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A MULTI-DISCIPLINARY DESIGN OPTIMIZATION FOR CONCEPTUAL DESIGN OF GENERAL AVIATION AND HYBRID-ELECTRIC AIRCRAFT

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Abstract. *The design of an aircraft has become increasingly complex, since it depends on technological advances and integration between modern engineering systems. These systems are multidisciplinary, i.e., any process or division of design of any aircraft design produces effects in all others, making the definition of each parameter a great challenge. In this context, this work presents a multidisciplinary design optimization (MDO) for general aviation and hybrid-electric aircraft design integrating boxes of aerodynamics, structures, flight mechanics and performance. The methodologies used in each box are presented. The aircraft designed is in the general aviation market and the requirements are specified according to its competitors, and the objective-functions are minimize fuel consumption and maximum takeoff weight.*

Keywords: *aircraft design, multidisciplinary design optimization, general aviation aircraft, hybrid-electric aircraft*

1. INTRODUCTION

Current practice in industry is to move the design of complex equipment away from a process involving a sequence of specialist departments and to emphasize its multidisciplinary nature through the use of integrated product teams. Although there are tools and software that are highly effective individually, the challenge now is to provide the right tools to support this integrated approach (Sobieszczanski-Sobieski and Haftka, 1997).

Multidisciplinary design optimization (MDO) allows the incorporation of all relevant disciplines simultaneously using optimization method to solve design problems (Venter and Sobieszczanski-Sobieski, 2004), increasing the efficiency of designs to be optimized, resulting in a reduction of time cycles and costs of the project (Raymer, 2002).

The use of multiple simulations is a key concept of MDO. This involves several tools, such as fluid flow solvers, structural analysis, cost modelling and tools for design and reliability (Yu and Du, 2006). At a general level, when considering the overall mission performance of an aircraft, the tool is used during the early stages of the project. However, currently most MDO applications are based on major simplifications in mathematical modelling (Kroo *et al.*, 1994), such as beam structural models or panel methods for aerodynamics.

Along those lines, this work presents a MDO method for conceptual design of general aviation and hybrid-electric aircraft. The MDO comprises boxes of engineering that includes aerodynamics, flight mechanics, structures, and performance, integrating all of them. After validating the accuracy of the aerodynamic and flight mechanics solvers, a case study is presented. A serial powertrain is used as propulsive architecture, along with distributed propulsive system. The design requirements are specified based on the aviation market for an aircraft of four passengers. It is assumed short ranged flights to enable the use of batteries and consequent electrical configuration (Hepperle, 2012). Finally, a discussion of the results is presented, describing the main geometric characteristics and weight breakdown of the aircraft obtained.

2. MULTIDISCIPLINARY OPTIMIZATION

The MDO proposed integrates the boxes of performance, aerodynamics, flight mechanics, and structures. Having the market and operational requirements, the MDO uses a genetic algorithm technique to optimize the aircraft in order to find the "best" aircraft that fits the constraints and the objective function.

For each aircraft under analysis, the Fig. 1 presents the convergence process of an iteration. With the definition of requirements and design variables, the analysis begins with the constraint diagram. The determination of the wing loading (W/S) allows the calculation of the wing geometries. Then, the aerodynamic coefficients are evaluated through the aerodynamic solver.

The static stability box takes care of the sizing of the horizontal tail, vertical tail, and elevator using an iterative method integrated with the aerodynamic solver. Then, wing and empty weights are obtained through the structural box. Next, having the complete geometry defined, the aircraft's drag polar is computed, considering a trim condition, in order to feed the mission module. The propulsive system is determined based on the power required and energy spent in each flight phase. Finally, the value of the maximum takeoff weight (MTOW) is updated, including all the weight breakdown (payload, powertrain, wing, battery, fuel, and airframe), and it goes back to the system so that a new wing geometry is defined.

At each iteration, this procedure repeats until there is a convergence in the MTOW value, aircraft geometry and configurations. The methodology applied in each box is presented in the following sections.

