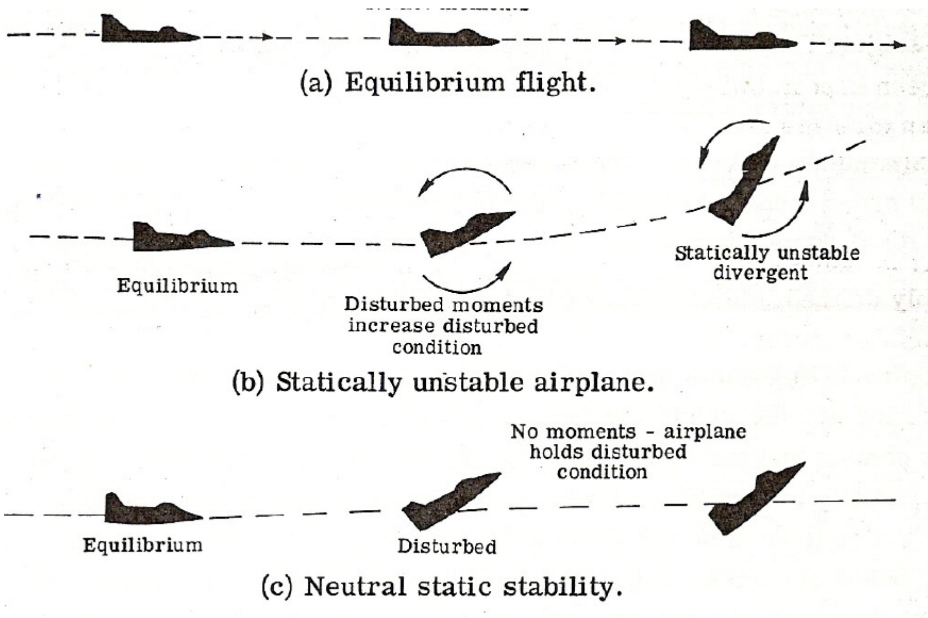


Figure 61 - Statically stable and unstable airplane

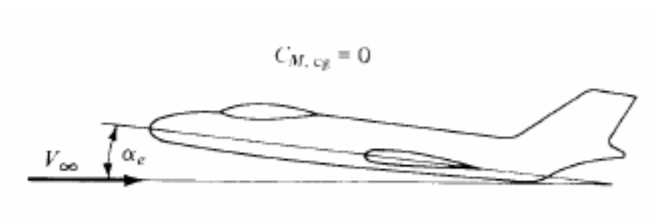


Figure 62 - Static Equilibrium at Cm = 0

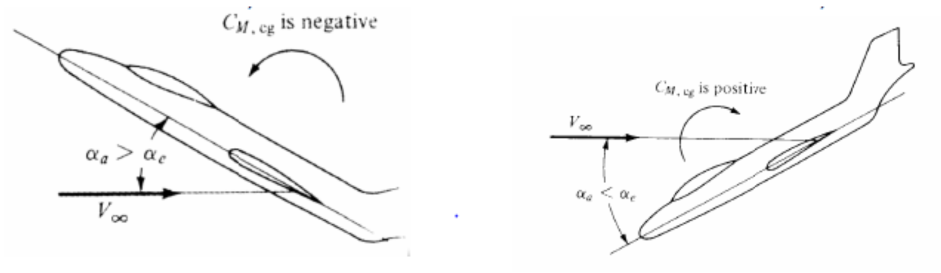


Figure 63 – Static Equilibrium at Cm < 0 nad Cm > 0

That means that the "stability derivative", defined as the partial derivative of pitching moment respect to changes in angle of attack, has to be negative according to conventional body axis (see figure 64).

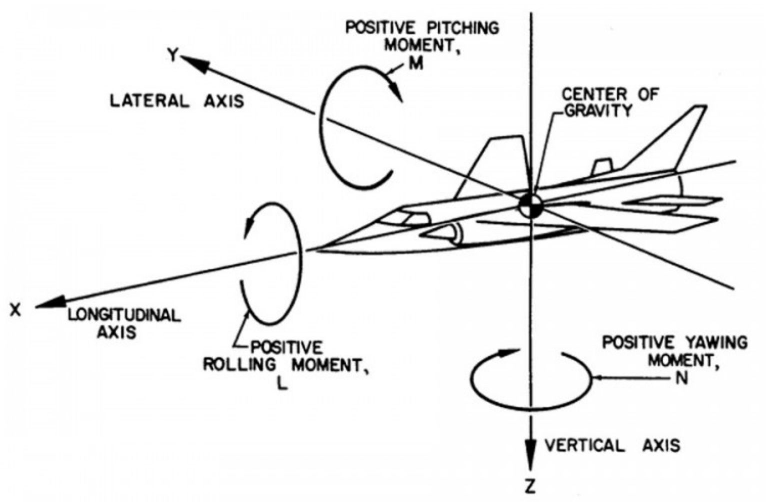


Figure 64 - Body Axes

Regarding Minibee static stability, it is important to underline that the aerodynamic center typically does not change with loading or other changes, but the c.g. does:

- If the c.g. moves forward, the airplane becomes more stable (greater moment arm between the a.c. and the c.g.), and if too far forward will cause the airplane to be difficult for the pilot to bring nose-up as for landing.
- If the c.g. is too far aft, the moment arm between it and the a.c. diminishes, reducing the inherent stability of the airplane.

The fix equipment of Minibee allows to move the c.g. aft and forward, in particular the batteries position, and this is a significant advantage because permits to choose between Conventional static Stability and Relaxed Static Stability (RSS), see figure 65.

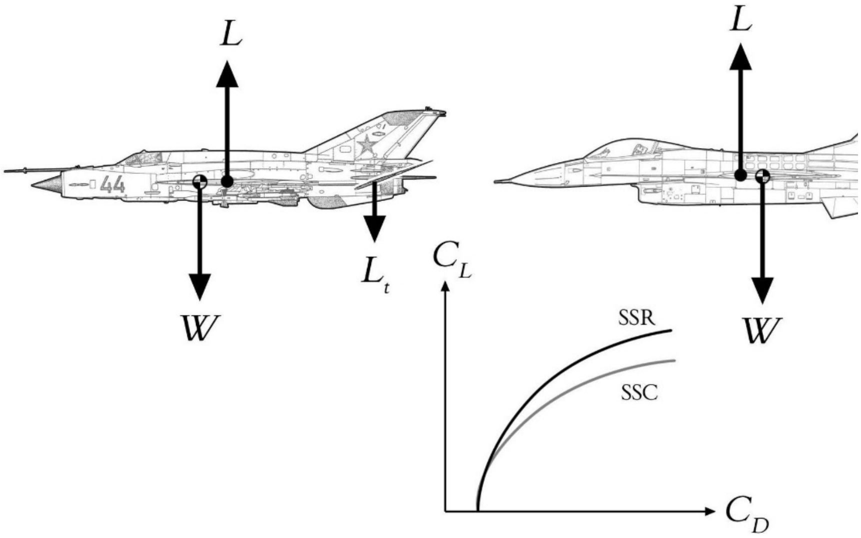


Figure 65 - Conventional static stability and Relaxed Static Stability

- In Conventional Static Stability the center of gravity is placed before the point of application of the lift while the horizontal tail plane generates a downforce (negligible in modulus respect to lift and to weight force , but with a remarkable arm) that balances the pitching moment of the wing
- In relaxed Static Stability the aerodynamic center is located before the center of gravity: the aircraft will tend to raise the nose and tail will produce lift to keep it in level flight.

The choice is focused on RSS because produces efficiencies in cruise from a more rearward c.g. and a physically lighter tail structure; in fact conventional downward lifting tails add weight to the aircraft by virtue of the “down-lift” they generate (and also drag, the by-product of that lift) so the main wing has to produce additional lift in compensation, and consequently produces more drag itself.
The main disadvantage is that RSS is “less” stable than conventional configuration: this means that MiniBee needs flight computers and autopilot to fly safely.

5.7.2. Forces Equation

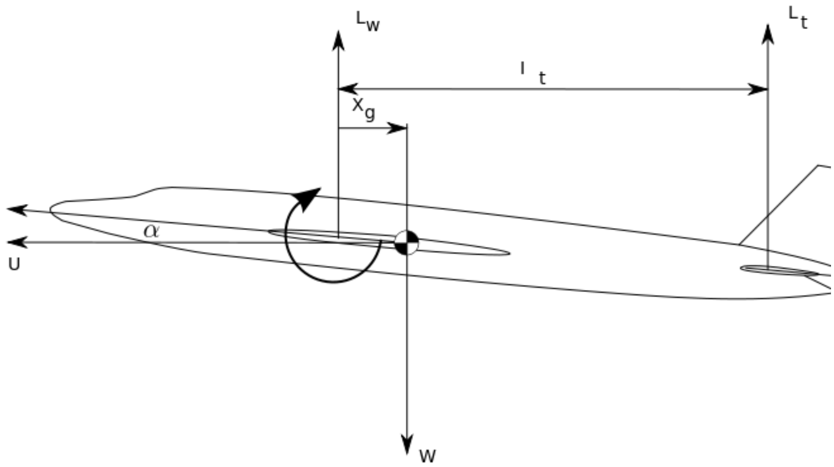


Figure 66 - Cruise Equilibrium

Near the cruise condition – figure 66, most of the lift force is generated by the wings, with ideally only a small amount generated by the fuselage and tail. The longitudinal static stability can be analyzed considering the aircraft in equilibrium under wing lift, tail force, and weight.

In vertical direction:

$$W = Lw + Lt$$

Where W is the weight, Lw is the wing lift and Lt is the tail force.
For Minibee airfoil at low angle of attack, the wing lift is proportional to the angle of attack according to:

$$Lw = \frac{1}{2}\rho v^2 S \frac{\partial Cl}{\partial \alpha}(\alpha - \alpha 0)$$

Equation 16 - Lift Definition

Where ρ is the air density, v is the cruise speed, S is the wing surface, Cl is the wing lift coefficient, α is the angle of attack and α0 is included to account cambered airfoil.
The same definition can be applied to the tail force but we have to consider that air doesn't arrive at equal angle of attack, because of downwash angle and elevator deflection (data not available during the preliminary phase of the project).

However, from the moment equation about the center of gravity:

$$M = Lw * xg - (lt - xg) * Lt$$

It could be possible to estimate Xg position, observing that in equilibrium condition, the pitching moment M has to be M=0.
As told before, there aren't enough specific data, especially aerodynamics data to find exact position of Xg. But the nature of stability can be examined considering the increment in pitching moment with change in α at trim condition (i.e. M=0).

5.7.3. Static Margin

Differentiating the moment equation with α:

$$\frac{\partial M}{\partial \alpha} = xg * \frac{\partial Lw}{\partial \alpha} - (lt - xg) * \frac{\partial Lt}{\partial \alpha}$$

Equation 17 - Stability derivative

It can be convenient in order to simplify the equation to treat total lift as acting at distance *h* from the center of gravity:

$$M = h * (Lw + Lt)$$

$$\frac{\partial M}{\partial \alpha} = h * (\frac{\partial Lw}{\partial \alpha} + \frac{\partial Lt}{\partial \alpha})$$

Hence:

$$h = xg - lt * \frac{\frac{\partial Lt}{\partial \alpha}}{\frac{\partial Lw}{\partial \alpha} + \frac{\partial Lt}{\partial \alpha}}$$

Equation 18 - Static Margin Definition

Considering typical value for aerodynamic coefficients, the expression for h can be written more compactly and approximately, according to Piercy, as:

$$h = xg - 0.5 * c * Vh$$

Equation 19 - Static Margin by Piercy

where Vh is the horizontal tail volume coefficient and c is the mean aerodynamic chord.

