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A MDO PROCESS FOR PRELIMINARY DESIGN OF A REMOTELY PILOTED AIRCRAFT

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Abstract: *In this work, a multidisciplinary optimization technique focused on preliminary design of small remotely piloted aircraft is discussed. The multidisciplinary design optimization (MDO) problem currently in study is composed of three sub problems. In the aircraft design, it is considered a maximum total mass, defined as design requirement, and thus it is searched the minimum power-to-weight ratio that is necessary to comply with a series of performance constraints. Next, the wing spar parameters are optimized, aiming wing weight minimization. The maximum allowable stress at the wing root is the constraint used in the proposed procedure. At the end, tail parameters are optimized, aiming also minimization of mass. The framework is organized in a way that other constraints and objective functions can be easily added to the formulation. It is promising for future developments in automated preliminary design process.*
Keywords: *RPA, design point, wing stress, optimization, MDO*

1. INTRODUCTION

Aircraft design can be divided into three phases before the manufacturing starts: conceptual, preliminary and detailed design. According to Pahl and Beitz (1996), the conceptual design aims to identify the essential problems and define the global function and sub-functions of the product. The process of aircraft conceptual design involves a large number of parameters and equations. Traditionally, the values of various parameters in the areas of stability, aerodynamics, propulsion and structure are estimated based on known models that meet the design requirements. Thus, it is necessary to sketch a concept, analyze it, evaluate it, and compare its performance to others, limiting the accuracy, validity, and flexibility of this product design phase. Methodologies for automatic definition of the optimum design point are desired, aiming reduction of later redesign phases. To reduce the rate of uncertainty at this stage of the project, it is necessary to replace the known models by others detailed multidisciplinary models based on mathematical techniques that indicate the optimal solution. The preliminary design sets the configuration arrangement from the conceptual sketch according to Raymer (1992). During this phase, the design and analysis of each discipline is performed. The Multidisciplinary Design Optimization (MDO) is a method that helps finding a satisfactory design point for several disciplines simultaneously, in complex designs still in the conceptual phase or not, since the design changes have a lower cost than in the subsequent phases.

One of the most important considerations when running a MDO process is to adequately organize the analysis models, surrogate models (if they exist) and the optimization software together with the problem formulation in a way that an optimum design point might be achieved. Such combination of the problem formulation with the organizational strategy is called MDO architecture.

Sobieszczanski-Sobieski and Haftka (1996) presented a revision about multidisciplinary optimization in aerospace design, focusing in the multiple ways that engineers deal with the main challenges of a MDO process: computational cost and organizational complexity. Schweiger *et al.* (2002) described a work in progress at the time where different approaches for MDO application were studied to improve the performance of experimental aircraft. Agte *et al.* (2010) affirmed that the origins of MDO are found in the development of structural optimization. The inclusion of other disciplines into the optimization process was a natural evolution, since the aerodynamics, propulsion and performance variables, for example, are intrinsically related to the structural efficiency. Initial processes were developed in sequential levels, where variables and objective functions were uncoupled, and later it was possible to study in a single optimization block with as much as possible coupling between disciplines. Roglev (2013) applied MDO methods to the design of a box-wing aircraft. Due to the unconventional configuration, this concept of aircraft presented a strong coupling between structure and aerodynamic effects, and thus the MDO was considered as an ideal tool to obtain the best design.

Martins and Lambe (2013) explains in detail MDO architectures that have been presented in literature. Also a classification is provided citing the benefits and drawbacks of each category. The authors suggest that it is necessary to test multiple architectures on a given MDO problem to determine which one is most efficient for each case.

In the present work, an initial approach of MDO architecture is assembled for application in small remotely piloted aircraft (RPA) design. The multidisciplinary system is divided into three sub problems. In order to obtain some initial sizing values, Altman (2000) proposed a constraint diagram, showing the relationship between power-to-weight ratio (P/W) and wing loading (W/S) for the required mission. Such diagram is developed from performance equations describing different flight conditions. This methodology was later improved by the work of Landolfo (2008).

In aircraft design, it is considered a fixed maximum total mass, defined as a design requirement, and thus it is searched the minimum power-to-weight ratio that is necessary to comply with a series of performance constraints. The design variables are the wing chord and span. The constraint is established based on lift required by the airplane in cruise flight. Using the Prandtl lifting-line theory, it is possible to predict the lift forces acting on the wing for critical flight envelope points, and then compute the stresses at the wing root. Taking into consideration the yield strength of the material used and the safety factor required for the project, the maximum allowable stress is calculated. The second objective is to optimize the wing spar volume in order to minimize the wing spar mass. The design variables are thickness, height and width of the rectangular box section of spar. The tail mass can be written in relation to the distance between aircraft center of gravity (CG) and the aerodynamic center of the horizontal / vertical tail surface. The third aim is to minimize tail mass under a constraint of static longitudinal stability. The design variable is distance between CG and the aerodynamic center of the horizontal / vertical tail surface. A study case is presented and discussed. It considers a fixed aircraft concept of conventional type: front high wings and horizontal and vertical tail connected by a tail boom to the fuselage.