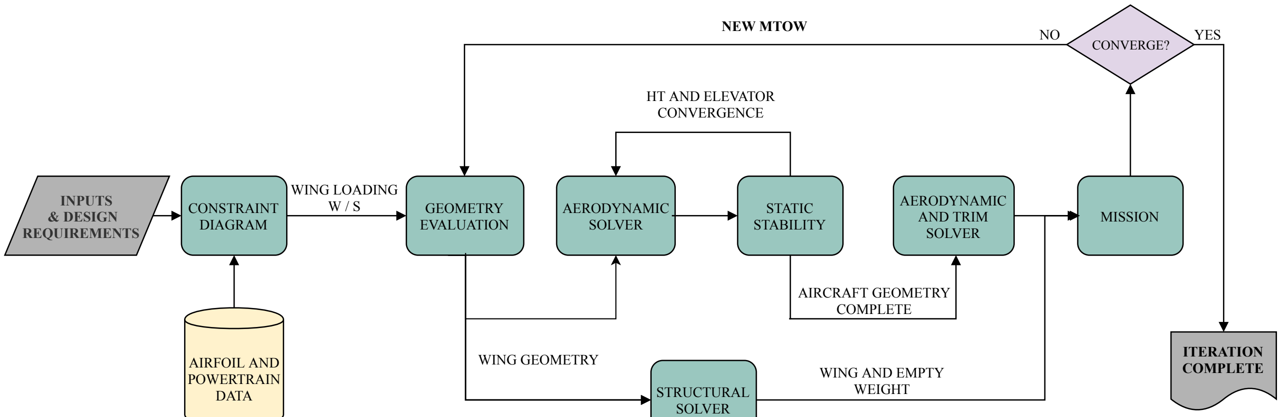


Figure 1: Illustration of the integration of the MDO boxes.

2.1 Constraint Diagram

The constraint diagram box evaluates the different performance constraints of the aircraft and generates a resulting feasible design space in terms of wing loading and thrust-to-weight ratio (or power loading). Then, from this feasible design space, it is possible to define the wing area and the installed power, which are, initially, the main step when designing an aircraft.

For this MDO, the mission requirements define the actual constraints. In addition, it is taken into account a distributed propulsive system, which comprises a set of electric propulsors usually installed along the wing, generating air-propulsive interactions that result in gains in propulsion and total-lift of the aircraft. Thus, after considering the "Deltas" of C_L and C_D, the optimum design point is chosen to generate the smallest wing; in other words, it means the point that delivers the highest wing loading (W/S). Finally, for a specific aircraft weight, this value of W/S feeds the geometry evaluation, which calculates the rest of the geometric parameters of the aircraft and so on.

2.2 Aerodynamics

The aerodynamic package will be composed by potential flow-based solvers, such as Lifting Line Theory (LLT), Vortex Lattice Method (VLM) and Non-Linear Vortex Lattice Method (NL-VLM). These solvers are integrated with object-oriented variables in order to share its results - aerodynamic loads, coefficients, and derivatives - with the remaining areas of project in a standard way.

The potential-flow methods are based on the solution of Laplace's equation through the distribution of vortex singularities in a simplified version of the geometry and the imposition of boundary conditions. Specifically for the VLM, the boundary condition imposed is of tangential flow at the 3/4 of the local chords. Once the singularities are placed and the boundary conditions are applied, a linear system is solved in order to compute the resulting aerodynamic loads. This methodology can only be applied for non rotational inviscid flows, though it is possible to approximate viscosity effects through the NL-VLM. Despite these restrictions, it presents low computational costs and enough accuracy to be used in optimization routines and preliminary projects.

The Non-Linear Vortex Lattice Method, firstly implemented by Carvalho (2018), is capable of simulating generic sets of lifting surfaces over subsonic symmetrical or asymmetrical flow and maneuver speeds, considering interference between surfaces. It is also able to predict the non-linear region of the lift slope, as well as accounting for viscous effects through the dechambering strategy proposed by Mukherjee and Gopalarathnam (2003) and adjusted by Vargas (2006). This strategy is based on the correction of the inviscid three-dimensional solution by means of previously known bi-dimensional viscous section data, such as numeric simulation or experimental results.

2.3 Structures

The structures box will adopt an analytical sizing approach that allows the estimation of the wing structural weight and the aircraft empty weight. The method for the wing estimation is able to generate more reliable results when compared with statistical methods traditionally used in this step, especially if it is related to innovative designs, because it does not depend on historical databases or tabulated coefficients.

Since the structure of a wing is highly complex for the application of analytical equations, simplifications had to be made to facilitate the sizing process. Thus, the structural idealization method proposed by Megson (2016) was used in order to idealize a real wing structure by transforming it into a simpler mechanical model with equivalent mechanical properties.

With the geometry, aerodynamic loads and the idealized structure, the normal forces and shear flow acting on the wing structural elements (spars, stiffeners, skin panels and ribs) could be calculated. To verify if the wing structure is capable to withstand these stresses, the von Mises criterion for limit load was applied. This method can calculate a σ_{VM} stress equivalent to all stresses tensors applied to the elements. Subsequently, these equivalent stresses were compared with the respective elements' limit stresses to check if they were not exhibited any plasticity or rupture.

There were also evaluated stability criteria that could generate a structural failure with loads lower than the limit stresses, caused by compression and shear forces in the form of buckling or crippling. The methods used in this work to calculate instabilities in wing structures are those proposed by Gerard and Becker (1957).