If the stability derivative $\frac{\partial M}{\partial \alpha}$ has to be negative, h has to be negative, which means:

$$h < 0 \leftrightarrow xg < 0.5 * c * Vh$$

Considering Minibee values for c and Vh:

c	1.875 m
Vh	0.70

$$xg < 0.66\ m$$

If the center of gravity position respects this margin, then the stability derivative is negative and the aircraft can be considered longitudinally stable.

This margin must be related to the moment equation and forces equations, as well as to the position of each component in the Minibee's configuration: this iterative process leads to the best mass distribution along the aircraft but it needs more accurate aerodynamic and structural analysis.

5.8. Cost estimation

In a market with high demands on performance and affordability, one cannot simply look at the purchase price of the aircraft to validate investments: the producer has to take all cost that could occur during the life of an aircraft into account within the early design phases. This approach is called Life Cycle Cost Analysis. According to Roskam, the analysis of a typical system includes costs for:

1. Planning and conceptual design
2. Preliminary design and system integration
3. Detail design and development
4. Manufacturing and acquisition
5. Operation and support
6. Disposal

It is important to note that most of the cost impacts are defined in the earliest design phases, whereas the bigger part of the outgoings are during the operation and support phase, as can be seen in figure 67.

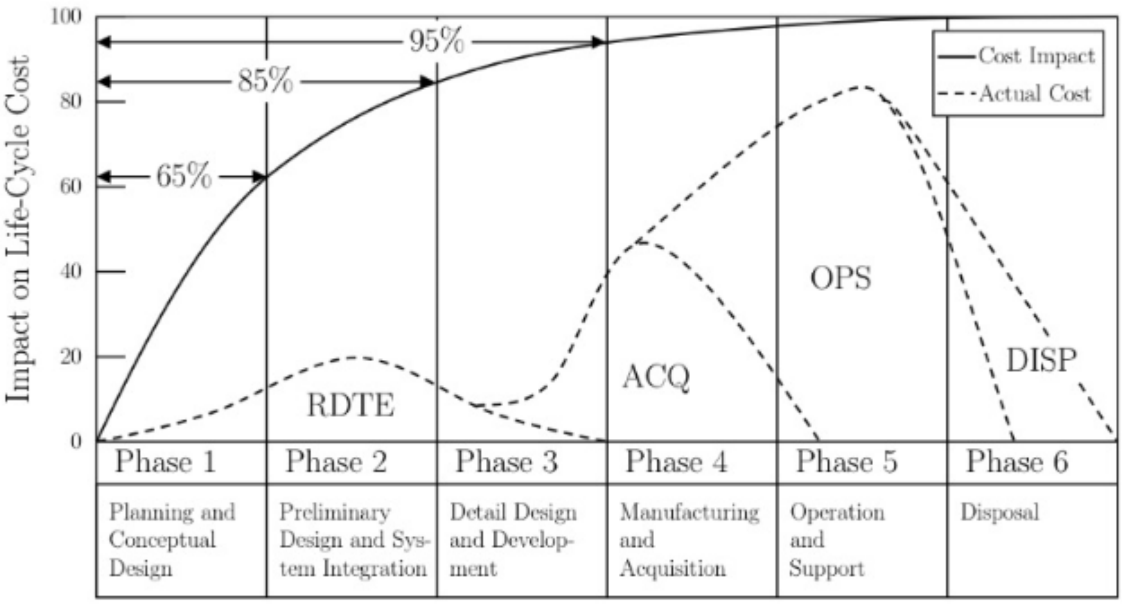


Figure 67 - Life Cycle Cost

For this reason, different design guidelines exist that integrate cost as an inherent element but The choice for Minibee is to follow a Cost Model proposed by J. Markish as it is an intuitive method suitable for the preliminary project phase.

5.8.1. Development cost

There are two primary categories of cost in aircraft programs: development and manifacturing cost. Development cost is the non-recurring effort required an aircraft concept to production. It includes preliminary design, detail design, tooling, testing and certification. The development cost model is made of several basic steps:

- The cost is divided into several processes for each aircraft part: each process is characterized in term of relative time and cost based on non-dimensional industry data for typical commercial aircraft.
  The graph in figure 68 shows a plot of person-hours used for the all period between program launch and first product.

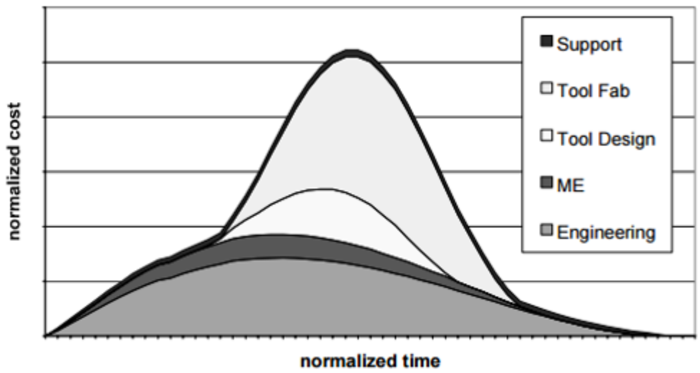


Figure 68 - Person-Hours Used vs normalized time

The hours are divided not by aircraft parts, but by separate processes: engineering; manufacturing engineering; tool design; tool fabrication; and support.
Note that the time profile for the entire process, as well as the individual process has the shape of a Bell curve—that is, a normal distribution.

- An estimate is made of total weight and non-recurring cost for a similar aircraft: the Cessna 182 Skylane.

To build a complete development cost model, it is necessary to assume that cost scales directly with operating empty weight.

- An estimate is made of total duration of the effort and of the fractional weight and cost breakdown, by parts for typical commercial aircraft.

According to data provided by industry sources, the cost breakdown can be assumed as in the figure:

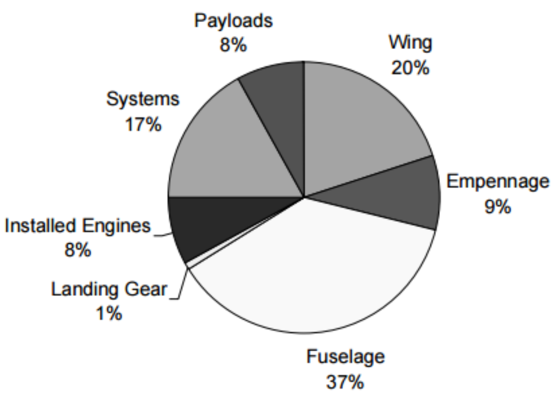


Figure 69 - Cost Breakdown

- Fractional cost and weight breakdowns for typical aircraft are combined with cost and weight estimates for the 182 Skylane to yield cost per kg estimates for each part.
  The resulting data is shown in Table 22 :

Cessna cost [€]	Cost breakdown	Weight breakdown	**€/kg**	Engineering [€/kg]	Manufactoring [€/kg]	tool design [€/kg]	tool fabrication [€/kg]	support [€/kg]
400000		Cessna 182		40%	10%	10,50 %	34,80%	4,70%
Wing	80000	106,6	***750,47***	300,19	75,05	78,80	261,16	35,27
Empennage	36000	28,12	***1280,23***	512,09	128,02	134,42	445,52	60,17
Fusolage	148000	181,44	***815,70***	326,28	81,57	85,65	283,86	38,34
Landing Gear	4000	59,87	***66,81***	26,72	6,68	7,02	23,25	3,14
Installed Engines	32000	247,21	***129,44***	51,78	12,94	13,59	45,05	6,08
System	68000	77,54	***876,97***	350,79	87,70	92,08	305,18	41,22
Payloads	32000	324,32	***98,67***	39,47	9,87	10,36	34,34	4,64

Table 22 - Cessna 182 Cost Breakdown

One complication requires a modification of the model: the effect of commonality. In fact considering an aircraft with high degree of commonality with aircrafts already in production (for example, it may use the same wing, or the same fuselage section, or even both), In such a case, the non-recurring cost for those components that already exist should be significantly lowered from the all-new design.
According to a set of assumptions based on reasonable first-order estimates the cost reduction factors can be estimated as in the table below:

	Engineering	Manufactoring	tool design	tool fabrication	support
Wing	20%	50%	10%	10%	50%
Empennage	20%	50%	10%	10%	50%
Fusolage	20%	50%	10%	10%	50%
Landing Gear	20%	50%	10%	10%	50%
Installed Engines	20%	50%	10%	10%	50%
System	20%	50%	10%	10%	50%
Payloads	20%	50%	10%	10%	50%

Table 23 - Cost Reduction Factors

Then using these data, Minibee development cost can be estimated:

	Cost [€]
Wing	158027,31
Empennage	63594,54
Fuselage	174905,88
Landing Gear	7645,59
Installed Engines	43620,85
System	358339,13
Payloads	9091,07
Total cost	***815224,37***

Table 24 - Minibee Cost

5.8.2 Manufacturing cost

Like the development cost model, the manufacturing cost model is centered around the assumption of relationship between cost and weight.
However the nature of manufacturing cost is recurring: the cost is incurred repeatedly for each unit.
This phenomenon is characterized in aircraft production by a significant decrease in unit cost when additional aircraft are built- see figure 70. The decrease is most noticeable early in the production, and eventually decays to a negligible level, when unit cost remains roughly constant.

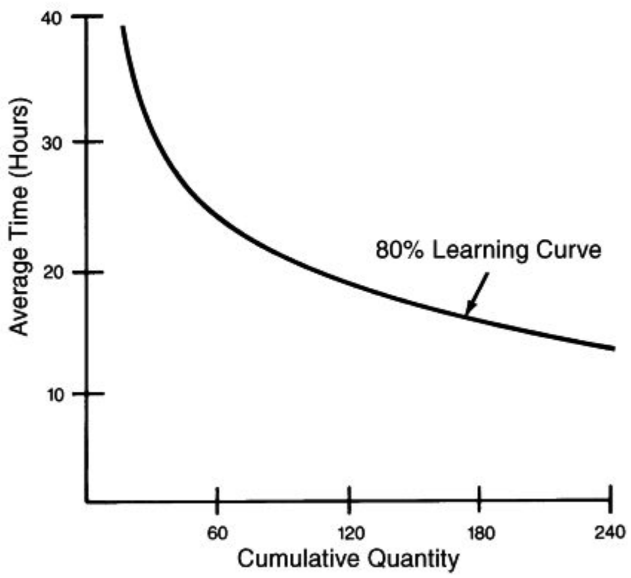


Figure 70 - Learning Curve

However, given the complexity of its nature, the manufactoring cost requires a more accurate model, that considers the geometric of the parts and the material used, as well as non-linear cost estimating relationships.

For these reasons, the necessary information is not already available: the manufacturing cost represents, therefore, the next step in Minibee's design process.

5.9. Batteries estimation

The batteries pack must be able to sustain the quadcopter system at full power for 10 minutes, to ensure a safe time window for vertical takeoff.
The previous analysis shows that quadcopter system will need about 800KW to work at full throttle: note that it is the same power demand of helicopters of the same class and same MTOW of the MiniBee (as the AW119 Koala that requires a 747KW power system [14]).
Assuming a simple general power definition as below, the total energy that the batteries pack must store is estimated:

$$P = \frac{E}{t}$$

Equation 20 - Estimation of power

P	Power	800KW	Power estimation for the quadcopter system
E	Energy	To be defined	Energy storage capacity estimated for the batteries pack
t	Time	10min = 600s	Total operating time of the system

Table 25 - Synthesis of equation 6

The result shows that the batteries pack must be able to store at least 480MJ of energy to support the quadcopter system for 10 minutes. In raw power, that translates in a 133,33KW/h batteries pack. Below in the figure 22 is presented a chart with different kinds of batteries cells.

