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**HYBRID-ELECTRIC AIRCRAFT: CONCEPTUAL DESIGN,
STRUCTURAL AND AEROELASTIC ANALYSES**



UNIVERSIDADE FEDERAL DE UBERLÂNDIA
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STRUCTURAL AND AEROELASTIC ANALYSES**

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Projeto de conclusão de curso
APROVADO pelo Colegiado do Curso de
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To our moms and dads.

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Alexandre Acerra Gil

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ABSTRACT

This work presents the conceptual design of two hybrid aircrafts (four and six passengers) and the structural and aeroelastic analyses for the wing and empennages. The requirements in the conceptual phase were established by the aeronautical design competition held annually by the American Institute of Aeronautics and Astronautics (AIAA). The initial characteristics and geometries of a conventional aircraft were estimated through the study of historical trends. Then, the hybridization characteristics were evaluated using the series architecture, which affects mainly the range equations and the definition of the propulsive system elements that better fit the project, e. g., the internal combustion engine and batteries. To explore the multidisciplinary characteristics of the aircraft development, it was implemented an optimization procedure based on a genetic algorithm fully integrated with an aerodynamic module with performance and stability constraints, searching for the best values of the geometrical design variables. After finishing the conceptual design, the structural evaluation of wing and empennages (horizontal and vertical) was performed considering only static loads, but the failure index and the buckling instability were checked. The evaluated loads were extracted from aerodynamic models at the extreme conditions of flight envelope presented in V-n diagrams. First, the structural components were defined in aluminum 7050-T651 and later using composite material, specifically the carbon-epoxy Hexcel 8552 NMS 128/2. Additionally, only the aeroelastic analysis for the wing was evaluated, because in this work the fuselage analysis wasn't done. The results showed no flutter speed and no divergence phenomena within the aircraft's flight envelope, which is justified by the torsional rigidity obtained in the main box structure.

KEYWORDS: *aeronautical design, hybrid aircraft, optimization, aerodynamic loads, aeronautical structures, aeroelasticity.*

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RESUMO

O presente trabalho apresenta o projeto conceitual de duas aeronaves híbridas (quatro e seis passageiros) e as análises estruturais e aeroelásticas para a asa e empenagens. Os requisitos na fase conceitual foram estabelecidos pela competição de projeto aeronáutico realizada anualmente pelo *American Insitute of Aeronautics and Astronautics* (AIAA). A características e geometrias iniciais de uma aeronave convencional foram estimadas através do estudo de tendências históricas. Assim, as características de hibridização foram avaliadas utilizando a arquitetura em série, a qual afeta principalmente as equações de alcance e a definição dos elementos do sistema propulsivo que melhor se adequam ao projeto, isto é, o motor a combustão interna e as baterias. Para explorar as características multidisciplinares do desenvolvimento das aeronaves, foi implementado um processo de otimização baseado em um algoritmo genético totalmente integrado com um módulo aerodinâmico, juntamente com limitações de desempenho e estabilidade, buscando os melhores valores das variáveis geométricas do projeto. Após finalizar o projeto conceitual, a avaliação estrutural da asa e das empenagens (horizontal e vertical) foi feita considerando somente cargas estáticas, mas o índice de falha e instabilidade de flambagem foram checados. As cargas analisadas foram extraídas dos modelos aerodinâmicos nas condições extremas de voo apresentadas nos diagrama V-n. Primeiramente, os componentes estruturais foram definidos em alumínio 7050-T651 e, posteriormente, foi utilizado material compósito, especificamente o carbono-epoxy Hexcel 8552 NMS 128/2. Além disso, apenas a análise aeroelástica da asa foi avaliada, pois nesse trabalho a análise da fuselagem não foi realizada. Os resultados não apresentaram os fenômenos de *flutter* e de divergência no envelope de voo da aeronave, o que é justificado pela rigidez torcional obtida na estrutura principal da asa.

PALAVRAS CHAVE: projeto aeronáutico, aeronave híbrida, otimização, cargas aerodinâmicas, estruturas aeronáuticas, aeroelasticidade.

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List of Symbols

α	Angle of Attack
Γ_W	Wing Dihedral Angle
δ_a	Aileron Deflection
δ_e	Elevator Deflection
δ_r	Rudder Deflection
η_{bat}	Battery Efficiency
η_{cont}	Controller Efficiency
η_{em}	Electric Motor Efficiency
η_{eng}	ICE Efficiency
$\eta_{generator}$	Generator Efficiency
η_{inv}	Inverter Efficiency
η_{prop}	Propeller Efficiency
η_{wiring}	Wiring Efficiency
λ_{HT}	Horizontal Tail Taper Ratio
λ_{VT}	Vertical Tail Taper Ratio
λ_W	Wing Taper Ratio
ν	Poisson's Ratio
ν_{12}	Poisson's Ratio Direction 12
Λ_W	Wing Sweep Angle
ρ_{air}	Air Density
σ_Y	Yield Strength

ϕ	Bank Angle
ϕ	Supplied Power Ratio
ψ	Degree-Of-Hybridization in Energy
a	Trend Constant
$a_{D_{LND}}$	Constant for D_{LND} Estimate
$a_{D_{TO}}$	Constant for D_{TO} Estimate
$a_{fuselage}$	Constant for Fuselage Estimate
$a_{S_{wet}}$	Constant for S_{wet} Estimate
a_T	Constant for T Estimate
$a_{W/S}$	Constant for W/S Estimate
a_{W_0}	Constant for W_0 Estimate
AR_{HT}	Horizontal Tail Aspect Ratio
AR_{VT}	Vertical Tail Aspect Ratio
AR_W	Wing Aspect Ratio
b_{HT}	Horizontal Tail Span
b_{VT}	Vertical Tail Span
b_W	Wing Span
c	Trend Constant
$c_{fuselage}$	Constant for Fuselage Estimate
$c_{l\beta}$	Rolling Moment Coefficient Related to β
$c_{mac_{HT}}$	Horizontal Tail Mean Aerodynamic Chord
$c_{mac_{VT}}$	Vertical Tail Mean Aerodynamic Chord
c_{mac_W}	Wing Mean Aerodynamic Chord
$c_{m\alpha}$	Pitching Moment Coefficient Related to α
$c_{n\beta}$	Yaw Moment Coefficient Related to β
c_P	Power-Specific Fuel Consumption
$c_{r_{HT}}$	Horizontal Tail Root Chord
$c_{r_{VT}}$	Vertical Tail Root Chord
c_{r_W}	Wing Root-chord
$c_{S_{wet}}$	Constant for S_{wet} Estimate
c_T	Constant for T Estimate

c_T	Thrust Coefficient
$c_{W/S}$	Constant for W/S Estimate
c_{W_0}	Constant for W_0 Estimate
C_D	Aircraft Drag Coefficient
C_{D0}	Parasite Drag Coefficient
C_{fe}	Friction Coefficient
C_L	Aircraft Lift Coefficient
$C_{L_{clean}}$	Aircraft Lift Coefficient without Flaps
$C_{L_{cruise}}$	Cruise Lift Coefficient
$C_{L_{max}}$	Maximum Lift Coefficient
$C_{L_{LND}}$	Landing Lift Coefficient
$C_{L_{TO}}$	Take-off Lift Coefficient
$d_{fuselage}$	Fuselage Diameter
$dCm_{\delta \text{ elevator}}$	Pitching Moment Variation Related to Elevator Deflection
$dCl_{\delta \text{ aileron}}$	Rolling Moment Variation Related to Aileron Deflection
$dCl_{\delta \text{ rudder}}$	Rolling Moment Variation Related to Rudder Deflection
$dCn_{\delta \text{ aileron}}$	Yaw Moment Variation Related to Aileron Deflection
$dCn_{\delta \text{ rudder}}$	Yaw Moment Variation Related to Rudder Deflection
$dCL_{\delta \text{ elevator}}$	Lift Coefficient Variation Related to Elevator Deflection
D	Drag
e_w	Wing Oswald's Coefficient
E	Endurance
$E_{battery}$	Battery Energy
$E_{electric}$	Electric Energy
E_{em}	Total Supplied Electric Motor Energy
$E_{em_{total}}$	Electric Motor Total Energy
$E_{gasturb}$	Total Supplied Gas Turbine Energy
$E_{shaft_{total}}$	Shaft Total Energy
E_{start}	Aircraft Initial Stored Energy
E_{total}	Total Energy
E_{11}	Young's Modulus Direction 11
E_{22}	Young's Modulus Direction 22
g	Gravity

G_{12}	Shear Modulus Direction 12
G_{23}	Shear Modulus Direction 23
G_{13}	Shear Modulus Direction 13
$H_{battery}$	Battery Energy Density
H_E	Degree of Hybridization for Energy
H_{fuel}	Fuel Energy Density
H_P	Degree of Hybridization for Power
i_w	Wing Incidence Angle
I_{xx}	Moment of Inertia
I_{yy}	Moment of Inertia
I_{zz}	Moment of Inertia
k_2	Induced Drag Constant
$l_{fuselage}$	Fuselage Length
L / D	Aerodynamic Efficiency
M_{cruise}	Number of Mach in Cruise
$n_{z_{max}}$	Maximum Positive Load Factor
$n_{z_{max} Gust}$	Maximum Positive Load Factor with Gust
$n_{z_{min}}$	Maximum Negative Load Factor
$n_{z_{min} Gust}$	Maximum Negative Load Factor with Gust
P_a	Available Power
P_{bat}	Battery Power
P_{cont}	Controller Power
$P_{electric}$	Electric Power
P_{em}	Electric Motor Power
P_{eng}	ICE Power
P_{fuel}	Fuel Power
$P_{generator}$	Generator Power
P_{inv}	Inverter Power
P_{total}	Total Power
P_{wiring}	Wiring Power
Q_f	Fuel Flow Rate

R	Range
R_x	Constant for Moment of Inertia Estimate
R_y	Constant for Moment of Inertia Estimate
R_z	Constant for Moment of Inertia Estimate
S	Degree-of-hybridization in Power
S_{HT}	Horizontal Tail Area
S_{VT}	Vertical Tail Area
S_w	Wing Area
S_{wet}	Wetted Area
t_{final}	Final Mission Time
t_{start}	Initial Mission Time
T	Thrust
V	Velocity
V_A	Acrobatic Speed
V_C	Cruise Speed
V_{cruise}	Cruise Speed
V_D	Dive Speed
V_G	Gust Speed
V_{HT}	Horizontal Tail Volume Coefficient
V_{stall}	Stall Speed
V_{VT}	Vertical Tail Volume Coefficient
W_{cruise}	Cruise Weight
W_{final}	Final Weight
W_{fuel}	Fuel Weight
$W_{payload}$	Payload Weight
W_{start}	Initial Weight
W_0	Gross Weight
W_1	Initial Climb Weight
W_2	Initial Cruise Weight
W_3	Final Cruise Weight
W_4	Final Descent Weight
W_5	Landing Weight
W_6	Final Mission Weight

x_{CG}	Center of Gravity Position
x_{HT}	Horizontal Tail Position
$x_{Landing\ gear}$	Landing Gear Position
x_{VT}	Vertical Tail Position
x_w	Wing Position
x_C/d	Relation Between Cabin Length and Diameter
x_L/d	Relation between fuselage length and diameter
x_N/d	Relation Between Nose Length and Diameter
x_T/d	Relation between tail length and diameter
y_{engine}	Engine Position
y_{mac_w}	Wing Mean Aerodynamic Chord Position

List of Abbreviations and Acronyms

4PAX	4 Passengers Aircraft
6PAX	6 Passengers Aircraft
AGAPE	ACARE Goals Progress Evaluation
AGL	Above Ground level
AIAA	American Institute of Aeronautics and Astronautics
AR	Aspect Ratio
ATAG	Air Transport Action Group
AVL	Athena Vortex Lattice
CFR	Code of Federal Regulations
CG	Center of Gravity
DESPPS	Dual-Energy Storage-Propulsion-Power System
DoH	Degree-of-Hybridization
EAS	Equivalent Airspeed
FAA	Federal Aviation Administration
FEA	Finite Element Method
FL	Flight Level
ICE	Internal Combustion Engine
IFR	Instrument Flight Rules
ISA	International Standard Atmosphere
KEAS	Equivalent Airspeed in Knots
LHS	Left-hand Side
MIT	Massachusetts Institute of Technology
MTOW	Maximum Take-Off Weight
NACA	National Advisory Committee for Aeronautics
PK-NL	PK-method No Looping

RBE	Rigid Body Elements
RFP	Request for Proposal
RHS	Right-hand Side
ROC	Rate of Climb
SFC	Specific Fuel Consumption
SOL	Solution
TAS	True Airspeed
TZFW	Total Zero Fuel Weight
VFR	Visual Flight Rules
YEIS	Year Entry-Into-Service

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CHAPTER I

INTRODUCTION

*“When everything seems to be going
against you, remember that the airplane
takes off against the wind, not with it.”*

Henry Ford

The study of fully or partially electric aircraft has been the subject of discussion between researchers and engineers at universities and the aeronautics industry over the last years. The need to develop ever more efficient and greener aircraft leads to the motivation to expand technologies and move toward previously unfeasible concepts.

Currently, most General Aviation aircraft typically use internal combustion engines (ICE) as a power source. These engines burn fossil fuels with high energy densities, making this type of raw material very advantageous for aviation. However, they are highly polluting, since their burning generates the production of carbon dioxide (CO₂), which is the main responsible gas for global warming. The Air Transport Action Group (ATAG) points out that 2% of anthropometric carbon dioxide emissions come from aviation, and this number only tends to increase along with the number of aircraft in operation [1,2]. Under those circumstances, several targets have been defined in Vision 2020 and AGAPE 2020 for the next few years [3].

Besides the problem of emissions and pollution, the amount of fossil fuels available in the world is limited. Even not knowing about its availability and scarcity in

the future, the fossil fuel prices themselves tend to increase in the coming decades due to the fast growing worldwide energy demands and the uncertain political situation in the Middle East [4].

Thereby, all the situations mentioned above lead to consider and rethink in alternative ways of the power supply. Thus, the introduction of electric propulsion systems has been a great option. First of all, batteries can be used as a power source instead of conventional fuels. However, the battery use itself brings challenges such as the weight on board and its specific energy. Regarding the first one, aircraft are very sensitive when it comes to weight because as it increases on board, the available payload diminishes. Regarding the second one, the batteries currently have low specific energy when compared to high specific energy of conventional fuels. However, studies have shown great results for batteries in the future. Lithium-air batteries may show an impressive theoretical specific energy of 11,680 Wh/kg [5]. But in more realistic numbers, Zn-O₂ batteries with specific energy of 400 Wh/kg in 2025 are expected [6]. It is also worth remembering the challenge related to the international aviation regulations, which require that the minimum level of safety compared to the batteries be guaranteed.

Moreover, engines have a lower efficiency and power-to-weight ratio when compared to electric motors [7]. Hence, hybrid-electric systems are proposed to balance the advantages of both engine and motor systems, improving the performance. Such systems have potential advantages including low fuel costs, lower vibrations, lower pollution, and reduced noise.

After all, it is not an easy task to balance all these interests and to develop a wholly or partially electric propulsion system, but it is of great importance and necessity to carry on the study and improvement of physical limitations, fulfilling the requirements and guidelines for the future.

Along those lines, when designing an aircraft, it is important to seek the best aerodynamic efficiency, good stability control, lower associated weight, better aeroelastic behavior, and ease of manufacturing. In this work, it is done a conceptual design of two hybrid-electric aircraft whose design requirements were determined through a Request for Proposal (RFP) published by the committees of the annual design competition sponsored by The American Institute of Aeronautics and Astronautics (AIAA). Nonetheless, the structural and aeroelastic analyses for the wing and empennages of both aircraft are presented.

CHAPTER II

LITERATURE SURVEY AND THEORY

*“Inventar é imaginar o que ninguém pensou;
é acreditar no que ninguém jurou;
é arriscar o que ninguém ousou;
é realizar o que ninguém tentou.”*
Alberto Santos Dumont

This chapter presents a literature review of the current papers and books that discuss new concepts and designs of hybrid aircraft, including their respective system architectures, advantages, and perspectives for the aviation future. Besides, the aircraft design philosophies adopted by most manufacturers will be presented.

2.1. Hybrid-Electric Aircraft

Triggered by the need for developing new sustainable technologies, the aerospace industry has recently spent a lot of effort on the designing of greener aircraft. Such alternative among the new concepts and linked to this perspective is the hybrid aircraft.

A hybrid aircraft is an aircraft which has a combination of two or more sources capable of generating power into a single power system. The commonly used term “hybrid-electric” describes a system that utilizes one or more heat engines together with one or more electric-motors in a specific configuration [8].

Earlier to the hybrid aircraft concept, it is worth coming up with the electric propulsion system of aircraft. There are many advantages and enough motivation to move toward the development of such systems. The air traffic is growing (Figure 1) and, likewise, the emission of carbon dioxide (CO₂), the main responsible gas of the global warming. Thus, introducing electric propulsion systems brings a new level of efficiency, in addition to the possibility of being zero-emission (no pollution), becoming environmentally friendly and reducing issues such as the noise that comes from the conventional engine nozzle.

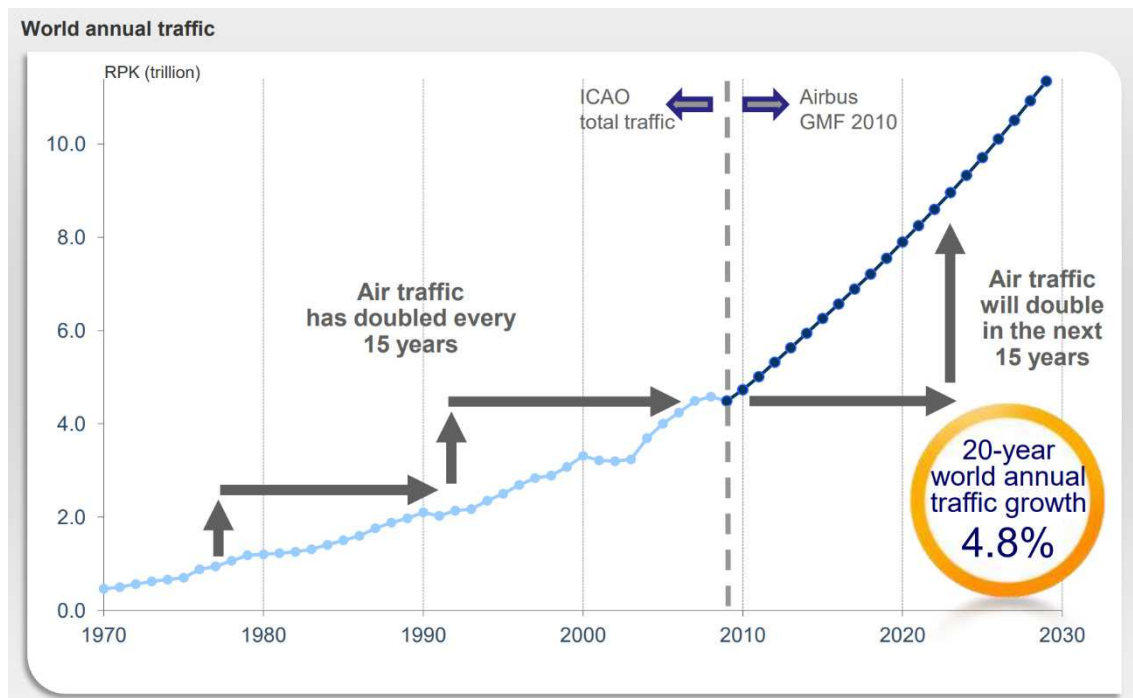


Figure 1 - History and prediction for the air traffic. Source: Airbus Global Market Forecast 2010-2029¹.

Since fully electric aircraft do not use fossil fuels as energy source, the operational costs related to fuels and maintenance are reduced considerably. Moreover, the aircraft maneuverability and performance improve, considering the lack of attitude and altitude effects. Furthermore, the aircraft can reach a lower level of vibration and present a reliable start-up.

But not everything in the garden is rosy. The amount of battery to provide a reasonable endurance and the minimum required power of the aircraft would be enormous, since the energy capacity of the batteries is currently very small, which would result in a large weight of

¹ Available at: <https://issuu.com/esmaaviationacademy/docs/airbus-global-market-forecast-2010->. [Accessed 5 September 2017].

batteries on board. In addition, the associated costs, availability, and project maturity are the major constraints.

Therefore, a synergy between the qualities of both conventional and full electric aircraft result in what is called hybridization, which means the integration of the propulsion system with energy stored source (batteries) and fuel source (conventional engine). The Degree-of-Hybridization (DoH) express the percentage of total power required by the aircraft that comes from the electric system [9]. Most commonly used in literature are the degree of hybridization for energy (H_E) and power (H_P) [17], defined as:

$$H_P = \frac{P_{electric}}{P_{total}} = S \quad (1)$$

and

$$H_E = \frac{E_{electric}}{E_{total}} = \psi \quad (2)$$

However, the degree of hybridization of power is not really a good parameter to measure how hybrid a design is [18]. Bogaert asserts:

“For example, having a large electric motor that is only for a short while will results in a large degree of hybridization of power while only a very small part of the mission is ‘hybrid’. In that regard, the degree of hybridization of energy is a better parameter. However, it is also not ideal since the specific energy of fuel is much larger compared to the battery specific energy, and the efficiency of the electrical systems much higher than that of the gas turbine. This results in values of H_E being generally quite low (< 0.2) even though the total supplied electric motor energy (E_{em}) might be higher than the total supplied gas turbine energy ($E_{gasturb}$).”

For this reason, another parameter is introduced: the supplied power ratio (Φ) [17]. This is defined as the total electric motor power over the entire mission in relation to the total shaft power over the entire mission.

$$\Phi = \frac{E_{em_{total}}}{E_{shaft_{total}}} \quad (3)$$

The advantage of this parameter is that it is more intuitive than the degree of hybridization of energy. A value of $\Phi = 0$ represents a conventional aircraft while a value of $\Phi = 1$ represents a fully electric aircraft. When using a constant power Split over the entire mission the supplied power ratio will be approximately the same as this constant power split [18].

Figure 2 displays an example functional correlation between installed H_P and H_E for a dual-energy storage-propulsion-power system (DESPPS) based upon kerosene and batteries as energy carriers targeting YEIS 2035 and were derived from assumed step values of Φ .

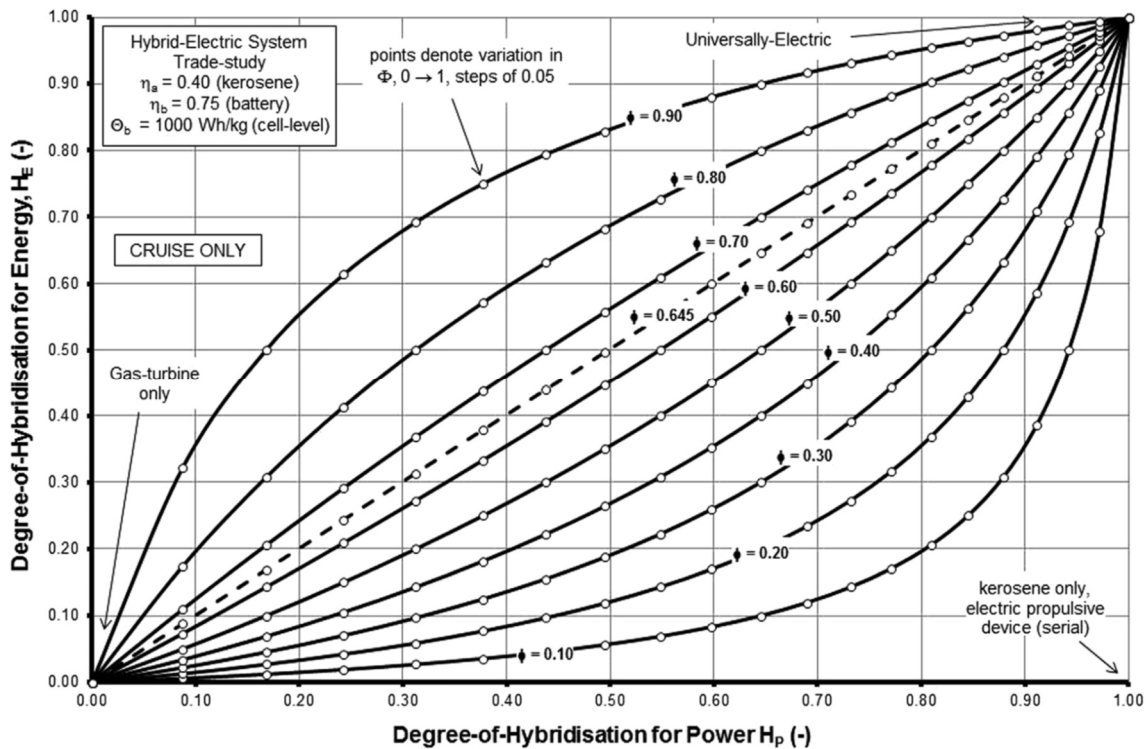


Figure 2 - Example of a Degree-of-Hybridization trade-study conducted for a hypothetical Dual-Energy Storage-Propulsion-Power System (YEIS 2035). Source: Isikveren et al. [17]

When designing a hybrid electric propulsion system, there are many distinct possible architectures [17]. The series and parallel hybrid architectures are the most commonly used. But, to choose which one of them is the best for a certain project, it depends on the applications and limitations of the project that they are being designed for. In road vehicles, for example, the series hybrid architecture has the lowest fuel consumption, as presented by Caiying Shen [11]. However, Hung points out that it has a larger weight than a parallel architecture, which is crucial and determinant in aerospace applications [12]. In a design study of light aircraft,

Friedrich and Robinson assert that the parallel configuration provides the highest efficiency for aerospace applications [13]. The series and parallel configurations are further discussed in the following sections.

2.1.1. Series Architecture

Series architecture involves an internal combustion engine (ICE), a generator, a battery pack, an inverter, a controller and electric motors, as shown in the simplified diagram in Figure 3. The ICE is used to drive a generator, which in the sequence provides power to the controller. This controller also receives power from an inverter, which drains power from a battery pack. Thus, the controller combines both powers from the ICE and the battery pack, and finally provides power to the electric motors.

The main benefit of a series architecture is that the ICE driving the generator can be designed to operate at a consistent and optimum engine speed [8], because the ICE is not directly mechanically linked to the propellant of the aircraft. Also, the arrangement can be installed in different positions on the aircraft layout system. A drawback to a series architecture is that the electric motor must be sized based on the capability to provide the maximum power output the aircraft requires.

If so designed, the aircraft can operate fully electric using the battery pack, turning the ICE off, operating at its maximum efficiency point, leading to improved fuel efficiency and lower carbon emissions compared to other configurations.

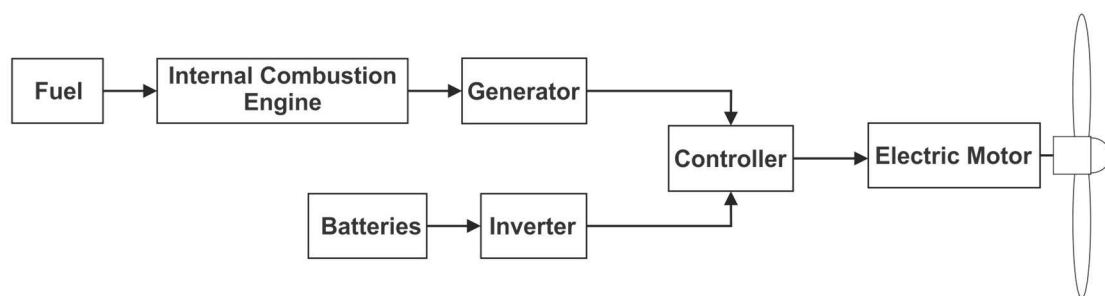


Figure 3 - Series architecture diagram.

2.1.2. Parallel Architecture

The parallel architecture comprises a turboshaft engine, a battery pack, an inverter, and an electric motor, as shown in the simplified diagram in Figure 4. Fuel is used to power a turboshaft engine, and batteries are used to power an electric motor. Both turboshaft engine and electric motor power the drive train coupled to the propellers.

Good performance is possible, in this case, because the power is generated with both engines [14]. Different control strategies are used in a preferred approach. If the power required by the transmission is higher than the output power of the turboshaft engine, the electric motor is turned on so that both engines can supply power to the transmission. If the power required by the transmission is less than the output power of the ICE, the remaining power is used to charge the battery pack [8]. Moreover, mechanical and electric power could be decoupled, and the system has a high operating flexibility enabling three modes of operation: purely combustion; purely electric and hybrid.

On the other hand, Miller comes up with other similar definitions, but in a different perspective. He splits the parallel hybrid-electric systems into three major types: mild, power-assist, and dual-mode [16]. The types are nominally classified based on the sizing and participation of the electric motor. In the mild system, for example, the electric motor is relatively small and it is used to aid in acceleration and utilizes excess power to recharge the batteries. The power-assist parallel system uses a larger electric-motor and a more substantial battery pack, what warrants a modest downsizing of the engine. Likewise, the dual-mode parallel system utilizes a yet even larger electric motor and larger energy storage bank (battery). In other words, the higher the participation of the electric-motor and the battery pack, the lower the turboshaft engine sizing.

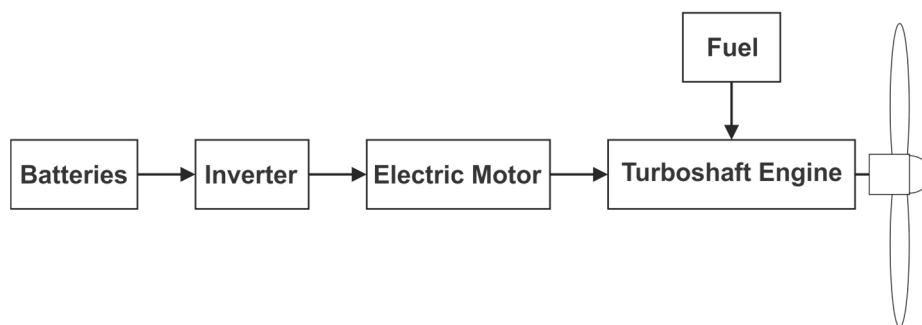


Figure 4 - Parallel architecture diagram.

3.2. Aircraft Design Theory

According to the convention commonly used by several manufacturers, and presented by Raymer [20], the aircraft development process is composed of the following sequential phases presented in Figure 5:

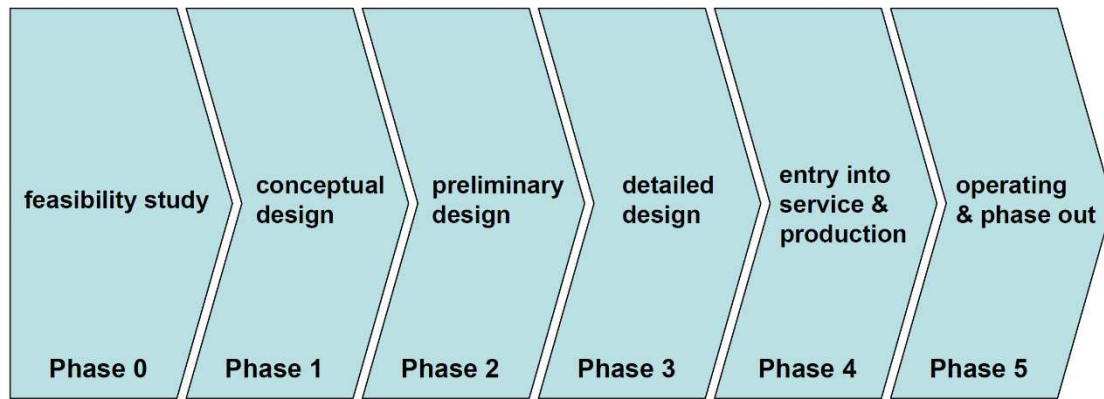


Figure 5 - Aircraft development process chart. Source: Adapted [19].

Phase 0 - Feasibility Study

The development of a new aircraft begins by defining the main flight requirements, e.g., range, payload, passenger capacity, cruise speed and service ceiling, which are defined through several analyzes of market, operators, buyers, and competitors. Next, a technical, economic and commercial feasibility study is made.