In order to generate an efficient structure, a sizing optimization routine was implemented with the objective of finding the optimum thickness for each structural element that minimizes the wing final weight. Thereupon, wing structures with random elements' thicknesses were generated and their structural stresses were compared to the sizing criteria. If all criteria were respected, this configuration is considered as valid and its weight is calculated by the objective function. Consequently, the output of the structural sizing box will be the lowest wing structure weight admissible for the geometry and aerodynamic load inputs provided by the multidisciplinary optimization. In addition, an estimate of the aircraft empty weight is performed here using the methodology defined by Nicolai and Carichner (2010).

Figure 2 presents the calculation routine of the structure box.

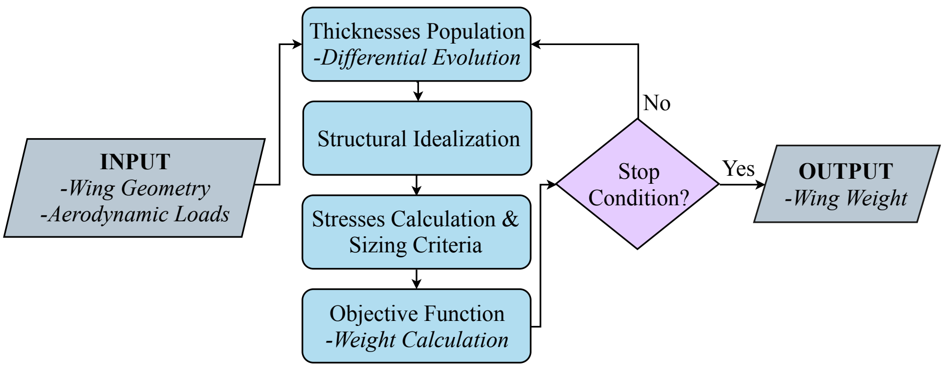


Figure 2: Structure box flowchart.

2.4 Flight Mechanics

The Flight Mechanics box will be responsible for the tail and control surfaces design. It is based on an analytical procedure with minimal dependence on historical data. The main constrains will come from the type of mission that the aircraft is meant to accomplish and the desired stability characteristics. Since this is an early design procedure, few geometrical data are available and, therefore, simplifications had to be applied.

The implemented methodology to size both the aircraft tail and its elevator is adapted from the one proposed by Resende (2019). Based on the fuselage length and the wing position, it is possible to estimate the horizontal tail position. Then, the horizontal and vertical tails areas are evaluated, followed by their other geometric parameters. These evaluations are based on the wing aerodynamic data, provided by the Aerodynamic package, and on desirable characteristics for the aircraft's behavior in flight. Figure 3 presents an overlay of the tail sizing procedure. From this step, it is possible to design the airplane's control surfaces.

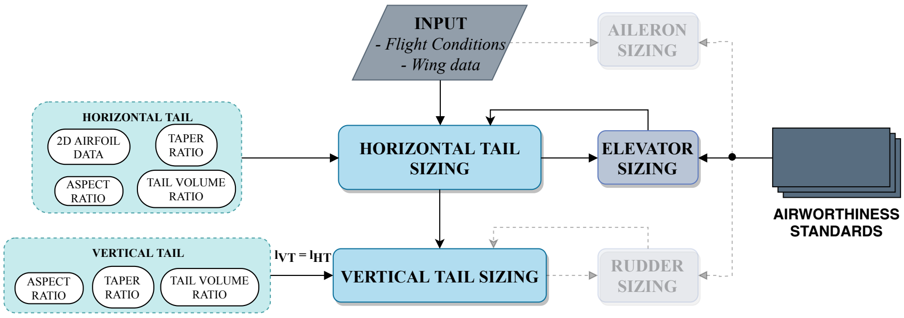


Figure 3: Tail and elevator flow diagram.

The elevator sizing is based on guarantee that the aircraft will be able to be in a trim condition at critical flight conditions (e.g. at the stall velocity, for a longitudinal trim). At this point of the design, however, one does not have the control surface dimensions, which means that their effectiveness parameter (τ) are also unknowns. But, as the control deflection at critical conditions is expected to be at its extreme value, one still remains with the same quantity of unknowns. Therefore, by modifying the trim equations, it is possible to estimate the effectiveness parameter of the elevator and, then, calculate its area (Nelson *et al.*, 1998). Since the ailerons and rudder would not play a significant contribution to the aircraft weight, their design are not considered during this MDO process in order to improve computational performance.