	Cell Maker	Chemistry	Capacity	Configuration	Voltage	Weight	Volume	Ener dens	Spec Ener	Used in:	
		Anode/Cathode	Ah		V	Kg	liter	Wh/liter	Wh/kg	Company	Model
1	AESC	G/LMO-NCA	33	Pouch	3.75	0.80	0.40	309	155	Nissan	Leaf
2	LG Chem	G/NMC-LMO	36	Pouch	3.75	0.86	0.49	275	157	Renault	Zoe
3	Li-Tec	G/NMC	52	Pouch	3.65	1.25	0.60	316	152	Daimler	Smart
4	Li Energy Japan	G/LMO-NMC	50	Prismatic	3.7	1.70	0.85	218	109	Mitsubishi	i-MiEV
5	Samsung	G/NMC-LMO	64	Prismatic	3.7	1.80	0.97	243	132	Fiat	500
6	Lishen Tianjin	G-LFP	16	Prismatic	3.25	0.45	0.23	226	116	Coda	EV
7	Toshiba	LTO-NMC	20	Prismatic	2.3	0.52	0.23	200	89	Honda	Fit
8	Panasonic	G/NCA	3.1	Cylindrical	3.6	0.048	0.018	630	233	Tesla	Model S

Figure 71 - Comparison between different kinds of batteries cells

As MiniBee is an aeronautical project, low weight and high energy density are the variables prioritized to choose a proper battery cell. Utilizing Panasonic G/NCA batteries cells, the same one that the American automaker Tesla uses, it is possible to create a battery pack with about 570kg and a volume of 211L, which can support the quadcopter system of the MiniBee.

5.10. Modeling of the fuselage

Fuselage is a very important part during the design of an aircraft because it should housing both payload and systems and it is also the structural element who takes together all the aircraft's parts.

There are many constraints to be taken in account on the design of the fuselage, but it is important to remember that many constraints can lead to a poor job.

The right way to approach this task is to find the right balance between the demands, constraints and the reality. The first thing to do, is to keep in mind the purpose for which the aircraft is beeing projected, so we can better understand which are the most important criterias. To cope with this problem it was decided to introduce matrix of choice (Table 26). In the first column there are constrains, which concern the fuselage, and are presented, in the second column, instead, there's a number which represents the quantity of possibles solutions (1: only a possible solution, 3: large number of possibles solutions).

Pressurization	1
Pilot position and visibility	2
Payload	3
Aerodynamic	3
Access door	2
Engine position	1
Boarding system	2
Wings and electric propeller position	1
Structure	2
Vertical and horizontal tail	2
Low cost	1

Table 26 - Choice Matrix of the fuselage

5.10.1. Pressurization

In the literature, there are three possible pressurisation levels:

- Low differential pressure: Usually used in combat aircraft in which pilots are used to use the oxygen masks or in vehicles traveling at an altitude of more than 2km, to provide greater comfort to passengers. In this configuration, the pressure difference is lower than 0.27 bar.

- Normal (high) differential pressure: it is usually used in commercial flights that are used to travel at very high altitudes. The difference in pressure in these cases is ca. 0:58 bar. In this configuration, as in the previous, for structural reasons it would be desirable to avoid the flat surfaces. The aircraft that operate in these categories tend to have the cross section (perpendicular to the axis x of the aircraft) circular in shape.

- No pressurisation: for aircraft that operate below 2.44km it is not necessary to pressurize the cabin. This characteristic leads to a substantial change of the cross section (perpendicular to the x axis of the aircraft). In these circumstances it is no longer necessary to avoid flat surfaces, for this reason rectangular shape sections are preferred as much as possible to facilitate the internal arrangement in the aircraft.

For an aircraft such as that defined by this project cabin pressurisation is not necessary, because it is not the task of this aircraft flying at an altitude above 2.44 km. Moreover the pressurization system could lead to a great complication in the propulsive system because the engine should also provide the necessary power to the compressor required for pressurization.
Thanks to this consideration it is not needful to avoid flat surface for this reason it is a good idea to choose a rectangular in shape cross section to facilitate the internal payload and systems packaging.

5.10.2. Pilot Vision and visibility

Another very important factor that determines the layout of the fuselage is the position of the pilot. There are laws that govern the visibility that the pilot must be able to ensure. Obviously this affects the shape of the nose of the fuselage, and in our case also the location of the electrics propellers that do not obstruct the view of the pilot under any circumstances. It is also very important to conduct a study on the correct posture of the pilot and the space he needs.
We can take as an example a dummy used as a reference of the average man used in AvP970, figure 72:

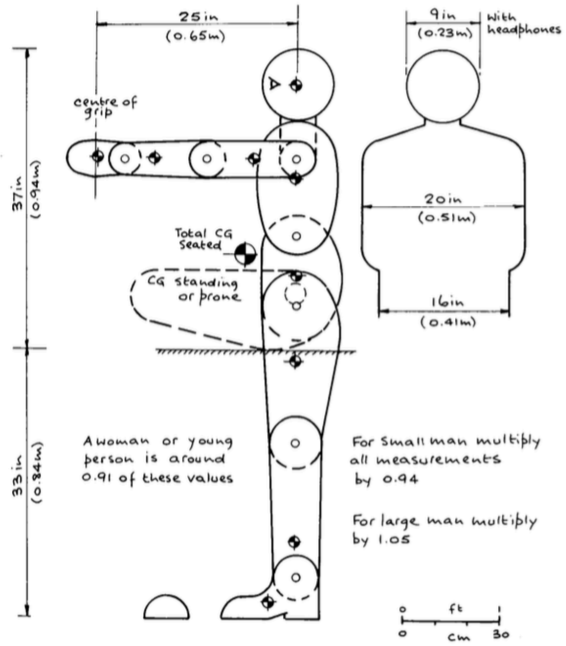


Figure 72 - Reference Average man

The pilot must reach easily the command and must have a comfortable position. The figure below show some reference dimensions for the pilot seat dimensioning.

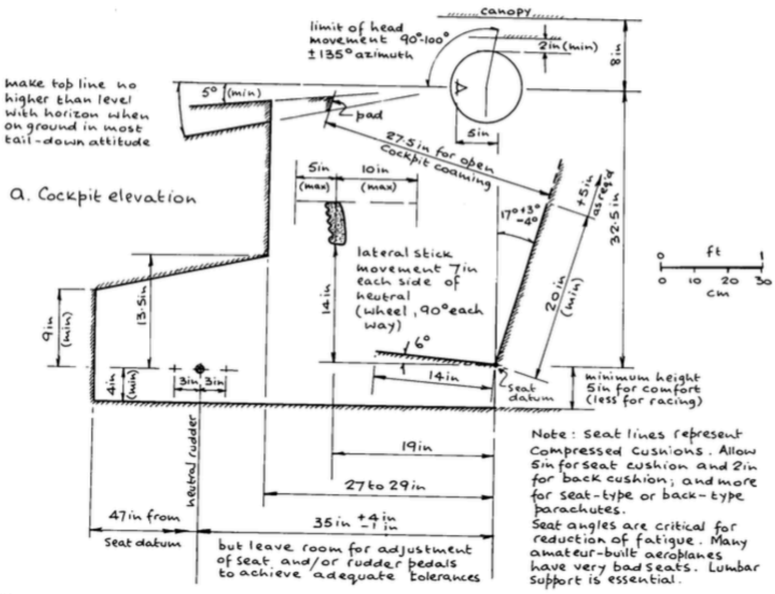


Figure 73 - Pilot's field of vision

5.10.3. Payload

In this category are considered both passengers and their luggages. This requirement is not very strict, there are multiple configurations to provide passengers and luggage.

This vehicle is also designed to be used for the transport of patients from one hospital to another, in fact, in this eventuality would be ideal a means capable of taking off from small surfaces and with a cruising speed sufficiently high. The interior design should therefore be very modifiable and contemplate the possibility of different arrangements.

For the well-being of the passengers it would be good thing that their seats are facing in the direction of acceleration of the aircraft to reduce the impact of changes in speed.

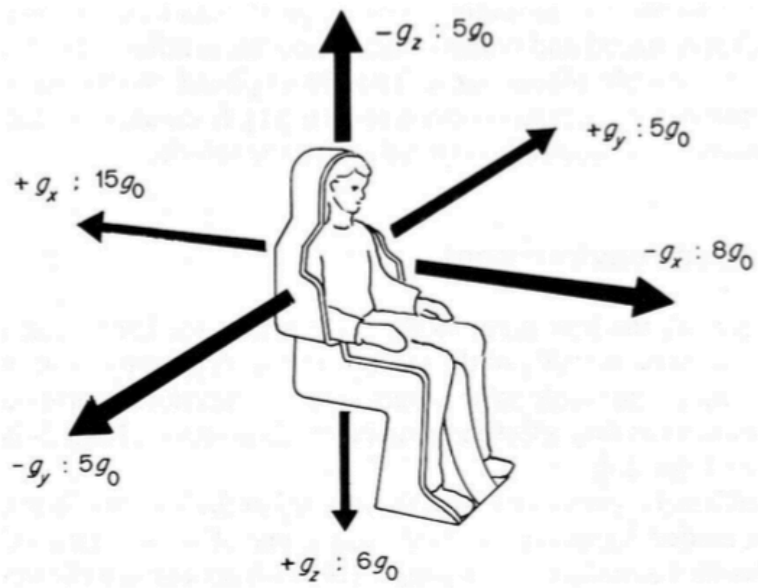


Figure 74 - Human maximum acelerations

It is necessary to ensure the space required for a comfortable journey to each passenger, in the configuration of passenger transport, both in the case this aircraft is used as flight ambulance.

5.10.4. Aerodynamics

The aerodynamics factor affects the outer shape of the aircraft. It would be appropriate to have an outer surface with as low as possible drag coefficient because a low drag would lead to a reduction of fuel consumption.

The aerodynamics of a vehicle affects mainly on the longitudinal section (perpendicular to the y axis). The best thing to do would be to avoid abrupt changes in section and make a profile as clean as possible. This is not always easy, it is also not always possible to predict the behavior of the flow around the aircraft. For this reason for the aerodynamic study, after a first approximation of the profile, it is appropriate to use aerodynamic analysis performed with the help of the computer aerodynamic analysis cfd.

Even if the influence on the cross section shape is smaller it should be noted that for aerodynamic reasons edges should be avoided also in this section. It is also very important, from the aerodynamic point of view, to reduce as much as possible the dimension of the cross section of the aircraft because this one is an important factor in the calculation of the drag. It is necessary to operate an arrangement between the cross section dimension and the necessary volume.

Although, this factor does not have the highest priority over all others, the aim of this project is in fact an aircraft able to carry out certain activities and not an aircraft with the best aerodynamics parameters.

5.10.5. Entrance door

The positioning of the entrance door is affected by factors such as the position of the wings and that one of electric rotors, but also by the aircraft purpose. Taking into account the possibility of using it for the transport of patients (stretcher), it is better to provide the aircraft of a back door. This decision is also supported by the fact that given the presence of electric rotors and the top wing, the alternative would be making an opening just below the wing, but from the structural point of view this is inadvisable because creating an opening in that position would mean an excessive strengthening of the area which would follow an increase in the weight.

This choice influences therefore the shape of the rear fuselage, which must be shaped to ensure the presence of the opening.

5.10.6. Engine Position

Regarding the position of the motor there is not much choice, as you can see from the literature and with simple avionic considerations (see chapter engines) the best configuration in case you have a single heat engine is to put it in the front of the aircraft. Motor dimensions obviously influence the shape of the fuselage in that part, which must already subject to the constraints imposed by the visibility of the pilot.

5.10.7. Boarding system

The task of the fuselage is also host board systems necessary for navigation and operation of the aircraft. The characteristics of the systems impose constraints position for some of them, the fuselage must therefore be shaped so as to be able to house them. Relevant in terms of weight and position in the airplane are the batteries. Having a high weight would be appropriate to place them as close as possible to the center of gravity of the airplane. But at this stage we do not know the exact location of the center of gravity, but we know that this will be for reasons of stability near the aerodynamic center and then close the wings. Also for reasons of packaging would be a good idea to place the batteries under the passenger cabin.