Phase 1 - Conceptual Design

In this phase, geometric and performance estimates are made for the aircraft through graphs, tables and analyzes of historical trends. But, throughout the conceptual design, new ideas and problems emerge as the project goes into details, making the process iterative, i.e., values are recalculated with each analysis.

In a simplified way, the aircraft's geometric and performance estimates follow a logical flow of calculation, as presented through the conceptual design diagram shown in Figure 6. The first stage of evaluation within the conceptual design usually involves the estimates of the appropriate weights of the aircraft. Next, there are design calculations of wing, empennage,

performance ratios (thrust-to-weight ratio and wing loading), control surfaces, ending with all dimensions of the aircraft.

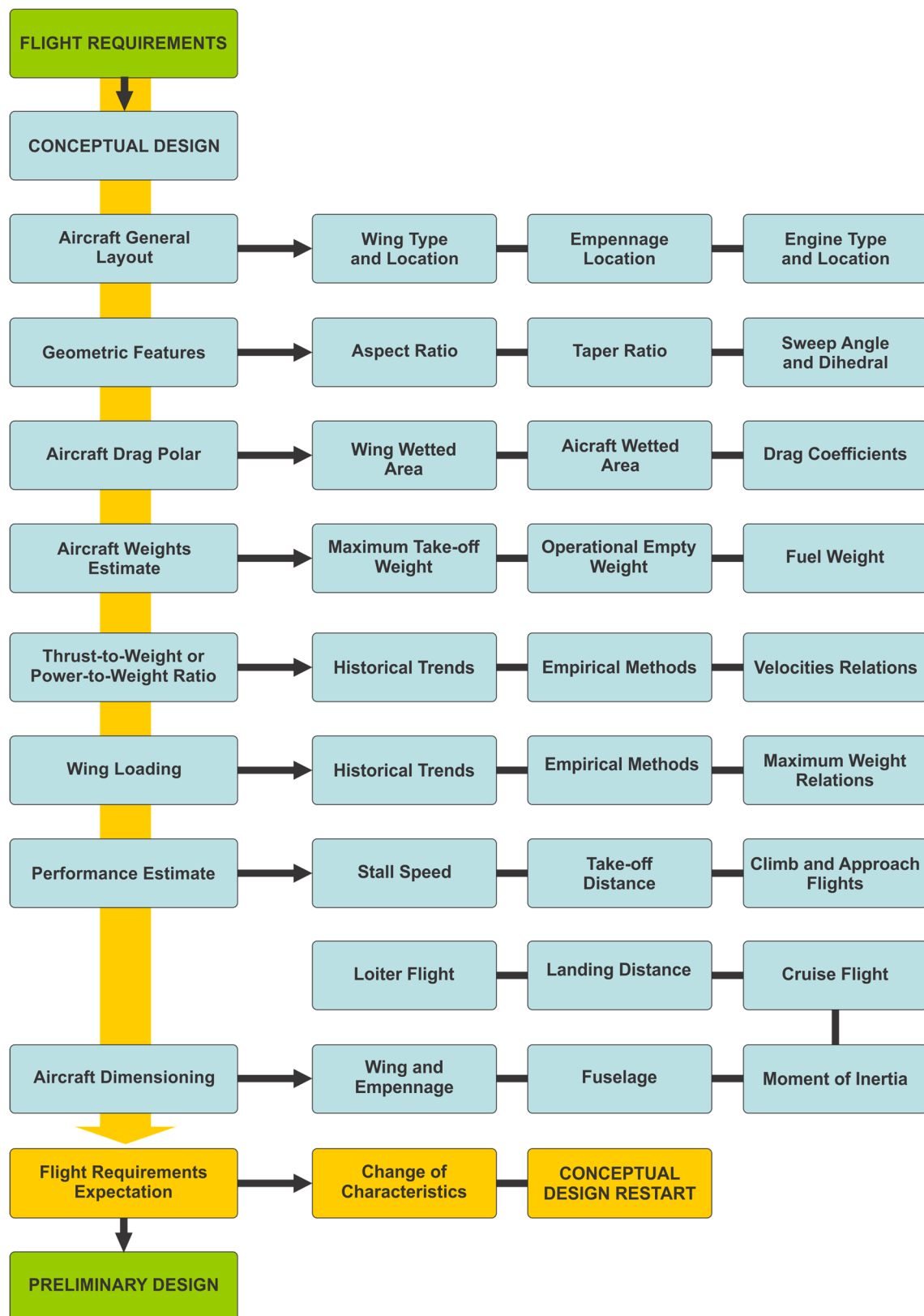


Figure 6 – Conceptual design diagram. Source: Venson [19].

Usually, aircraft commonality strategy is also evaluated within the conceptual design, which allows different models of the same family to have common characteristics, as well as other advantages such as reducing the development time of new aircraft, sharing production and maintenance tools and reduction of aircraft certification time.

Phase 2 - Preliminary Design

The preliminary design of the aircraft initially involves the aerodynamic and structural design and the systems integration project. At the beginning of this stage, although the structural calculation has not yet been carried out, concepts regarding the materials to be used are already defined such as the use of aluminum and/or composite materials.

In this phase, the predominant loads in the structure are defined and the stability and control criteria of the aircraft are evaluated. In addition, wind tunnel tests are performed to determine loads, stresses and also engine integration parameters, considering the chosen motor-propulsion group.

One of the most important stages of aircraft development occurs during the preliminary design, which is the "freezing" of the aircraft configuration, i.e., from that point on the design and the propulsion system do not change anymore (it can even change, but it compromises the project).

Phase 3 - Detailed Design

In this phase, 2D drawings are made for the fabrication of the parts of the aircraft structure, and the 3D drawings for the assembly, including the structure along with the components and systems. The drawings and specifications involve the complete structures detailing, manufacturing materials, manufacturing tolerances, type of heat treatment, surface finish and applicable standards. The production tooling and assembly plan are elaborated, together with the construction of the flight simulators (iron birds) and the prototype aircraft.

One of the main activities of the detailed design phase is the development of the functional test benches, mainly used to assess the structural and operational strength of the aircraft. Functional tests are defined by international regulations and involve some types of tests such as bird-strike, noise, ice, tires, and wing and fuselage fatigue tests (Figure 7).

Parallel to the functional tests, the flight test campaigns can be carried out with the prototype aircraft for qualification and certification. In some projects, some prototypes never fly; they are only used for static ground testing.



Figure 7 – Structural fatigue test in the fuselage. Source: Aviation Week².

At the end of the detailed design phase, after completing the entire flight test campaign for aircraft certification, the aeronautical authorities shall issue the type certificate, authorizing their operation.

Phase 4 - Entry Into Service & Production

With the conclusion of the detailed design and certification of the aircraft by the aeronautical authorities, the marketing phase begins, marked by its first delivery. Usually the company that receives the first aircraft had already been defined during the detailed design, when the orders and the purchase options began.

² Aviation Week (2015). Boeing Begins Analysis of 787 Fatigue Tests. Retrieved from <http://aviationweek.com/commercial-aviation/boeing-begins-analysis-787-fatigue-tests>. [Accessed 21 September 2017]

As aircraft are sold, new ones are delivered, and new purchase orders are made. In addition, the companies can also place purchase options order requests, which do not formalize the application, only demonstrate their intention to acquire more aircraft.

The purchase options are a vital part of the trading process. They show trends that allow the manufacturers to be prepared for a significant increase or decrease in their production.

Phase 5 - Operating Life & Phase Out

During the detailed design phase, an aircraft usage expectation is defined, called airplane useful life (or airplane service life). The life of the aircraft can be defined by their age, the number of hours flown, the number of operating cycles or even technical or operational unfeasibility.

Concerning technical unfeasibility, the current environmental legislations which differ from the legislations of the project year, limiting or prohibiting the use of motors with high noise levels and emissions of pollutants.

Regarding the operational unfeasibility, there are high operating costs associated with the high fuel consumption of old engines, maintenance costs of old aircraft, and airport taxes for engines with high noise level.

Even for an aircraft that has reached the end of its useful life, defined for example by their structure age, some devices can still be used in other applications. Since the life of different components is not the same, most aircraft, whether commercial or military, are not immediately discarded, reusing some of their parts and components.

CHAPTER III

AIAA DESIGN REQUIREMENTS AND PROPOSALS

“It always seems impossible until it's done.”

Nelson Mandela

The aircraft designed in this work is a hybrid-electric aircraft whose design requirements were determined through a Request for Proposal (RFP) published by the committees of the annual design competition sponsored by The American Institute of Aeronautics and Astronautics (AIAA)³.

Founded in 1963, the AIAA is a professional society for the field of aerospace engineering. It has earned an international reputation for being responsible for organizing and hosting the aerospace industry's most important conferences and events, where aerospace professionals exchange information, present findings, publish papers, and collaborate with each other [22].

Every year, the AIAA sponsors several design competitions, which offer an opportunity for students to participate in a simulated real-world problem, designing an aircraft, engine, or

³ AIAA (2017). Undergraduate Team Aircraft Design – Hybrid-Electric General Aviation Aircraft (HEGAA). Retrieved from <http://www.aiaa.org/2018UndergradTeamAircraftDesign%E2%80%93HEGAA/> [Accessed 15 September 2017]

solving other task determined by the committees. Thus, the AIAA Technical Committees elaborate a Request for Proposal (RFP) and launch it on public so that students (graduate and undergraduate) from different places in the world get involved.

For the year 2017-2018, the RFP is for the design of two-member Hybrid-Electric General Aviation Aircraft family, one for 4 passengers and another one for 6 passengers. The year entry-into-service (YEIS) is 2028 for a 4-seat model with 1000 nmi of range and 2030 for the 6-seat model with 750 nmi of range. The intent is to have energy storage for takeoff, climb, go-around and emergencies via batteries and electric motors with an engine providing additional power and/or direct propulsion [23]. Moreover, the airframe and propulsion system commonality, by weight, between the 4-seat and 6-seat variant should be 75% or greater of the 4-seater's empty weight.

The other requirements [23] are presented in Tables 1, 2 and 3.

Table 1 - AIAA general requirements.

General Requirements	Capable of taking off and landing from different runways (dirt, grass, metal mat, gravel, asphalt, and concrete)
	Minimum cruise speed of 174 knots
	Target cruise speed: 200 knots or greater
	Capable of VFR and IFR flight with an autopilot
	Capable of flight in known icing conditions
	Meets applicable certification rules in FAA 14 CFR Part 23
	Use of engine(s) that will be in service by 2028
	Use of electric motor(s) that will be in service by 2028 and document battery energy and power density assumptions based on reasonable technology trends
	Show airframe and propulsion system commonality of at least 75% between the 4-seater and 6-seater by weight
	Show the emergency range to get to an alternate airport at the maximum feasible weight from an engine failure at 5000 ft AGL (ISA + 18 deg F) with electric power from batteries alone for both the 4- and 6-seat variants
	Provide systems and avionics architecture that could enable autonomous flight; otherwise, provide a market justification for choosing to either provide or omit this capability

Table 2 - AIAA requirements for the aircraft of 4 passengers.

4-Seat Variant Mission Requirements	Crew: 1 pilot
	3 passengers
	Passenger/pilot weight of 190 lb
	Baggage weight per passenger/passenger of 30 lb and volume of at least 4 cubic feet per passenger
	1000 nmi design range mission with IFR reserves
	Maximum takeoff and landing field lengths of 1,500' over a 50' obstacle to a runway with dry pavement (sea level ISA + 18oF day)
	Takeoff, and landing performance should also be shown at 5,000' above mean sea level (ISA + 18oF) as well as for grass & concrete fields at sea level (ISA+18oF)
	Initial climb rate at sea level (ISA+ 18oF) at least 1500 fpm with both electric and fossil fuel propulsion operating
	Meet 14 CFR 23.67 Climb: One engine inoperative requirements with either propulsion type inoperative if it will be treated as a twin-engine airplane

Table 3 - AIAA requirements for the aircraft of 6 passengers.

6-Seat Variant Mission Requirements	Crew: 1 pilot
	5 passengers
	Passenger/pilot weight of 190 lb
	Baggage weight per passenger/pilot of 30 lb and volume of at least 4 cubic feet per passenger
	750 nmi design range mission with IFR reserves
	Maximum takeoff and landing field lengths of 1,800 ft over a 50 ft obstacle to a runway with dry pavement (sea level ISA + 18oF day)
	Takeoff, and landing performance should also be shown at 5,000 ft above mean sea level (ISA + 18 deg F) as well as for grass & concrete fields at sea level (ISA + 18 deg F)
	Initial climb rate at sea level (ISA + 18 deg F) at least 1300 fpm with both electric and fossil fuel propulsion operating
	Meet 14 CFR 23.67 Climb: One engine inoperative requirements with either propulsion type inoperative if it will be treated as a twin-engine airplane

CHAPTER IV

CONCEPTUAL DESIGN OF HYBRID-ELECTRIC AIRCRAFT

*“Science is about knowing;
engineering is about doing.”*

Henry Petroski

This work deals with the design of two hybrid-electric aircraft: one for four passengers (4PAX) and the other for six passengers (6PAX). Both aircraft are expected to have 75% or greater of communality. Therefore, the first aircraft to be designed was the 4PAX one, since the AIAA specifications require shorter field lengths for takeoff and landing, a higher rate of climb and a more significant range. Thus, after having that aircraft in hands, it is easier to adapt it to the 6PAX configuration, changing the empennage, but keeping the same wing.

The design of this hybrid-electric aircraft was obtained using the procedures and theories presented by Venson [24,25,26,27,28,29], and only the conceptual and preliminary design phases were evaluated.

4.1 Initial Design Estimate

First of all, in the conceptual design of the aircraft, several trend tables were used to obtain some initial estimates. But for that, it was necessary to assign some initial configurations to the aircraft. Thus, four main configurations were assumed: twin-engine with T-tail, twin-engine with conventional tail, single-engine with T-tail, and single-engine with conventional tail.

The algorithm used to estimate the design of the aforementioned aircraft considers several coefficients and characteristic values of conventional aircraft presented in historical tables present in the literatures [24,25,26,27,28,29]. Most of them are represented by a function defined by:

$$f(a, W_0, c) = aW_0^c \quad (4)$$

where $f(a, W_0, c)$ represents the characteristic to be calculated such as wetted area (S_{wet}), a and c are constants that depend on the type of aircraft under analysis, and W_0 is the gross weight estimate of the aircraft, which is iterated throughout the project.

This process of iteration and updating of the aircraft weight is performed considering the weight variation and performance during the phases of flight which the fuel consumption is higher: cruise and loiter. In [28,29] the weight ratios per phase are introduced, setting a correlation between them during the aircraft mission. The simplified scheme of iteration flow is presented in Figure 8 (first iteration) and Figure 9 (other iterations).

First Iteration

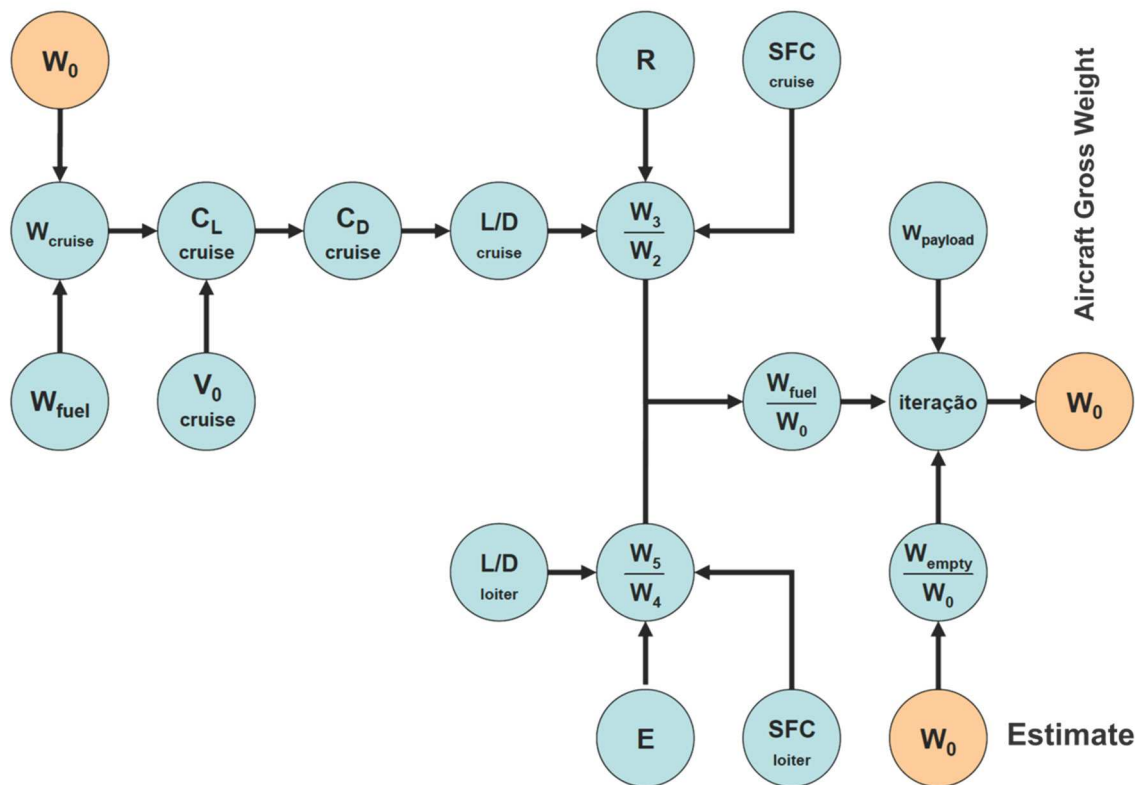


Figure 8 - First iteration for aircraft gross weight estimate. Source: Venson [26].

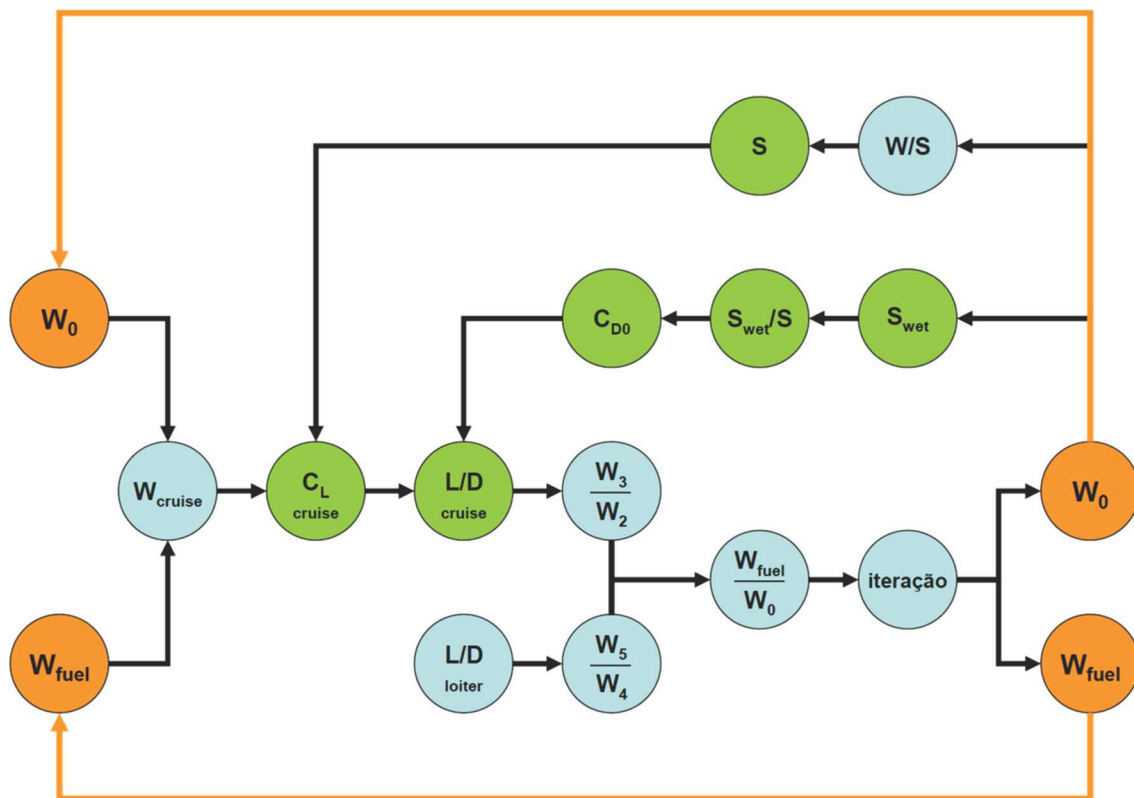


Figure 9 - Other iterations for aircraft gross weight estimate. Source: Venson [26].

Hence, the values used as respective inputs for each aircraft in the algorithm are presented in Table 4. It is worth remembering that these values come from historical tables mentioned above.

Table 4 - Algorithm inputs for each aircraft configuration.

	Twin-engine aircraft with T-tail	Twin-engine aircraft with conventional tail	Single engine aircraft with T-tail	Single engine aircraft with conventional tail
AR_W	7.8	7.8	7.2	7.2
λ_W	0.75	0.75	0.75	0.75
Λ_W [°]	22	22	22	22
e_w	0.82	0.82	0.83	0.83
$C_{L_{max}}$	1.7	1.7	1.7	1.7
AR_{HT}	4.4	4.3	4.1	39
λ_{HT}	0.5	0.5	0.5	0.5
V_{HT}	0.96	0.84	0.60	0.45
AR_{VT}	1.0	1.4	1.2	1.5
λ_{VT}	0.4	0.4	0.5	0.5
V_{VT}	0.071	0.050	0.030	0.030
$a_{S_{wet}}$	0.2933	0.2933	0.6762	0.6762
$c_{S_{wet}}$	0.5632	0.5632	0.4884	0.4884
a_{W_0}	766	0.766	0.892	0.892
c_{W_0}	-0.20	-0.020	-0.047	-0.047
$a_{W/S}$	1.512	1.512	0.408	0.408
$c_{W/S}$	0.664	0.664	0.804	0.804
a_T	0.0116	0.0116	0.0116	0.0116
c_T	0.4789	0.4789	0.4789	0.4789
S_{Wet}/S_{Ref}	4.0	4.0	4.5	4.5
C_{fe}	0.055	0.055	0.055	0.055
W_1/W_0	0.984	0.984	0.990	0.990
W_2/W_1	0.990	0.990	0.992	0.992
W_4/W_3	0.992	0.992	0.993	0.993
W_6/W_5	0.992	0.992	0.993	0.993

$a_{D_{TO}}$	9.68	9.68	8.23	8.23
$a_{D_{LND}}$	1.463	1.463	1.524	1.524
$a_{fuselage}$	0.4088	0.4088	0.4088	0.4088
$c_{fuselage}$	0.3140	0.3140	0.3140	0.3140
x_L/d	8.3	8.3	8.3	8.3
x_N/d	1.3	1.3	1.3	1.3
x_C/d	0.56	0.56	0.56	0.56
x_T/d	2.3	2.3	2.3	2.3
R_x	0.34	0.34	0.25	0.25
R_y	0.38	0.38	0.38	0.38
R_z	0.39	0.39	0.39	0.39
$C_{L_{clean}}$	1.6	1.6	1.5	1.5
$C_{L_{TO}}$	1.9	1.9	1.8	1.8
$C_{L_{LND}}$	2.3	2.3	2.0	2.0
Aileron wing Ratio	0.05	0.05	0.07	0.07
Elevator Tail Ratio	0.44	0.44	0.41	0.41
Rudder Tail Ratio	0.38	0.38	0.36	0.36
Main gear at mac	0.52	0.52	0.50	0.50
Nose gear at length	0.17	0.17	0.13	0.13
Inner engine span	0.34	0.34	0.30	0.30

Besides the inputs from Table 4, the algorithm needed an engine to be considered during the performance analysis.

Since both aircraft were going to be hybrid, the strategy used here was to get started choosing an electric motor instead of a conventional engine. Searching for commercial and available electric motors, the Siemens' SP260D was chosen due to its application in electric aircraft, such as the Extra 330LE, also developed by Siemens. Moreover, this electric motor provides 260 kW along with a significant efficiency of 95%, which would be sufficient for both aircraft, and weighs only 50 kg, saving a lot of weight. Further details and technical information are available in [30].



Figure 10 – Electric motor SP260D (on the left) and Siemens Extra 330LE (on the right). Source: Endless Sphere⁴.

Thus, after running the algorithm, it was released the following outputs for each aircraft, shown in Table 5.

Table 5 - Algorithm outputs for each aircraft configuration.

	Twin-engine aircraft with T-tail	Twin-engine aircraft with conventional tail	Single engine aircraft with T-tail	Single engine aircraft with conventional tail
MTOW [kg]	1880.10	1880.10	1466.20	1466.20
Fuel Weight [kg]	297.60	297.60	233.07	233.07
Empty Weight [kg]	1183.30	1183.30	833.97	833.97
Payload Weight [kg]	399.20	399.20	399.20	399.20
Structural Weight [kg]	487.97	487.19	433.03	433.14
b_w [m]	11.83	11.83	10.74	10.74
c_{r_w} [m]	1.73	1.73	1.70	1.70
c_{mac_w} [m]	1.53	1.53	1.50	1.50
y_{mac_w} [m]	2.82	2.82	2.56	2.56
S_w [m ²]	17.94	17.94	16.01	16.01

⁴ Reprinted from *Endless Sphere*, by LockH, 2017, Retrieved from <https://endless-sphere.com/forums/viewtopic.php?t=81667>. [Accessed 12 November 2017]

$C_{L_{cruise}}$	0.26	0.26	0.22	0.22
CD_0	0.02	0.02	0.02	0.02
k_2	0.05	0.05	0.05	0.05
x_w [m]	2.65	2.72	3.21	3.10
Service Ceiling [ft]	12000.00	12000.00	12000.00	12000.00
Mach (cruise)	0.28	0.28	0.28	0.28
V_{stall} [m/s]	31.96	31.96	29.88	29.88
W_1/W_0	0.98	0.98	0.99	0.99
W_2/W_1	0.99	0.99	0.99	0.99
W_3/W_2	0.89	0.89	0.89	0.89
W_4/W_3	0.99	0.99	0.99	0.99
W_5/W_4	0.98	0.98	0.98	0.98
W_6/W_5	0.99	0.99	0.99	0.99
Take-off Distance [m]	268.05	268.05	290.45	290.45
Landing Distance [m]	747.39	747.39	680.47	680.47
Range [km]	1228.00	1228.00	1094.60	1094.60
ROC [ft/min]	2483.90	2483.90	1377.20	1377.20
$l_{fuselage}$ [m]	8.52	8.52	8.52	8.52
$d_{fuselage}$ [m]	1.70	1.70	1.70	1.70
S_{HT} [m ²]	7.71	6.75	3.53	3.53
$c_{r_{HT}}$ [m]	1.77	1.67	1.16	1.31
b_{HT} [m]	5.83	5.39	3.80	3.71
$c_{mac_{HT}}$ [m]	1.37	1.30	0.95	1.00
x_{HT} [m]	6.75	6.85	7.36	7.21
S_{VT} [m ²]	4.42	3.11	1.51	1.51
$c_{r_{VT}}$ [m]	3.00	2.13	1.58	1.39
b_{VT} [m]	2.10	2.09	0.87	0.87
$c_{mac_{VT}}$ [m]	2.23	1.58	1.74	1.76
x_{VT} [m]	5.52	6.39	6.94	7.13
$x_{Landing.gear}$ [m]	4.43	4.43	4.26	4.26
y_{engine} [m]	1.45	1.45	0.00	0.00

I_{xx} [kg·m ²]	7602.20	7602.20	2640.20	2640.20
I_{yy} [kg·m ²]	4926.80	4926.80	3842.30	3842.30
I_{zz} [kg·m ²]	393790.00	393790.00	213500.00	213500.00
x_{CG} (empty aircraft) [m]	3.81	3.82	4.15	4.14
x_{CG} (loaded aircraft) [m]	3.67	3.68	3.89	3.88
Sale Price [US\$]	1062300.00	1061200.00	893590.00	893750.00
Operating Costs per Year [US\$/year]	35092.00	35092.00	33082.00	33082.00
Operating Costs per Hour [US\$/hr]	116.97	116.97	116.74	116.74

Thus, gathering the results from Table 5, the following aircraft configurations were obtained and are presented in Figure 11.

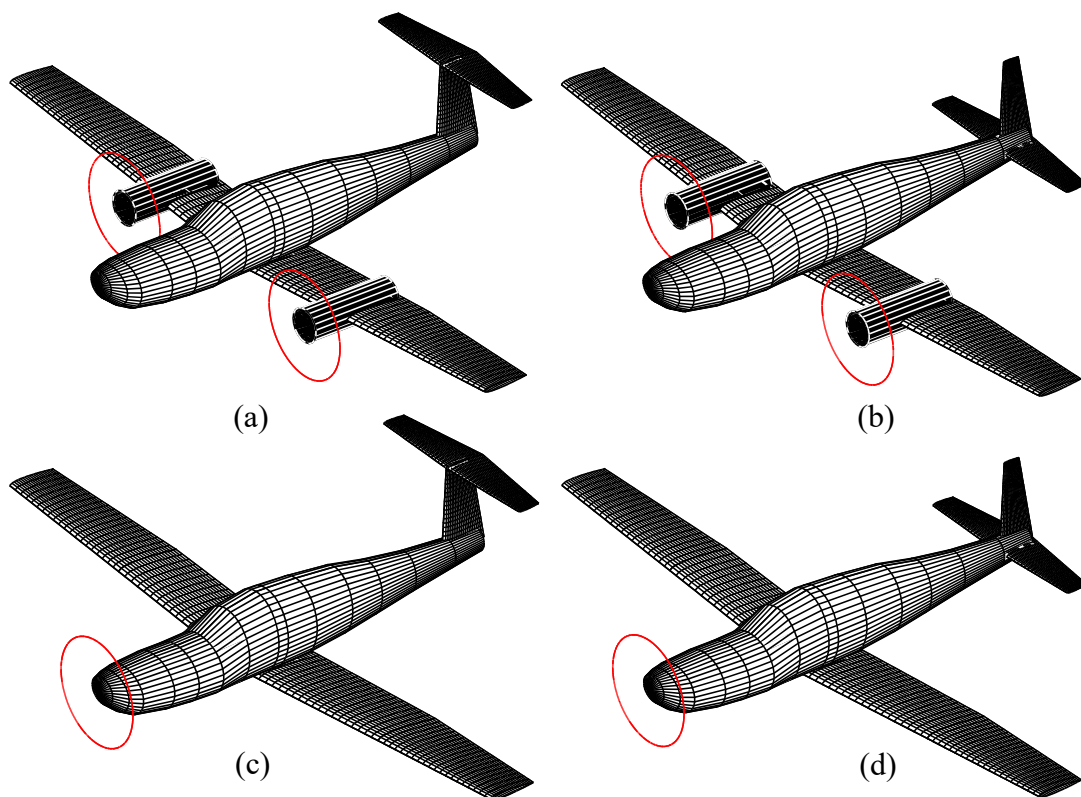


Figure 11 – Sketch of the 4 aircraft released by the initial concept design estimate: (a) twin-engine with T-tail, (b) twin-engine with conventional tail, (c) single engine with T-tail, (d) single engine with conventional tail.

Analyzing the AIAA design requirements with the results from Table 5, the aircraft that best satisfies the specifications are the single engine aircraft, either with conventional or T-tail. In both cases, the single engine provides sufficient power for the aircraft to fly and reaches the required performance. So, it was easy to choose only one engine. But for the tail, it goes beyond that.

The design of an empennage for any aircraft is extremely important, since it affects the aircraft mass and center of gravity, i.e., the static and dynamic stability. Then, it is necessary to consider the effects of both empennages arrangement.

The conventional tail provides appropriate stability and control, and also leads to the most lightweight construction in most cases, so much so that approximately 70% of aircraft are fitted with it [31]. Moreover, for this configuration, the stabilizer trim is relatively less complex and the vertical tail is usually larger. However, engines cannot be coupled to the rear of the aircraft, what is useful for static stability. Furthermore, spin characteristics can be bad in the case of conventional tail due to the blanketing of the vertical tail, in addition to the downwash of the wing being relatively large in the area of the horizontal tail.