2.5 Mission

The mission box is responsible to evaluate a typical mission of the aircraft. From operational requirements, the box evaluates the powertrain necessary to meet constraints of takeoff, climb, cruise, climb, descent, loiter, and approach. In this process, it is computed the required power for each flight phase, and then it is estimated the corresponding weight of electric motors and engines. For a certain combination of degree-of-hybridization throughout the typical mission, the energy supplied by the batteries is estimated, and, consequently, its weight. More details of the mission estimation is described in the work of Silva (2019).

Finally, the box estimates the weight breakdown of the aircraft, i.e., the empty weight, payload, powertrain weight, and the fuel consumption during the mission. The MDO may be set to find the best configuration of the aircraft that results in lower fuel consumption, for example. Other objective function in terms of performance could be to carry more payload, and so on. Figure 4 presents the calculation routine of the structure box.

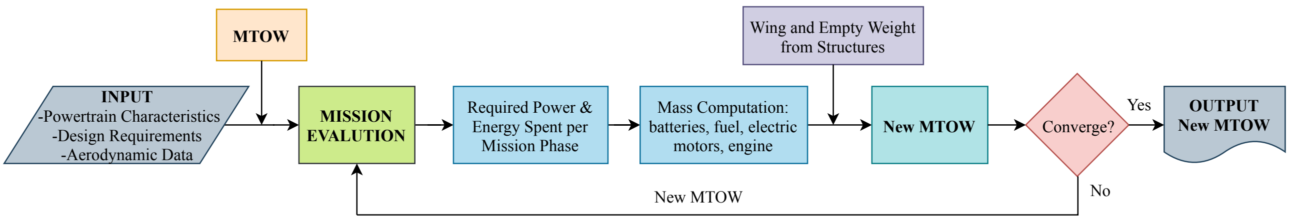


Figure 4: Overlay procedure structure.

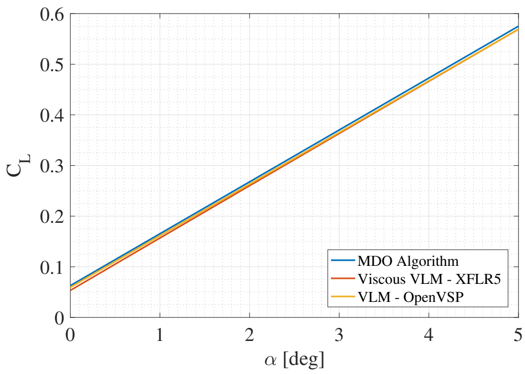
3. VALIDATION

In order to validate the accuracy of the aerodynamic and flight mechanics solvers, their results were confronted with two others software: XFLR5 and OpenVSP. A base-line model, composed by a wing with $Æ_w = 15.0$, $\lambda_w = 0.40$ and area of 8.64 m^2 and by a horizontal tail with $Æ_{HT} = 5.8$, $\lambda_{HT} = 0.38$ and area of 1.39 m^2. The wing and horizontal tail airfoils were the NACA23012 and the AH21 (with inverted camber), respectively. The CG position was fixed 0.214 m from the wing leading edge, considered at its root chord, and the horizontal tail is placed 4.72 m from the same reference.

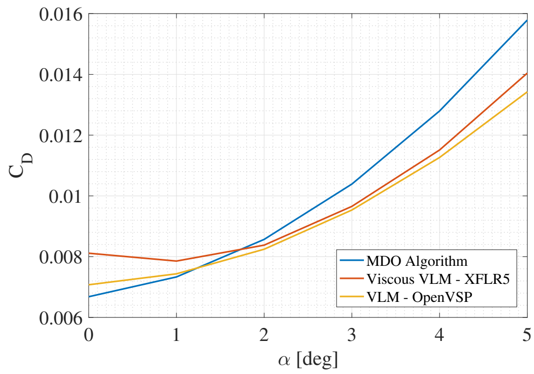
For this test case, the freestream velocity was set to 82.3 m/s and the Reynolds number to 3.8×10^6, simulating a cruise flight condition for general aviation aircraft. The aerodynamic solver evaluated this simulation as described in section 2. 2 Both the XFLR5 and OpenVSP performed a VLM calculation with similar mesh. The XFLR5, however, considered a correction model to account for viscous effects.

Figure 5 shows the aerodynamic coefficients obtained by each solver, and Tab. 1 presents their numeric values at $\alpha = 0^o$ and $\alpha = 5^o$. As expected, the lift coefficient had a good agreement between the methods. The drag coefficient, however, is more sensitive to the adopted methodology on each software. Despite of that, the differences encountered were considered to be acceptable.