5.10.8. Wings and electrical motors position

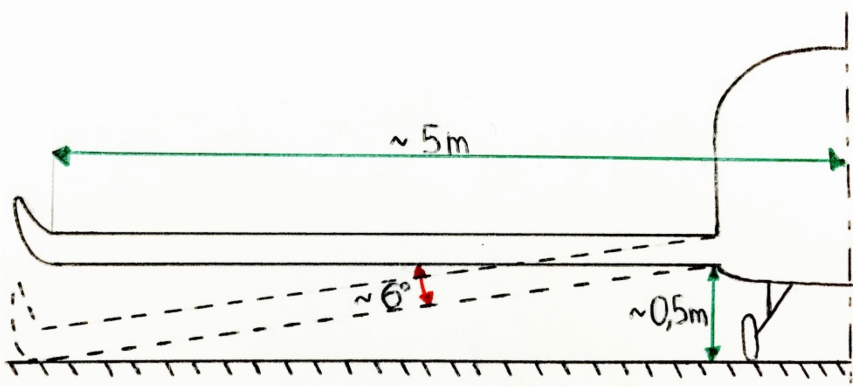
The positioning of the wing interferes much, as already seen in the case of the placement of the entrance, with the design of the fuselage.

There are three different configurations of wing:

- midi-wing: this one is the best from the point of view of reduction in the drag, but it is very detrimental to the structural level. The best thing would be to have a wing continues to not have excessive burdening of the fuselage at the point of attack of the wings, if you choose a midi-wing and you do not want the increase in weight you should ensure continuity of the wing, leaving it to go through the fuselage. This can be done in the case of military

vehicles in which the wing crosses the weapons bay that does not require continuity, can not be done, however, in the case of our project where the wing would be forced to cross the passenger cabin.

- low-wing: from the structural point of view is a good choice because we can ensure the continuity of the wing below the bottom of the cabin. The real problem of the wing lower is the risk that this will be damaged during the phases of takeoff and landing. Assuming a cart of 0.5m in height and wingspan of 10m and neglecting the dihedral angle of the wings, with low-wing the aircraft lateral tilt can not be more then 6° because with the inclination the wing tip touch the ground. low-wing design can tilt



- High-wing: provides greater visibility to the pilot during takeoff and landing, factor quite useful given the presence of the VTOL in the aircraft in question. In this configuration, there is also a greater stability to dihedral effect.

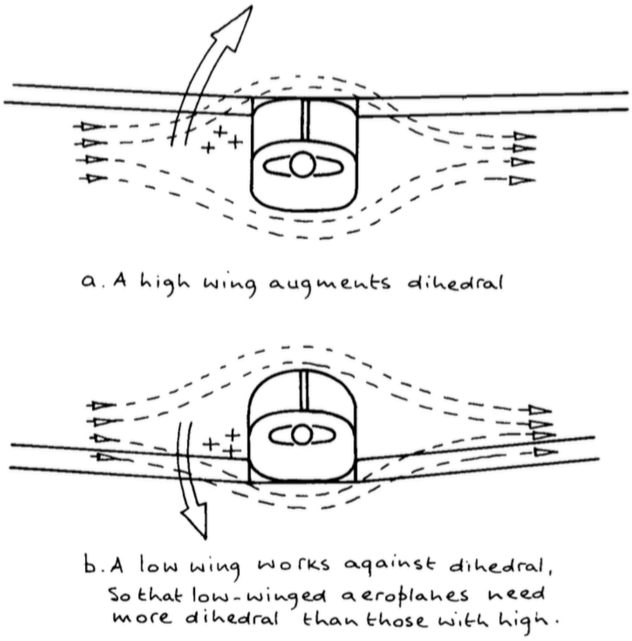


Figure 75 - Comparison Between High and low wing

	Mid wing	Low wing	High wing
aerodynamics	3	2	2
stability	2	1	3
structure	1	3	3
visibility	2	1	3
	Mid wing	**Low wing**	**High wing**
landing inclination	2	1	3
	10	8	14

Table 27 - Matrix of choice for the wing height

Finally It is not a good idea select the first configuration, for structural reasons, it is necessary instead to avoid the second for security reasons relating to the take-off and landing, taking also into account the fact that the aircraft in question will have to have the VTOL. Thanks to this consideration the best choice is the high-wing configuration.

Another factor that influences the choice is also the presence of electric rotors that for reasons of interference must not be positioned at the same height of the wings. At the same time, however, the electric rotors can not be too close to the ground to prevent to be damaged by debris raised, and to avoid an excessive ground interaction.

The most plausible solution is so high wing and medium rotor.

5.10.9. Structure

The main problem relates to structures regards the hight weight these ones can reach without the necessary considerations. As we saw in the previous categories (access door, pressurization, positioning wings) often structural principles impose constraints.

Good choices made by the structural point of view lead to a reduction of the weight of the aircraft, at a simplification in the process of construction and maintenance. It is good to keep in mind which are the areas most stressed at the structural level (as for example the area of attachment of the load-bearing surface) to avoid further stress in the same areas.

5.10.10. Vertical and horizontal tail

For Vertical tail the resolution used is a single surface located at a relatively high distance from the gravity center of the aircraft to maximise the moment arm.

For sizing the tail See chapter to it on. As it regards the horizontal tail it is important to taken into account the downwash that is affected by the vertical distance between the wings and the horizontal tail. We must therefore maximize this distance.

Given the presence of the top wing and rotor, which by their presence would decrease the horizontal tail efficiency, the appropriate solution is that of having the tail (v + h) in the classic shape of a T. This choice, however, leads the need for a more resistant structure in the vertical tail to cope with the presence at the tip of horizontal tail.

5.10.11. Cost

This is a constrain that is reflected in the choices made in the previous categories. This constraint leads to prefer simplicity in choices. With an infinite budget we can do fantastic things, we can provide some of the constraints that we have set ourselves in the best possible way. This choice, however, would be payed with the impossibility of entering the real market. We can therefore see this policy as one that requires us to look at reality from time to time.

5.11. Fuselage Aerodynamic study

To begin to test the aerodynamic fuselage quality, in this first phase of the aircraft project, it has been decided to analyze a 2D cross-section fuselage CFD analysis. The result of this analysis consists of data and images that should be analyzed in a phase that takes the name of post-processing. The purpose of the images is to show the trend of a characteristic along the analyzed profile. The features that will be used for the analysis of the aircraft will be:

- Pressure: the study of the pressure distribution alongside the profile. It is needed for the calculation of the lift capacity, flow resistance and to make considerations about the energy of the flow. If the whole high part of the profile has a higher pressure than the lower part the profile, turns out to be load-bearing. This can be useful to understand which are the points to be modified to obtain a variation of lift. In the case of MiniBee, figure 27, the profile has a higher pressure on the back, this suggests that the profile is producing a downforce. This result also has shown up through numerical results quantitatively as can be seen on tab1. The importance of the images is the capability to analyze and identify areas that need to be changed. At the same time you can observe the sudden increase of pressure in correspondence to the stopping point (in red in figure 27) the intensity of which is closely linked to aerodynamic drag. In the stop point the relative speed between the aircraft and the flow is zero, the pressure therefore corresponds to the total pressure, and this sudden stoppage leads to a dissipation of energy. Although is impossible to have a profile without of stop points but it is possible to reduce the intensity of the energy dissipated. The orange area immediately after the stop point is also interesting. This area is responsible for a further slowing in the flow, it would therefore be appropriate to reduce it as much as possible. In this case it would be necessary to review the profile to decrease the pressure on the back and trying to reduce the size of the 'red-areas', but this process should be carried out taking into account the restrictions set above.

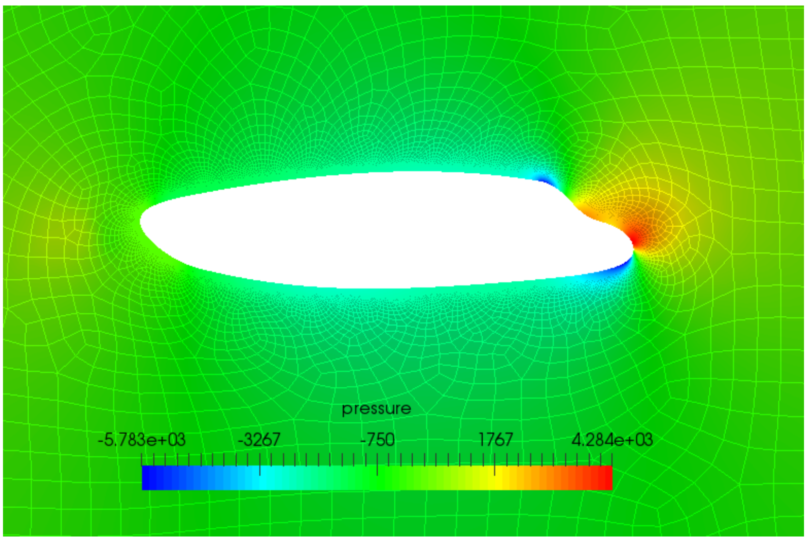


Figure 76 - Pressure distribution around the fuselage

Forces - Direction Vector (0 1 0)

Zone	Forces (n) Pressure	Viscous	Total
wall	-6552.2061	6.002285	-6546.2038
Net	-6552.2061	6.002285	-6546.2038

Table 28 - Numerical evaluation of pressure distribution

- Velocity Magnitude: this image represents the tendency of the modulus of the velocity vector. In the case of a flow which can, in a good approximation, be considered incompressible the relationship between velocity and pressure it is quite narrow. The stop point of which has been widely discussed in the previous image can be easily identified even in this image because it corresponds to a zone of very low speed (blue in figure 28). In this image is interesting to note the low flow velocity in the rear of the profile, the fact that in that area the flow is almost stationary mean that the vortices are present in the rear. This fact can be studied with more clarity thanks to the use of the stream line shown on figure 29. These are the virtual reproductions of the flow of smoke introduced into the wind tunnel to see the trajectories followed by flowing. They are used to highlight special or strange behavior of the flow investing vehicle. The color of the lines once again represents the intensity of the speed, their shape instead represents the trend of the flow. With this picture it is clear that in the profile rear part vortices are present, these in addition to dissipating much energy are responsible for increased aerodynamic profile drag. For these reasons, once again it would be appropriate to modify the profile so as to reduce as much as possible the detachment of the flow with consequent formation of vortices.