On the other hand, the T-tail is heavier than the conventional tail because the vertical tail has to support the horizontal tail. However, the T-tail has advantages that partly compensate this important disadvantage (weight). Because of the end plate effect, the vertical tail can be smaller. In addition, the horizontal tail is more effective because it is positioned out of the airflow behind the wing and is subjected to less downwash. Therefore, it can be smaller. For the same reason, the horizontal tail is also subject to less tail buffeting. As the T-tail creates space at aircraft's rear, there is enough space to fix the engines, improving static stability.

Hence, since the aircraft is going to have a lightweight engine at the nose, considering the information above and aiming an easier structural analysis, the conventional empennage was chosen for this aircraft design.

Thus, at this point an overall configuration for the aircraft had been defined. However, it was necessary to improve and refine the aircraft adjusting equations and algorithms for it to become a hybrid-electric one. In other words, the performance equations had to be solved using the hybrid-electric architectures presented in Section 2.1, what would imply in new results and configurations. But, first of all, the propulsion system architecture had to be determined.

4.2 Hybrid-Electric Aircraft Design

Several papers indicate the parallel architecture as the most appropriate arrangement for large aircraft due to the lower weight associated, since the batteries only feed an electric generator, which helps the main shaft of a turbojet engine, for example. However, the aircraft designed here will be small, carrying four and six passengers, what makes the architecture simpler. Therefore, the series-architecture was chosen for the aircraft, since the engine is already an electric-motor, requiring only electric power to move it. Figure 12 depicts the propulsion system architecture proposed.

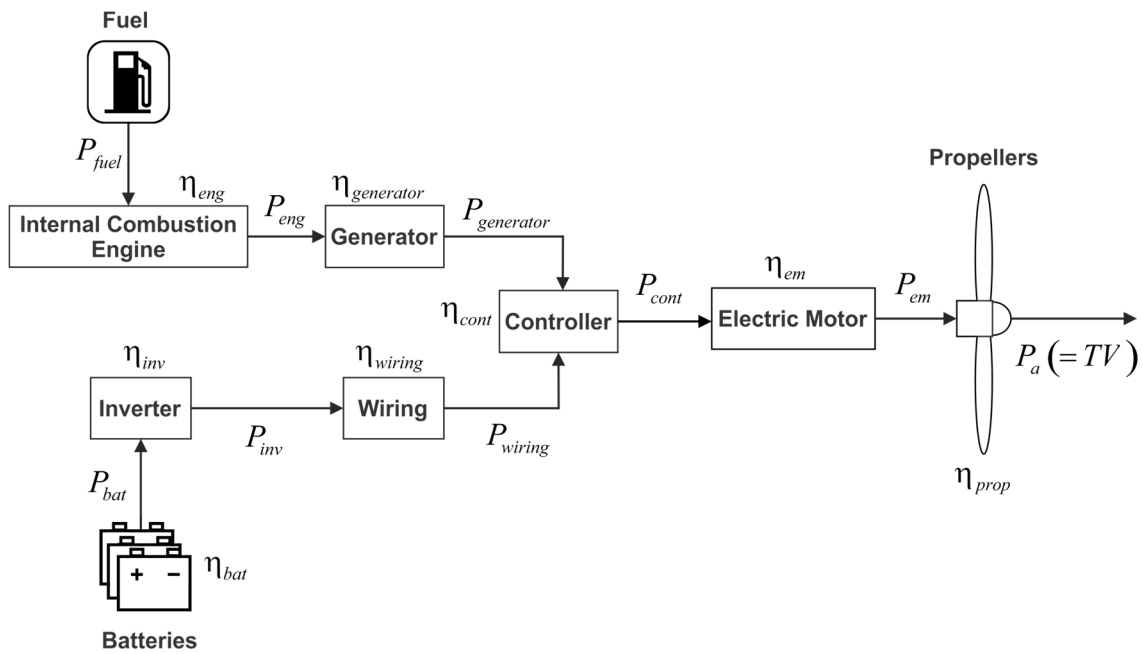


Figure 12 - Series-architecture used in both 4PAX and 6 PAX aircraft.

For the architecture above, all efficiencies were assumed base on literatures and papers, as presented in Table 6.

Table 6 - Efficiencies for the hybrid-electric propulsive system.

Efficiency	Value
η_{bat}	90% ^[35]
η_{inv}	95% ^[36]
η_{wiring}	97% ^[37]
η_{cont}	99% ^[38]
η_{eng}	60% ^[39]
$\eta_{generator}$	95% ^[40]
η_{em}	95% ^[40]
η_{prop}	85% ^[41]

Now it is necessary to come up with the new performance analysis. First, the classic Bréguet's range is formulated for a conventional fuel powered aircraft at zero wind conditions, and thrust vector parallel to the airspeed vector, which defines the following:

$$R = \int_{W_{final}}^{W_{start}} \frac{V}{c_T} \frac{C_L}{C_D} \frac{1}{W} dW \quad (5)$$

where c_T means the specific fuel consumption.

The solution to the integral depends on the flying strategy used (e.g., gradual climb at a constant airspeed and angle of attack) and the models used to represent the propulsion system and the aerodynamic characteristics. Solutions to the most common flying strategies can be found in many standard textbooks on aircraft performance. But, fundamentally, these solutions use the idea that weight changes gradually throughout the flight. Therefore, the range equation cannot be applied to the hybrid aircraft to be designed here, which uses an electric system (batteries) as energy supply.

Thus, Voskuijl presents in [32] a new formulation for the range equation relating the energy stored to the consumption of a hybrid aircraft with parallel architecture. Since the architecture of both aircraft under analysis here is going to be in series, the formulation is reevaluated for the scheme available in Figure 12, which is presented in the sequence.

Fundamentally, the range can be determined by integrating speed over time:

$$R = \int_{t_{start}}^{t_{final}} V dt \quad (6)$$

The energy stored in batteries and fuel can be related to the time to solve this. As times goes by, the energy stored in the batteries or as fuel reduces. The power (energy per second) consumed by the aircraft over the time can be expressed using both variation in time:

$$\frac{dE}{dt} = \frac{dE_{fuel}}{dt} + \frac{dE_{bat}}{dt} \quad (7)$$

Analyzing the first addend, the time rate of change of the energy stored in the fuel can be related to the fuel flow and the energy density of the fuel. Furthermore, the shaft power delivered by the turboshaft engine is related to the fuel flow through the power-specific fuel consumption.

$$Q_f = -\frac{dW_{fuel}}{dt} = -\frac{g}{H_{fuel}} \frac{dE_{fuel}}{dt} \quad (8)$$

$$Q_f = c_p P_{eng} \quad (9)$$

The power provided by the internal combustion engine (P_{eng}) can be specified in terms of its participation in the total aircraft power required, degree of hybridization and the respective efficiencies:

$$P_{eng} = (1-S) \frac{P_a}{\eta_{generator} \eta_{cont} \eta_{em} \eta_{prop}} \quad (10)$$

Thus, the time rate of energy stored in fuel can be expressed as a function of the aircraft power available:

$$\frac{dE_{fuel}}{dt} = -\frac{c_p}{\eta_{generator} \eta_{cont} \eta_{em} \eta_{prop}} \frac{H_{fuel}}{g} (1-S) P_a \quad (11)$$

Analyzing the second addend along the same lines, the time rate of energy stored in the batteries can be expressed in terms of the power available:

$$\frac{dE_{bat}}{dt} = -\frac{S}{\eta_{inv} \eta_{wiring} \eta_{cont} \eta_{em} \eta_{prop}} P_a \quad (12)$$

Combining both Eqs. 11 and 12, an equation for the change of total energy in time is presented:

$$\frac{dE}{dt} = -\left(\frac{c_p}{\eta_{generator} \eta_{cont} \eta_{em} \eta_{prop}} \frac{H_{fuel}}{g} (1-S) + \frac{S}{\eta_{inv} \eta_{wiring} \eta_{cont} \eta_{em} \eta_{prop}} \right) P_a \quad (13)$$

In quasi-steady and quasi-rectilinear flight, the thrust must equal drag and lift equal weight. Assuming the thrust vector is parallel to the airspeed vector, the following relation is obtained:

$$P_a = TV = DV = \frac{C_D}{C_L} WV \quad (14)$$

Now the basic range equation for hybrid-electric cruise flight can be created by combining Eqs. 6, 13 and 14.

$$R = \int_{E_{final}}^{E_{start}} \frac{1}{\left(\frac{c_p}{\eta_{generator}\eta_{cont}\eta_{em}\eta_{prop}} \frac{H_{fuel}}{g} (1-S) + \frac{S}{\eta_{inv}\eta_{wiring}\eta_{cont}\eta_{em}\eta_{prop}} \right)} \frac{C_L}{C_D} \frac{1}{W} dt \quad (15)$$

It is assumed that the power split and angle of attack are kept constant throughout the cruise flight. Furthermore, for the range of flight speeds and altitudes of interest, the power-specific fuel consumption and all efficiencies are considered constant.

In case of a flight at constant airspeed, this results in a gradual climb. Since the fuel consumption is lower than for conventional aircraft, the weight variation is lower, resulting in a “less steep” climb. If it were to consider a full electric aircraft, there would not be climb, because there is no weight variation, allowing a horizontal flight profile.

In Eq. 15, the aircraft weight is the sum of the following components:

$$W = W_{empty} + W_{payload} + W_{battery} + W_{fuel} \quad (16)$$

Rewriting the $W_{battery}$ and W_{fuel} in terms of energy based on their respective energy densities, it is obtained:

$$W = W_{empty} + W_{payload} + \left(\frac{E_{battery} H_{fuel} + E_{fuel} H_{battery}}{H_{battery} H_{fuel}} \right) g \quad (17)$$

Inserting the degree of hybridization in terms of energy (Section 2.1), and relating the participation of the batteries and the fuel in the total aircraft energy, it comes:

$$W = W_{empty} + W_{payload} + \left(\frac{\psi H_{fuel} + (1-\psi) H_{battery}}{H_{battery} H_{fuel}} \right) gE \quad (18)$$

The energy densities, payload weight, empty weight and degree of hybridization in terms of energy are constant. Hence, substituting Eq. 18 into Eq. 15, and solving the integral, the final hybrid range equation is obtained:

$$R = \frac{\eta_{cont}\eta_{em}\eta_{prop}}{g \left(\frac{c_p}{\eta_{generator}} \frac{H_{fuel}}{g} (1-S) + \frac{S}{\eta_{inv}\eta_{wiring}} \right)} \frac{C_L}{C_D} \frac{H_{battery}H_{fuel}}{\psi H_{fuel} + (1-\psi)H_{battery}} \ln \left(\frac{\left(\psi H_{fuel} + (1-\psi)H_{battery} \right) g E_{start} + W_{empty} + W_{payload}}{H_{battery}H_{fuel}} \frac{W_{empty} + W_{payload}}{W_{empty} + W_{payload}} \right) \quad (19)$$

4.3. Design Assumptions

Before finishing the concept design, some aspects were added and settled for the development of both aircraft (4PAX and 6PAX), such as: typical aircraft mission profile, the batteries and an internal combustion engine (ICE) for the hybrid system.

4.3.1. Typical Aircraft Mission Profile

Based on historical data, FAR – Part 23 and AIAA specifications, some mission characteristics were chosen and are presented in Table 7. The flight level FL120 was chosen to avoid any pressurization issues, since the regulation 14 CFR 91 requires supplemental oxygen system for aircraft flying over FL120.

Also, the minimum cruise speed required by the AIAA is 174 knots. Thus, the chosen cruise speed for this project was chosen to be 185 knots, since similar aircraft, such as the Diamond DA42, usually fly around this velocity during cruise. Besides, the rate of climb of 1500 and 1300 fpm were kept the same as specified, and the rate of descent were assumed 900 fpm from trends in [50]. The ranges for both aircraft are also fixed by AIAA and the estimate of endurance in cruise is obtained dividing these ranges by the respective cruise speed. Additionally, the total endurance is an estimate of the time to reach the AIAA requirements (climb and cruise), the assumptions (descent) and the FAR Sec. 125 regulations (taxi, take-off, approach, loiter and landing).

Table 7 - Chosen mission characteristics for both aircraft design.

Mission Characteristics	4PAX	6PAX
Service ceiling [ft]	12000	12000
Cruise speed [knots]	185	185
Rate of climb [ft/min]	1500	1300
Rate of descent [ft/min]	900	900
Range in cruise [km]	1860	1340
Endurance in cruise [hr]	5.43	3.9
Total endurance [hr]	7.22	5.79

Thus, evaluating the time spent during all phases of flight, reaching the parameters in Table 7, typical flight mission for the 4PAX and 6PAX aircraft regarding altitude and flight time are shown in Figure 13 and Figure 14, respectively. For example, climbing at a rate of 1500 fpm would take 2% of the total endurance to reach the service ceiling of 12000 ft.

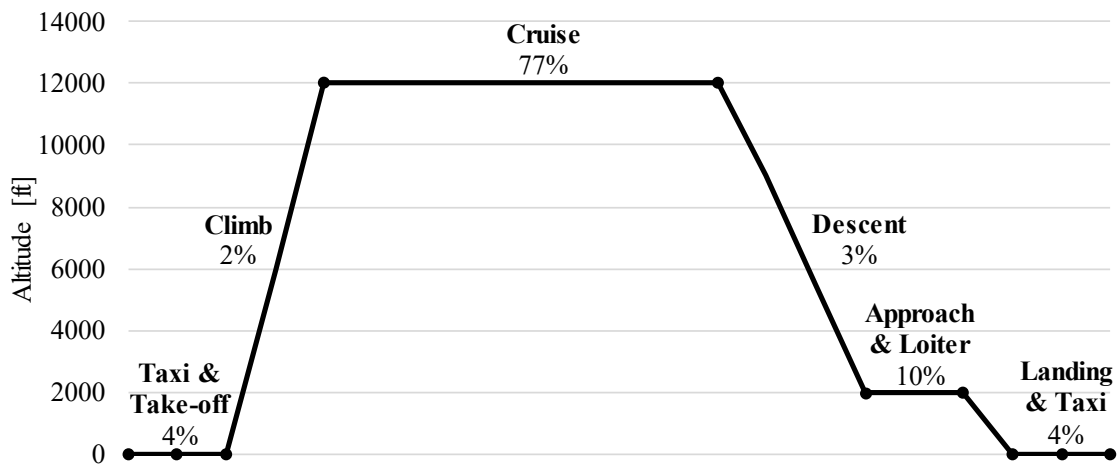


Figure 13 – 4PAX aircraft typical flight mission.

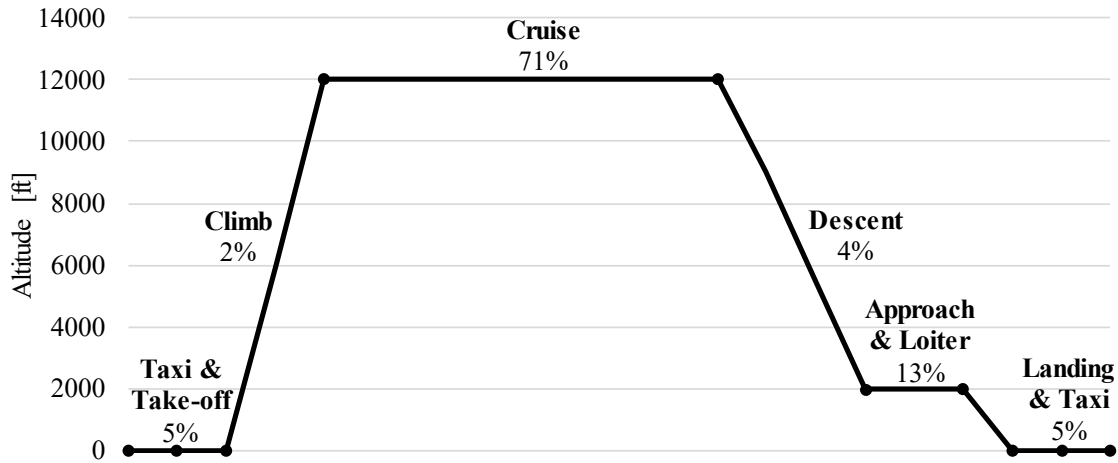


Figure 14 - 6PAX aircraft typical flight mission.

4.3.2. Battery Design

The choice of batteries to be used in the aircraft hybrid system was based on the estimates proposed by Hepperle [42] and Girishkumar [43] for the technology available in 2025, which are shown in Table 8. The theoretical specific energy values are not truly feasible in actual applications. The amount of energy that the batteries are able to provide are much less than the expected. In other words, there is an efficiency associated to these chemical reactions involved. Thus, when choosing a battery system, it is very important to consider these effects.

Table 8 - Specific energy density of current and future chemical battery systems. Source: Adapted [42].

System	Theoretical specific energy density	Actual battery energy density expected in 2025
Li-Ion	390 Wh/kg	250 Wh/kg
Zn-O ₂	1090 Wh/kg	400-500 Wh/kg
Li-S	2570 Wh/kg	500-1250 Wh/kg
Li-O ₂	3500 Wh/kg	800-1750 Wh/kg

Later, it was performed an assessment of the feasibility of using such batteries in both aircraft in 2028, as required by AIAA specifications.

According to Bruce [44] and Ji [45], the researches with Li-S and Li-O₂ batteries are in full development and implementation. However, the reliability and safety of these types of batteries are still unknown, which makes them unviable for aerospace applications expected in 2028.

On the other hand, Zn-O₂ batteries have been developed since the 1960s and already have medical and telecommunication applications [46]. In addition, companies such as Teck Cominco Metals Ltd.[®] and Tesla Motors, Inc.[®] hold patents for the application of this type of battery in the automotive industry [48,49]. Thus, due to its energy density (higher than for Li-Ion), maturity of their behavior and safety, which is crucial for aerospace applications, the type of battery chosen for both aircraft was the Zn-O₂. Using a more conservative approach, the energy density that will be considered for this battery will be 400 Wh/kg.

4.3.2. Internal Combustion Engine (ICE)

To choose an internal combustion engine (ICE) that better fits the aircraft selected in Section 4.1, it was compared the viability of two turbo-engines. The first one is a 4-cylinder engine with rated power below the power required for cruising (145 kW) and climb (215 kW), while the second one is a 6-cylinder engine with power output greater than those required in both cruise and climb conditions. Table 9 shows the main characteristics of each ICE.

Table 9 - Main characteristics of Austro AE330⁵ and Lycoming IO-580⁶ engines.

Characteristic	Austro AE330	Lycoming IO-580
Number of Cylinders	4	6
Rated Power [kW]	134	220
Mass [kg]	186	201
Dimensions [m]	0.74 x 0.85 x 0.57	0.99 x 0.87 x 0.53
Fuel consumption [kg/hr]	31.20	47.68

⁵ Austro Engine (2017). AE330 Fact Sheet. Retrieved from http://austroengine.at/uploads/pdf/mod_products9/AE330FactSheet.pdf. [Accessed 02 December 2017]

⁶ Lycoming (2015). IO-580-B1A Operation and Installation Manual. Retrieved from <https://www.lycoming.com/content/operator%27s-manual-IO-580-B1A-60297-28>. [Accessed 02 December 2017]

The degree of hybridization was considered so that the ICE used its maximum capacity during taxi & take-off, climb and cruise phases. For the remaining phases, aiming to reduce noise during approach, loiter and landing, the ICE would stay in idle, and all the power required for that phase would be supplied by the batteries.

Table 10 shows the comparison of the 4-cylinder engine with the proposed hybridization, and Table 11 shows a 6-cylinder engine with a higher power than required and without hybridization.

Table 10 - 4-cylinder engine with proposed hybridization.

Phase of Flight	S	Time [hr]	Battery Weight [kg]	Fuel Weight [kg]
Take-off + Climb	0.4126	0.13	31.37	4.16
Cruise	0.1092	5.52	226.62	172.41
Loiter	1.0000	0.50	46.83	0.00
Descent + Landing	1.0000	0.22	20.81	0.00
TOTAL	-	6.37	325.63	176.57

Table 11 - 6-cylinder with higher power required and without hybridization.

Phase of Flight	S	Time [hr]	Battery Weight [kg]	Fuel Weight [kg]
Take-off + Climb	0.0000	0.13	0.00	6.32
Cruise	0.0000	5.52	0.00	266.48
Loiter	0.0000	0.50	0.00	23.68
Descent + Landing	0.0000	0.22	0.00	10.52
TOTAL	-	6.37	0.00	307.01

Analyzing costs, the 4-cylinder engine has an estimated cost of US\$106.64 in fuel [51] and US\$25.84 for the total recharge of the batteries [52], resulting in a total cost of US\$132.48 per flight, while the 6-cylinder engine has a total cost of US\$185.42 per flight [51].

Therefore, there is a saving of US\$52.94 per flight when using the Austro AE330 engine. In addition, the 4-cylinder engine is 38 kg lighter and has a 27% lower volume than the 6-cylinder engine, which justifies its application in conjunction with the hybridization of the aircraft.

4.4. Genetic Algorithm Implementation

After formulating the new range equation for hybrid-electric aircraft, an optimization algorithm was implemented in MATLAB to find the best aircraft configuration.

The type of optimization implemented was based on the differential evolution algorithm proposed by Rainer Storn [53]. In this method, 300 generations were created, where each generation contained 10 members per population. However, because of the several iteration variables, the genetic algorithm requires too much processing, which generates a huge computational cost, spending many hours to reach the final result. In this case, for example, it took an average of 15 hours to end the optimization, which justifies the low value of 10 members per population.

The input variables for the objective function were generated based on the values of geometry, weight and performance from the results presented in Section 4.1. Table 12 shows the maximum and minimum values used in each optimization variable.

Table 12 - Maximum and minimum values for each optimization variable.

Parameter	Max	Min
Takeoff Engine Power [kW]	200.000	260.000
Climb Engine Power [kW]	200.000	260.000
Wing position along the aircraft's longitudinal axis [m]	3.238	5.397
Wing span [m]	8.033	13.388
Wing root-chord [m]	1.313	2.188
Wing taper ratio	0.525	0.875
Wing swept angle [°]	0.000	10.000
Wing breaking-point [m]	1.339	2.231
Wing dihedral [°]	0.000	5.000
Wing incidence angle [°]	0.000	3.000
Wing twist [°]	0.000	3.000
Wing profile	1.000	5.000
Horizontal tail span [m]	1.134	1.890
Horizontal tail taper ratio	0.525	0.875
Horizontal tail root-chord [m]	0.794	1.324
Horizontal tail swept angle [°]	0.000	5.000
Horizontal tail profile	1.000	3.000
Vertical tail span [m]	1.126	1.876
Vertical tail taper ratio	0.525	0.875
Vertical tail root-chord [m]	0.883	1.472
Vertical tail swept angle [°]	0.000	5.000
Vertical tail profile	1.000	2.000
Fuel Weight [kg]	54.348	380.437

Thus, the genetic algorithm used these inputs to find the best configuration option for the aircraft, meeting the AIAA requirements and achieving a lower gross weight as optimization variable.

Different airfoil profiles were used for the objective function to be used in the wings and empennage. All of them were also optimization variables:

- Wing: NACA 23010, NACA 23012, NACA 23015, NACA 23016 and NACA 63415;
- Horizontal Tail: NACA 0012, NACA 0009 and NACA 2412;
- Vertical Tail: NACA 0012 and NACA 0009.

For the trimming, aerodynamics and dynamics stability analyses, it was necessary to use a program to aid in the process. For this genetic algorithm, the chosen program was the AVL (Athena Vortex Lattice).

AVL is a program developed at MIT for the aerodynamic and flight-dynamic analysis of rigid aircraft of arbitrary configuration. As described on its website [33]: “it employs an extended vortex lattice model for the lifting surfaces, together with a slender-body model for fuselages and nacelles. General nonlinear flight states can be specified. The flight dynamic analysis combines a full linearization of the aerodynamic model about any flight state, together with specified mass properties”. An example of how the AVL deal with aircraft geometries is presented in Figure 15.

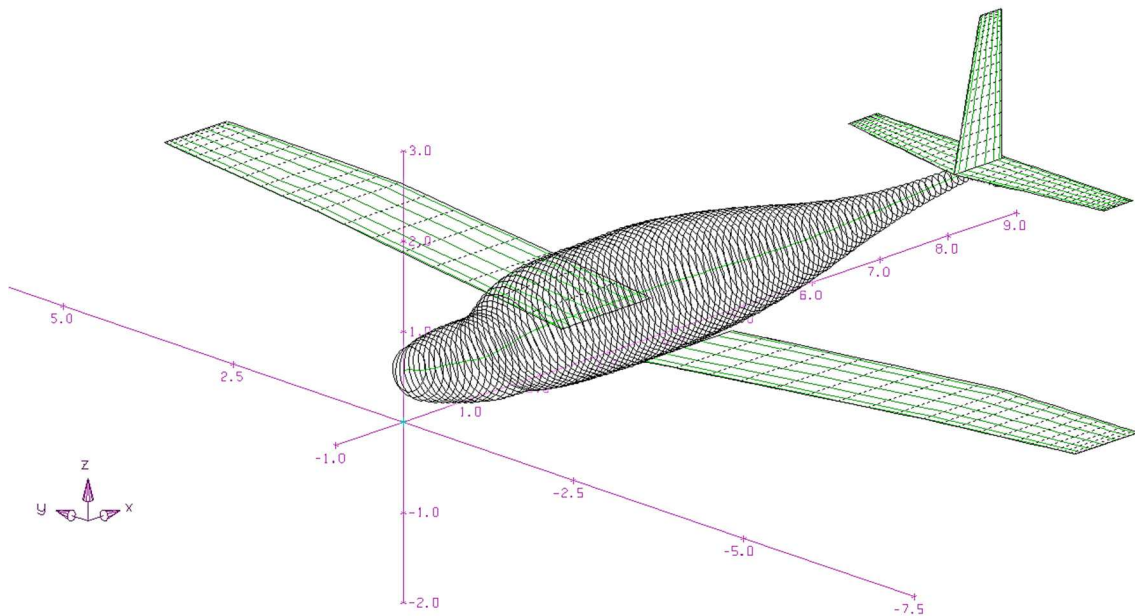


Figure 15 - 4PAX aircraft AVL model.

Therefore, the AVL provides aerodynamics and stability results to the genetic algorithm. In addition, it is possible to calculate the performance characteristics using the hybridization ratios. So, it compares these values to the specified criteria shown in Table 13.

Table 13 - Specified criteria for genetic algorithm.

		4PAX		6PAX	
		Min	Max	Min	Max
Geometrical	AR_W	-	13	-	13
	AR_{HT}	-	7.5	-	7.5
	AR_{VT}	1.5	5.5	1.5	5.5
Performance	Takeoff Length [m]	-	457.20	-	548.64
	Landing Length [m]	-	457.20	-	548.64
	Rate of Climb [ft/min]	-	1500	-	1300
	Angle of Climb [°]	5	30	5	30
	Range [km]	1852	-	1389	-
	$E_{\max}$	10	20	10	20
Stability	$c_{m\alpha}$	-	-0.40	-	-0.40
	Empty Static Margin	10	40	10	40
	MTOW Static Margin	10	40	10	40
	Trimmed Angle of Attack [°]	0	5	0	5
	Elevator Deflection for Trimmed Condition [°]	-10	10	-10	10
	$c_{n\beta}$	0.05	-	0.05	-
	$c_{l\beta}$	-	-0.04	-	-0.04

During the optimization, if the aircraft do not meet all the criteria presented, a penalty is added to an optimization variable. Thus, these aircraft with penalties are discarded over time by the objective function.

Figure 16 shows the flowchart used in the optimization process described above. The numbers in the process are associated to the variables available in Table 14.

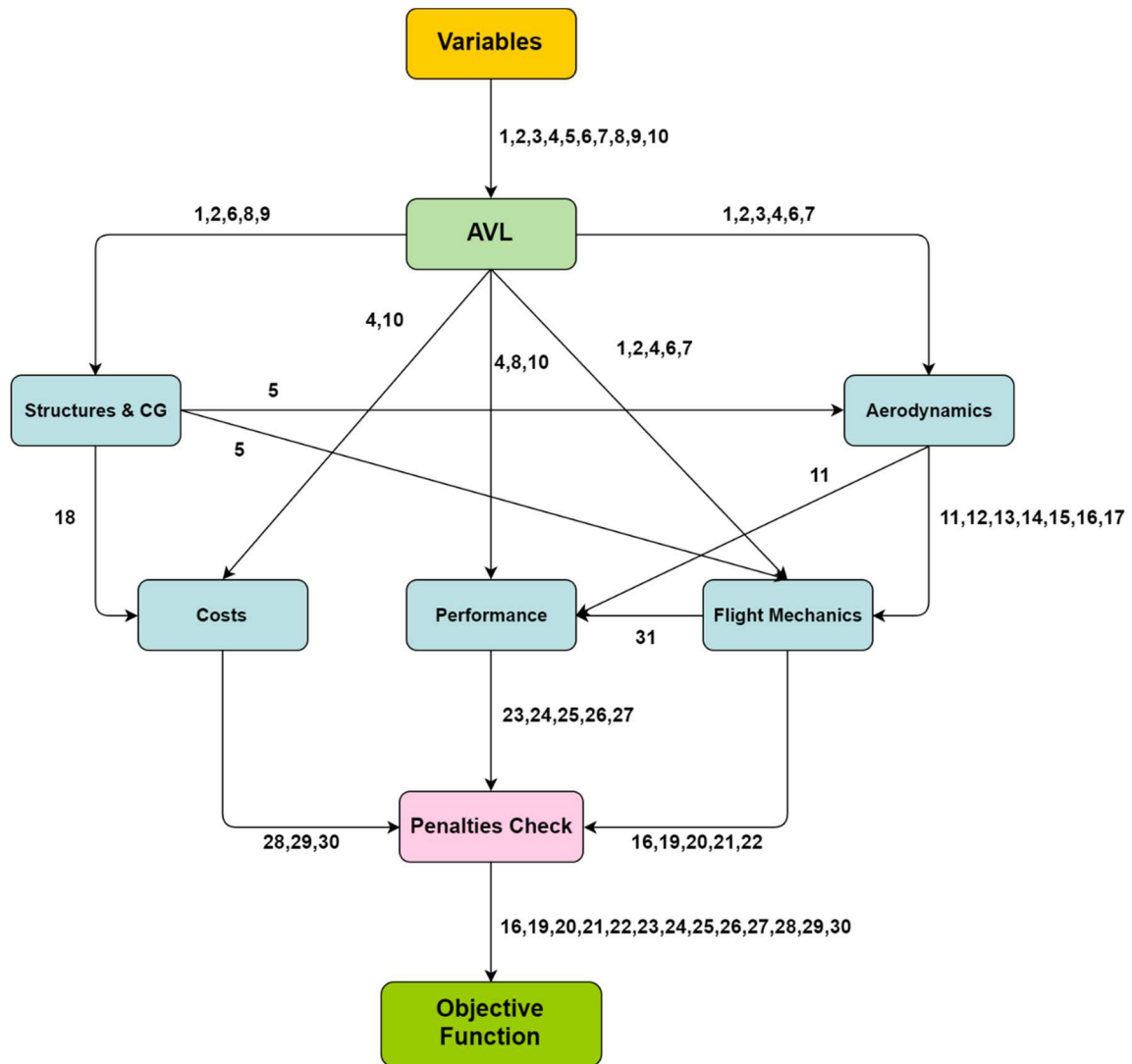


Figure 16 - Genetic algorithm flowchart.

Table 14 - Variables used in the genetic algorithm and represented in Figure 16.