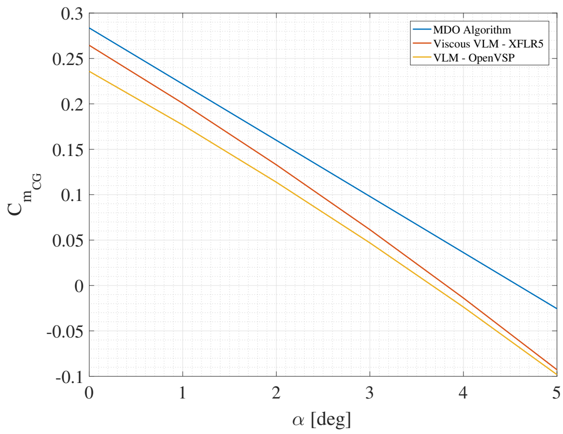
From Fig. 5c, it is clear that all solvers evaluated similar slopes for the pitching moment coefficient, which is a reasonable indicator that the flight mechanics package will be feed with coherent data for the tail and elevator sizing. The variation between the curves may be explained by how each software evaluates the airfoils pitching moment around their aerodynamic center, which would cause a shift in the C_{m_0} (pitching moment coefficient at $\alpha = 0^o$) of the assembly wing-horizontal tail. Furthermore, a static stability calculation was performed on each software to evaluate the assembly neutral point (X_{NP}). As shown in Tab. 2 the flight mechanics solver had good agreement with the two different software computations of the pitching moment slope (C_{m_α}) and the X_{NP}.



(a) C_L x α curve.



(b) C_D x α curve.



(c) $C_{m_{CG}}$ x α curve.

Figure 5: Aerodynamic solver validation.

Table 1: Numeric values for aerodynamic solver validation.

	MDO	XFLR5	OpenVSP	MDO	XFLR5	OpenVSP
	$\alpha = 0^o$			$\alpha = 5^o$		
C_L	0.0628	0.0533	0.0595	0.5752	0.5692	0.5692
C_D	0.0067	0.0081	0.0071	0.0158	0.0140	0.0134
$C_{m_{CG}}$	0.2836	0.2645	0.2358	-0.0256	-0.0928	-0.0982

Table 2: Numeric values for aerodynamic solver validation.

	MDO	XFLR5	OpenVSP
X_{NP}	3.676	3.660	3.658
C_{m_α} [rad^{-1}]	-3.543	-4.094	-3.385

4. CASE STUDY

For this case study, it is selected a general aviation aircraft to cover a mission of 300 km with the design requirements and specifications listed in Tab. 3, comparing an all-electric and a hybrid-electric aircraft, named E-4P and H-4P, respectively. The aircraft comprises a distributed propulsive system along the wing and a serial powertrain architecture, as illustrated in Fig. 6, and it ends the mission with a battery final state of charge of 25% and a fuel tank level of 10%, at least.

Table 3: Design requirements and specifications.

Parameter	Value	Unit
Number of passengers	3	-
Number of pilots	1	-
Number of engines	1	-
Number of electric propulsors	8	-
Stall speed	60	
Cruise speed	160	KTAS
Climb/descent speed	105	EAS
Climb rate	500	fpm
Descent rate	-350	fpm
Takeoff field length	325	m
Cruise altitude	8000	ft
Loiter altitude	1000	ft
Loiter time	45	min
Battery specific energy	500	Wh/kg
Fuel specific energy	11900	Wh/kg

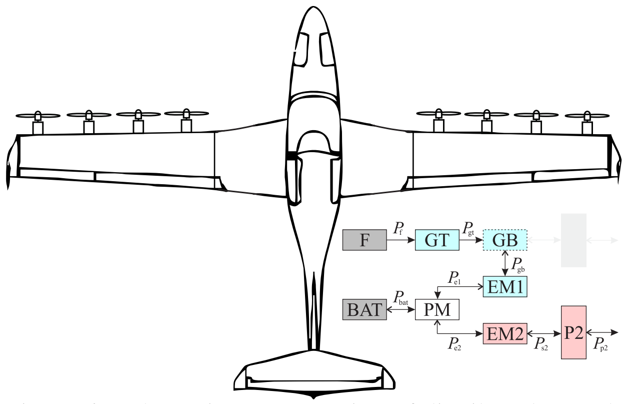


Figure 6: Schematic representation of distributed propulsion along the wing and serial powertrain architecture. Legend: "F" = fuel, "GT"= gas turbine, "GB"= gearbox, "P"= propulsor, "BAT"= batteries, "EM"= electrical machine (i.e. electric motor or generator), "PM"= power management.