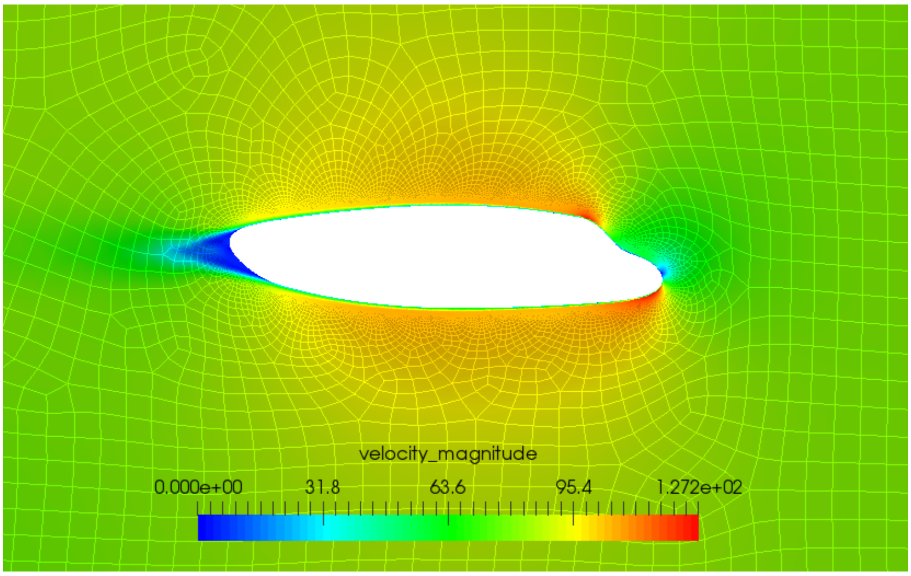


Figure 77 - Velocity Magnitude around the fuselage

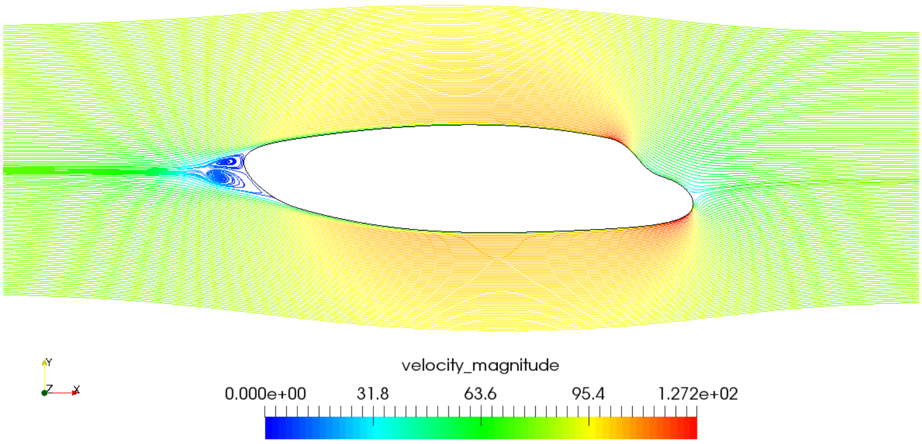


Figure 78 - Stream Lines around the fuselage

- Turbulent Kinectic energy: in a turbulent motion the local fluid characteristics (speed, pressure …) vary in a continuous way with a characteristic frequency. The turbulent kinetic energy is an index related to the mean square of the speed, it represents the amount of energy exchanged between the large vortical structures and small ones which then are dissipated into heat. A high value of this parameter leads to a considerable loss of energy in the form of heat, it should therefore seek to limit it. In figure 30 we can observe the amount of energy dissipated due to the vortices of the rear part, as mentioned above the intensity and the size of these vortices should be reduced as much as possible changing the rear part profile.

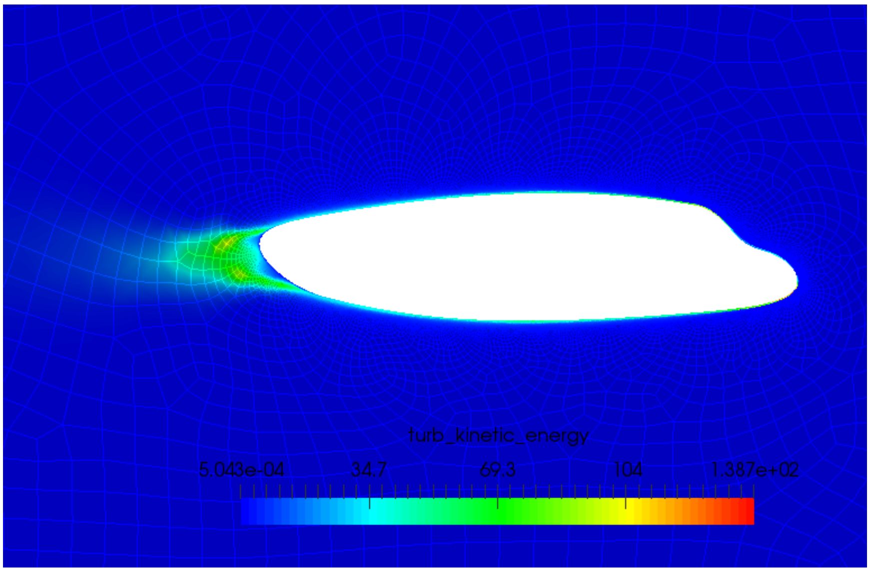


Figure 79 - Turbulent kinectic energy

6. PARAMETRIC CAD AND 3DEXPERIENCE

6.1. 3Dexperience introduction

It is decided to proceed with the MiniBee project development using 3DEXPERIENCE. 3DEXPERIENCE is a business platform created to follow a project in its complete development, starting from the first planning and design to the development of the marketing for the product itself.
The platform 3DEXPERIENCE attends this path through dedicated software, such as Catia for the design phase.

3DEXPERIENCE is not only a software package but has built in it a new way of project management, its main purpose is to ensure communication between the various sections of the project. This object is achieved through applications for communication and exchange of ideas internal to the platform itself. The communication between the various stages of the project allows a continuous change and adaptation to each part at any time and guarantees a continuous loop of renovation.

Another factor definitely positive in this platform is the wall where the most important information and the latest news can be exposed in order to quickly reach all the people working on the project as shown in figure 80. This all seems very inspired to today's social network, nothing is totally foreign, it is only to publish news about our work and not about our lives.

The platform is then associated with a cloud that allows you to have all the files and all information about the project everywhere and always.

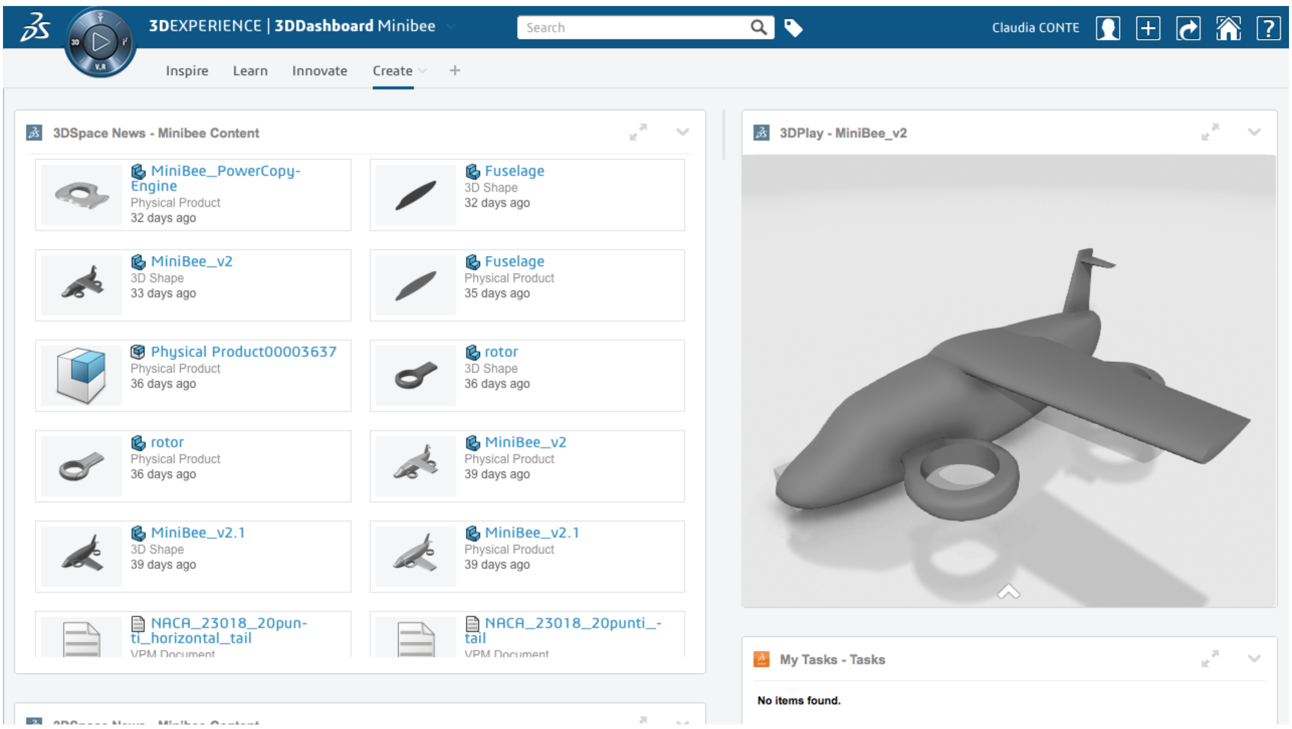


Figure 80 - Example of 3Dexperience dashboard

3DEXPERIENCE is an innovative way to manage global business that certainly,
if well used, could lead to a market improvement in the design process resulting in a reduction of the time to set production.
In contrast, however, to work with the CAD platform has brought some problems, it sometimes stop to work suddenly. Probably this problem is linked to the outdated computers used.

Another problem regards how to use the wiki section, it is not intuitive, no one know how to delete blank pages. It is a good idea to have this section, it would be very useful if it was fixed.
In the end the experience with the platform 3DEXPERIENCE can be considered positive, it takes a little preparation to be able to accept change in the management of the project but the simplification that follows is relevant.

6.2. Parametric CAD

Within CAD design it has decided to introduce from the beginning the idea of parametrisation.
A parametric CAD allows to embed an object, during its creation, of a wide range of information. Its creation is seen as a series of actions that they are saved in the object history and they can be edited and rearranged easily. As the various information are related to each other, CAD is home to a continuous reconstruction and revision during the design.
This process provides for flexibility in modelling. It is a creation system that associates each object with the information about its geometry and its interaction with other parts of the project. This information can be changed at any time by adjusting the parameters that define it.
Following the change in one or more dimensions, in fact, the system reconstructs the parametric CAD model according to what requested by the user. Very important within this environment it is also the definition of the constraints which must accurately define each element of the project but should not be redundant.

These can be divided into three main types:

- geometrical (parallelism, coincidence, tangency ...)
- on property and operators (symmetry, equality between areas ...)
- non-geometric (relationships between physical ...)

There is also the possibility to easily connect together the parameters that express the size in the project through functions or equalities, using constant values or by importing values from an Excel spreadsheet. The possibility of a parametric CAD design, as you can see, are vast. For MiniBee project will be of particular importance flexibility in changing the parameters, the possibility of designating some parameters such as driving parameters and the ability to impose geometric constraints to guide surface.

This choice has led to a greater expense of time in the preparation of the first CAD model, but at the same time to a dramatic reduction of the work needed for future modifications of the same.
We can compare the parametric model to a computer program created to solve a problem, its creation takes time, but once created it will be able to solve easily a considerable number of problems similar to each other, unlike the simple problem resolution would lead at first to a gain in terms of time, but would not allow flexibility in change.

It is only at the beginning of the project so it seem appropriate to ensure CAD flexibility as much as possible in order to be able to accompany all developments of the project without having to start over each time, which is why we chose to parameterise the model as much as possible.

6.3. MiniBee specific use

The first step in Cad design is to find a way to introduce, in the CAD program, the profile for the wing. When we first sketched it the profile had already been chosen, but it was not definitive.
To address this problem it was decided to create some parameters connected, through relationships, to an easily editable Excel spreadsheet:

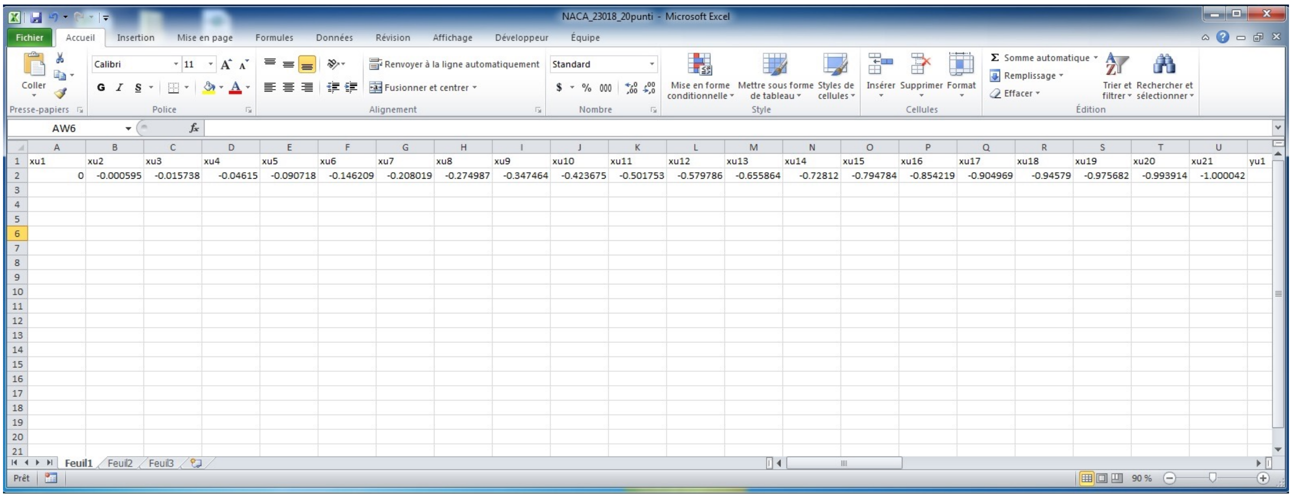


Figure 81 - EXCEL TABLE WITH PROFILE POINTS COORDINATES

This way we simply load a new folder excel in the program to switch from one profile to another without having to re-introduce the coordinates of points.