2. AIRCRAFT DESIGN PROCESS

The equations considered in the proposed design process are described in this section, for later development of the optimization procedure. First, it is necessary to establish the equations that are used to define the design space as described by Landolfo (2008). Next, the wing loading computation and resulting spar root stresses are defined. Also, simplified tail sizing equations are defined. One design requirement is the maximum total mass, which includes the aircraft mission.

2.1 Aircraft Performance

In order to define a range of feasible values for the initial design of the small RPA, a numerical study was carried out to analyze the first point of the design: the power-to-weight ratio (P/W) in different flight conditions. According to Altman (2000) and Landolfo (2008), the main conditions of flight that must be considered are Maximum Load / Turn, Endurance, Cruise and Takeoff Distance, described, respectively, by the following equations, which are written already as function of the wing chord c and span b:

$$\frac{P}{W} = \frac{1}{\eta_p}\left[\frac{1}{2}\rho\, v^3\, C_{D0}\frac{b\,c}{W} + 2\frac{n^2\,W}{\pi\, e\, b^2\,\rho\, v}\right] \tag{1}$$

$$\frac{P}{W} = \frac{4}{\eta_p}C_{D0}^{1/4}\left(\frac{c}{2\,\pi\, e\, b}\right)^{3/4}\left(\frac{2W}{\rho\, b\, c}\right)^{1/2} \tag{2}$$

$$\frac{P}{W} = \frac{2}{\eta_p}C_{D0}^{1/4}\left(\frac{c}{\pi\, e\, b}\right)^{3/4}\left(\frac{2\,W}{\rho\, b\, c}\right)^{1/2} \tag{3}$$

$$\frac{P}{W} = \frac{2.44}{\eta_p\, g\, d_{to}}\left(\frac{W}{\rho\, C_{L_{max}}\, b\, c}\right)^{3/2} \tag{4}$$

In Eqs. (1) to (4), the term ρ is the air density, g is the acceleration due to gravity, m is the aircraft mass, η_p is the propeller efficiency coefficient, e is the Oswald coefficient, n is the load factor, C_{D0} is the induced drag coefficient, $C_{L_{Max}}$ is the maximum lift coefficient, d_{t0} is the take-off distance and v is the cruising speed.

All these equations can be plotted together in a constraint diagram, leading to a design space definition.

2.2 Wing Loading and Stress

The wing lift is the force that balances the aircraft weight and maintains it aloft. It is produced by the flow passing on the wing mainly, and is proportional to the square of the aircraft speed in subsonic flight. Lift is a function of lift coefficient C_L, air density ρ, wing area S and cruise velocity v as follows

$$L = \frac{1}{2}C_L\, S\,\rho\, v^2. \tag{5}$$

Assuming that only the wing produces lift, for flight balanced condition, the lift must be equal to weight W multiplied by load factor n as follows

$$L = n\, W = n\, m\, g. \tag{6}$$

Equation (6) can be rewritten as

$$L - n\,W = L - n\,m\,g = 0. \tag{7}$$

For the structural sizing and aerodynamic analysis of an aircraft to be performed properly, it is of utmost importance to determine the lift distribution along the wingspan. The model shown in Fig. 1 is derived from the Prandtl line theory and represents a particular case applied to elliptical shaped wing, called the elliptical lift distribution. From this theory, it is possible to approximate the lift distribution on a wing with no elliptical geometric shape.

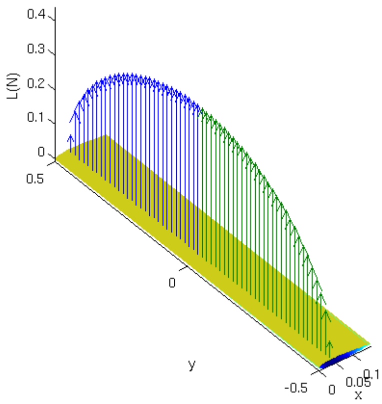

Figure 1: **Wing lift distribution on a rectangular planform wing.**

By the application of this theoretical model, it is possible to estimate the circulation distribution $\Gamma(y)$ along the wingspan. Besides this, applying the Kutta-Joukowski theorem, it is possible determine the lift force that acts in each section of the wingspan. Therefore, the distribution of the circulation along the wingspan is calculated directly by the following equation:

$$\Gamma(y) = \Gamma_0 \sqrt{1 - \left(\frac{2y}{b}\right)^2}, \tag{8}$$

where Γ_0 is a constant and represents the circulation at the midpoint of the wing under study. Equation 8 shows that $\Gamma(y)$ reaches its maximum value at the midpoint of the wing, with the coordinate of position $y = 0$ and decreases to zero at the tips of the wing where $y = \pm b/2$. According to the Prandtl line theory, the circulation at the midpoint of the wing is obtained by

$$\Gamma_0 = \frac{4L}{\rho\,v\,b\,n}. \tag{9}$$

By determining the value of Γ_0 (m^2/s), the distribution of circulation along the entire wingspan is calculated. Then, the lift force acting on each section can be calculated by Kutta-Joukowski's theorem as follows

$$L(y) = \rho\,v\,\Gamma(y). \tag{10}$$

Such methodology allows to obtain, in a simplified way, the lift distribution along the wingspan and thus a more adequate stress distribution.

In order to start the design of the wing spar, it is proposed the concept of a box spar, with outer dimensions a_1, horizontal, and a_2, vertical, and thickness t_1 and t_2, in the same order. For the wing structural sizing, it should be considered that the aerodynamic lift force generates bending and shearing stresses in aircraft structure. For the aerodynamic lift distribution given by Kutta-Joukowski theorem, the wing root is the critical region. Due to that, the stresses are computed only at this point in the present work.

The equation for the maximum stress at the wing spar, i.e. wing root, is given by

$$\sigma = \frac{M_{root}\,h/2}{I_x}, \tag{11}$$

where M_{root} is the resulting bending moment at the wing root, I_x is the cross sectional inertia moment at the root and h is half height of the beam.

The maximum bending stress σ_{adm} on the wing spar due to the sum of the aerodynamic lift forces is

$$\sigma_{adm} = \frac{\sum M_{lift}\,a_2}{2\,I}, \tag{12}$$

where M_{lift} is the moment caused by aerodynamic lift force on wing.

The shear stress τ_{spar} in the cross section of the wing spar is

$$\tau_{spar} = \frac{\sum F_{lift}}{A_{spar}}, \tag{13}$$

where F_{lift} is the aerodynamic lift force on wing and A_{spar} is wing spar cross section area.

The admissible bending and shear stresses allowed in the spar are, respectively,

$$\sigma_{adm} \leq \frac{\sigma_{esc}}{SF} \qquad \tau_{spar} \leq 0.75\,\sigma_{esc}, \tag{14}$$

where σ_{esc} is the material yield stress and SF is the safety factor. Since the optimization problem includes the minimization of the wing weight, it is necessary to define the wing parameters. Initially, the wing will be divided into 3 parts for mass computation: spar, ribs and skin. In this work, only the mass of the spar will be calculated. The aileron and the second spar will be considered in future calculations.

2.3 Empennage Sizing

Tail sizing is one of the least accurate parts of aircraft design. The horizontal and vertical surfaces must be adequate to guarantee stability and control of the aircraft. The equations applied to estimate the tail use two dimensionless variables called horizontal tail volume V_h and vertical tail volume V_v. The coefficients for the proposed RPA may be found by researching comparatives UAVs (Unmanned Aerial Vehicles).

Landolfo (2008) uses V_v = 0.0375 and V_h = 0.675. These coefficients are out the limits considered by Rodrigues (2013) based on single engine aircraft, but are close enough to be taken into account.

The equations used to determine these coefficients of the vertical and horizontal tails can be rewritten as

$$S_h = \frac{V_h\,\bar{c}\,S}{l_h} \qquad S_v = \frac{V_v\,b\,S}{l_v}, \tag{15}$$

where l_h is the distance between aircraft CG and the aerodynamic center of the horizontal tail surface, l_v is the distance between aircraft CG and the aerodynamic center of the vertical tail surface, S_h is the required area for horizontal tail surface, S_v is the required area for vertical tail surface and $\bar{c}$ represents the mean aerodynamic chord of the wing.

The tail configuration for the proposed RPA is the conventional one. The horizontal empennage is only composed by the elevator. Assuming the elevator geometric shape is a rectangle, it's necessary to estimate an aspect ratio ($A.R._h$) to calculate the elevator chord c_h and span b_h. The same procedure is applied to the rudder.

2.4 Stability

According to Loureiro (1979), the location of the center of gravity of an airplane in relation to the center of pressure of aerodynamic forces in the wing and tail is important due to aerodynamic and structural reasons. The position of theses limits is usually given in terms of the percentage of the mean aerodynamic chord.