For the optimization problem, two objective functions are chosen to be minimized: W_{fuel} and $MTOW$. The selected design variables (x) are based on values of geometry and performance, i.e., wing profile, horizontal tail profile, wing aspect ratio (Æ_w), wing taper ratio (λ_w), horizontal tail aspect ratio (Æ_{HT}), horizontal tail root chord ($c_{\text{root},HT}$), and the degrees-of-hybridization per flight phase ($\psi_{\text{phase},i}$). Thus, the optimization problem proposed here is characterized as follows:

$$\begin{array}{c}\textit{Multi-objective}\\ \textit{optimization problem}\end{array} : \left\{ \begin{array}{l} \min\left(W_{fuel}\right) \quad \text{and} \quad \min\left(W_{TO}\right) \\ x : [\text{Airfoil}_w, \text{Airfoil}_{HT}, \text{Æ}_w, \lambda_w, \text{Æ}_{HT}, c_{\text{root},HT}, \psi_{TO}, \psi_{CL}, \psi_{CR}, \psi_{DS}, \psi_{LT}] \\ \text{Airfoil}_w = \text{NACA63}_1\text{412, SELIG1223, NACA2412, NACA23012} \\ \text{Airfoil}_{HT} = \text{NACA0012, AH21, NACA63-012} \\ 7 \leq \text{Æ}_w \leq 15 \\ 0.40 \leq \lambda_w \leq 1.00 \\ 3 \leq \text{Æ}_{HT} \leq 6 \\ 0.50 \leq c_{\text{root},HT} \leq 1.00 \\ 0 \leq \psi_{\text{phase},\, i} \leq 1 \end{array} \right. \tag{1}$$

Moreover, to find the Pareto solution, it is used the NSGA II, a multi-objective algorithm based on genetic algorithms and proposed by Deb *et al.* (2000). The algorithm was executed several times with a number of populations of 100, generation equal to 500, crossover index of 20, and mutations index of 20. These parameters were also changed, but all the results converged to the same.

5. RESULTS AND DISCUSSION

The Pareto-optimal front depicted in Fig. 7 show the solution for the optimization of the case study defined in Section 4. The result shows that to minimize the fuel consumption for this type of mission, it is necessary to increase the amount of batteries onboard. However, as the weight of the batteries increases, the weight of the aircraft increases as a whole, i.e., the maximum takeoff weight increases as well. Therefore, to reduce the fuel weight, the total weight of the aircraft is considerably increased. Regarding the design variables, the NSGA II algorithm found different combinations that delivered that result. Among those combinations, all the wing airfoil profiles were used, but only the AH21 and NACA63-012 were selected for the horizontal tail profile. The aspect ratio remained around 14.3 to 15 for the wing and 4.2 and 5.9

for the horizontal tail. Moreover, the wing taper ratio presented values from 0.40 to 0.70 and the horizontal tail root chord varied from 0.50 to 1.00.

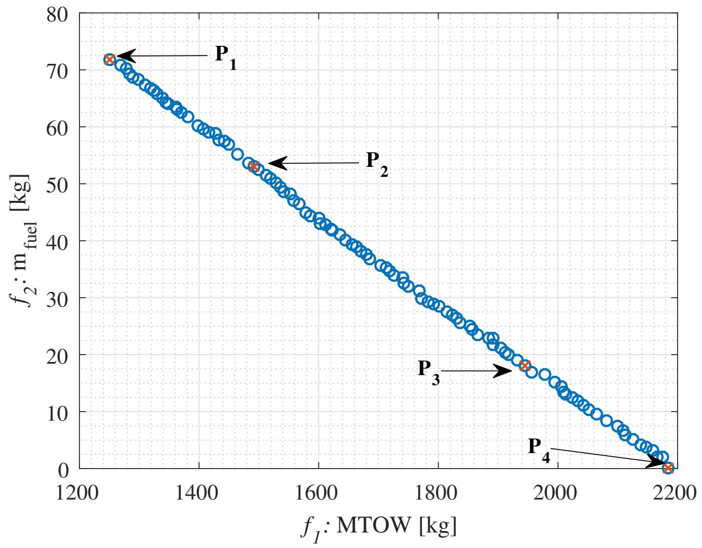


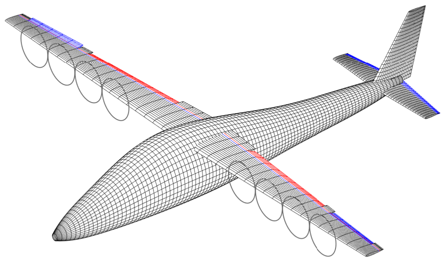
Figure 7: Pareto-front that represents the solution for the optimization problem.