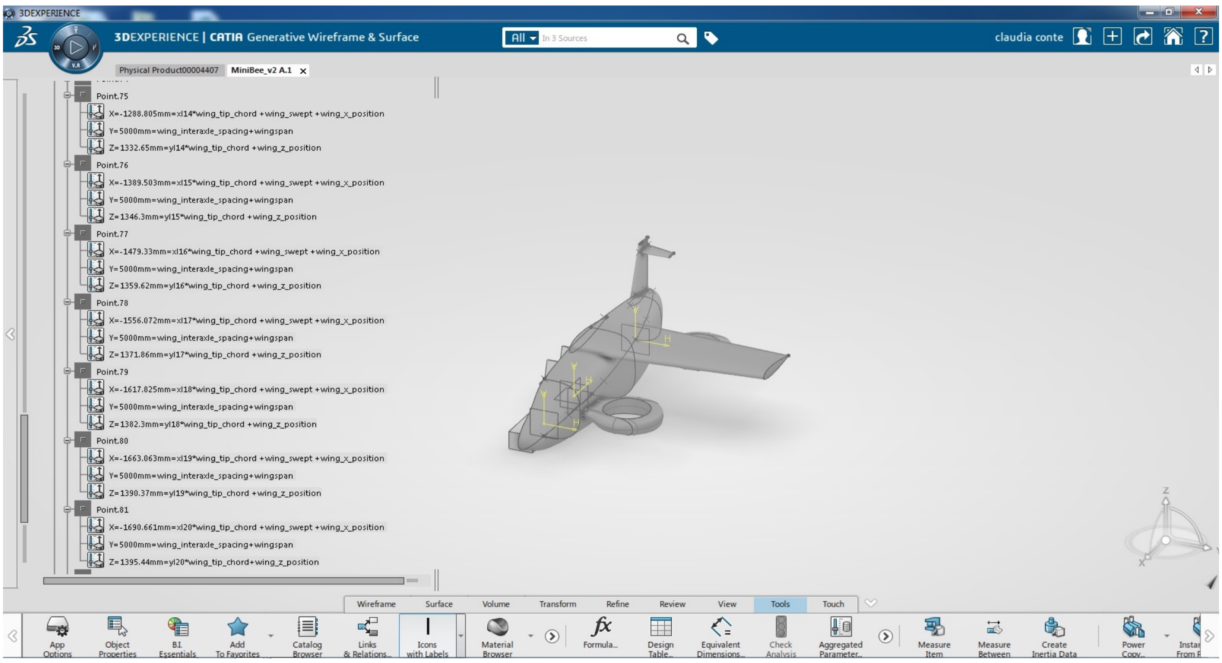


Figure 82 - FORMULAS DESCRIBE THE POSITION OF EACH POINT

In order to change the position and the profile chord have been added to other parameters in the definition of the coordinates of individual points, for example, we can update the chord or the position of the wing by simply changing the value of the parameter relating to them.

After defining the points of the profile contour, it was necessary to create the spline support for the wing surface. To do this it was necessary to analyse the curvature of the spline itself to avoid errors in future surface. It was necessary to modify the spline in order to obtain a curve without abrupt changes how it can be seen in figure 83.

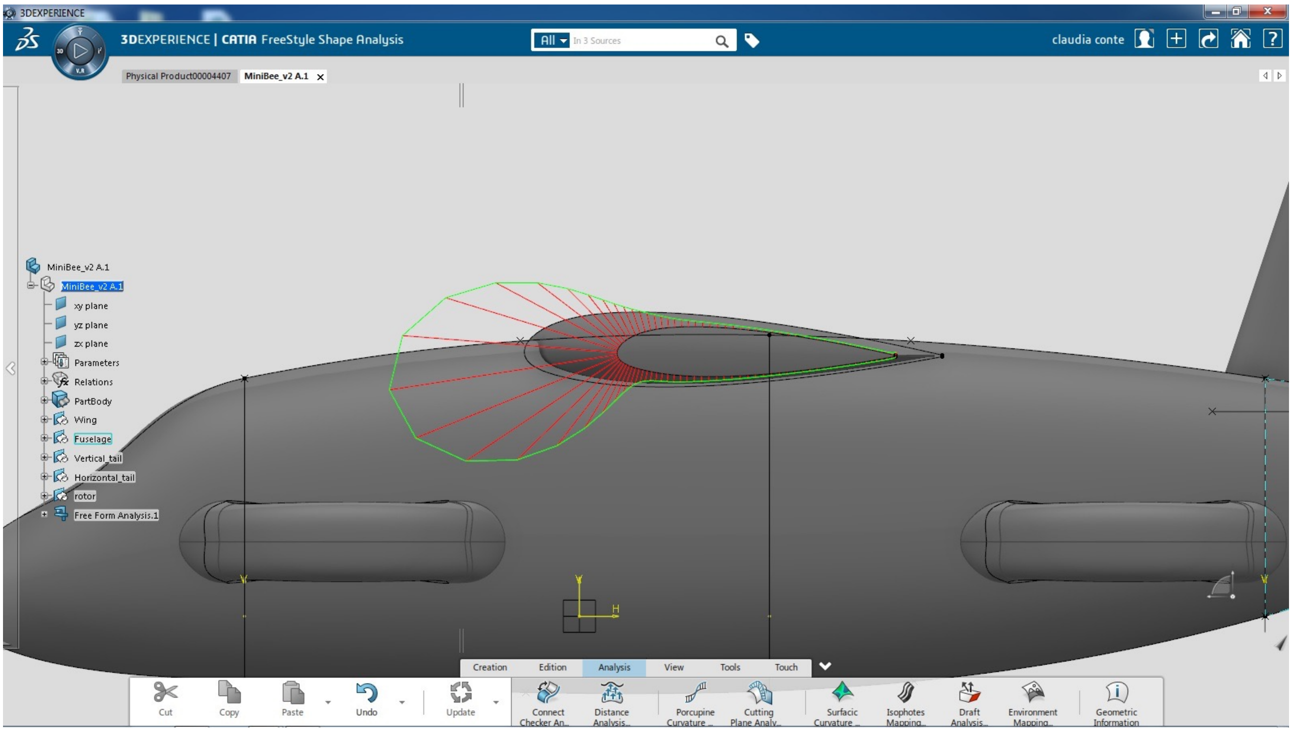


Figure 83 - CURVATURE ANALYSIS OF WING PROFILE

After getting a good profile it was quite simple create the wing surface.
Additional parameters have been added to allow:
-swept angle
-wing rastremation
-dihedral angle

After the wing determination it was considered appropriate to start working on the profile of the fuselage. For the definition of profile initial shape the notions explained in the chapter on the fuselage have been taken in to account. To ensure the approximate size relative to the packaging, 3D models of the seats and the battery pack were created.

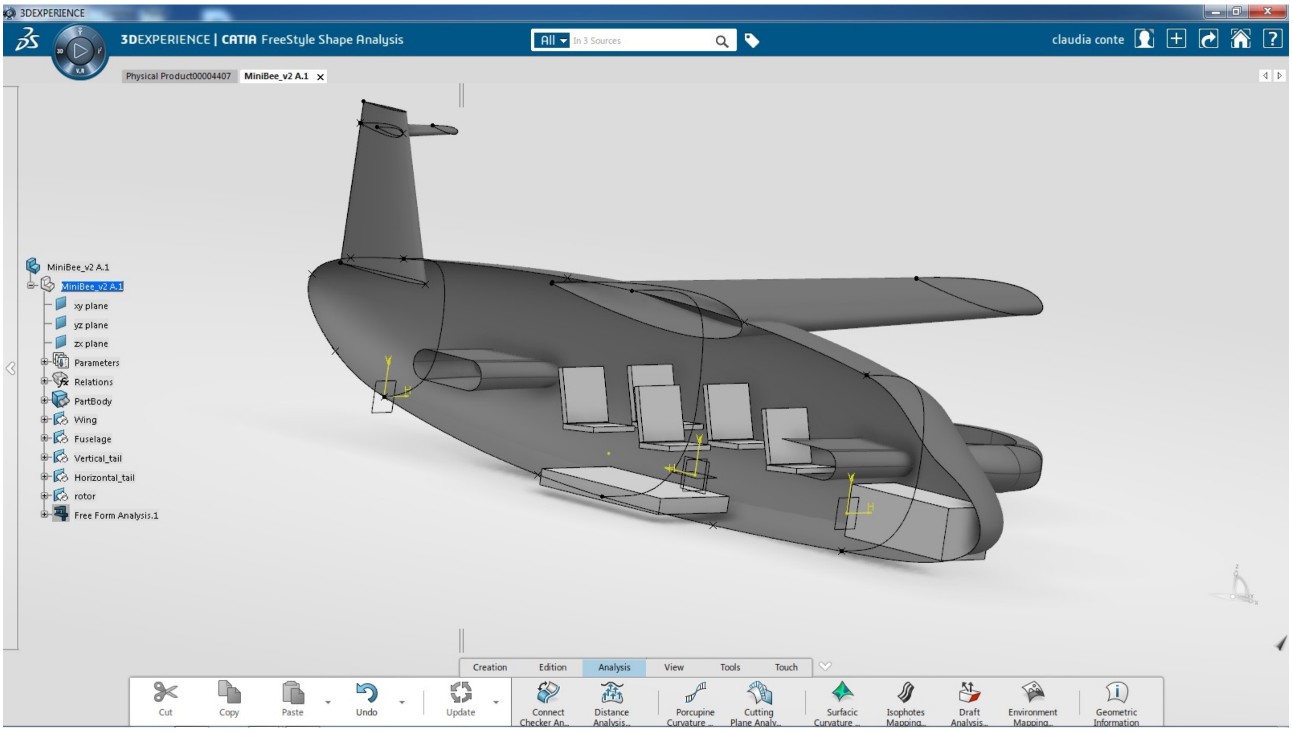


Figure 84- APPROXIMATIVE DIMENSION OF ENGINE, BATTERY PACK AND SEATS

This was necessary for a first definition of the fuselage.

Instead of the wing, given the non standard nature of the fuselage, the use of parametric is very reduced part. The fuselage is therefore the element more "not editable" of the entire project or at least one that needs more work in the event of changes.