Variable	Description
1	Wing and Empennage Positions
2	Wing, Empennage and Fuselage Dimensions
3	Wing and Empennage Sweep, Dihedral, Incidence, Twist Angles and Profiles
4	Sea Level Condition
5	CG Position
6	Flap and Control Surfaces Dimensions
7	Flap and Control Surfaces Positions
8	MTOW
9	Moment of Inertia
10	Engine Power/Consumption and Fuel Weight
11	Wing and Empennage CL_α
12	Non-trimmed Drag Polar
13	Wing and Empennage Cm_α
14	$dCl_{\delta \text{ aileron}}$ and $dCl_{\delta \text{ rudder}}$
15	$dCn_{\delta \text{ aileron}}$ and $dCn_{\delta \text{ rudder}}$
16	Cn_β and Cl_β
17	$dCL_{\delta \text{ elevator}}$ and $dCm_{\delta \text{ elevator}}$
18	Structural Weight
19	Full Aircraft Cm_α
20	Empty and MTOW Static Margins
21	Longitudinal Trim Conditions
22	Lateral–Directional Trim Conditions
23	Range
24	Endurance
25	Rate of Climb
26	Take-Off and Landing Length
27	Service Ceiling
28	Sale Price
29	Operating Costs Per Year
30	Operating Costs Per Hour
31	Trimmed Drag Polar

Finally, the genetic algorithm released an optimized aircraft configuration, which satisfies all requirements for the 4PAX configuration.

However, this aircraft had to be modified in order to better fit the 6PAX arrangement. A length of 0.75 m was added in the passenger cabin to accommodate the two extra passengers, in addition to the increase in payload. Thus, the optimizer was reshaped so that it provides just a new empennage (horizontal and vertical), since the wing was assumed to be the same.

Therefore, the final configurations for both aircraft generated by the genetic algorithm are presented in Table 15. A CATIA model for the 4PAX configuration is shown in Figure 17.

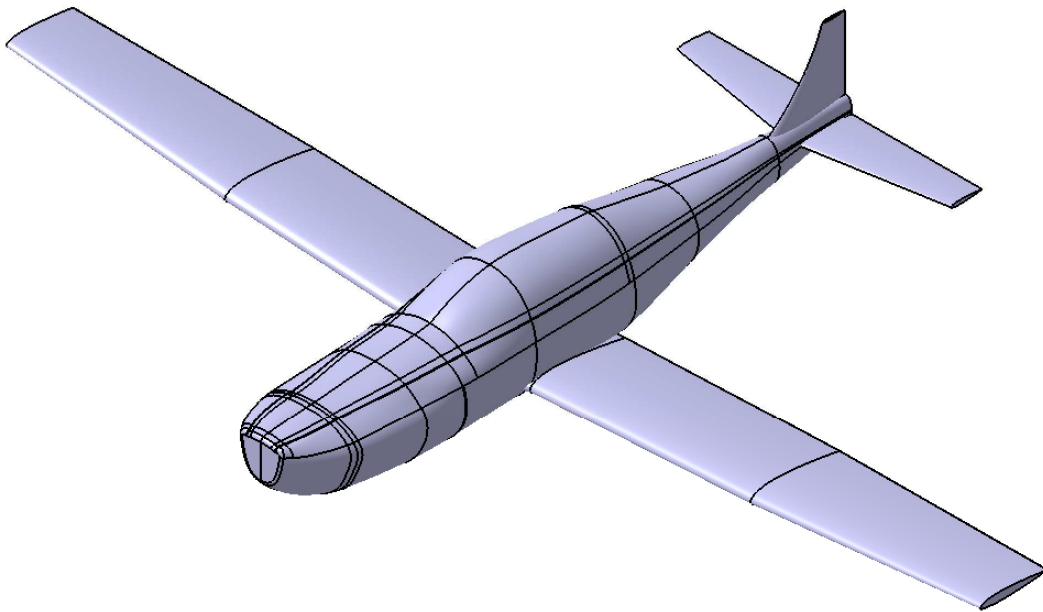


Figure 17 - 4PAX aircraft CATIA model.

Table 15 - Final optimized configurations for both hybrid aircraft.

		4PAX	6PAX
Weight	MTOW [kg]	1855.10	1966.20
	Fuel [kg]	176.57	130.63
	Empty [kg]	1279.30	1236.80
	Payload [kg]	399.20	598.80
	Structural [kg]	747.38	497.50
	TZFW [kg]	1713.80	1861.70
	Battery [kg]	370.63	309.71
Wing	b_w [m]	13.38	13.38
	c_{r_w} [m]	1.31	1.31
	c_{mac_w} [m]	1.19	1.19
	AR_w	12.84	12.84
	λ_w	0.70	0.70
	Λ_w [°]	0.080	0.080
	S_w [m ²]	13.95	13.95
	$C_{L_{max}}$	1.51	1.51
	C_{D_0}	0.024	0.024
	k_2	0.046	0.046
	x_w [m]	2.88	3.52
	Y-Position Break [m]	3.61	3.61
	Chord at Break [m]	1.23	1.23
	Γ_w [°]	4.70	4.70
	i_w [°]	2.00	2.00
	Twist [°]	0.00	0.00
	$C_{L_{max}}$ with take-off flap	2.00	2.00
Performance	Maximum Ceiling [m]	5486.40	5486.40
	V_{cruise} [m/s]	94.14	94.14
	M_{cruise}	0.28	0.28
	T / W	0.11	0.10

	V_{stall} [m/s]	33.20	34.18
	Take-off Distance [m]	442.16	533.62
	Range [km]	1869.80	1404.00
	Landing Distance [m]	442.16	533.62
	ROC [ft/min]	1558.50	1311.90
	Take-off Angle [°]	9.83	7.96
Fuselage	Length [m]	8.12	8.87
	Cabin [m]	2.40	3.15
	Nose [m]	2.21	2.21
	Tail [m]	3.51	3.51
	Cabin Diameter [m]	1.70	1.70
Horizontal Tail	S_{HT} [m ²]	2.23	1.84
	AR_{HT}	6.31	5.36
	$c_{r_{HT}}$ [m]	0.76	0.75
	λ_{HT}	0.57	0.55
	V_{HT}	0.65	0.52
	b_{HT} [m]	3.75	3.14
	$c_{mac_{HT}}$ [m]	0.59	0.59
	x_{HT} [m]	7.58	8.11
	Λ_{HT} [°]	0.00	0.00
Vertical Tail	S_{VT} [m ²]	0.84	0.63
	AR_{VT}	3.27	4.27
	$c_{r_{VT}}$ [m]	0.70	0.54
	λ_{VT}	0.44	0.41
	V_{VT}	0.022	0.016
	b_{VT} [m]	1.66	1.64
	$c_{mac_{VT}}$ [m]	0.50	0.38
	x_{VT} [m]	7.66	8.32
	Λ_{VT} [°]	0.00	0.00
Control Surfaces	Aileron Root Chord [m]	0.31	0.31

	Aileron Tail Chord [m]	0.29	0.29
	Aileron Span [m]	0.50	0.50
	Y-position Aileron [m]	5.99	5.99
	Elevator Root Chord [m]	0.19	0.18
	Elevator Tail Chord [m]	0.11	0.10
	Elevator Span [m]	3.75	3.14
	Rudder Root Chord [m]	0.35	0.27
	Rudder Tail Chord [m]	0.16	0.11
	Rudder Span [m]	1.66	1.64
Landing Gear	$x_{Landing.gear-Nose}$ [m]	1.11	1.15
	$x_{Landing.gear-Main}$ [m]	4.26	4.43
Electrical Engine	Take-off Power [kW]	240.00	230.00
	Climb Power [kW]	228.12	216.99
	Cruise Power [kW]	150.43	151.04
	Descent Power [kW]	24.56	31.98
	η_{prop}	0.85	0.85
Center of Gravity	Empty x_{CG} [m]	3.21	3.73
	MTOW x_{CG} [m]	3.35	3.96
	TZFW x_{CG} [m]	3.30	3.92
Stability	$c_{n\beta}$	0.054	0.066
	$c_{l\beta}$	-0.11	-0.11
	$c_{m\alpha}$	-0.38	-0.42
	Empty Static Margin	35.15	32.63
	MTOW Static Margin	16.72	12.89
	Trim Angle [°]	2.58	2.89
	Elevator Trim Deflection [°]	-1.79	-1.13

4.5 Aircraft of Comparable Role and Configuration

Searching for other aircraft of the same category, it was found models such as the Cessna TTx, Cirrus SR22 and Diamond DA42. These aircraft present similar characteristic to the hybrid-electric aircraft designed here. As an exemplification, they are compared to the hybrid-electric aircraft of 4PAX in Table 16. In addition, they are placed over each other to show the differences in size, as show in Figure 18.

Table 16 - Aircraft of comparable role and configuration.

Characteristics	Diamond DA42	Cirrus SR22	Cessna TTx	Hybrid 4PAX
Wing span [m]	13	11	11	13.38
Fuselage length [m]	8.56	8.28	7.68	8.12
Height [m]	2.49	2.72	2.74	2.96
Range [km]	1693	1289	2352	1875
Endurance [hr]	8.8	4.5	5.25	7.22
Cruise speed [km/h]	326	226	435	343
Take-off field [m]	776	600	390	442
Landing field [m]	620	620	805	442
MTOW [kg]	1999	1111	1633	1855
Passengers	3	3	3	3
Fuel [kg]	231	220	280	176
Service ceiling [m]	5500	4100	7620	3660
Climb Rate [ft/min]	1337	720	1400	1559

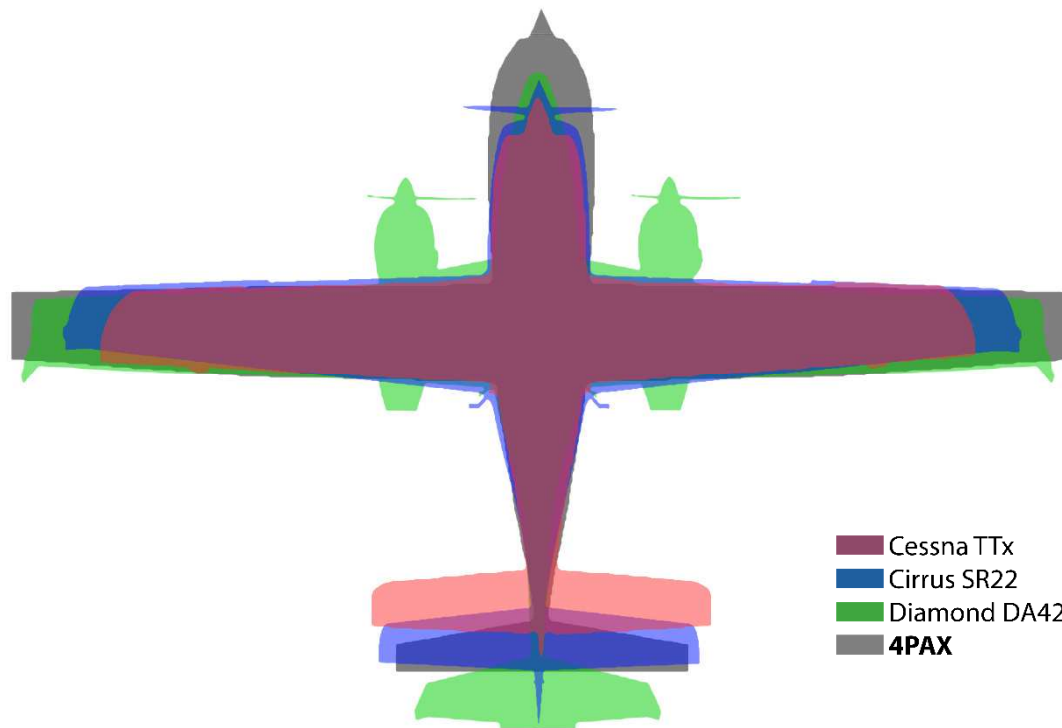


Figure 18 – Aircraft of comparable role and configuration overpainted.

Therefore, the aircraft developed in this work meet the AIAA requirements and also is competitive in the aviation market. Since it presents good general and performance characteristics, its main differential certainly would be lower fuel consumption because of the hybrid-electric system on board, attracting new customers. The aircraft presents a longer nose compared to the other aircraft because all the propulsive system, which includes batteries, inverter, ICE, generator, controller and electric motor were assumed to be placed at the front of the aircraft.

CHAPTER V

SCTRUCTURAL ANALYSIS

*“Some people dream of success,
while other people get up every morning
and make it happen.”*

Wayne Huizenga

Having both aircraft configurations (4PAX and 6PAX) all set, the next step is the structural analysis and sizing. In this work, only the aircraft's wing and empennage (horizontal and vertical) will be evaluated.

In any structural project, it is necessary to know the loads that the structure will be subjected. Thus, it is possible to better size the structure, avoiding failure and reducing weight, which is determinant for any aviation project.

When it comes to safety, there are several regulations that oblige the manufacturers to ensure that the structures will withstand various extreme loads conditions. The Federal Aviation Administration (FAA), an American authority with powers to regulate all aspects of civil aviation, is the one responsible to publish the regulation for civil aircraft loads. The 14 CFR - Part 23 [54] is the part that handles all

types of aircraft loads, including the design airspeeds, used to generate the well-known V-n diagram.

The V-n diagram provides a treasure trove of information regarding flight performance for pilots. Basically, it is a graph depicting the variation of load factor with the speed of the aircraft. An understanding of what the load factor represents is crucial to understand the V-n diagram. In many military movies with fighter jets, such as *Top Gun*, it is possible to notice the pilots calling out how many G's they are pulling. They are, in fact, talking about the load factor on the airframe.

Load factor provides a measure of the stress the airframe is subjected to, and it is a dimensionless quantity. However, in the aeronautics industry, it is represented by the number of G's pulled by the aircraft. 1G flight, for example, represents the condition of straight-level flight. In this state, the lift is equal to the weight and there is no vertical acceleration. Here, a man inside the aircraft would feel his weight the same as if he were sitting at home. A condition of free fall, or 0G, occurs when the airframe is generating no lift. In this case, the same man would feel weightless.

Following the procedures described in the FAA regulation [54], the V-n diagrams for both aircraft designed in this work are presented in Figure 19 and Figure 20.

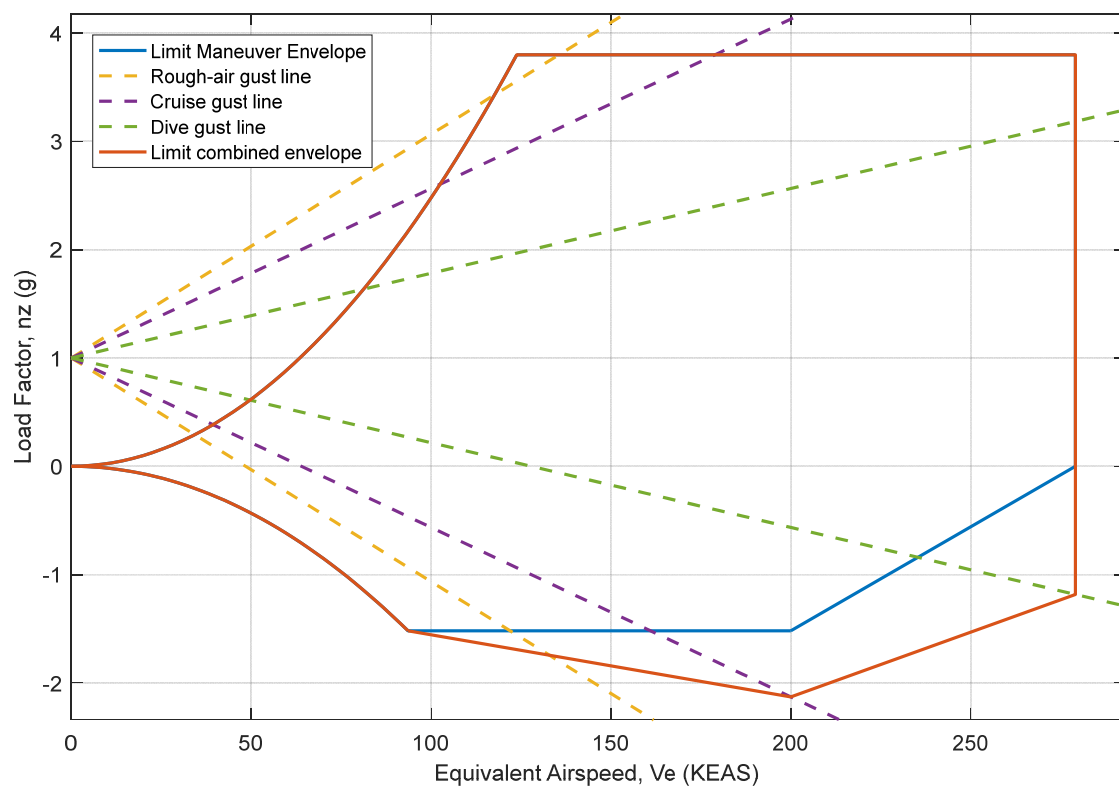


Figure 19 - V-n diagram for 4PAX aircraft.

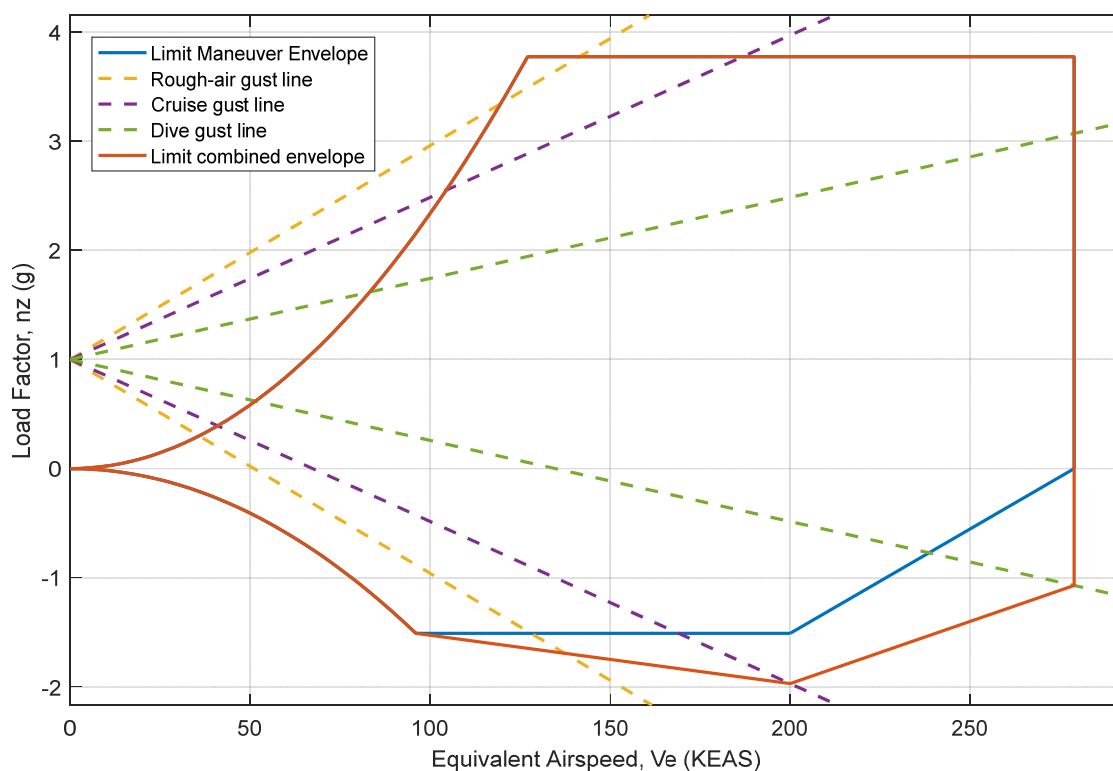


Figure 20 - V-n diagram for 6PAX aircraft.

The images give a never exceed speed and the limit load factors for a particular altitude of 12000 ft, which are arranged in Table 17. The parabolic curves, called the stall lines, are branching out from the origin on either side. Beyond around 280 KEAS and above 3.8G and below -1.0G, the airframe would be subject to structural damage.

Table 17 - Parameters from V-n diagrams.

Parameters	4PAX	6PAX
$n_{z_{\max}}$	3.8	3.8
$n_{z_{\max} \text{ Gust}}$	3.8	3.8
$n_{z_{\min}}$	-1.5	-1.5
$n_{z_{\min} \text{ Gust}}$	-2.1	-2.0
V_G	93.6 KEAS	96.0 KEAS
V_A	123.8 KEAS	127.0 KEAS
V_C	200.0 KEAS	200.0 KEAS
V_D	279.1 KEAS	278.9 KEAS

In addition to the limit maneuver envelope curve, the diagrams include the gust lines, which represent the loads experienced when the aircraft encounters a strong gust (when flying close to a thunderstorm or during a clear air turbulence encounter) and may exceed the maneuver loads. Therefore, it is necessary to create a new envelope combining the gust loads, as shown in the diagrams.

The assumption that an airplane instantly encounters a gust and this gust instantly effects the airplane is unrealistic. Gusts follow cosine-like intensity increase allowing aircraft more time to react. This reduces the acceleration experienced by the airplane.

Furthermore, it was necessary to ensure the structural design would handle all critical cases. Thus, some specific conditions had been chosen to evaluate the loads, such as: dive at V_C , dive at V_D , and bunt at V_C from V-n diagrams; and coordinate turn from the maneuver boundaries.

The maneuver boundaries are calculated using theory in [55] and shown in Figure 21 and Figure 22. Analyzing the curves, the extreme condition possible during a coordinate curve is at a velocity of 64.6 m/s for 4PAX and 64.8 m/s.

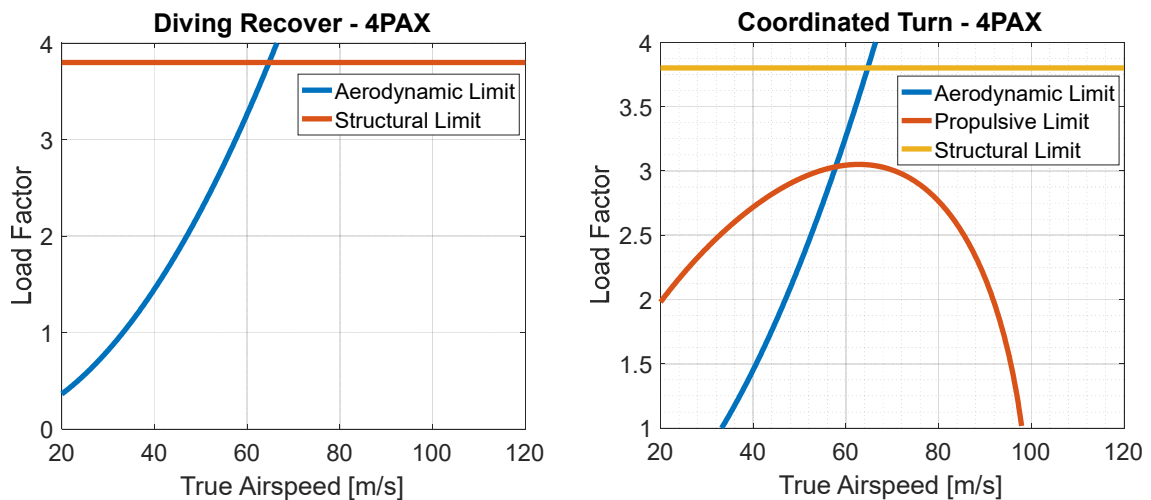


Figure 21 - Maneuver boundaries for 4PAX aircraft.

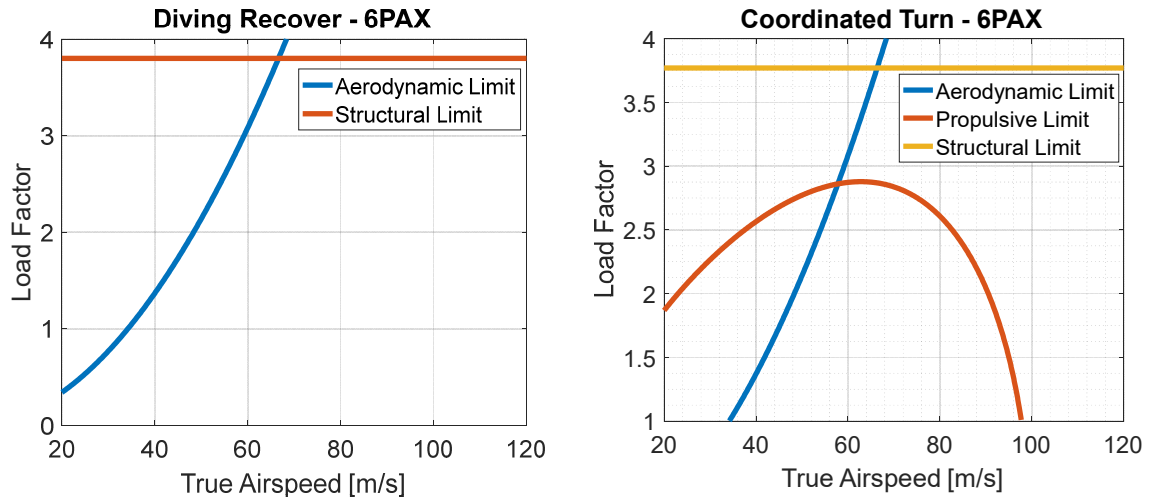


Figure 22 - Maneuver boundaries for 6PAX aircraft.

Thus, there are velocities and load factors for each of the aforementioned four conditions. Inserting these parameters into AVL program, it was possible to determine the aircraft bank angle and the trim angles for each of the control surfaces. From these results, the following Table 18 and Table 19 have been set up.

Table 18 - Trim conditions for 4PAX aircraft.

Condition	Velocity [m/s]	Altitude [ft]	ρ_{air} [kg/m ³]	n	α [°]	δ_e [°]	ϕ [°]	δ_a [°]	δ_r [°]
Coordinate Turn	64.6	ISA + 18°F	1.1839	2.7	5.34	-3.52	68.85	-4.41	11.63
Dive at V_D	143.80	ISA + 18°F	1.1839	3.8	4.24	-3.55	0	0	0
Dive at V_C	102.89	12000	0.6981	3.8	9.19	-8.20	0	0	0
Bunt at V_C	102.89	12000	0.6981	-1.5	-4.99	4.19	0	0	0

Table 19 - Trim conditions for 6PAX aircraft.

Condition	Velocity [m/s]	Altitude [ft]	ρ_{air} [kg/m ³]	n	α [°]	δ_e [°]	ϕ [°]	δ_a [°]	δ_r [°]
Coordinate Turn	64.8	ISA + 18°F	1.1839	2.5	5.20	-3.16	66.27	-4.45	11.78
Dive at V_D	143.63	ISA + 18°F	1.1839	3.8	4.87	-3.35	0	0	0
Dive at V_C	102.89	12000	0.6981	3.8	10.39	-7.15	0	0	0
Bunt at V_C	102.89	12000	0.6981	-1.5	-5.47	3.79	0	0	0

Having the aircraft trim conditions in hands, the next step was to calculate the lift and pitching moment distributions over the wing and empennage surfaces, for both aircraft. To do so, the aerodynamic analyses were evaluated through the software XFLR5, which is an analysis tool for airfoils, wings and planes operating at low Reynolds Numbers. An example of the XFLR5 analysis for the 4PAX configuration is shown in Figure 23.

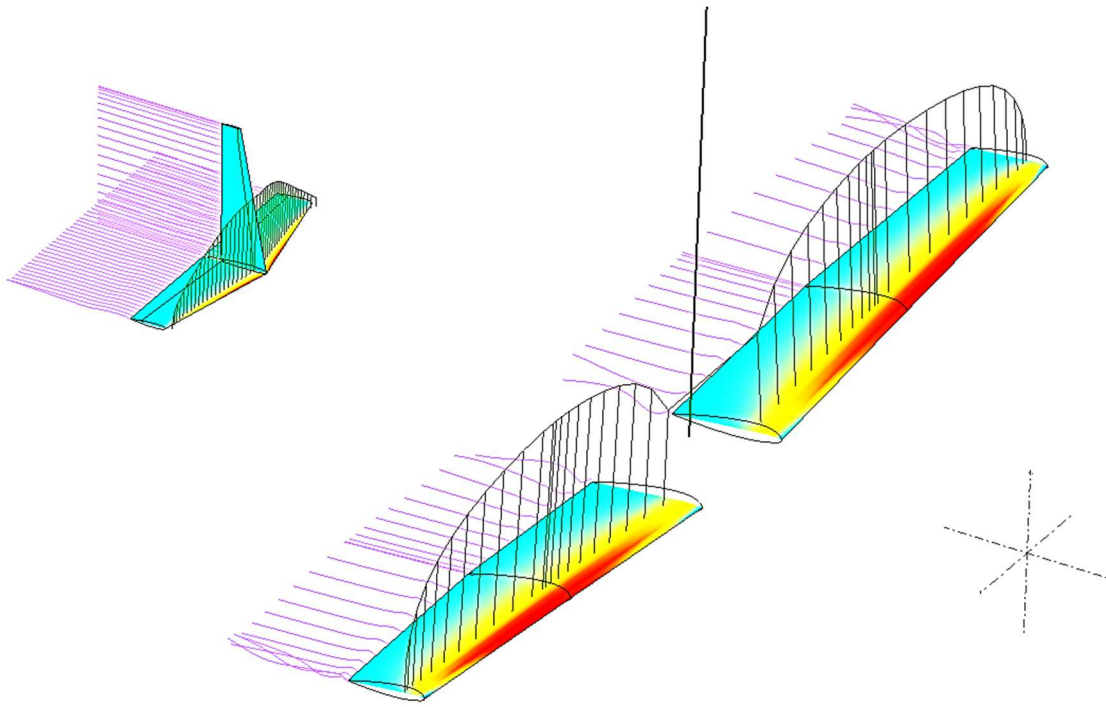


Figure 23 – Aerodynamic analysis in XFLR5.

From these results, and using theory in [56], the bending and torsion moments over the wing and empennage span were calculated and presented in Figure 24 and Figure 25.

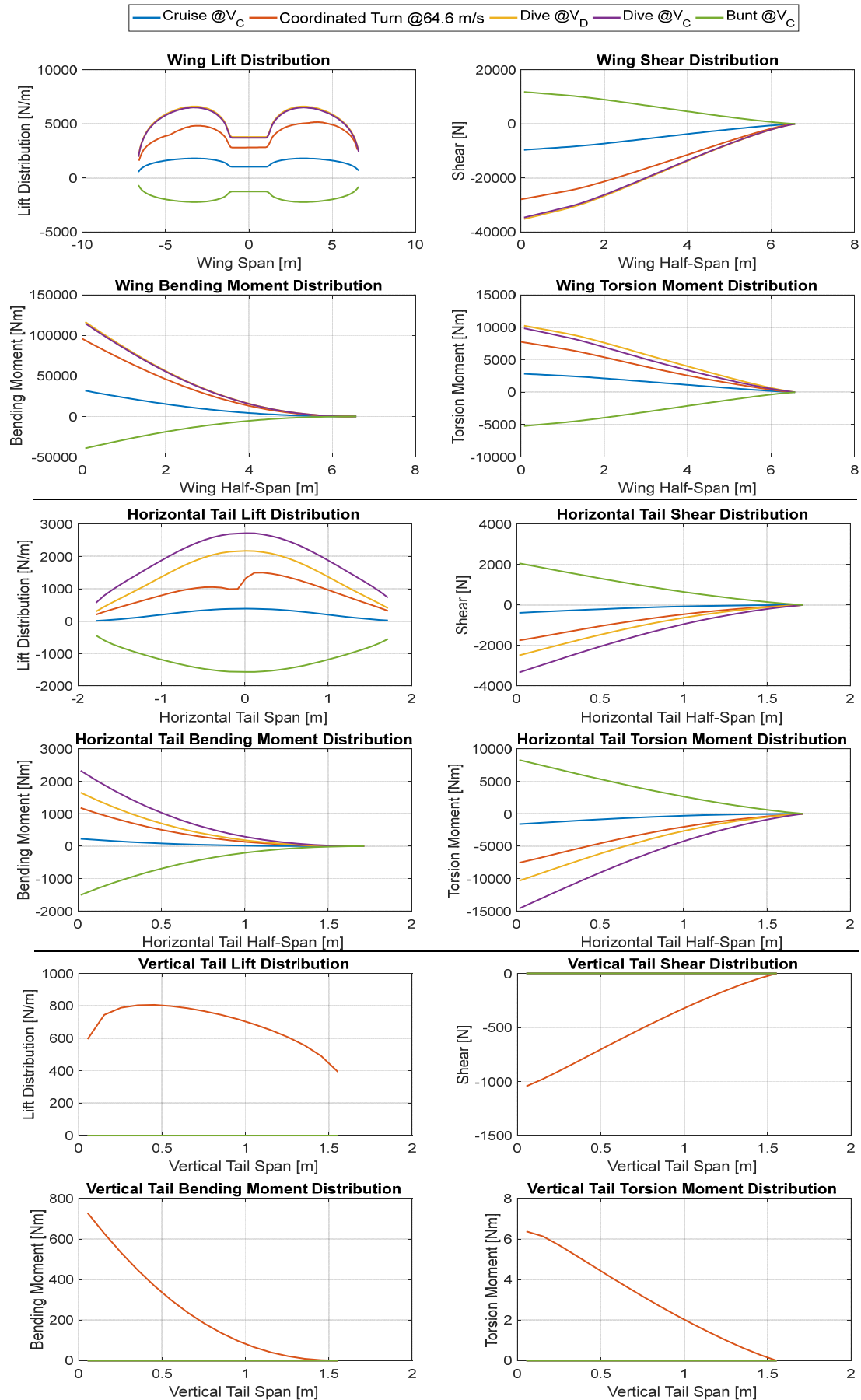


Figure 24 - 4PAX aircraft loads.