In Figure 7, four points of the solution were selected: P_1, P_2, P_3, and P_4. These points represent four different options for the aircraft design. P_1 is the condition where the aircraft has the highest fuel consumption, and it happens when all degrees-of-hybridization are equal to zero ($\psi_{\text{phase}_i} = 0$), i.e., in this case, the configuration represents a turboelectric aircraft. At the other end, P_4 is the condition where the aircraft consumes no fuel, i.e., it implies a full electric configuration; therefore, the aircraft energy is only supplied by batteries, increasing the takeoff weight to the maximum. The other points between P_1 and P_3 have different combinations of degrees-of-hybridization and geometry.

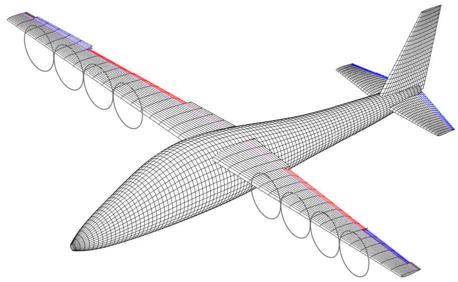
Table 4: Results of the optimization.

Parameter	P_1	P_2	P_3	P_4
Airfoil$_w$	NACA23012	NACA63$_1$412	NACA63$_1$412	NACA63$_1$412
Airfoil$_{HT}$	AH21	NACA63-012	NACA63-012	NACA63-012
$\AE_w$	15	15	15	15
λ_w	0.40	0.40	0.40	0.40
$\AE_{HT}$	5.6	4.5	4.6	4.2
c$_{\text{root},HT}$	0.57	0.98	0.99	0.79
ψ_{TO}	0.00	0.59	0.84	1.00
ψ_{CL}	0.00	0.21	0.79	1.00
ψ_{CR}	0.00	0.30	0.82	1.00
ψ_{DS}	0.00	0.00	0.35	1.00
ψ_{LT}	0.00	0.60	0.94	1.00
m_{empty} [kg]	711.0	771.6	888.7	949.7
m_{bat} [kg]	0.0	205.0	587.1	777.4
m_{fuel} [kg]	73.6	53.0	18.0	0.0
m_{PT} [kg]	81.8	77.7	66.6	64.5
m_{PL} [kg]	384.0	384.0	384.0	384.0
m_{TO} [kg]	1250.5	1491.3	1944.4	2175.5

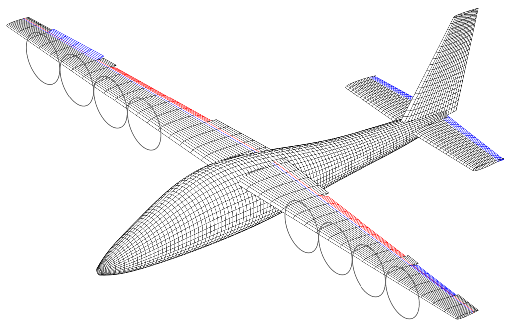
Furthermore, the details of the aircraft design optimization for the four points selected are displayed in Tab. 4. As one can see, the different combinations of the design variables resulted in different weight breakdowns for the aircraft. P_1, for example, requires 73.6 kg of fuel and none battery, since it is turboelectric. But moving forward through the other points, the correlation between fuel and battery mass is not linear, i.e., the amount one increases is not the same amount the other decreases. This happens because they are directly dependent of the degrees-of-hybridization selected by the algorithm of optimization, along the with the respective geometric parameters. The geometry differences, i.e, aspect ratio, wingspan, and tails sizes for the aircraft of points P_1, P_2, P_3, and P_4 are depicted in Fig. 8, and the weight breakdown in Fig. 9.



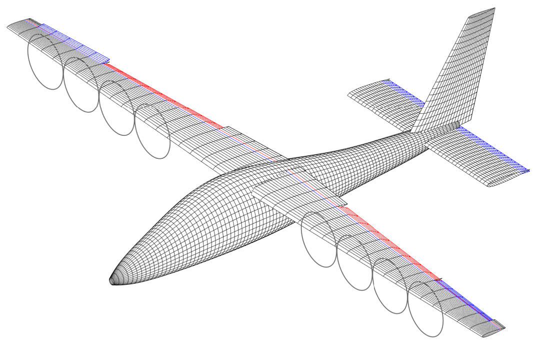
(a) Aircraft P_1 - Turboelectric.



(b) Aircraft P_2 - Hybrid-electric.

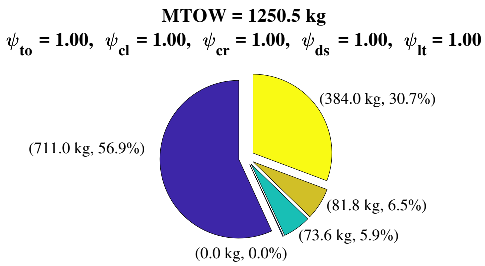


(c) Aircraft P_3 - Hybrid-electric.