The process of introduction of the vertical tail and the horizontal tail was very similar to that used for wing, also in this case parameters whose values have been directly linked with those present in a excel table have been introduced. As in the case of the wing, at first it was used a "temporary profile" which was subsequently changed to a later time.
The case of the control surface is slightly easier to set up because these surfaces profile is symmetrical, once drawn, therefore, one half of the profile, the other was obtained by symmetry. Also in this case we were introduced additional parameters to ensure:

- vertical tail rastremation
- vertical tail inclination
- horizontal tail rastremation
- horizontal tail swept
- horizontal tail dihedral angle

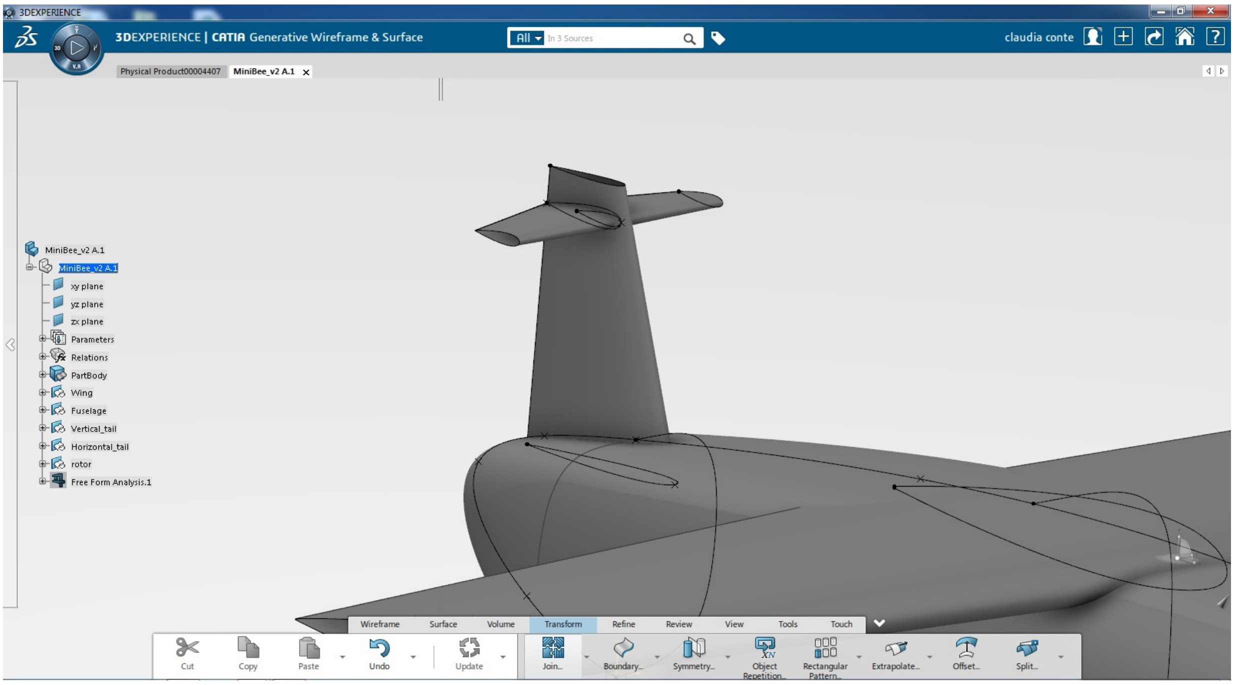


Figure 85 - POSITION AND SHAPE OF HORIZONTAL AND VERTICAL TAIL

As it regards the electric rotor it is preferred to draw the surfaces of coverage in a different CAD file and then add them only at a second time to the entire aircraft CAD file. This solution has been chosen as the definition of the form of the rotor requires a study to improve the aerodynamic shape. This study is required to reduce the interaction between the rotors and the wings but also to ensure that the resistance, due to the rotors, is a minimum.

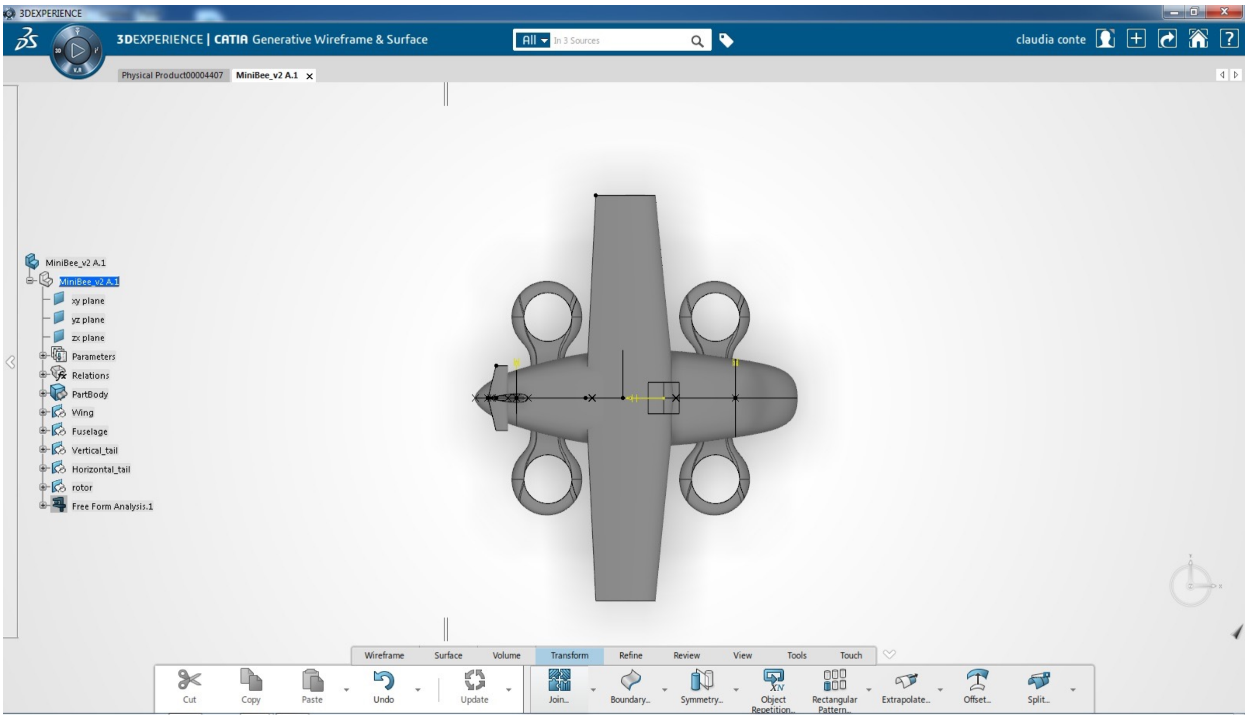


Figure 86 - ELECTRIC ROTORS POSITION IN YX PLANE

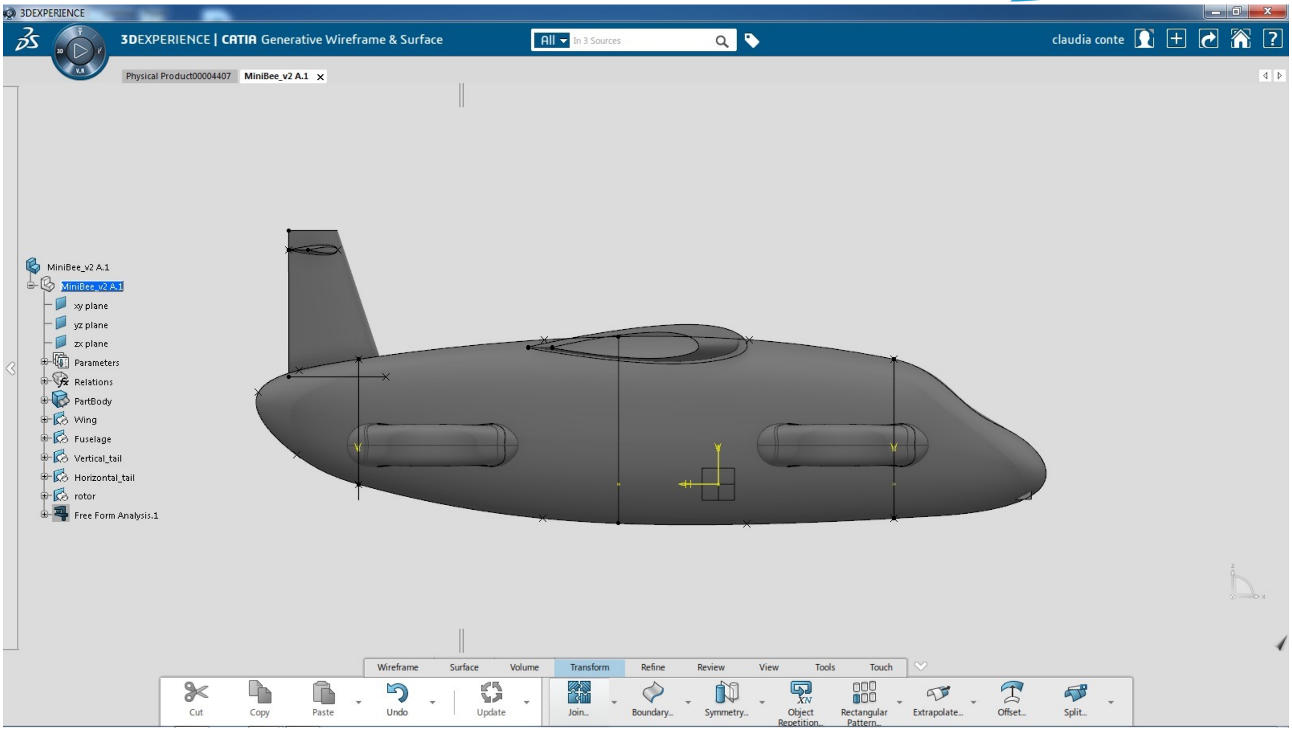


Figure 87 - ELECTRIC ROTORS POSITION IN ZX PLANE

After finalizing the shape they have been introduced in the entire aircraft CAD and positioned by means of parameters.

6.4. Potential of parametric

To show the potential of a parametric design we have produced 4 configurations of an aircraft with characteristics quite similar.
To obtain different configurations it was enough to change some parameter and recreate some surface. The power of the parametric is that for to create 4 models substantially different it was needed less than an hour of work, create each model starting again from the beginning, in the contrary, would have taken much longer.

6.4.1. Configuration 1

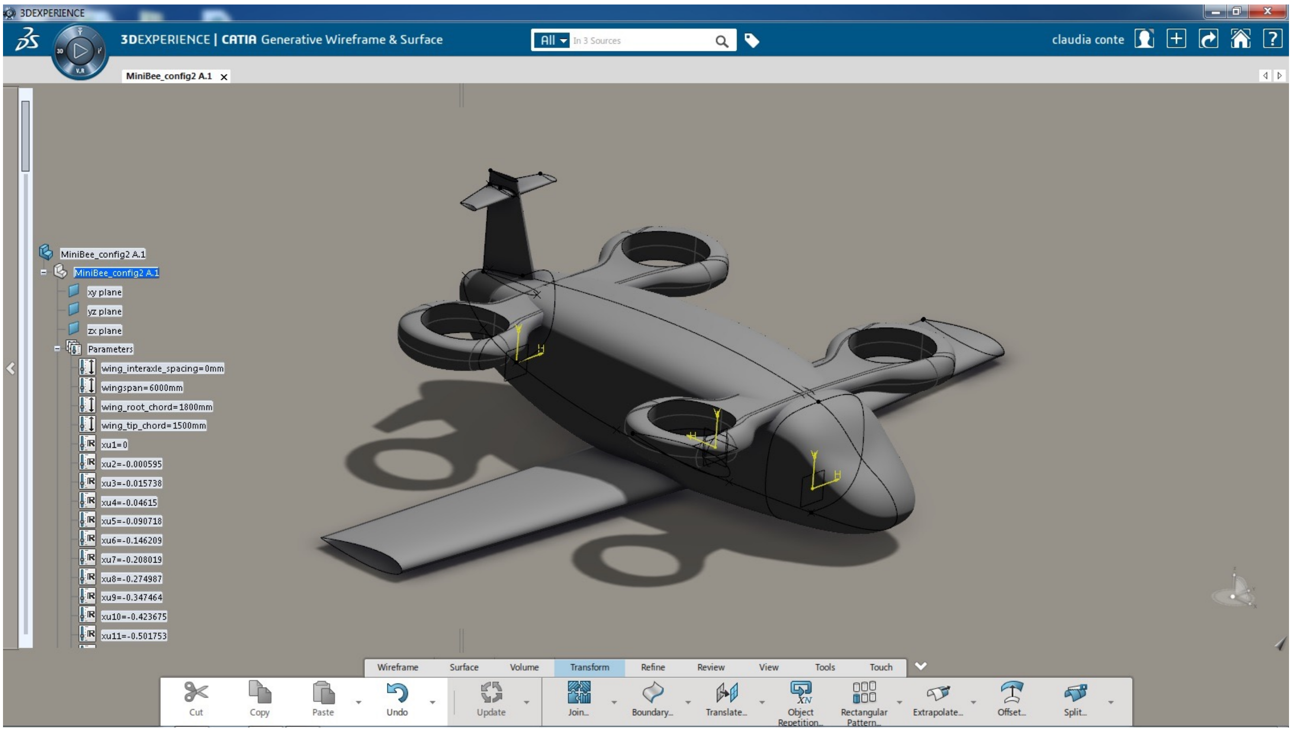


Figure 88 - Configuration 1

Configuration 1 is the first to be created, it is the most consistent with the form that the MiniBee, as a result of various considerations, should have. This configuration has a wing span of 4m and an high wing, 4 rotors positioned at medium height, a single central vertical tail. This is the starting configuration for the other configurations.