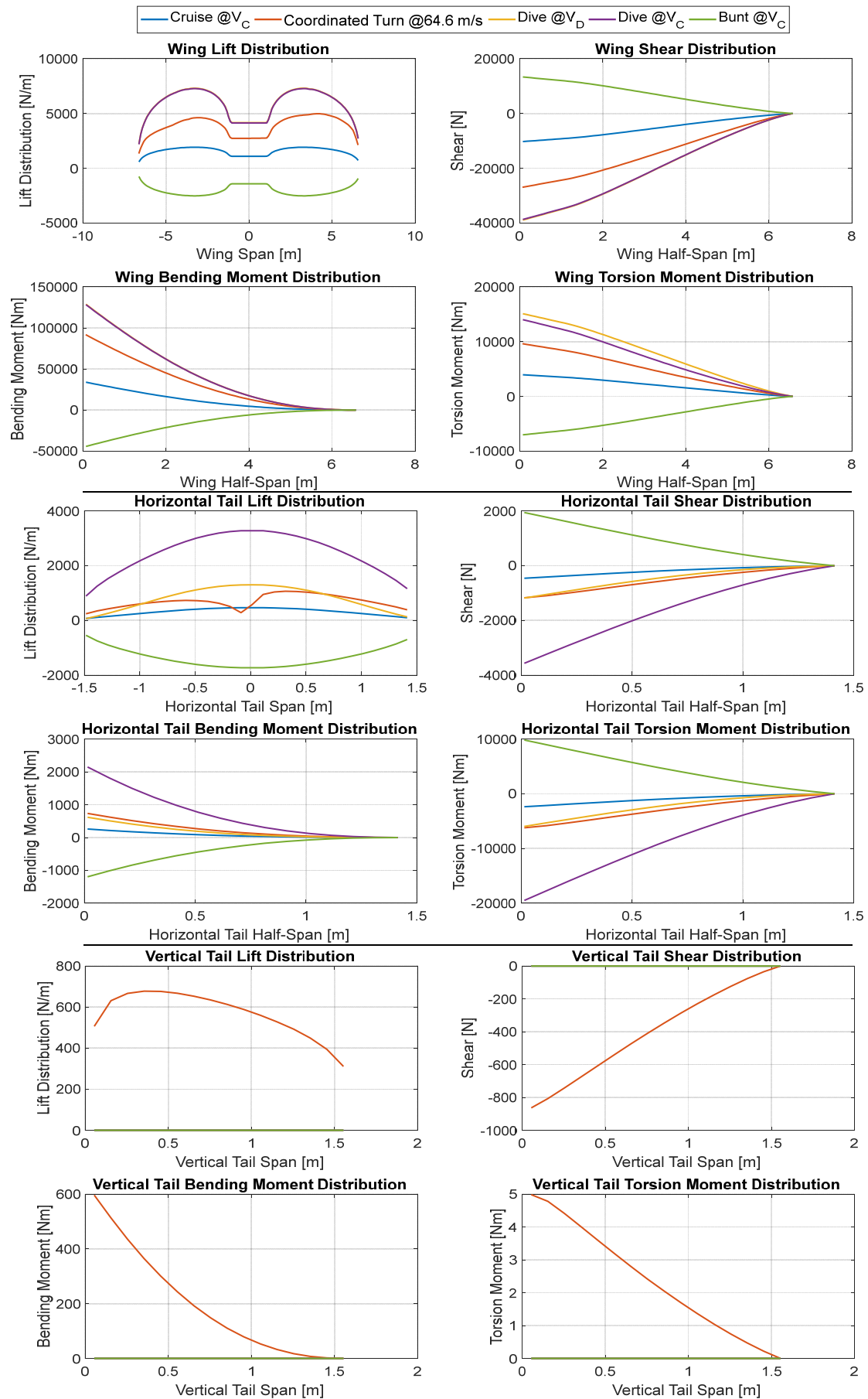


Figure 25 - 6PAX aircraft loads.

5.1 Structural Wing Design

For the wing design, the Embraer Phenom 100 was used as reference. Thus, observing its cutaway in Figure 26, it was possible to estimate the number and position of spars, ribs and stringers.

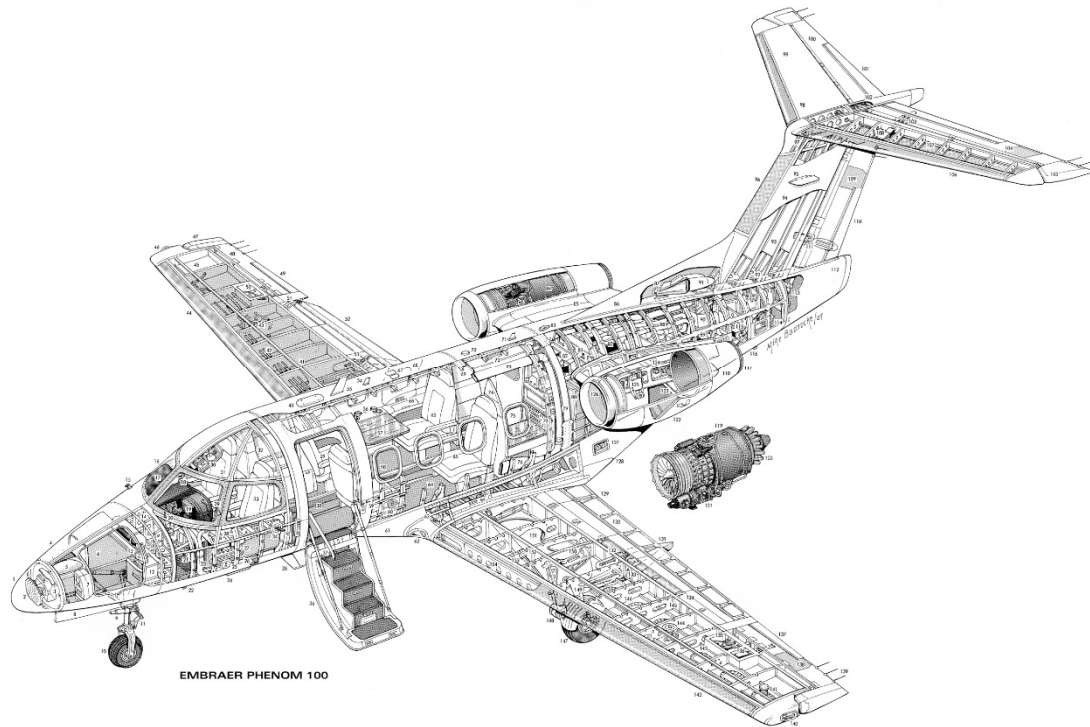


Figure 26 - Embraer Phenom 100 cutaway used as reference. Source: Patent US20130001356⁷.

The position and quantity of these elements were adapted to the actual aircraft's span and were not optimized due to the complexity of implementation, what led to the following the coordinates, as shown in Figure 27 and Figure 28.

⁷ Cutaway view of the Embraer Phenom 100 (drawing dated 2007, aircraft first flown 2007), from <http://img225.imageshack.us/img225/1192/embraerphenom1002007bad.jpg>

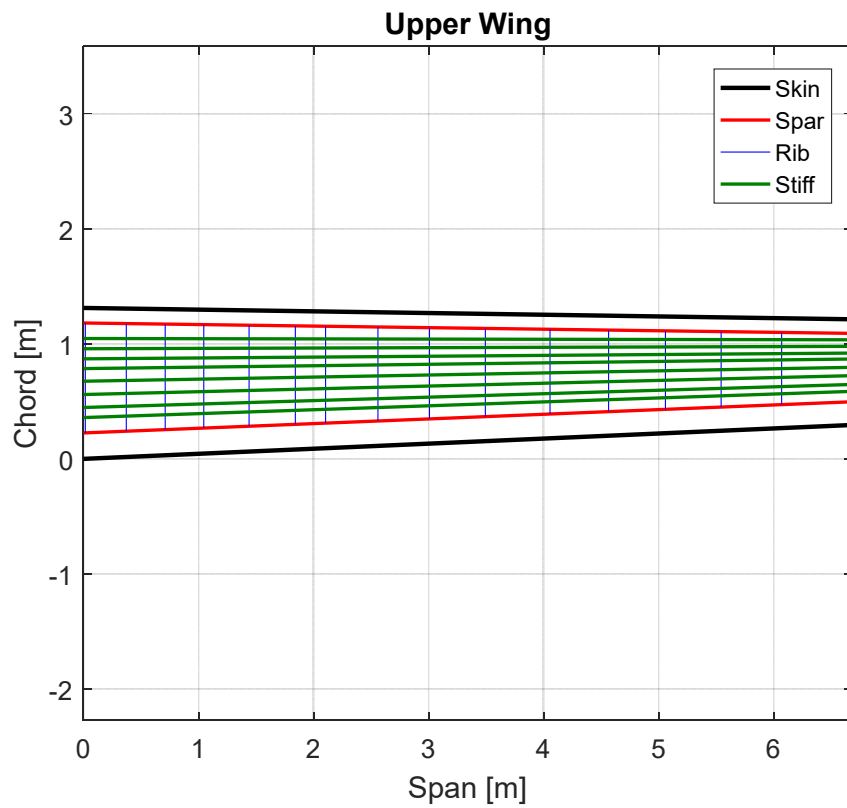


Figure 27 – Aircraft upper wing structure elements arrangement along the span.

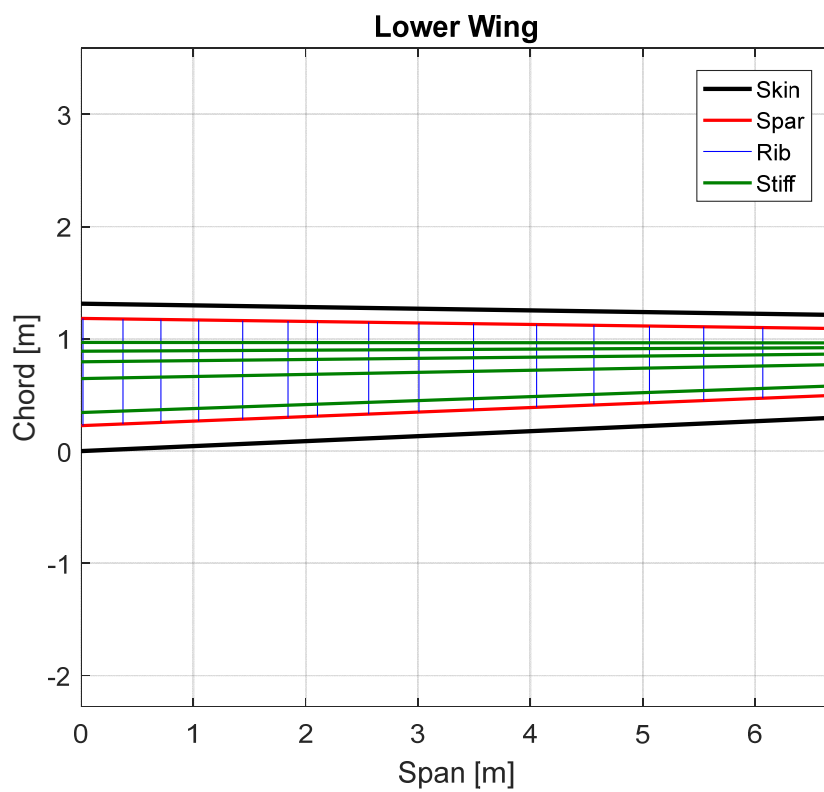


Figure 28 - Aircraft lower wing structure elements arrangement along the span.

Thus, an initial wing design was made using CATIA V5R21[®], so it was possible to refine the curvature and airfoil profile, as shown in Figure 29.

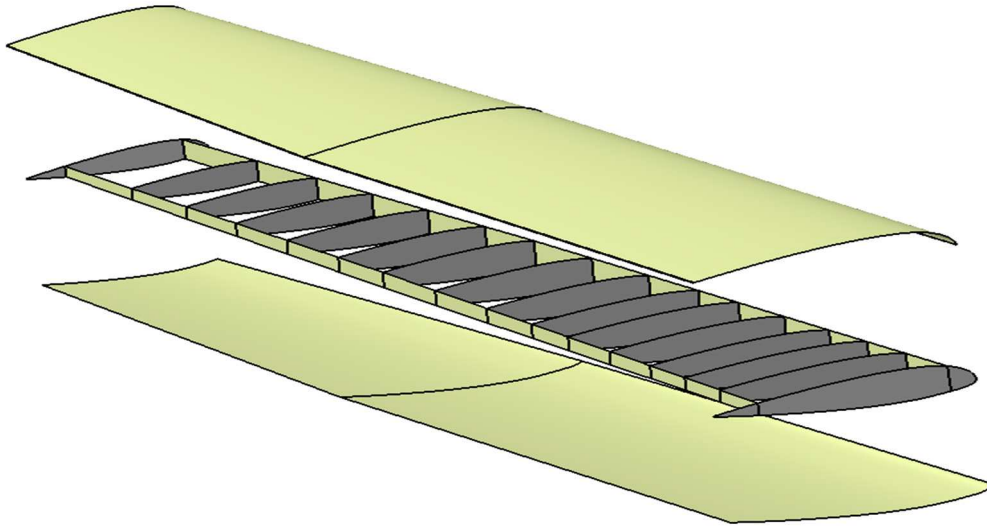


Figure 29 - Wing model drawn in CATIA.

Next, the model was exported to HyperMesh[®] to create the mesh. The mesh size was designed in order to allow a satisfactory finite element analysis and to avoid an excessive computational consumption. Thus, it was used the quadrilateral-mesh with an element size of 30 mm, generating a mesh of 19885 elements, as presented in Figure 30.

Moreover, for each rib it was created a RBE3 (rigid body element) to connect a single node to multiple nodes, allowing each rib to subject to a specific bending and torsion moments at 25% of the wing chord, depending on its location.

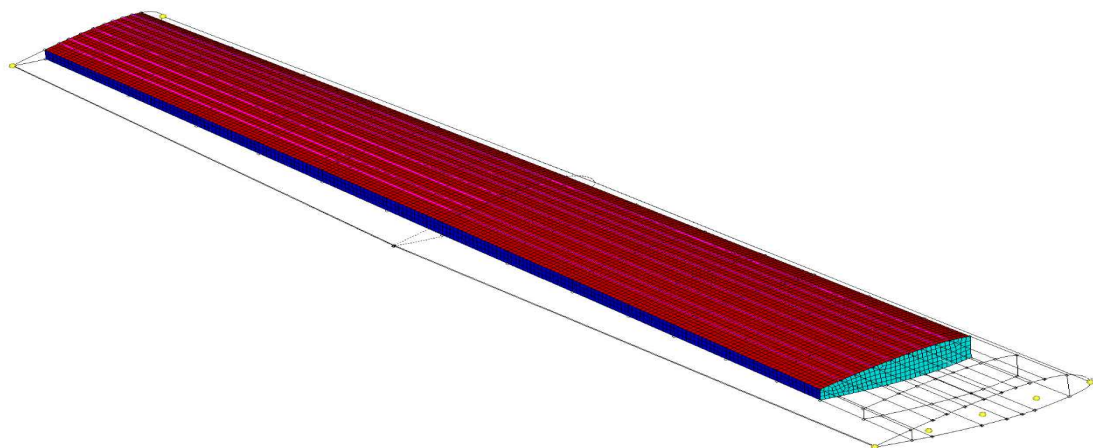


Figure 30 - Wing model after meshing in HyperMesh.

Having the model done, the wing structure was imported into Femap with NX Nastran®. Femap is a finite element modeling and post-processing environment developed by Siemens®. It interacts with solvers such as NX Nastran®, which is an actively-developed offshoot of the Nastran® code base and one of the most common solvers for finite element analysis (FEA).

Thus, in Femap it was possible to perform the actual sizing of the wing. Groups of components were created to make the analyses easier. These groups were split into the following components, as described in different colors in Figure 31:

- Upper skin (Red group);
- Lower skin (Blue group);
- Ribs (Green group);
- Front spar (Orange group);
- Back spar (Purple group);
- Upper stiffeners (Dark red group);
- Lower stiffeners (Dark blue group).

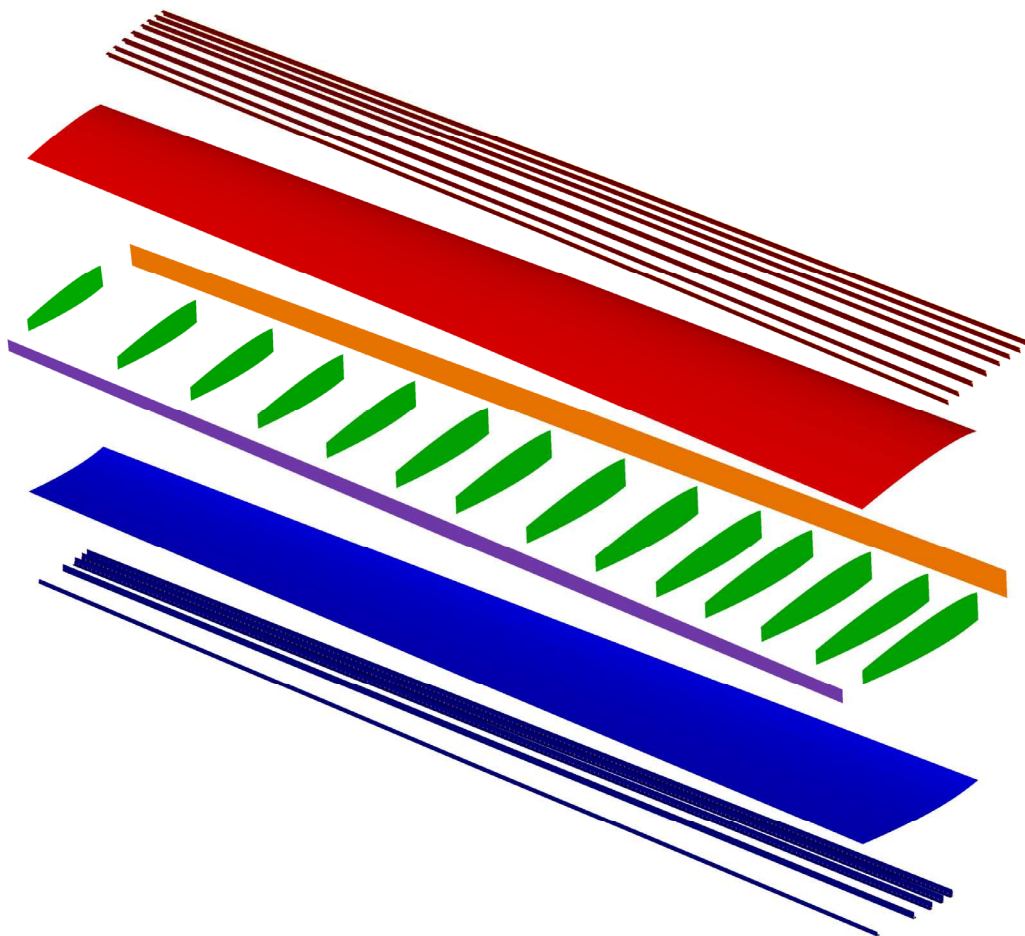


Figure 31 - Exploded wing and groups classified by colors.

To get started with the modeling, a material needed to be specified. For the entire model, the Aluminum 7050-T651 had been chosen due to its large applications and popularity in the aerospace industry. It is a heat treatable alloy that has very high mechanical properties and high fracture toughness, and offers good stress and corrosion cracking resistance and high strength at subzero temperatures. The Aluminum 7050-T651's general properties used in the model are presented in Table 20.

Table 20 - Aluminum 7050-T651 properties [58].

Property	Value
Young's Modulus, E	72000 MPa
Poisson's Ratio, ν	0.33
Mass Density, ρ	$2.7 \times 10^{-6} \text{ kg/mm}^3$
Yield Strength, σ_Y	490 MPa

In addition, it was created a property with random thickness for each element of each group, allowing change and adjustment individually. Some elements were created as plate type and others as beam type. The ribs and spars were initially made as plate type, but they are more appropriate as beam type. Thus, it was also created a new property to represent the spar and rib tabs, making them into beams. The classification of properties follows the pattern in Table 21.

Table 21 - Classification of aluminum properties in Femap.

Property ID	Property Type	Group of Elements
100000 – 109999	Plate	Upper skin
200000 – 209999	Plate	Lower skin
300000 – 309999	Plate	Ribs
400000 – 409999	Plate	Front spar
500000 – 509999	Plate	Back spar
600000 – 609999	Beam	Upper stiffeners
700000 – 709999	Beam	Lower stiffeners
800000 – 809999	Plate	Spar-tab
900000 – 909999	Plate	Rib-tab

Therefore, coloring each element with a different color, representing different properties, the following full model is presented in Figure 32.

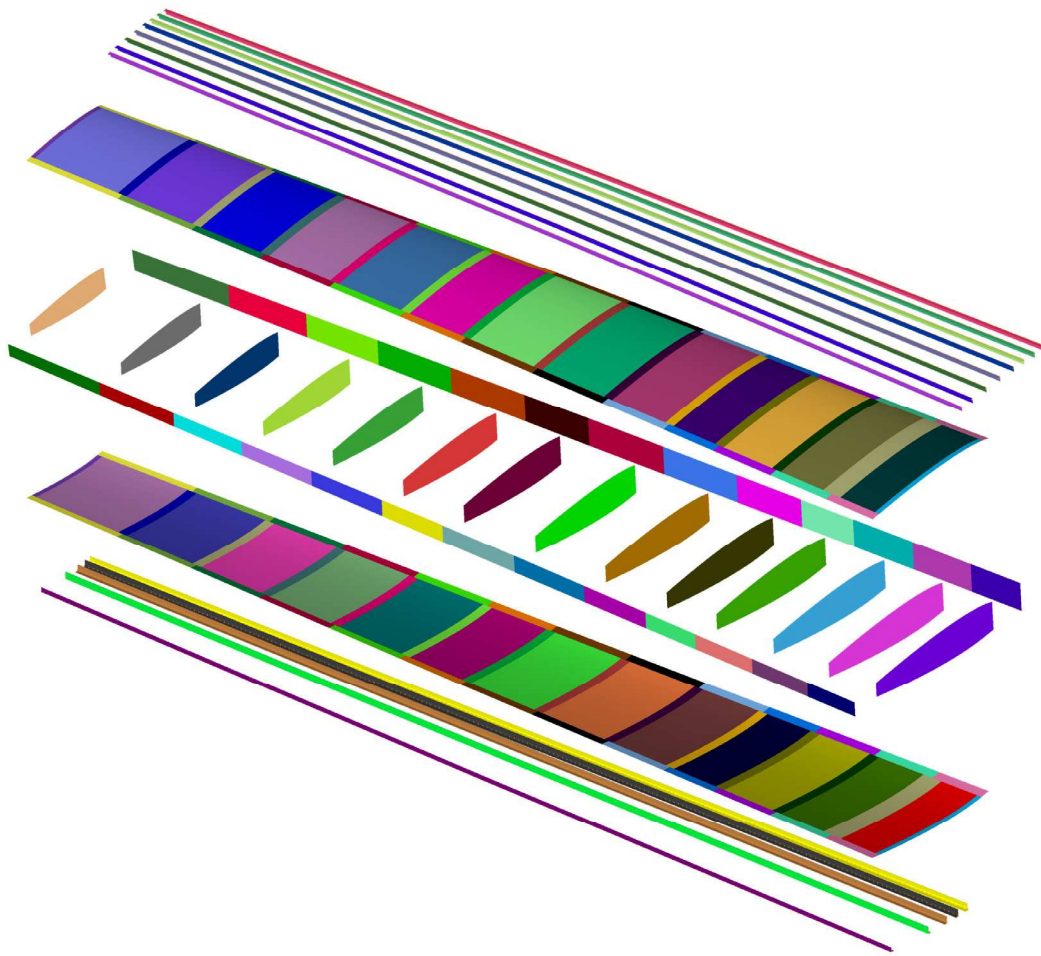


Figure 32 - Exploded wing and properties classified by colors.

Next, it comes up with the static analysis of the wing. The load cases evaluated in this analysis are those presented in Table 18 and Table 19. Each one of them was associated to a pattern code, so it was easy to study each load case in Femap. They are described in Table 22.

Table 22 - Load cases classifications in Femap.

Aircraft	Load Case	Number
4PAX	Dive at V_C	1
	Dive at V_D	2
	Bunt at V_C	3
	Coordinate turn - RHS	4
	Coordinate turn - LHS	5
6PAX	Dive at V_C	6
	Dive at V_D	7
	Bunt at V_C	8
	Coordinate turn - RHS	9
	Coordinate turn - LHS	10

Hence, for each load case it was necessary to determine and apply the specific loads in each RBE3. Thus, using the bending and torsion moments distributions from Figure 24 and Figure 25, the loads were calculated and corrected for the actual Femap reference axes.

Running the first static analysis for multiple load cases, the wing failed at these extreme conditions. Hence, the thicknesses of all elements in each group were manually modified so that the stresses throughout the wing were below 390 MPa, which is the Aluminum 7050's yield strength of 490 MPa with a safety factor of 1.2.

When analyzing the stress, the criteria of evaluation of the results were the maximum, minimum and shear stress for both plates and beams type elements. Thus, an envelope of results showing these stress values, for example, are presented in Figure 33, Figure 34 and Figure 35. On the right, it is displayed the stress values throughout all load cases, and on the left, it is displayed in which load case the maximum stress value was found.

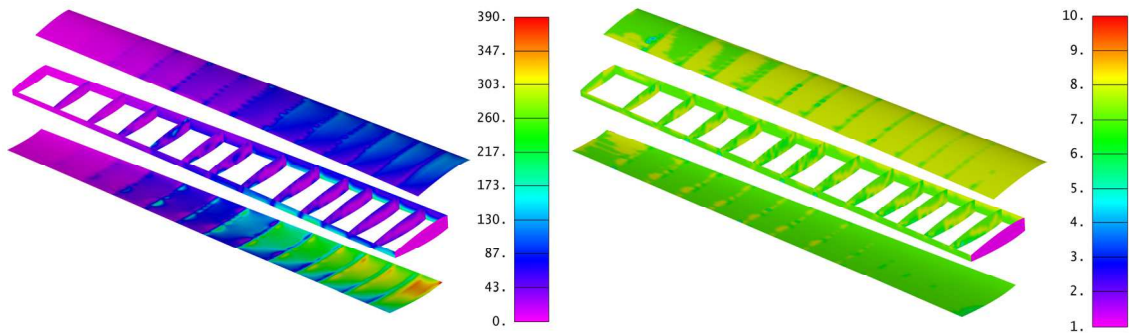


Figure 33 - Maximum stress values envelope in MPa (on the left) and respective load cases envelope (on the right).

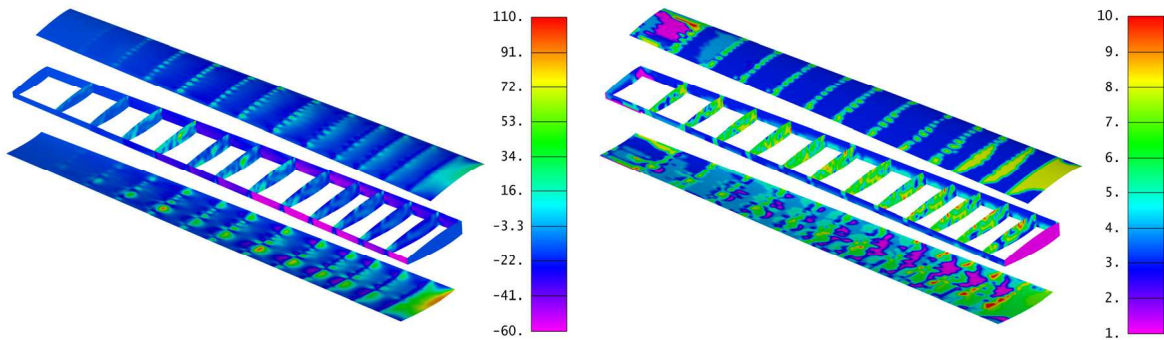


Figure 34 - Minimum stress values envelope in MPa (on the left) and respective load cases envelope (on the right).

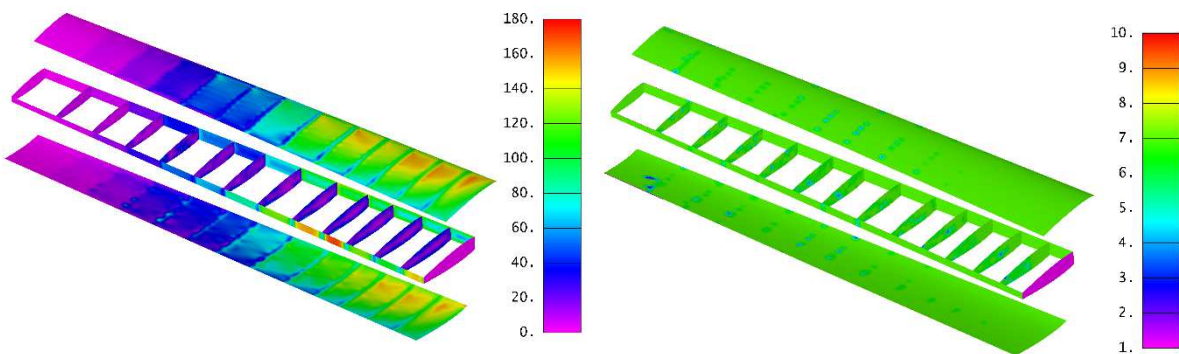


Figure 35 - Shear stress values envelope in MPa (on the left) and respective load cases envelope (on the right).

The stiffeners were modeled as beam type with a cross-section in “L”. Thus, they are not presented in Figure 33, Figure 34 and Figure 35, which show the results of the elements of plate type. However, it is worth pointing out that they were also designed

following the same process; in other words, changing cross-section dimensions and thicknesses. They are very important in the wing structural design because they increase the resistance of the skin panels, which is crucial against buckling.

Therefore, the static analysis is not limited only to these stresses analyzes. Because of the high bending moments, the wing skins are subjected to high stresses of compression. Therefore, a buckling analysis is crucial to avoid any unforeseen phenomena.

When running a Femap buckling analysis, the software shows what percentage of the current loads would be sufficient to buckle any part of the structure. Thus, the first buckling analysis in Femap showed that some skin plates, for example, would fail at 45.5% of those current loads, as presented in Figure 36.