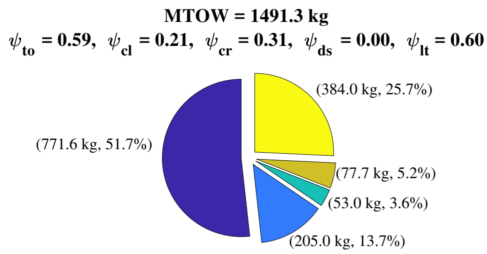


(d) Aircraft P_4 - Full-electric.

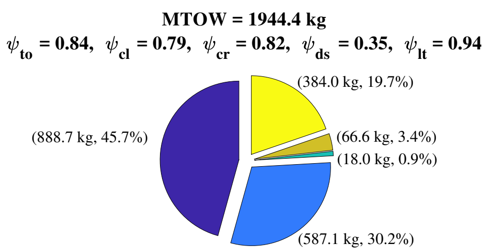
Figure 8: Illustration of the aircraft obtained from the select points of the Pareto-front.



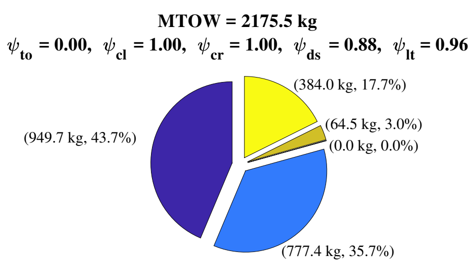
(a) Aircraft P_1 - Turboelectric.



(b) Aircraft P_2 - Hybrid-electric.



(c) Aircraft P_3 - Hybrid-electric.



(d) Aircraft P_4 - Full-electric.

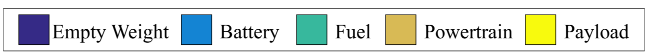


Figure 9: Weight breakdown of the aircraft obtained from the select points of the Pareto-front.

Notwithstanding, it is worth mentioning how the aircraft geometry is affected by the objective-function. Since the objective-functions were intended to reduce fuel consumption and takeoff weight, the algorithm of optimization searched for more efficient aircraft with better aerodynamic behavior; in other words, high values for wing aspect ratio were used. It happens due to the aerodynamic improvements, mainly from the aero-propulsive interactions due to the distributed propulsive system. This system provides a "Delta" in the lift coefficient, which allows a higher wing loading, resulting in smaller wing areas for a certain takeoff weight. Moreover, smaller wings result in lighter wings, and, consequently, lighter aircraft weight. In general, these correlations affect directly the aircraft drag polar, i.e., the parasite and induced drag in each flight phase, which directly affects the required powers and the energies spent throughout the flight. Thus, having those aerodynamic improvements, it is much easier to have an aircraft that fulfills the design requirements, but burning less fossil fuel.

Overall, any combination obtained from Fig. 7 results in a different aircraft, with different powertrain, fuel consumption, battery weight, takeoff weight and so on. Therefore, the choice of the point within the Pareto-front depends on what the designer wants to benefit the most.

6. CONCLUDING REMARKS

This work introduces a multi-disciplinary design optimization for conceptual design of general aviation and hybrid-electric aircraft. The study case presented satisfactory results, mainly in relation to the physical agreement of the models. Such an overarching and multidisciplinary optimization process may lead to mathematically coherent results, but physically meaningless. However, the MDO proposed here has proven to be robust enough against this problem.

The aerodynamic and flight mechanics solvers were validated using the XFLR5 and OpenVSP. Next, a study case was presented for a general aviation aircraft and a multi-objective problem was defined: minimize fuel consumption and MTOW. The Pareto optimal-front in Fig. 7 comprises aircraft with different propulsive architectures: turboelectric, hybrid-electric, and full-electric. Then, the NSGA II algorithm was able to optimize the aerodynamic characteristics in order to find the best aircraft configuration capable to meet the proposed requirements.

Since the degrees-of-hybridization per flight phase directly affect the final weight breakdown of the aircraft, one may realize that full-electric aircraft have their weight considerably increased, mostly because of the high battery demand to accomplish the flight mission. This overweight results in bigger lifting surfaces and, consequently, implies in more structural weight.

The MDO was developed as generic as possible in order to design aircraft to fly short-range missions. In summary, the mission profile is extremely important in the optimization problem. After all, the number of passenger, type of aircraft, total range, and loiter time are parameters that affect the overall required power and, consequently, the energy spent during the mission. In other words, in cases of intercontinental flights, where there is a great number of passengers and a long rang to be traveled, the energy required for the entire mission would be huge, which would imply in tons of batteries. Therefore, the current battery technology is not suitable for these type of mission, establishing the need for disruptive advances in technologies for the next years.

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