6.4.2. Configuration 2

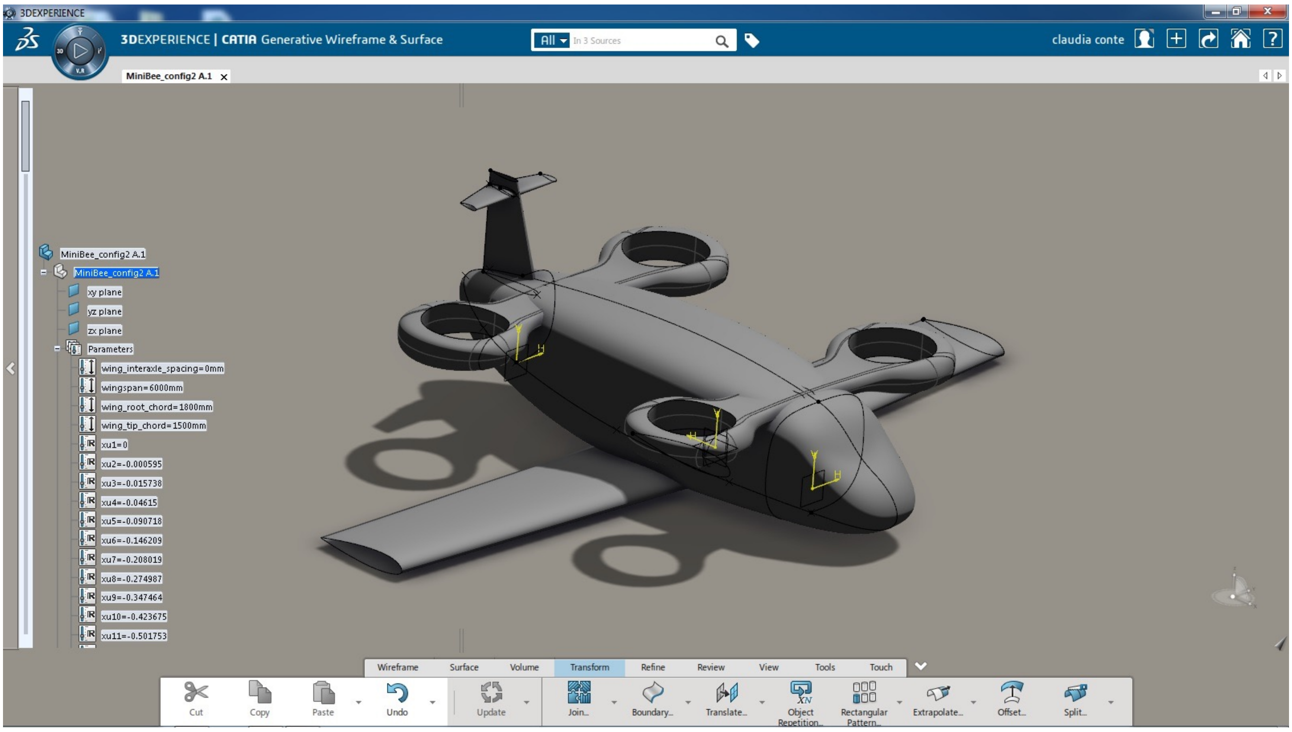


Figure 89 - Configuration 2

This configuration is not very different from the previous, the position of the wing is passed from high to low, and consequently it was possible to position the 4 rotor in the upper part of the fuselage.

The tail was te same. In addition to the relative position of the elements, also the shape of the wing has been modified, the span is now 5m which allows a reduction of the chord of the wing. This reduction, however, causes a greater overall dimensions of aircraft passing by ca. 10m to ca.12m.

6.4.3. Configuration 3

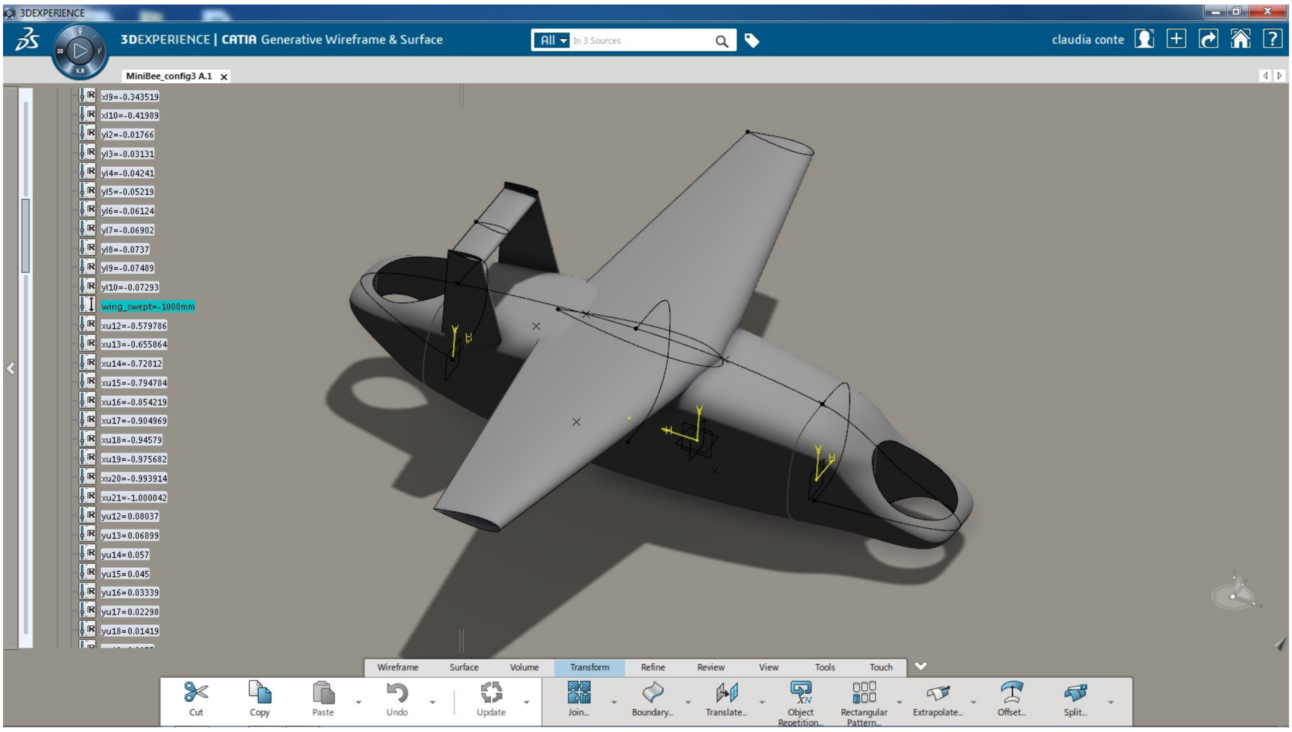


Figure 90 - Configuration 3

This configuration is very different from the others, the electric rotors passe from 4 to 2 and have been incorporated in the fuselage along the aircraft axis of symmetry. The presence of the central rear rotor imposed the scene of a double vertical tail. Wing rastremation and swept angle change.
Also in this case the changes, even though they have led to a very different pattern, have required a reduced time.

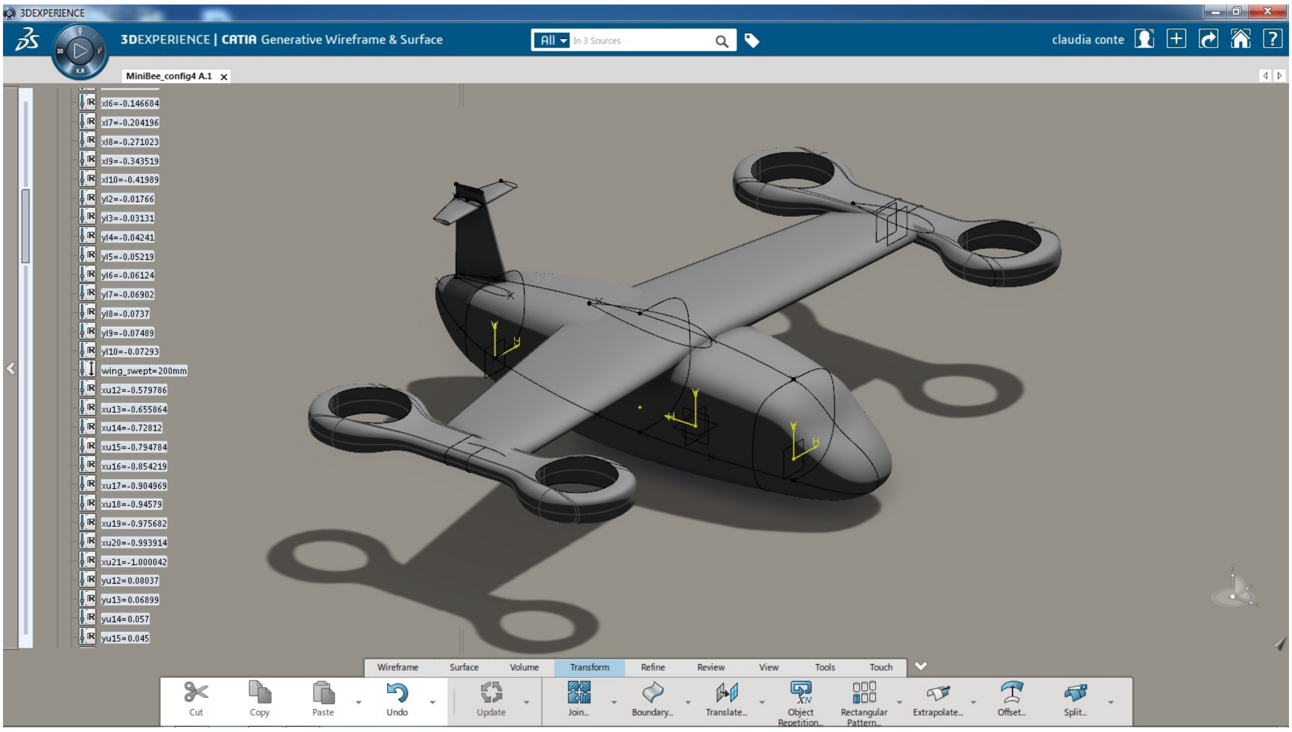


Figure 91 - Configuration 4

6.4.4. Configuration 4

In the last configuration it has changed the location of the 4 rotors. They were positioned to wing tip, away from the fuselage, also the surfaces have been rotated to ensure proper connection. there is also a visible change in the wing swept angle.

These configurations have been realized only to show the potential of the parametric model, the choices are maiden only for this purpose and they are not supported by physical considerations. For example, the reduction from 4 to 2 of the rotors of the electrical motors in config3 would not permit a plane of this dimensions to take off.

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