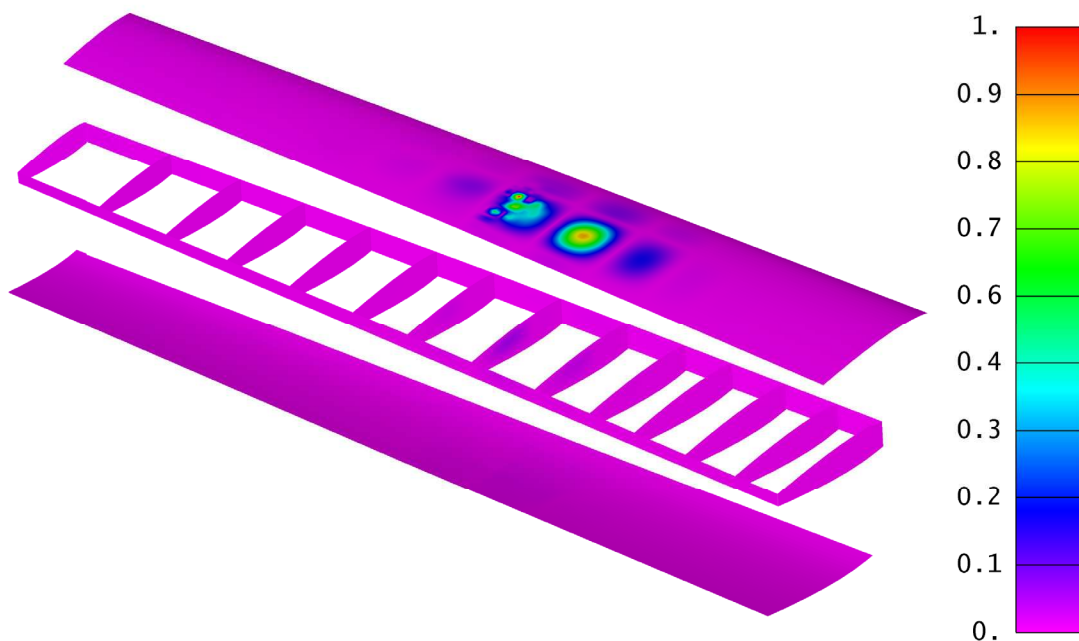


Figure 36 - Wing buckling analysis with failure at 45.5% of current loads.

After analyzing all the eigenvectors, new thicknesses were manually inserted into the properties of the elements involved, until the buckling analyses showed no failure. Now, the structure would buckle applying a load of 108.1% of the current loads, as presented in Figure 37; in other words, for the extreme loads evaluated in the aircraft envelope, there will not be any expected buckling phenomenon.

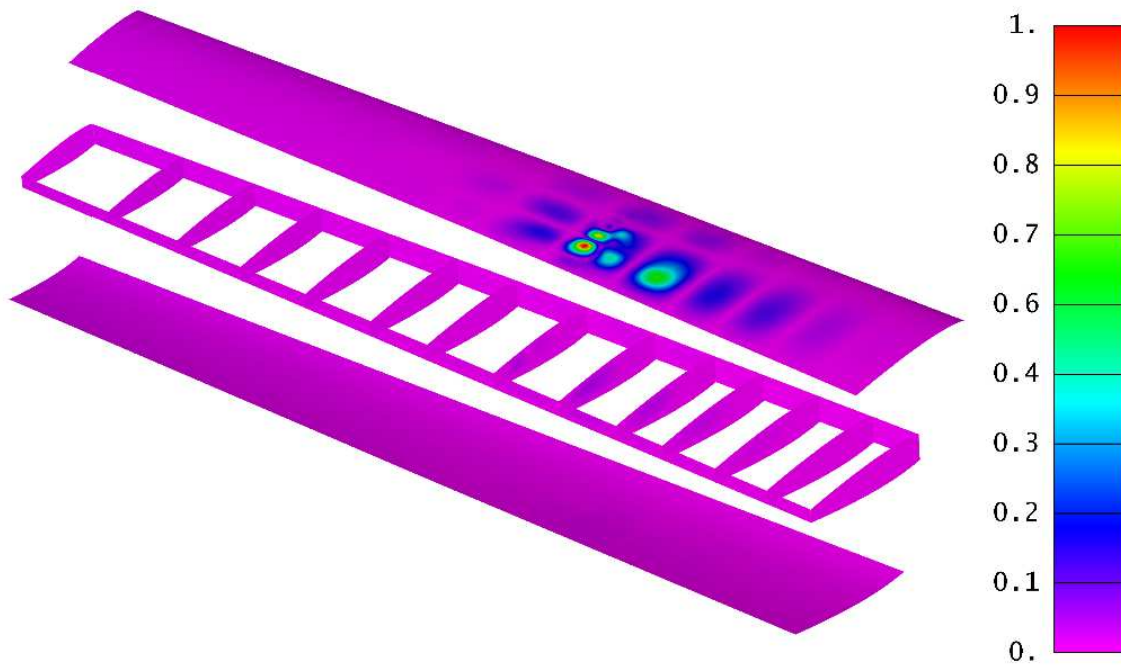


Figure 37 – Wing buckling analysis with failure at 108.1% of current loads.

Finally, the wing structure is well designed and dimensioned for the evaluated loads and buckling analyses. The thickness distribution along the wing is shown in Figure 38.

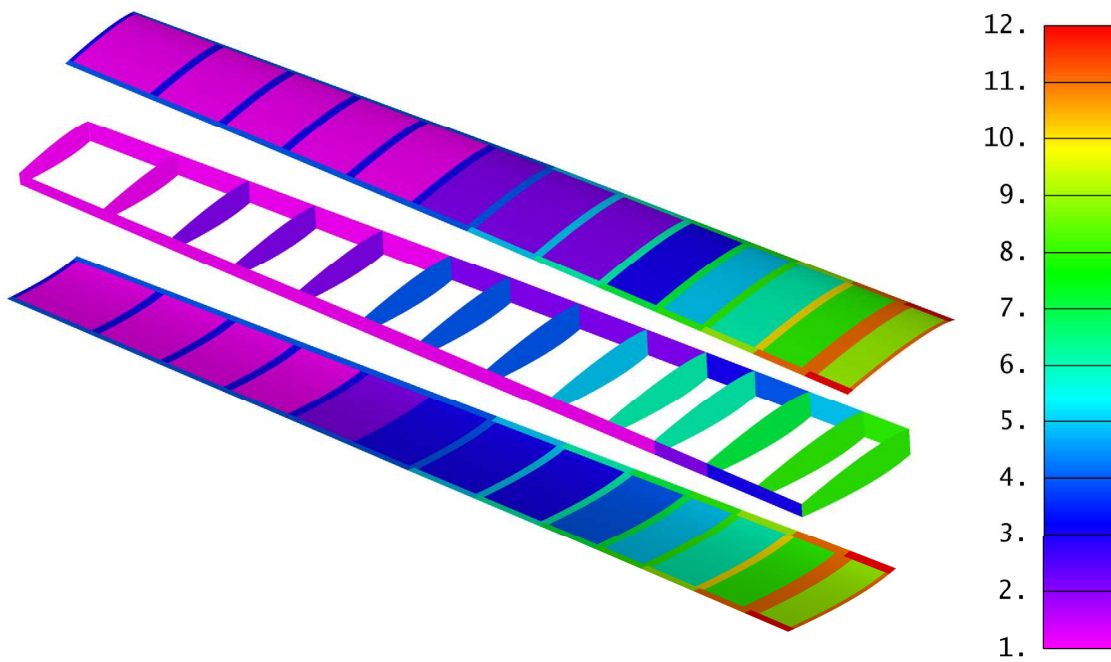


Figure 38 – Thickness distribution along the wing [mm].

The current wing has a weight of 154.84 kg. This weight could be optimized and reduced changing the thickness, geometry, location and quantity of each element (skin, ribs, stiffeners and spars) throughout the wing. However, this have not been done because of the complexity and computational processing time that would spend. Alternatively, a new material proposal has been discussed and analyzed: the carbon fiber epoxy Hexcel 8552 NMS 128/2.

The composite materials are fast becoming favorite materials for construction of aircrafts and spacecrafts [57]. They are widely used in the aerospace industry and have allowed engineers to overcome many structural obstacles. When combining materials, the final structural properties are highly improved, resulting in many advantages.

Weight reduction is the single greatest advantage of composite material usage and is the key factor in using it in aircraft structure. Fiber-reinforced matrix systems are stronger than traditional aluminum found on most aircraft, and they provide a smooth surface and increase fuel efficiency, which is a huge benefit.

Also, composite materials do not corrode as easily as other types of structures do. They hold up well in structural flexing environments and do not crack from metal fatigue. Composite designs also last longer than aluminum, which means fewer maintenance and repair costs.

However, it is important to point out some disadvantages. Because composite materials do not break easily, that makes it hard to tell if the interior structure has been damaged at all. This is the single most concerning disadvantage for using the composite material. In contrast, because of aluminum bends easily, it is quite easy to detect structural damage. Additionally, repairs can be much more difficult when a composite surface is damaged, which ultimately becomes costly.

Also, the resin used in composite material weakens at temperature as low as 150 °C, making it important for these aircraft to take extra precautions to avoid fires. Fires involved with composite materials can release toxic fumes and micro-particles into the air, causing health risks. Temperatures above 300 °C can cause structural failure. Finally, composite material can be expensive to manufacture.

Therefore, after all of these arguments, the structural analysis of the wing was reevaluated replacing the material from Aluminum 7050-T651 to composite prepreg material system Hexcel 8552 NMS 128/2. Its properties are listed in Table 23.

Table 23 - Composite material (Hexcel 8552 NMS 128/2) properties [58].

Property	Value
E_{11}	148 GPa
E_{22}	10.3 GPa
ν_{12}	0.27
G_{12}	5.9 GPa
G_{23}	5.9 GPa
G_{13}	5.9 GPa
X_{1t}	2439 MPa
X_{2t}	66 MPa
X_{1c}	2013 MPa
X_{2c}	381 MPa
S_{12}	78 MPa
S_{Bonding}	34.7 MPa
ρ	1577 kg/m ³

The carbon-epoxy plates are made of laminates with 0.2 mm of thickness, and their orientation start with 0° and varies with an increment of 45° for each layer. Thus, different plates were created following this pattern.

In addition, a new set of properties was added in Femap for each composite material thickness, as specified in Table 24. Now it is not specified per each structural element (e.g., skin, rib, spars and stiffeners), it is specified by thickness.

Table 24 - Classification of composite material properties in Femap.

Property ID	Property Type	Group of Elements
101	Laminate Plate	1 mm
102	Laminate Plate	2 mm
103	Laminate Plate	3 mm
104	Laminate Plate	4 mm
105	Laminate Plate	5 mm
106	Laminate Plate	6 mm
107	Laminate Plate	7 mm
108	Laminate Plate	8 mm
109	Laminate Plate	9 mm
110	Laminate Plate	10 mm
111	Laminate Plate	11 mm
112	Laminate Plate	12 mm

Thus, following the same process presented previously, the static analysis for the multiple set of critical load cases was evaluated for the new wing of composite material.

When analyzing structures in composite, it is not the maximum and minimum stress that are evaluated. Now it is the Tsai-Wu failure criterion [59]. Thus, an envelope of the results of this analysis using this composite fail criterion is presented in Figure 39.

The ribs were not substituted to composite material due to the high compression loads they are subjected to. Hence, when they were analyzed in composite, the thickness required to withstand the loads would be huge, resulting in a weight much greater if it were used aluminum. Therefore, they were kept in aluminum to save more weight. Their analysis is presented in Figure 40.

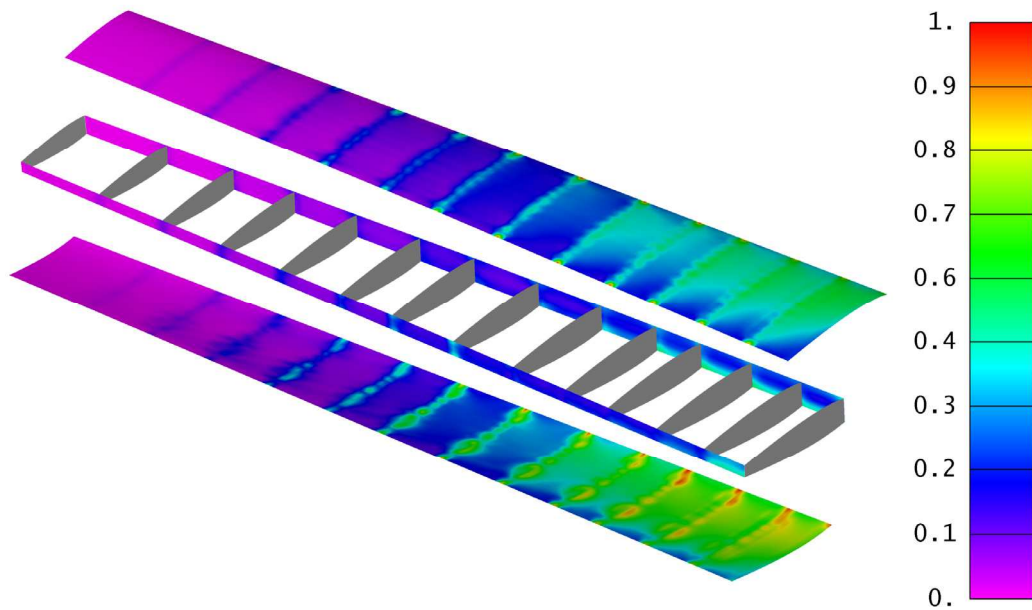


Figure 39 - Tsai-Wu failure criterion analysis.

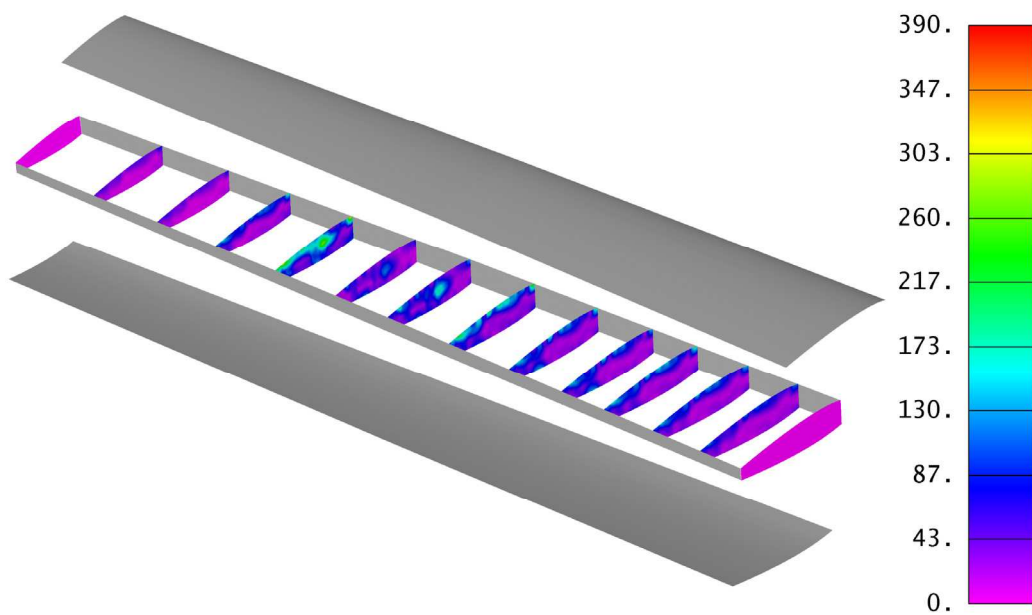


Figure 40 - Ribs maximum stress analysis [MPa].

The thicknesses of each element were manually readjusted in order to avoid any composite material failure. Next, a new buckling analysis was performed to ensure all elements were safe. The result of this analysis showed that some skin of composite material, for example, would fail for those current loads. Setting for the last time the elements' thicknesses, the buckling failures were fixed. Therefore, a final designed wing is obtained, which is presented in Figure 41.

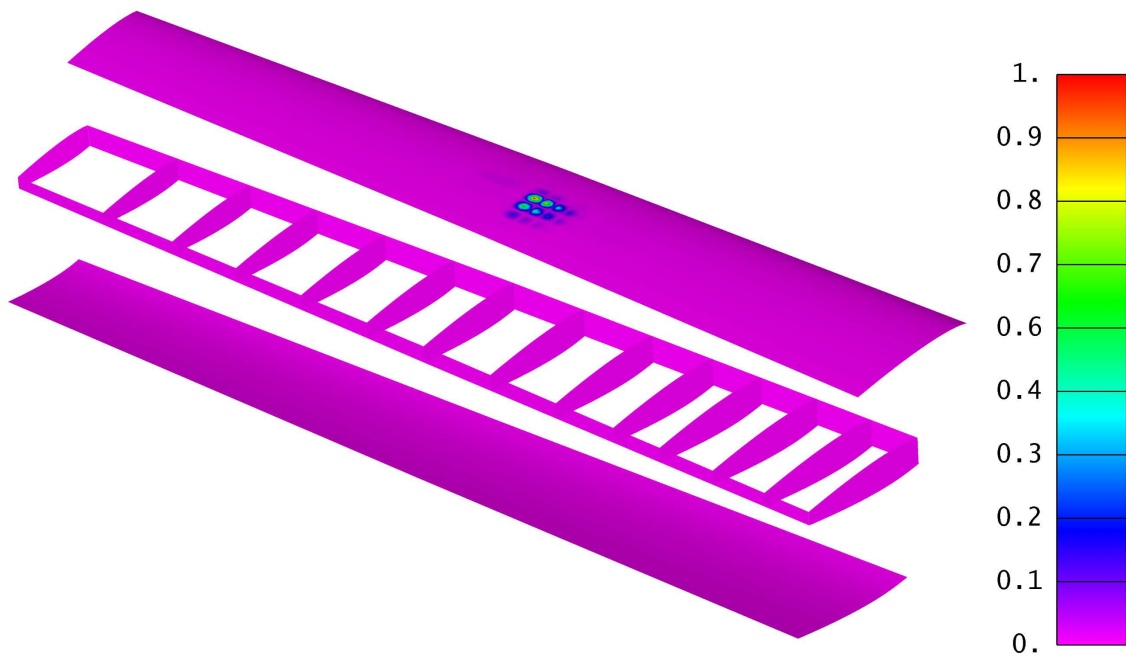


Figure 41 - Buckling analysis of the wing in composite material with failure at 106.3% of current loads.

The final wing weight is 115.21 kg, saving 39.63 kg when using composite material. The Figure 42 presents the thickness distribution of the final wing structure using carbon-epoxy.

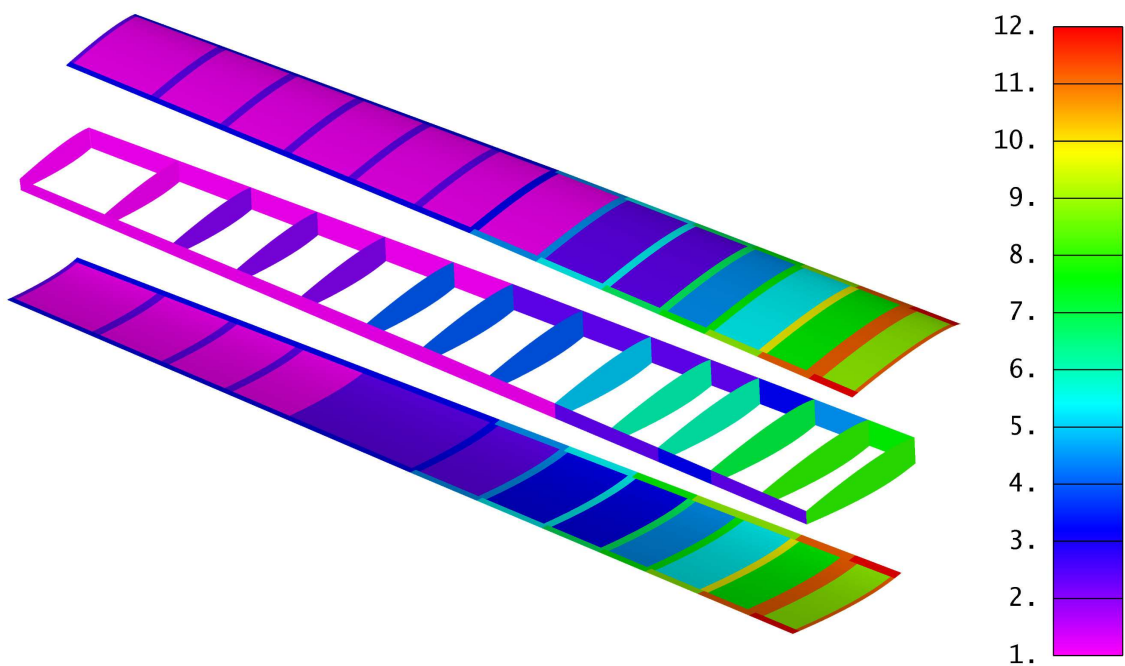


Figure 42 - Final thickness distribution of the composite material layers along the wing structure [mm].

From Figure 42, it is noticed that the layers at the root are very thick. This is necessary because the wing has a high AR and there is a large stress concentration in that region. This is a recurring issue on wings like that, but its reduction of aerodynamic drag, specifically induced drag, justifies its application.

5.2 Structural Empennage Design

The empennage design followed the same project philosophy presented in Section 5.1 for the wing design. Summarily, the Embraer Phenom 100's empennage was used as reference to estimate the number and position of the ribs and spars. The horizontal tail comprises ribs and spars, while the vertical tail is a multiple spars structure.

Each one of the aircraft (4PAX and 6PAX) have a different empennage, because each aircraft has different fuselage lengths and wing positions, what implies in different empennage sizes. Therefore, for each aircraft configuration it was designed in CATIA a specific vertical and horizontal tail, as presented in Figure 43. Their dimensions are available in Table 15.

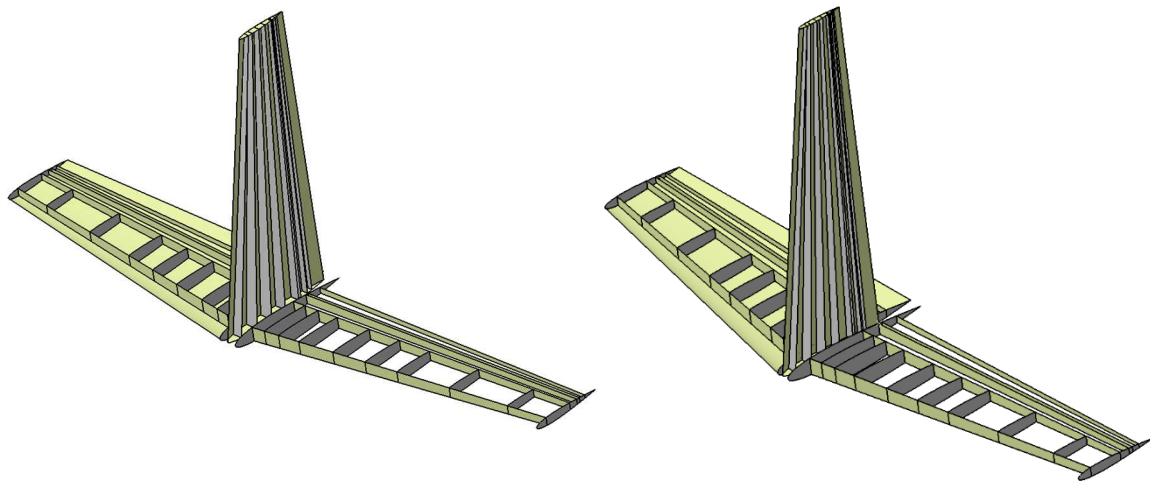


Figure 43 - Empennages of the 4PAX (on the left) and 6PAX (on the right) aircraft.

The structural empennage design and analyses for both aircraft are presented in the following sections.

5.2.1 Structural Horizontal Tail Design

Similar to the structural wing design, the initial horizontal tail models for both aircraft were made using CATIA V5R21[®], so it was possible to refine the curvatures and airfoil profiles. Next, the models were exported to HyperMesh[®] to create the meshes. The mesh size was designed in order to allow a satisfactory finite element analysis and to avoid an excessive computational consumption. Thus, it was used the quadrilateral-mesh with an element size of 10 mm, generating a mesh for the horizontal tails of 21274 elements for the 4PAX and 19456 elements for the 6PAX, as presented in Figure 44.

Moreover, for each rib it was created a RBE3 (rigid body element) to connect a single node to multiple nodes, allowing each rib to subject to a specific bending and torsion moments at 25% of the tail chord, depending on its location.

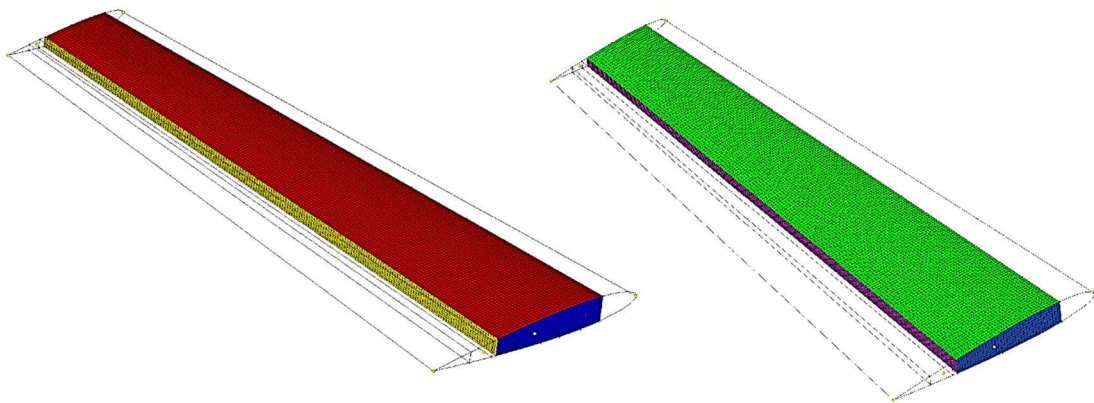


Figure 44 - 4PAX (on the left) and 6PAX (on the right) horizontal tails after meshing in HyperMesh.

Having the models done, they were imported in Femap with NX Nastran[®], where some groups of components were created to make the analyses easier. The groups were split into the following components, as described in different colors in Figure 45:

- Upper skin (Red group);
- Lower skin (Blue group);
- Ribs (Green group);
- Spar I (Orange group);
- Spar II (Light blue group);
- Spar III (Purple group);
- Upper stiffeners (Dark red group);
- Lower stiffeners (Dark blue group).

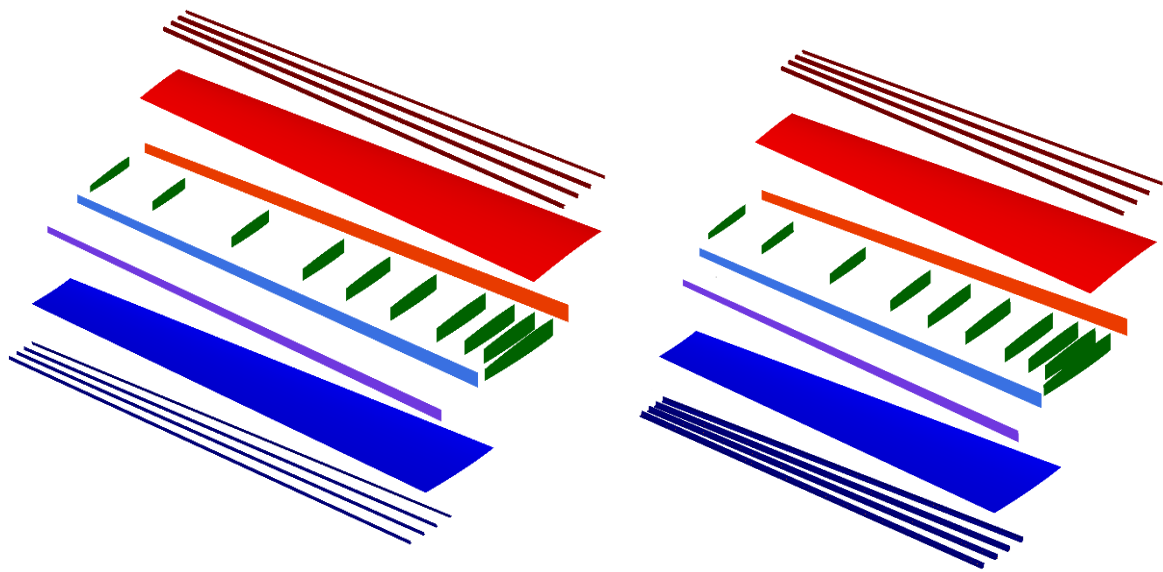


Figure 45 - 4PAX (on the left) and 6PAX (on the right) horizontal tails exploded and groups classified by colors.

Initially, the Aluminum 7050-T651 was chosen as the material for both horizontal tails, likewise in wing design. Its properties are presented in Table 20.

In addition, it was created a property with random thickness for each element of each group, allowing change and adjustment individually. Some elements were created as plate type and others as beam type. The ribs and spars were initially made as plate type, but they are more appropriate as beam type. Thus, it was also created a new property to represent the spar and rib tabs, making them into beams. The classification of properties follows the pattern in Table 25.

Table 25 - Classification of properties in Femap.

Property ID	Property Type	Group of Elements
110000 – 119999	Plate	Upper skin
210000 – 219999	Plate	Lower skin
310000 – 319999	Plate	Ribs
410000 – 419999	Plate	Spar I
510000 – 510999	Plate	Spar II
511000 – 511999	Plate	Spar III
610000 – 619999	Beam	Upper stiffeners
710000 – 719999	Beam	Lower stiffeners
810000 – 819999	Plate	Spar-tab
910000 – 919999	Plate	Rib-tab

Therefore, coloring each element with a different color, representing different properties, the following full models are presented in Figure 46.

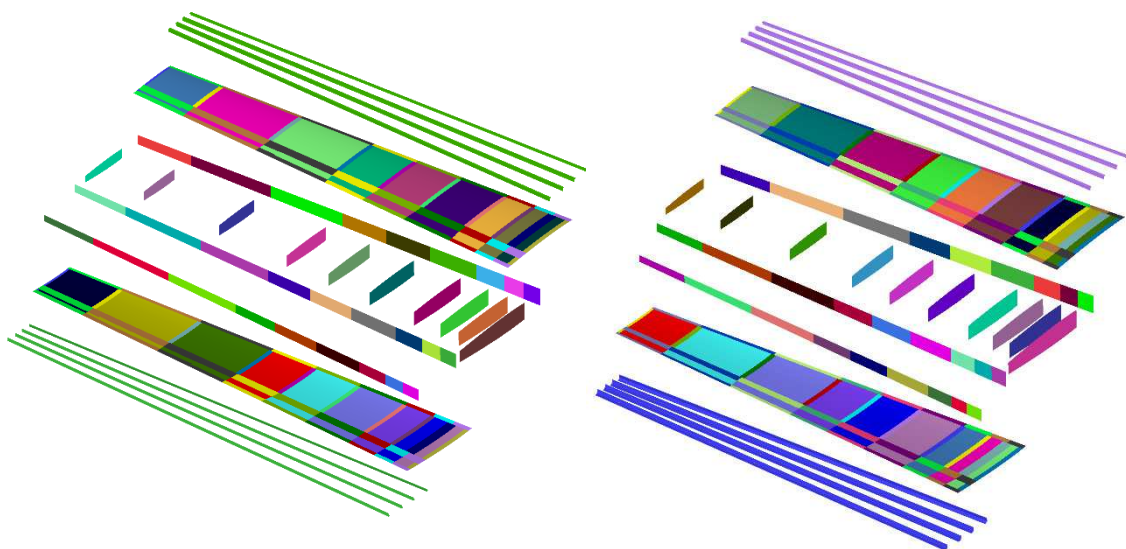


Figure 46 - 4PAX (on the left) and 6PAX (on the right) horizontal tails exploded and properties classified by colors.

Next, it comes up with the static analysis of both horizontal tails. The load cases evaluated in this analysis are those presented in Table 18 and Table 19. Each one of them was associated to a pattern code, so it was easy to study each load case in Femap. They are described in Table 26.

Table 26 - Load cases classifications in Femap.

Aircraft	Load Case	Number
4PAX	Dive at V_C	1
	Dive at V_D	2
	Bunt at V_C	3
	Coordinate turn - RHS	4
	Coordinate turn - LHS	5
6PAX	Dive at V_C	1
	Dive at V_D	2
	Bunt at V_C	3
	Coordinate turn - RHS	4
	Coordinate turn - LHS	5

Hence, for each load case it was necessary to determine and apply the specific loads in each RBE3. Thus, using the bending and torsion moments distributions from Figure 24 and Figure 25, the loads were calculated and corrected for the actual Femap reference axes.

The static analysis for both horizontal tails was performed equally to the wing case, as presented in Section 5.1. In other words, the thicknesses were readjusted during the entire process, so the structures did not fail, even for buckling. The final designed horizontal tails in aluminum have a weight of 15.63 kg and 15.39 kg, for the 4PAX and 6PAX respectively.

Next, the material was substituted for the same composite used in Section 5.1, the Hexcel 8552 NMS 128/2. When the wing structure was designed, the ribs were not substituted to composite material due to the high compression loads, so they were kept in aluminum. Now, both empennage ribs presented good results when applied composite material, so they were kept in carbon-epoxy.

Thus, once more both structures were redesigned changing the thickness and appliance of layers in each element group so that they meet all the fail criteria. The Tsai-Wu failure criteria envelopes of results are displayed in Figure 47 and Figure 48, for 4PAX and 6PAX, respectively.

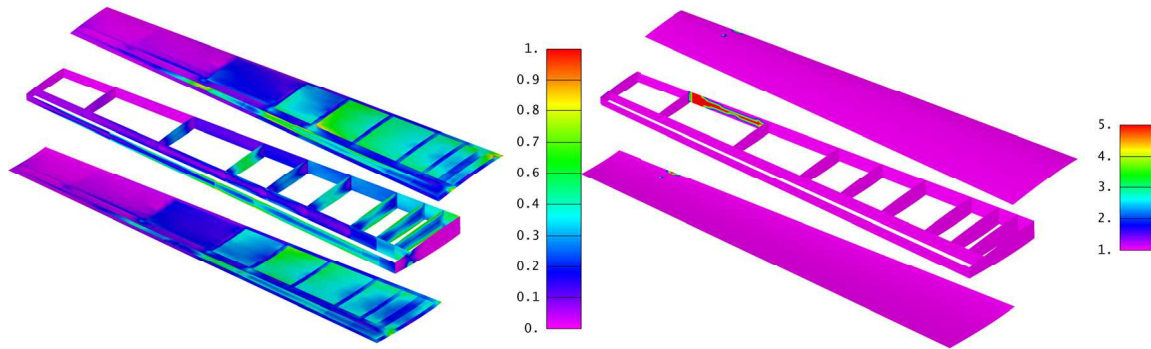


Figure 47 - Tsai-Wu failure criterion envelope (on the left) and load cases envelope (on the right) for the 4PAX horizontal tail.

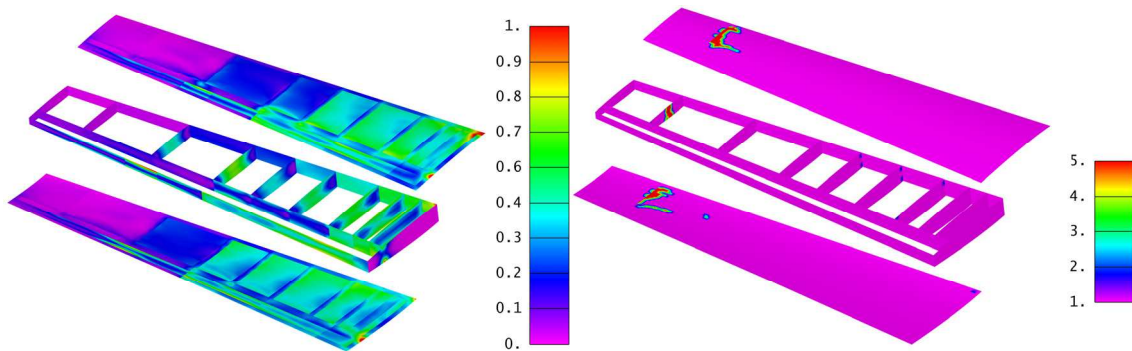


Figure 48 - Tsai-Wu failure criterion envelope (on the left) and load cases envelope (on the right) for the 6PAX horizontal tail.

Next, the buckling analysis was performed for both horizontal tails. Since some elements were failing, the thicknesses were increased until the structures did not fail anymore. Then, after the adjustment, the 4PAX and 6PAX horizontal tails would buckle applying a load of 104.5% and 106.4% of the current load, respectively. The buckling criteria results are presented in Figure 49.

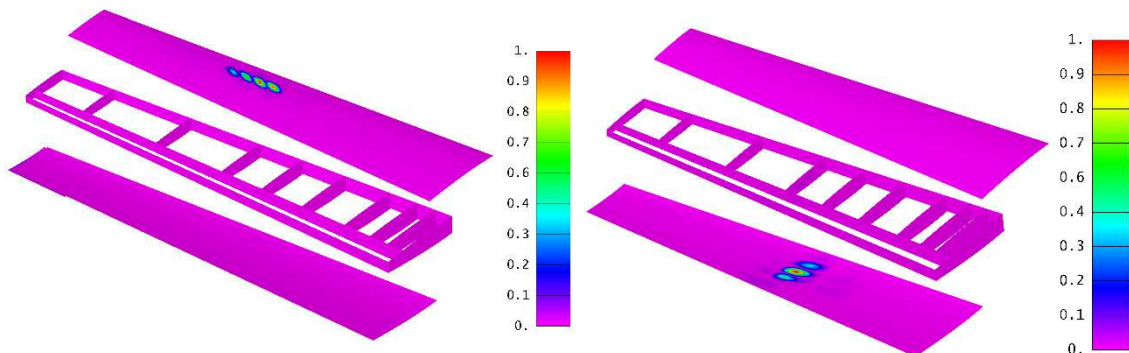


Figure 49 - Buckling analysis of the 4PAX (on the left) and 6PAX (on the right) horizontal tails in composite material with failure at 104.5% and 106.4% of current loads, respectively.

Finally, both horizontal tails are well designed for the current loads, resulting in the following thickness distributions, presented in Figure 50. The change of material from aluminum to composite resulted in a reduction of 4.36 kg and 4.65 kg for 4PAX and 6PAX, respectively. Therefore, they ended weighting 11.27 kg and 10.72 kg, respectively.

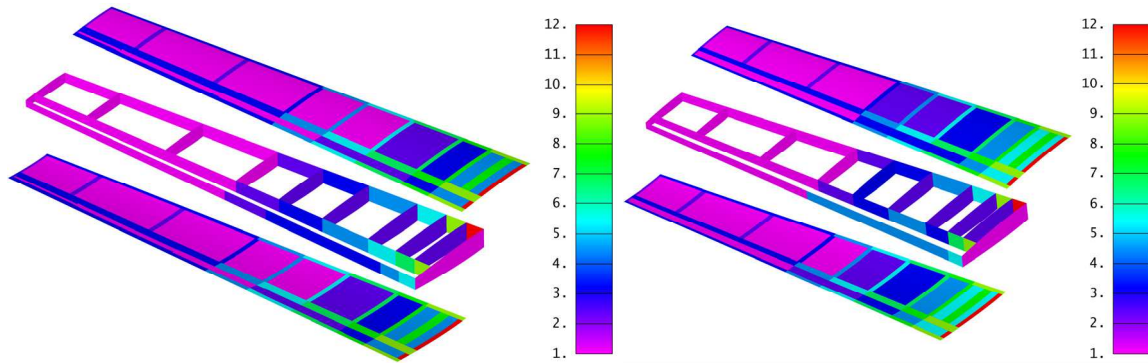


Figure 50 - Final thickness distribution of the composite material layers along the 4PAX (on the left) and 6PAX (on the right) horizontal tails.

The horizontal tails, in addition to concentrating stresses at the root due to the high load values, also suffer with the influence of twisting, unlike the wing structure. It is possible to notice that the sixth rib (counting from the root) is thicker than the root due to the twist in that location.

5.2.2 Structural Vertical Tail Design

Similar to the structural wing design, the initial vertical tail models for both aircraft were made using CATIA V5R21[®], so it was possible to refine the curvatures and airfoil profiles. Next, the models were exported to HyperMesh[®] to create the meshes. The mesh size was designed in order to allow a satisfactory finite element analysis and to avoid an excessive computational consumption. Thus, it was used the quadrilateral-mesh with an element size of 10 mm, generating a mesh for the vertical tails of 21514 elements for the 4PAX and 16555 elements for the 6PAX, as presented in Figure 51.

Since the vertical tails have no ribs, only multiple spars, four rigid points were created to apply locally the bending and torsion moments. Thus, for each of these points it was created a RBE3 (rigid body element).

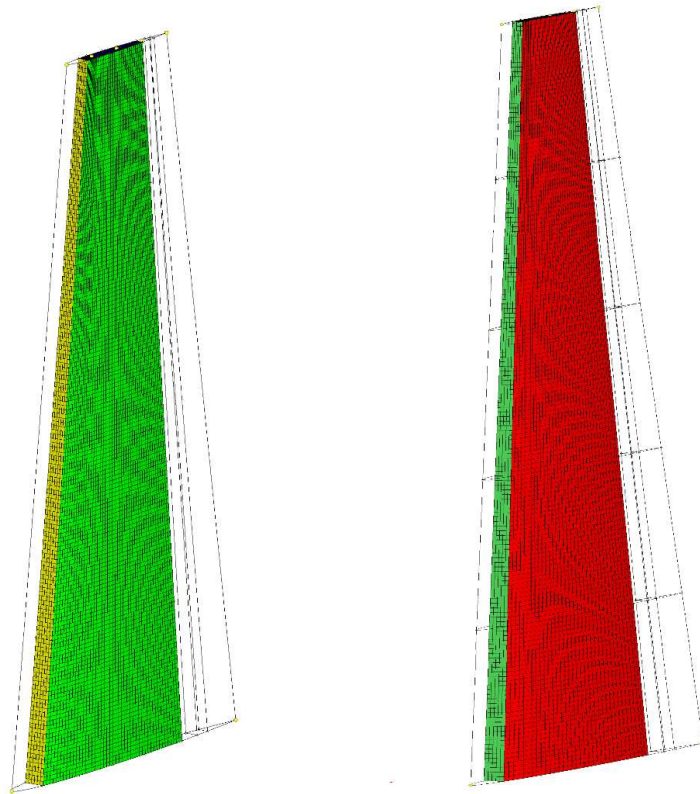


Figure 51 - 4PAX (on the left) and 6PAX (on the right) vertical tails after meshing in HyperMesh.

Having the models done, they were imported in Femap with NX Nastran[®], where some groups of components were created to make the analyses easier. The groups were split into the following components, as described in different colors in Figure 52:

- Left skin (Blue group);
- Right skin (Red group);
- Spar I (Purple group);
- Spar II (Light blue group);
- Spar III (Light brown group);
- Spar IV (Light pink group);
- Spar V (Orange group);
- Spar VI (Dark blue group);
- Ribs (Green group).

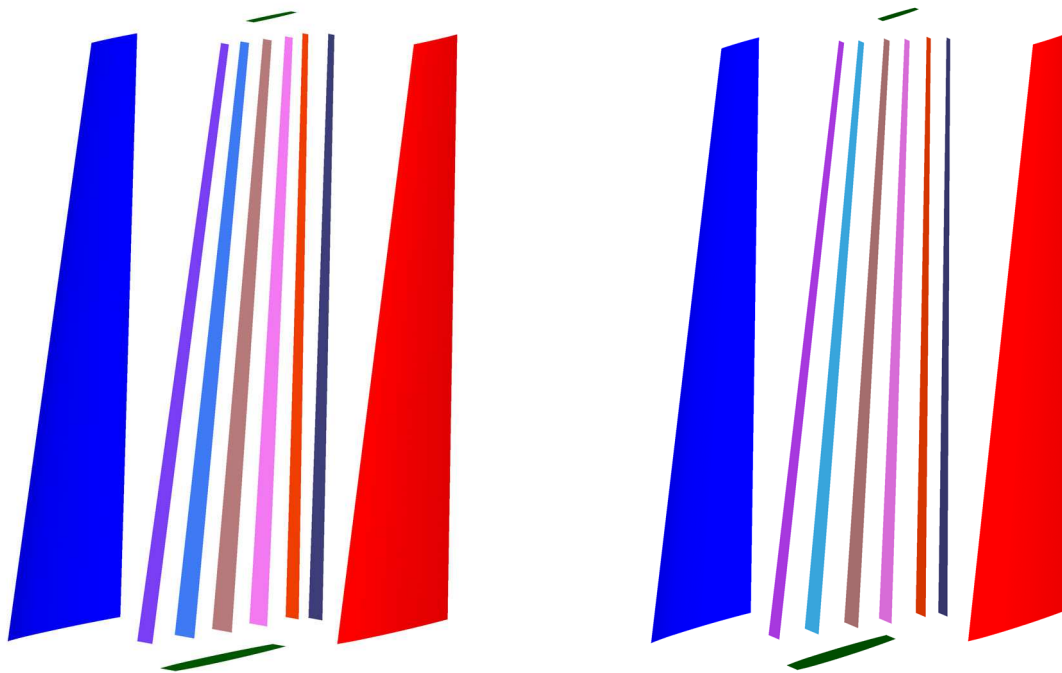


Figure 52 - 4PAX (on the left) and 6PAX (on the right) vertical tails exploded and groups classified by colors.

Initially, the Aluminum 7050-T651 was chosen as the material for both vertical tails, likewise in wing design. Its properties are presented in Table 20.

In addition, it was created a property with random thickness for each element of each group, allowing change and adjustment individually. All elements were created as plate type. But the spars are more appropriate as beam type. Thus, it was also created a new property to represent the spar tabs, making them into beams. The classification of properties follows the pattern in Table 27.

Table 27 - Classification of properties in Femap.

Property ID	Property Type	Group of Elements
120000 – 129999	Plate	Upper skin
220000 – 229999	Plate	Lower skin
320000 – 329999	Plate	Spar I
420000 – 429999	Plate	Spar II
520000 – 520999	Plate	Spar III
620000 – 629999	Plate	Spar IV
720000 – 729999	Plate	Spar V
820000 – 829999	Plate	Spar VI
920000 – 929999	Plate	Spar-tab

Therefore, coloring each element with a different color, representing different properties, the following full models are presented in Figure 53.

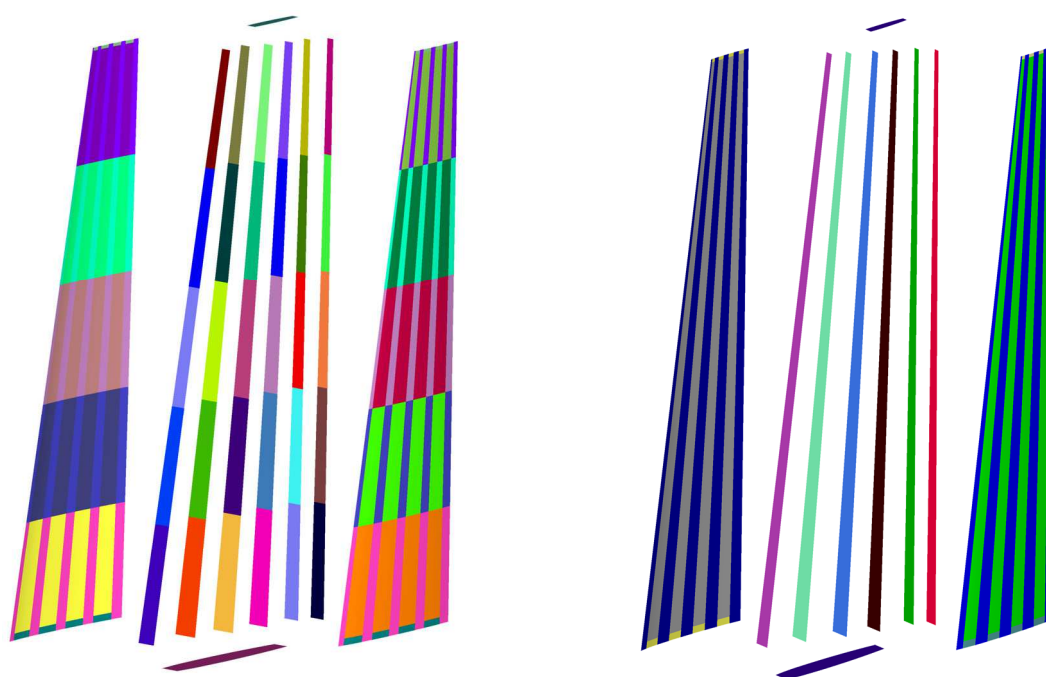


Figure 53 - 4PAX (on the left) and 6PAX (on the right) vertical tails exploded and properties classified by colors.

Next, it comes up with the static analysis of both vertical tails. The load cases to be evaluated in this analysis are those presented in Tables Table 18 and Table 19. Since

the vertical tail is only affect by coordinate turn condition, only this load case was evaluated. Its Femap code is described in Table 28.

Table 28 - Load cases classifications in Femap.

Aircraft	Load Case	Number
4PAX	Coordinate turn - RHS	1
	Coordinate turn - LHS	2
6PAX	Coordinate turn - RHS	1
	Coordinate turn - LHS	2

Hence, for these load cases it was necessary to determine an apply the specific loads in each RBE3. Thus, using the bending and torsion moments distributions from Figure 24 and Figure 25, the loads were calculated and corrected for the actual Femap reference axes.

The static analysis for both vertical tails was performed equally to the wing case, as presented in Section 5.1. In other words, the thicknesses were readjusted during the entire process, so the structures did not fail, even for buckling. The final designed horizontal tails in aluminum have a weight of 5.72 kg and 4.23 kg, for the 4PAX and 6PAX respectively.

Next, the material was substituted for the same composite used in Section 5.1, the Hexcel 8552 NMS 128/2. Thus, once more both structures were redesigned changing the thickness and appliance of layers in each element group so that they meet all the fail criteria. The Tsai-Wu failure criteria envelopes of results are displayed in Figure 54 and Figure 55, for 4PAX and 6PAX, respectively.

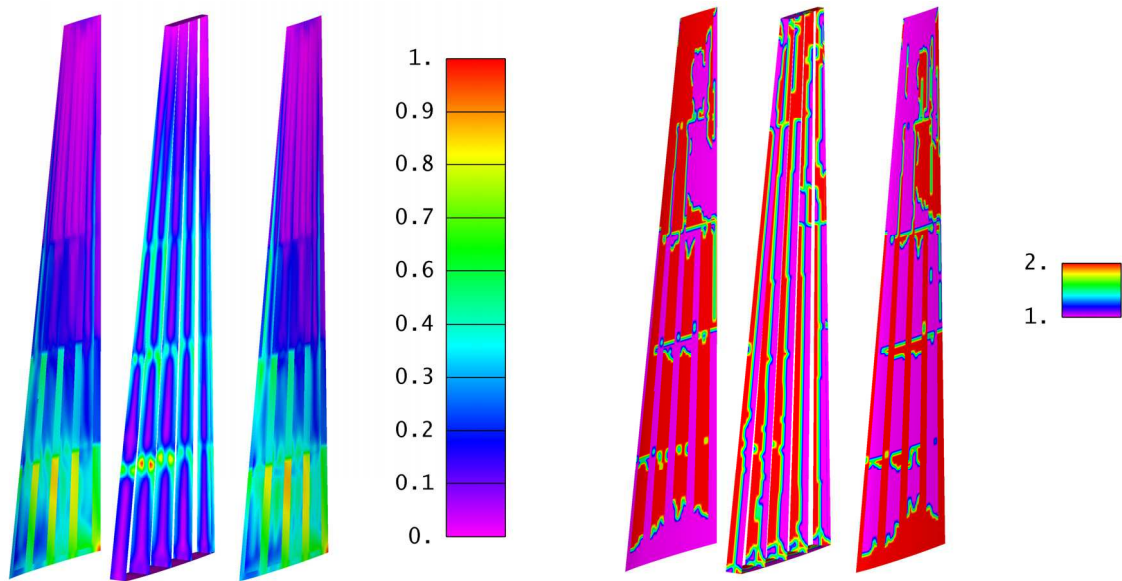


Figure 54 - Tsai-Wu failure criterion envelope (on the left) and load cases envelope (on the right) for the 4PAX vertical tail.

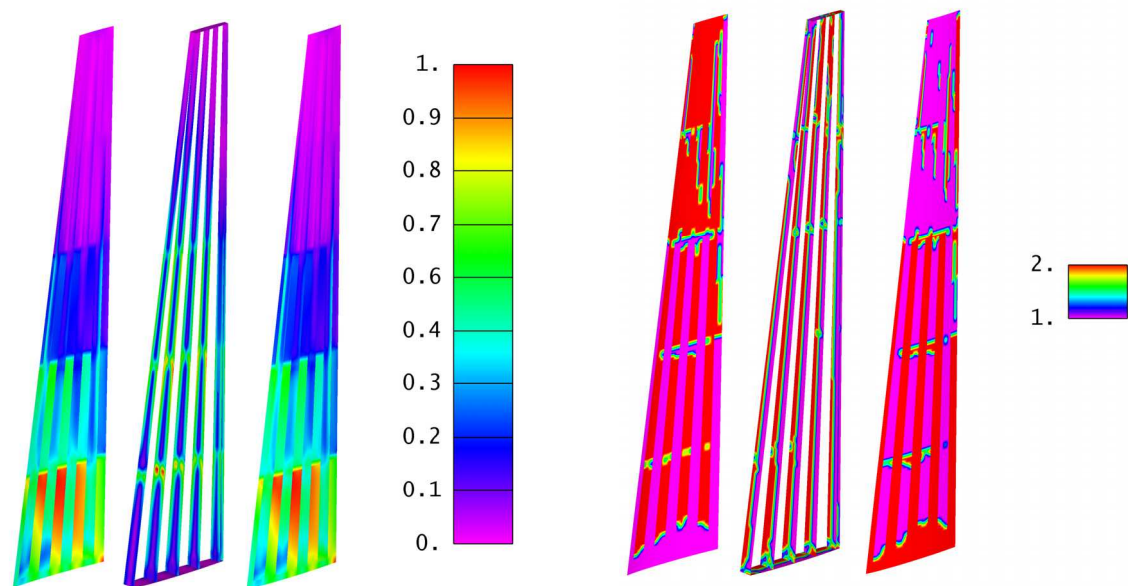


Figure 55 - Tsai-Wu failure criterion envelope (on the left) and load cases envelope (on the right) for the 6PAX vertical tail.

Next, the buckling analysis was performed for both vertical tails. Since some elements were failing, the thicknesses were increased until the structures did not fail anymore. Then, after the adjustment, the 4PAX and 6PAX vertical tails would buckle applying a load of 138.6% and 148.0% of the current load, respectively. The buckling criteria results are presented in Figure 56.

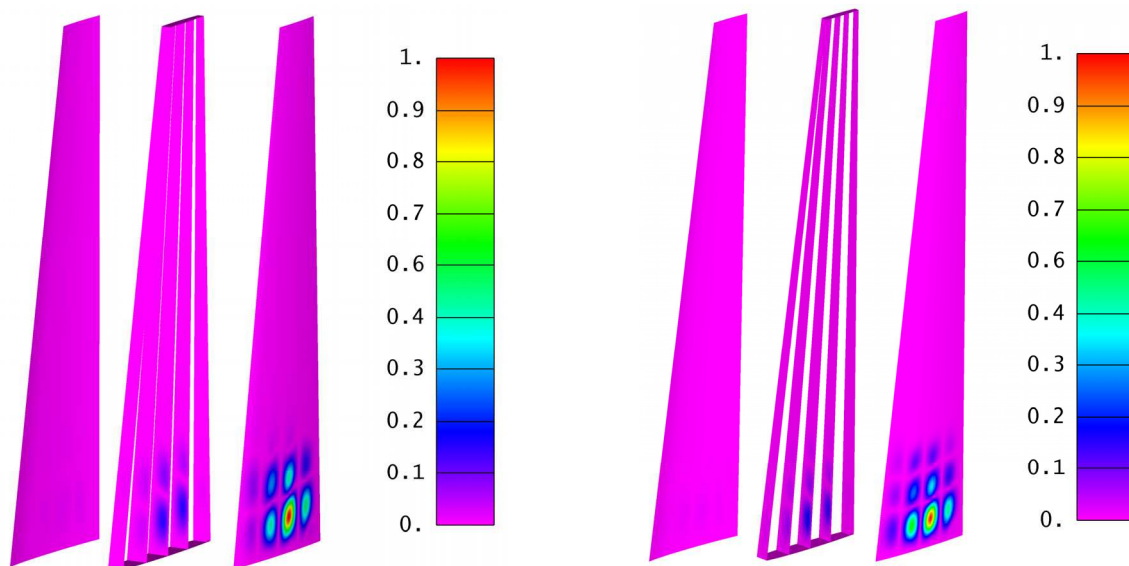


Figure 56 - Buckling analysis of the 4PAX (on the left) and 6PAX (on the right) vertical tails in composite material with failure at 138.6% and 148.0% of current loads, respectively.

Finally, both vertical tails are well designed for the current loads, resulting in the following thickness distributions, presented in Figure 57. The change of material from aluminum to composite resulted in a reduction of 2.26 kg and 1.28 kg for 4PAX and 6PAX, respectively. Therefore, they ended weighting 3.46 kg and 2.95 kg, respectively.

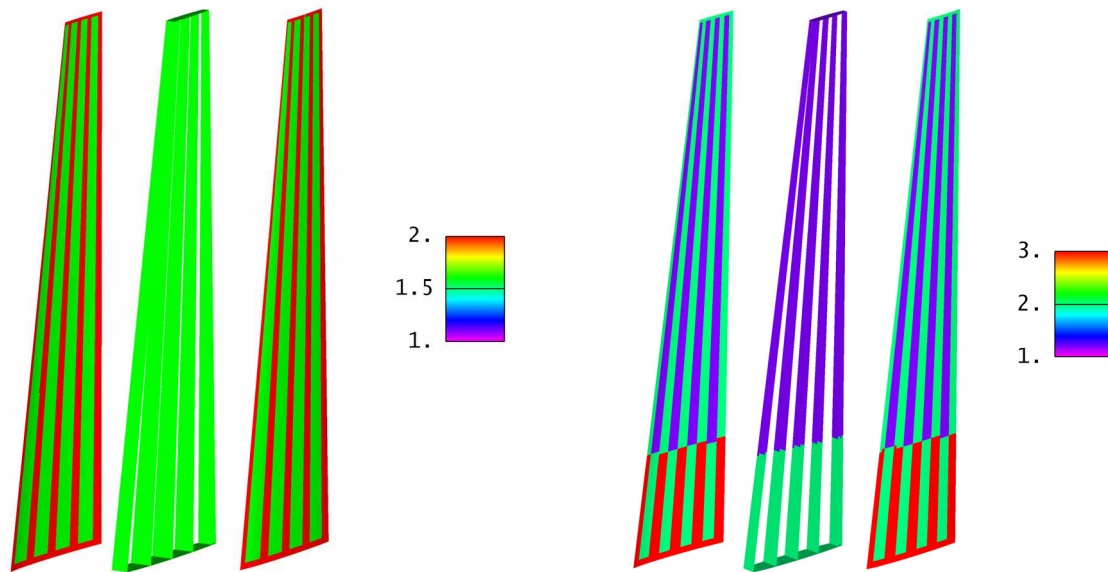


Figure 57 - Final thickness distribution of the composite material layers along the 4PAX (on the left) and 6PAX (on the right) vertical tails.

The vertical empennage thicknesses ended thinner because the loads applied to the rudder are relatively lower and its size is relatively small. Its structure is made of multiple spars, increasing considerably its resistance to bending, which is the main stress the rudder is subjected to.

CHAPTER VI

AEROELASTIC ANALYSIS

*“Any sufficiently advanced technology
is indistinguishable from magic.”*

Arthur C. Clarke

With the increase in worldwide aviation over the past few decades, aircraft aeroelastic analysis has become essential to ensure safety and competitiveness in new aeronautical projects.

The aerospace industries spend their effort trying to develop new, large, economically and ecologically competitive civil transport aircraft, improving the size and payload capacities beyond the current limits of existing aircraft. But this can bring up undesirable aeroelastic effects [60].

Negative aeroelastic effects like increased drag at cruising speed due to changes in wing twist, aileron reversal or flutter at a late stage of the product development, e.g. in ground vibration tests or even in-flight tests, would cause severe economic problems. These may only be overcome by considerable changes in structural design or mass distribution, resulting in weight penalty or even unacceptable limitations of the flight envelope.

A major redesign to improve the aeroelastic properties with an optimized and least-weight design is impossible because of the huge number of interdependencies of the different disciplines, departments or even companies involved in the design of a new aircraft.

The only way to avoid these problems is to meet the uncertainties of the influences of aeroelastic behavior on structural weight, flight performance, handling characteristics and finally economy as soon as possible within the aircraft design process [61]. That means, aeroelastic properties have to be taken into account in the conceptual or at least early preliminary design phase. Therefore, it is very important to implement new and more sophisticated models in the conceptual design and rely on statistical data of some aircraft. This option becomes even more attractive, since the introduction of new technologies and materials promise to help reducing the weight of structures and systems. But their benefit and potential risks have to be assessed thoroughly.

Thus, this chapter will present an aeroelastic analysis of the wing designed in Chapter 3. The empennage (horizontal and vertical) will not be evaluated due to its lack of a model that truly represents the boundary conditions, since the empennage interacts directly with the rear fuselage, what considerably changes the solution, releasing false results.

Firstly, it was evaluated the displacement and rotation values for the carbon-epoxy wing when applying the loads specified in Table 22. These values are displayed in Figure 58.

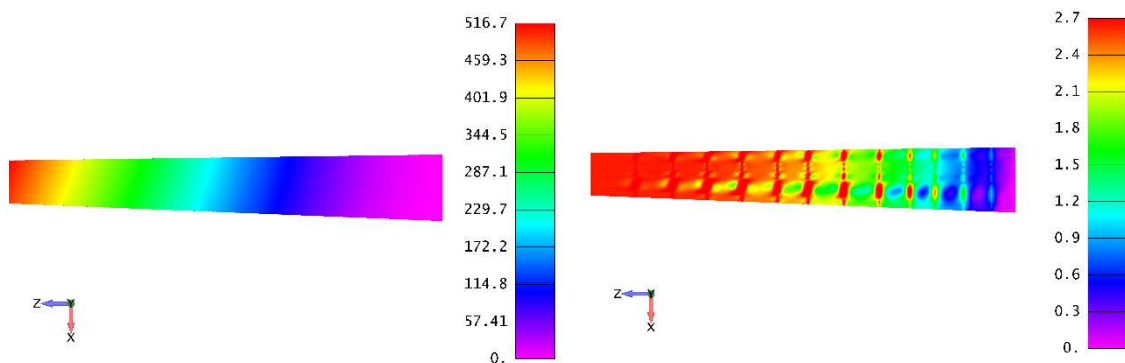


Figure 58 - Total displacements in mm (on the left) and total rotation in degrees (on the right) for the carbon-epoxy wing when applying the loads from Table 22.

According to literature [62], the displacement presented in Figure 58 are among the theoretical values expected for this type of wing. However, the rotation values are much lower, which is curious. It is explained basing on how the wing has been designed.

A bunch quantity of ribs in aluminum was applied in the structure, since the reference was the Embraer Phenom 100, as presented in Section 5.1. Hence, an improved torsion resistance is obtained. If an optimization had been implemented, probably it would be necessary less ribs in the wing design.

Next, a modal analysis of the wing model was firstly performed in Femap in order to find the natural frequencies, which are presented in Table 29. The first four modes are mostly bending-modes, as observed in Figure 59. The twisting-mode only shows up in the fifth mode (Figure 60), which is explained by high strength of the wing related to torsion moments, as mentioned before.

Table 29 - Natural frequencies obtained from wing modal analysis.

Mode Shapes	Natural Frequencies (Hz)
1	10.66
2	32.80
3	43.04
4	90.27
5	100.03
6	125.84
7	129.43
8	141.03
9	142.25
10	183.44

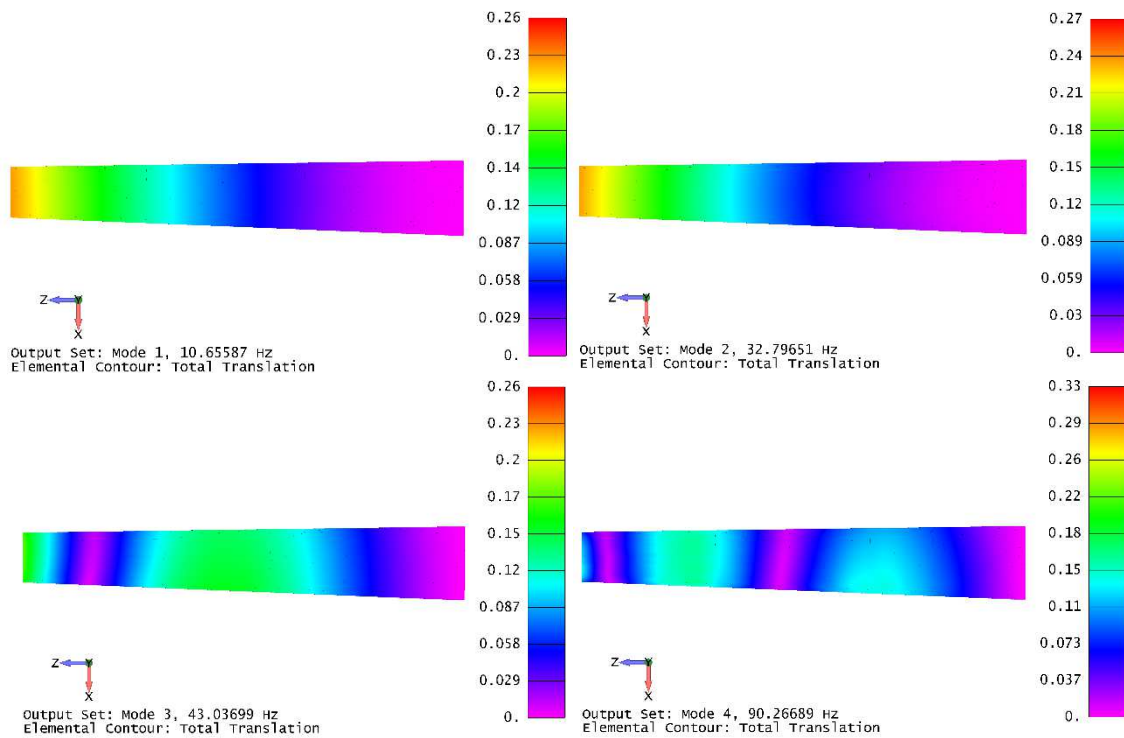


Figure 59 - First four mode shapes of the wing.

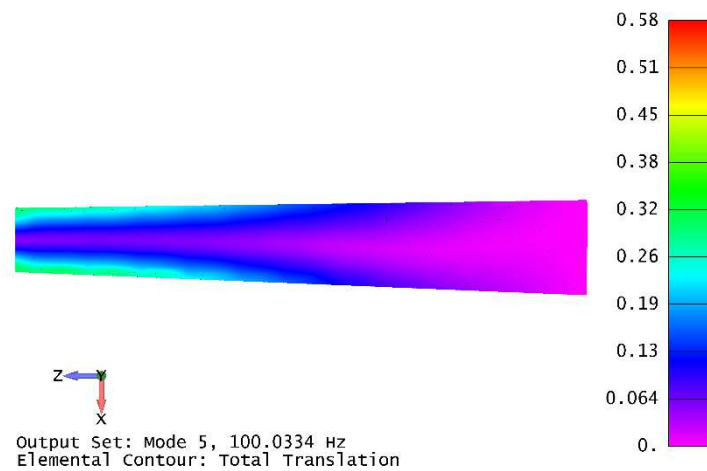


Figure 60 - Fifth mode shape of the wing.

Next, a divergence and flutter analysis was performed using the NX Nastran[®] SOL 145. First of all, the aerodynamic mesh was created to emulate the air flow through the wing (Figure 61). It seems to be bigger than wing structure, but it has that size to represent the entire wing area, since the wing structure in the model is only a wing box structure.

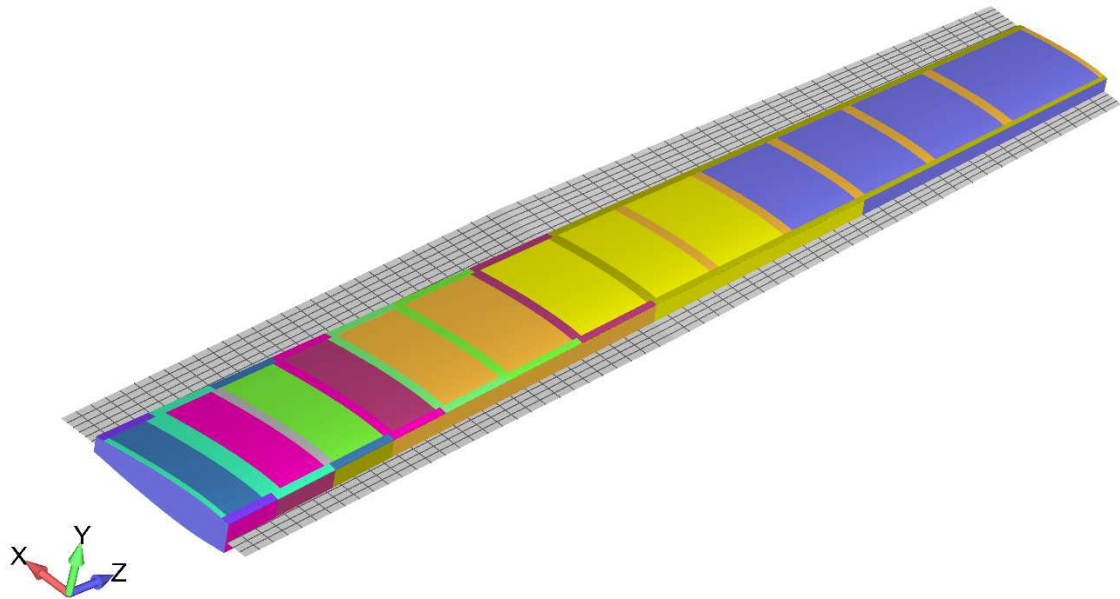


Figure 61 - Aerodynamic and structural meshes in Femap.

The association of both aerodynamic and structural meshes was made through a SPLINE1 connection on the nodes that intersect the following structures: upper skin, ribs and stiffeners. This allows to distribute the aerodynamic loads over the entire wing elements, similarly to the RBEs in the structure design presented in Section 5.1. These nodes are presented as small blue dots in Figure 62.

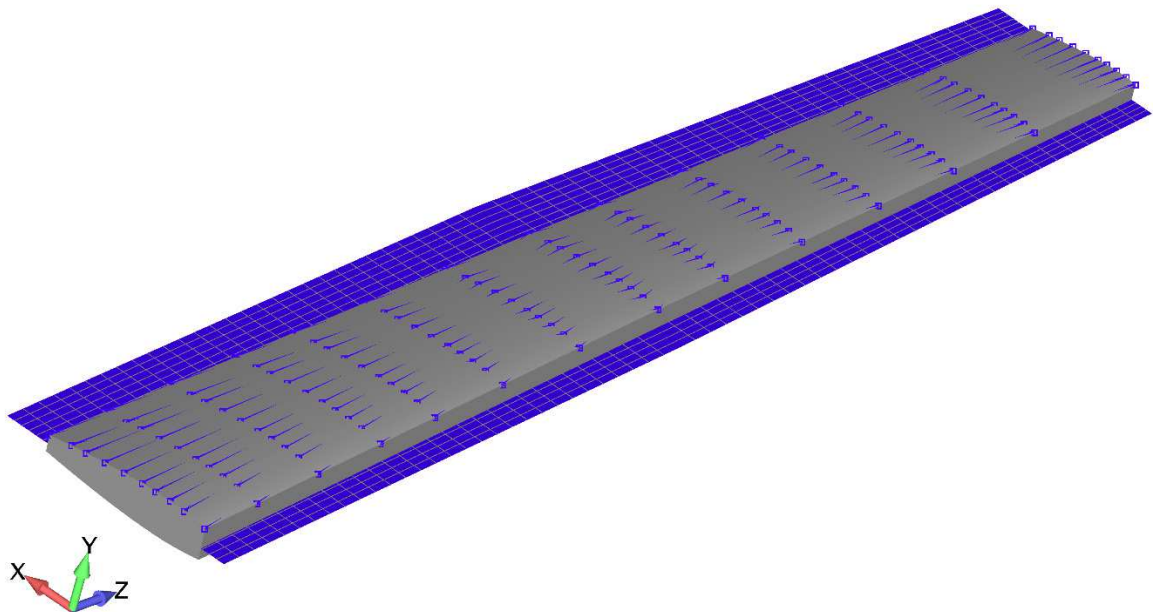


Figure 62 - Nodes associated with the SPLINE1.

Moreover, it was considered the first five mode shapes of the structure, and method used was PK-NL. The air density of 0.818 kg/m^3 was the equivalent of a flight altitude of 12000 ft (cruise), and the density of reference was taken at sea-level (ISA + 10°C). The airspeed was evaluated from 20 to 200 m/s TAS, representing the flight speed envelope of the aircraft.

This analysis released the following results, presents as V-g-f plots in Figure 63.

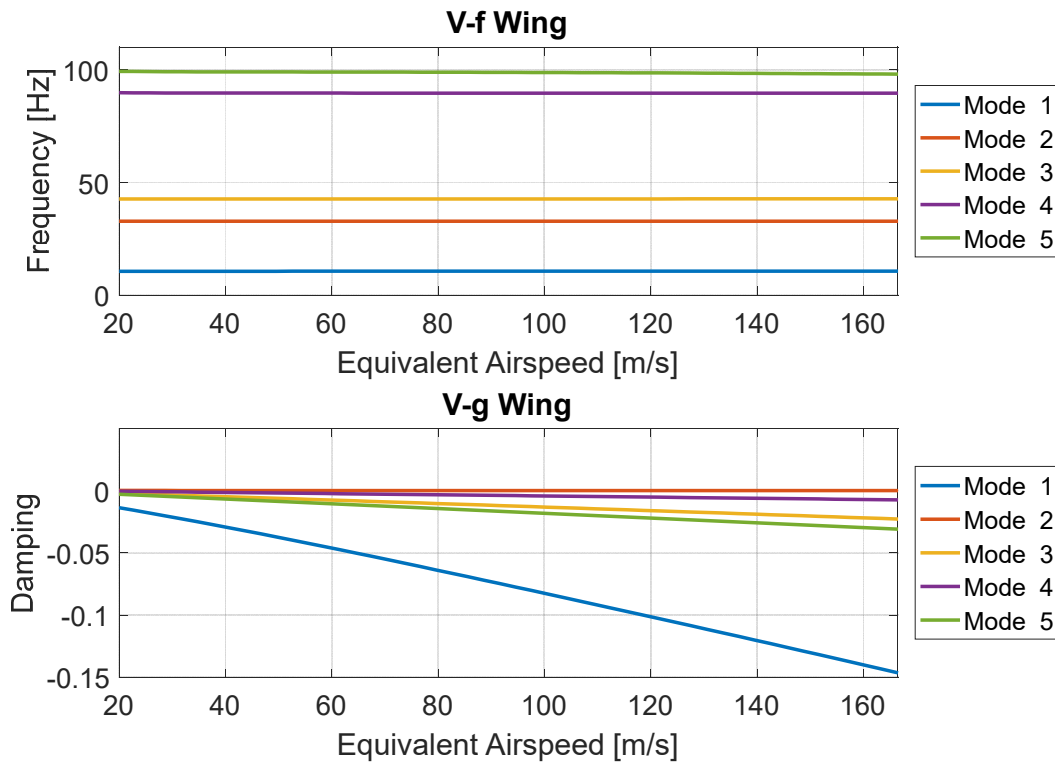


Figure 63 - V-g-f diagram for the aircraft velocities envelope.

Analyzing the V-g and V-f plots, there is no flutter and divergence phenomenon for the aircraft velocities envelope. This is due to the high natural frequencies and mode shapes that need very high speed to couple and, consequently, bring the system do instability. In addition, the composite material (Hexcel 8552 NMS 128/2) applied contributes to increase the stiffness of the wing, improving the aeroelastic responses.

To verify when the instabilities occur, the interval of velocities was expanded to 1000 m/s TAS. Hence, a new analysis was evaluated, resulting in the following V-g-f plots presented in Figure 64.

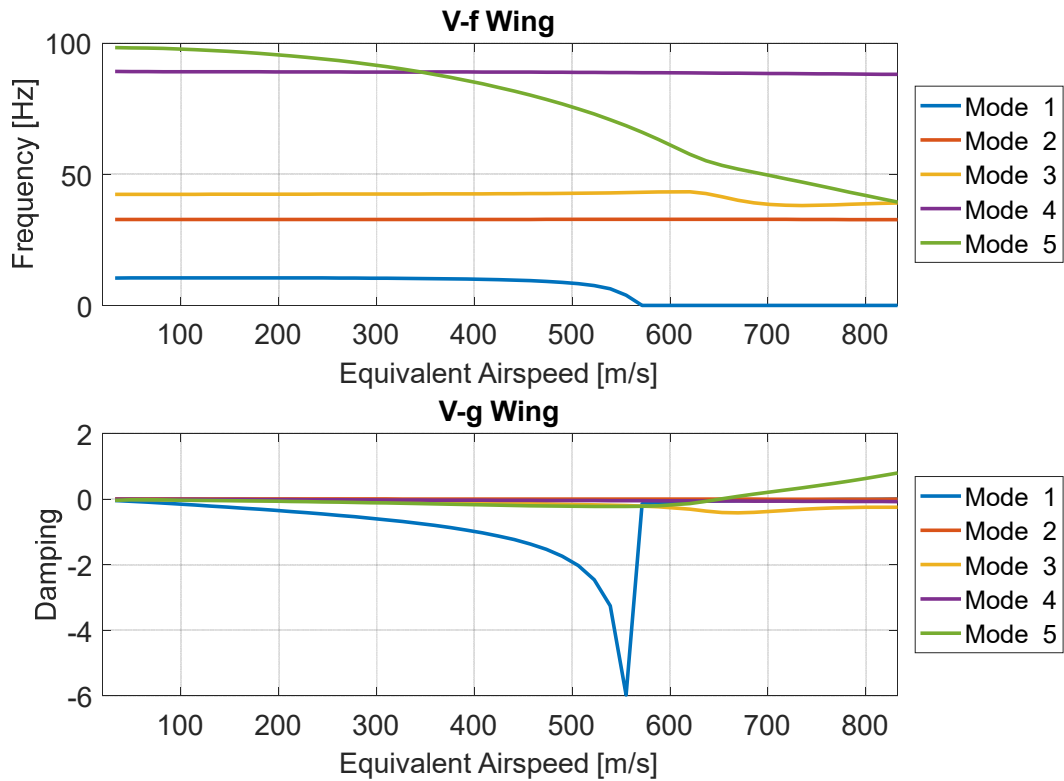


Figure 64 - V-d-f diagram for the expanded aircraft velocities envelope.

Now, it is possible to observe the flutter showing up at 650 m/s EAS, where the third (bending) and fifth (twisting) modes are coupled, as shown in V-f plot. Consequently, the damping of mode 5 becomes positive ($g > 0$) at this speed, as presented in V-g plot.

The flutter speed is very high because the wing structure has a great strength, delaying the coupling of natural frequencies of the bending and twisting modes.

CHAPTER VII

CONCLUSION AND PERSPECTIVES

This work is based on the development of two hybrid-electric aircraft of 4 and 6 passengers, respectively. The requirements for the project were proposed by the annual design competition sponsored by the AIAA.

The design methodology presented by Raymer and Venson allowed to create an initial aircraft estimate using several historical data. At this stage, four main different configurations of aircraft were considered: twin-engine with T-tail, twin-engine with conventional tail, single-engine with T-tail, and single-engine with conventional tail. The configuration that most met the required specifications was the single-engine with conventional tail.

Two architectures of hybridization were presented: the architecture in series e in parallel. It was considered the complexity, efficiency and weight of the configurations for the project, what led to the architecture in series to be chosen for the propulsive system, since it is simpler and is easier to implement in small aircraft. Thus, it was developed a new formulation of equations for range from the analysis of the energy stored variation during the flight time.

To complete the conceptual project, some additional characteristics were proposed to reach the requirements. The typical mission of both aircraft was created using the AIAA specifications, historical trends and aeronautical regulations. Next, a study was

performed to choose the best battery system, which would make up the hybrid system of the aircraft, where it was evaluated the feasibility of different batteries components and their applicability in 2025. Thus, the Zn-O₂ battery was chosen due to its energy density (higher than for Li-Ion), maturity of behavior and safety, which is crucial for aerospace applications.

The selection of the ideal ICE was also evaluated comparing the utilization of the hybridization in turbo-engines of different rated powers. It was verified that the use of an ICE of lower rated power, associated to a hybrid-electric system, would be more efficient and economical when compared to a engine with higher rated power used by the competitors, justifying the hybrid architecture chosen for these aircraft.

Having the initial dimensions in hands and the assumptions fixed for both aircraft, a genetic algorithm was created using the methodology of Differential Evolution presented by Rainer Storn. Thus, many variables of optimization were selected to evaluate the best aircraft configuration that would meet all the project criteria, where the objective function was the maximum reduction of the aircraft weight. To do so, the software ALV was used as a tool to analyze the characteristics of different configurations proposed by the algorithm e generate optimum aircraft.

In the preliminary design phase, it was performed a structural analysis of the wing and empennages. The procedures proposed by aeronautical authorities were followed in order to create an envelope of loads, displayed in V-n diagrams. Using performance and stability data, it was possible to evaluate several critical conditions inside the flight envelope, allowing to calculate the loads acting on the structures.

The position of all structural elements was based on the Embraer Phenom 100 cutway, since optimizations in this direction would be unfeasible due to the great computation cost. The structural models were drawn and analyzed in a software of finite elements, where the wing and empennages were dimensioned in aluminum, following failure and buckling criteria. The use of carbon-epoxy was a very adequate alternative as a way to reduce the structural weight of the aircraft. Studies for its design were made using the same criteria mentioned above and this material was used in most of the structural components of the aircraft.

Finally, an aeroelastic analysis was performed on the wing evaluating the possible instability phenomena in the aircraft velocities envelope. A high torsional rigidity was observed in the wing, generating torsional vibration modes only at high frequencies. This

caused a great resistance of the wing to instabilities, where the phenomenon of flutter occurred only at very high speeds.

At the end of the development of this work, from the analysis carried out and the results obtained, the following suggestions are proposed for its continuity:

- Improve the genetic algorithm used to generate the optimized aircraft, generating at the same time both hybrid-electric aircraft and inserting better aerodynamic and stability analysis;
- Consider the elastic deformations when calculating the wing and empennages loads;
- Implement a new genetic algorithm to reduce weight, finding the better thicknesses, position and quantity of the following structural elements: ribs, spars and stiffeners;
- Evaluate new approaches with aeroelastic tailoring to improve the aeroelastic behavior of the structures, using techniques such as: tow steered composites, functionally grade metallic, and curvilinear wing reinforcements;
- Create an aeroelastic model that is capable of representing the coupling of the fuselage with the empennage, and later the entire aircraft.

BIBLIOGRAPHY

- [1] Air Transport Action Group (ATAG), The Right Flightpath to Reduce Aviation Emissions, Durban, South Africa, 2010.
- [2] “Annual Energy Outlook 2012,” DOE/EIA-0383(2012), U.S. Energy Information Administration, U.S. Dept. of Energy, Washington, D.C., 2012
- [3] R. Muller, ACARE Goals (AGAPE) Progress Evaluation, Project Final Report Publishable Summary, 2010.
- [4] Seeckt, K. Performance assessment of part-electric General Aviation aircraft. Hochschule für Angewandte Wissenschaften Hamburg, Hamburg, 2006.
- [5] Girishkumar, G., McCloskey, B., Luntz, A.C., Swanson, S., Wilke, W.: Lithium-air battery: promise and challenges. *J. Phys. Chem. Lett.* 1(14), 2193–2203 (2010)
- [6] Hepperle, M. Electric Flight – Potential and Limitations. German Aerospace Center. Institute of Aerodynamics and Flow Technology. Braunschweig, Germany. 2012.
- [7] Friedrich, C., and Robertson P.A. “Hybrid-Electric Propulsion for Aircraft”, *Journal of Aircraft*, Vol. 52, No. 1 (2015), pp. 176-189.
- [8] Mengistu, I. H. Small Internal Combustion Engine Testing for a Hybrid-Electric Remotely-Piloted Aircraft. Air Force Institute of Technology. Wright-Patterson Air Force Base, Ohio. 2011.
- [9] Pornet, Cl., Isikveren A.T., Conceptual design of hybrid-electric transport aircraft. Elsevier. Germany, 2015.
- [10] Chau, K.T., Wong, Y.S.: Overview of power management in hybrid electric vehicles. *Energy Convers. Manag.* 43, 1953–1968 (2002)
- [11] Cagatay Bayindir, K., Gozukucuk, M.A., Teke, A.: A comprehensive overview of hybrid electric vehicle: powertrain configurations, powertrain control techniques and electronic control units. *Energy Convers. Manag.* 52(2), 1305–1313 (2012)
- [12] Hung, J.Y., Gonzales, L.F.: On parallel hybrid-electric propulsion system for unmanned aerial vehicles. *Prog. Aerosp. Sci.* 52(1), 1–17 (2012)

- [13] Friedrich, C., Robertson, P.A.: Design of a hybrid-electric propulsion system for light aircraft. In: 14th AIAA Aviat. Technol., Integration, and Operations Conference, Atlanta, Georgia (2014)
- [14] Kamil Çağatay Bayindir, Mehmet Ali Gözükcük, Ahmet Teke. A comprehensive overview of hybrid electric vehicle: Powertrain configurations, powertrain control techniques and electronic control units. *Energy Conversion and Management*. 2011;52(2):1305-1313
- [15] Katrašnik T, Tranc F, Rodman Oprešnik S. Analysis of the energy conversion efficiency in parallel and series hybrid powertrains. *IEEE Transactions on Vehicular Technology*. 2007;56(6/2):3649-3659
- [16] M. John Miller, *Propulsion Systems for Hybrid Vehicles*. London: The Institution of Electrical Engineers, 2004.
- [17] A.T. Isikveren, S. Kaiser, C. Pornet, P.C. Vratny, Pre-design Strategies and Sizing Techniques for Dual-Energy Aircraft, *Aircr. Eng. Aerosp. Technol. J.* 86(6) (2014) 525–542.
- [18] Bogaert, J.V., *Assessment of Potential Fuel Saving Benefits of Hybrid-Electric Regional Aircraft*. Delft University of Technology. 2015
- [19] Processo de Desenvolvimento de Aeronaves [Powerpoint slides]. Personal Collection of Giuliano Gardolinski Venson for Aircraft Design, Aeronautical Engineering Program, Faculty of Mechanical Engineering, Federal University of Uberlândia, Uberlândia, Minas Gerais, 2013.
- [20] Raymer, D. P. *Aircraft design: a conceptual approach*. 4th ed. Reston: AIAA Education, 2006.
- [21] Aviation Week (2015). Boeing Begins Analysis of 787 Fatigue Tests. Retrieved from <http://aviationweek.com/commercial-aviation/boeing-begins-analysis-787-fatigue-tests>.
- [22] AIAA. 2017. About AIAA. [ONLINE] Available at: <http://www.aiaa.org/AboutAIAA/>. [Accessed 4 October 2017].
- [23] AIAA. 2017. Undergraduate Team Aircraft Design – Hybrid-Electric General Aviation Aircraft (HEGAA). [ONLINE] Available at: <http://www.aiaa.org/2018UndergradTeamAircraftDesign%E2%80%93HEGAA/>. [Accessed 4 October 2017].
- [24] Layout Geral de Aeronaves [Powerpoint slides]. Personal Collection of Giuliano Gardolinski Venson for Aircraft Design, Aeronautical Engineering Program, Faculty

- of Mechanical Engineering, Federal University of Uberlândia, Uberlândia, Minas Gerais, 2013.
- [25] Estimativa da Polar de Arrasto de Aeronaves [Powerpoint slides]. Personal Collection of Giuliano Gardolinski Venson for Aircraft Design, Aeronautical Engineering Program, Faculty of Mechanical Engineering, Federal University of Uberlândia, Uberlândia, Minas Gerais, 2015.
- [26] Estimativa de Pesos de Aeronaves [Powerpoint slides]. Personal Collection of Giuliano Gardolinski Venson for Aircraft Design, Aeronautical Engineering Program, Faculty of Mechanical Engineering, Federal University of Uberlândia, Uberlândia, Minas Gerais, 2015.
- [27] Dimensionamento Inicial de Aeronaves [Powerpoint slides]. Personal Collection of Giuliano Gardolinski Venson for Aircraft Design, Aeronautical Engineering Program, Faculty of Mechanical Engineering, Federal University of Uberlândia, Uberlândia, Minas Gerais, 2015.
- [28] Balanceamento de Aeronaves e Distribuição de Pesos [Powerpoint slides]. Personal Collection of Giuliano Gardolinski Venson for Aircraft Design, Aeronautical Engineering Program, Faculty of Mechanical Engineering, Federal University of Uberlândia, Uberlândia, Minas Gerais, 2014.
- [29] Raymer, D. P. Aircraft design: a conceptual approach. 4th ed. Reston: AIAA Education, 2006.
- [30] Siemens. 2016. Aerobatic Airplane “Extra 330LE”. [ONLINE] Available at: <https://www.siemens.com/press/pool/de/events/2016/corporate/2016-12-innovation/inno2016-aerobatic-airplane-e.pdf>. [Accessed 14 September 2017].
- [31] Scholz, D., 2015. Aircraft Design. 1st ed. Hamburg, Germany: Hamburg University of Applied Sciences.
- [32] Voskuijl, M., 2017. Analysis and design of hybrid electric regional turboprop aircraft. *Journal of Aircraft*, 52, 11.
- [33] Drela, M. and Youngren, H. 2017. AVL. [ONLINE] Available at: <http://web.mit.edu/drela/Public/web/avl/>. [Accessed 30 November 2017].
- [34] Eshelby, M., 2000. Aircraft Performance: Theory and practice. 1st ed. Burlington, MA: Elsevier.
- [35] Ma, H., Wang, B., Fan, Y., & Hong, W. (2014). Development and characterization of an electrically rechargeable zinc-air battery stack. *energies*, 7(10), 6549-6557.

- [36] PennState. 2017. Efficiency of Inverters. [ONLINE] Available at: <https://www.e-education.psu.edu/eme812/node/738>. [Accessed 9 November 2017].
- [37] Photovoltaic-software. 2017. Electricity losses : AC and DC electrical wire voltage drop and energy losses. [ONLINE] Available at: http://photovoltaic-software.com/DC_AC_drop_voltage_energy_losses_calculator.php. [Accessed 24 November 2017].
- [38] Home Power. 2017. Input Voltage & Controller Efficiency. [ONLINE] Available at: <https://www.homepower.com/input-voltage-controller-efficiency>. [Accessed 24 November 2017].
- [39] International Journal of Engine Research. 2017. Maximum efficiencies for internal combustion engines: Thermodynamic limitations. [ONLINE] Available at: <http://journals.sagepub.com/doi/abs/10.1177/1468087417737700>. [Accessed 24 November 2017].
- [40] Transformative Vertical Flight Workshop. 2015. Electric propulsion components with high power densities for aviation. [ONLINE] Available at: <https://nari.arc.nasa.gov/sites/default/files/attachments/Korbinian-TVFW-Aug2015.pdf>. [Accessed 29 November 2017].
- [41] EPI, Inc. 2017. Propeller Performance Factors. [ONLINE] Available at: http://www.epi-eng.com/propeller_technology/selecting_a_propeller.htm. [Accessed 6 September 2017].
- [42] Hepperle, M. Electric Flight–Potential and Limitations. German Aerospace Center. Institute of Aerodynamics and Flow Technology. Braunschweig, Germany. 2012.
- [43] Girishkumar, G., McCloskey, B., Luntz, A.C., Swanson, S., Wilke, W.: Lithium-air battery: promise and challenges. *J. Phys. Chem. Lett.* 1(14), 2193–2203 (2010).
- [44] Bruce, P. G., Freunberger, S. A., Hardwick, L. J., & Tarascon, J. M. (2012). Li-O₂ and Li-S batteries with high energy storage. *Nature materials*, 11(1), 19-29.
- [45] Ji, X., & Nazar, L. F. (2010). Advances in Li–S batteries. *Journal of Materials Chemistry*, 20(44), 9821-9826.
- [46] Toussaint, G., Stevens, P., Akrou, L., Rouget, R. & Fourgeot, F. Development of a rechargeable zinc-air battery. *ECS Trans.* 28, 25–34 (2010).
- [47] Linden, D. & Reddy, T. B. *Handbooks of Batteries* (McGraw-Hill, New York, 2002).
- [48] Zhang, X. G. (2007). U.S. Patent No. 7,291,186. Washington, DC: U.S. Patent and Trademark Office.

- [49] Hermann, W. A., Straubel, J. B., & Beck, D. G. (2017). U.S. Patent No. 9,559,532. Washington, DC: U.S. Patent and Trademark Office.
- [50] IVAO Aero. 2017. Top of Descent Calculation. [ONLINE] Available at: https://www.ivaao.aero/training/documentation/books/SPP_APC_Top_of_descent.pdf. [Accessed 29 September 2017].
- [51] IATA. 2017. Fuel Price Analysis. [ONLINE] Available at: <http://www.iata.org/publications/economics/fuel-monitor/Pages/price-analysis.aspx>. [Accessed 8 December 2017].
- [52] U.S. Energy Information Administration. 2017. Average Price of Electricity to Ultimate Customers by End-Use Sector. [ONLINE] Available at: https://www.eia.gov/electricity/monthly/epm_table_grapher.php?t=epmt_5_6_a. [Accessed 8 December 2017].
- [53] Storn, R., & Price, K. (1997). Differential evolution—a simple and efficient heuristic for global optimization over continuous spaces. *Journal of global optimization*, 11(4), 341-359.
- [54] U.S. Government Publishing Office. 2002. 14 CFR 23.335 - DESIGN AIRSPEEDS.. [ONLINE] Available at: <https://tinyurl.com/designairspeeds>. [Accessed 17 November 2017].
- [55] Eshelby, M., 2000. *Aircraft Performance: Theory and practice*. 1st ed. Burlington, MA: Elsevier.
- [56] MIT. 2017. Wing Bending Calculations. [ONLINE] Available at: <https://ocw.mit.edu/courses/aeronautics-and-astronautics/16-01-unified-engineering-i-ii-iii-iv-fall-2005-spring-2006/systems-labs-06/spl10.pdf>. [Accessed 2 November 2017].
- [57] Nayak, N. V., 2014. Composite Materials in Aerospace Applications. *International Journal of Scientific and Research Publications*, [Online]. 4, 10. Available at: <http://www.ijsrp.org/research-paper-0914/ijsrp-p3342.pdf> [Accessed 23 November 2017].
- [58] Krupa, E., Cooper, J., Pirrera, A., & Silva, G. H. C. (2016). Aeroelastic Tailoring for Enhanced Aerodynamic Wing Performance. In *5th Aircraft Structural Design Conference* Royal Aeronautical Society.
- [59] Tsai, S. W., & Wu, E. M. (1971). A general theory of strength for anisotropic materials. *Journal of composite materials*, 5(1), 58-80.

- [60] Garrick, L. E. (1976). Aeroelasticity-frontiers and beyond. *Journal of Aircraft*, 13(9), 641-657.
- [61] Livne, E. (2003). Future of airplane aeroelasticity. *Journal of Aircraft*, 40(6), 1066-1092.
- [62] Wright, J. R., & Cooper, J. E. (2008). *Introduction to aircraft aeroelasticity and loads* (Vol. 20). John Wiley & Sons.