The forward and backward movement of the CG must be maintained within specified limits. Usually, the forward limit is determined by airplane control demand, for example: the pilot needs to be able to turn the tail down for aircraft landing. The airplane tends to get more stable when the CG is moved forward in relation to the aerodynamic center, and it demands larger deflections of the elevator to a change in the attitude of flight. When the CG of the airplane is moved backward, the stability of the airplane decreases, so that there is a posterior limit to the position of the CG to have positive stability.

By definition, the lift L and drag D forces actuate always perpendicular and parallel to the aerodynamic velocity direction, respectively. The action lines of theses forces are not fixed in the airplane, but they vary with the angle of attack. Then, it is convenient to decompose these forces in parallel components to the reference axis z and x that are fixed in the airplane. Solving the forces on the wing in parallel components in the z and x axis,

$$F_{zw} = L\,cos\,\alpha + D\,sin\,\alpha = \left(\frac{1}{2}\rho\,C_L\,S\,v^2\right) cos\,\alpha + \left(\frac{1}{2}\rho\,C_D\,S\,v^2\right) sin\,\alpha \tag{16}$$

and

$$F_{xw} = D\,cos\,\alpha - L\,sin\,\alpha = \left(\frac{1}{2}\rho\,C_D\,S\,v^2\right) cos\,\alpha - \left(\frac{1}{2}\rho\,C_L\,S\,v^2\right) sin\,\alpha, \tag{17}$$

where F_{zw} is the decomposition in z axis of the forces of lift and drag acting on the wing, F_{xw} is the decomposition in x axis of the forces of lift and drag acting on the wing and α is the angle of attack.

The moment due to the horizontal component of the empennage force F_{xt} is usually negligible, then it is general practice to consider only the F_{zt} component. The resulting moment in y axis of the airplane is

$$\sum M_y = -F_{zw}\,x_w + F_{xw}\,z_w + M_{ac} + M_f - F_{zt}\,l_t, \tag{18}$$

where M_{ac} is the aerodynamic moment, F_{zt} is the decomposition in z axis of the resulting forces acting on the horizontal tail, z_w is the vertical distance between CG and the aerodynamic center, x_w is the horizontal distance between CG and aerodynamic center, l_t is the horizontal distance between CG and the aerodynamic center of tail and M_f is the resulting moment due to several forces acting on the airplane.

3. PROPOSED OPTIMIZATION PROCEDURE

The optimization problem currently in study is divided in three and which one is composed by one objective function f_i. In the future, others sub problems can be added. Figure 2 presents the MDO architecture used. However, a multi-disciplinary system depends not only on the performance of the individual disciplines but also by their interactions. The proposed architecture is the first one tested on the simplified aircraft design process.

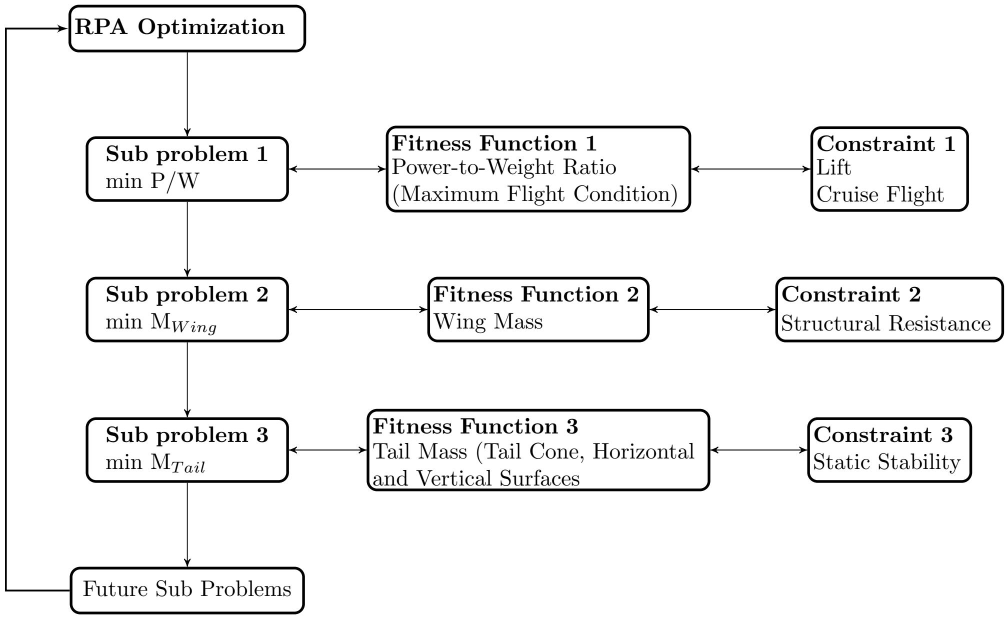

Figure 2: **Flowchart of the proposed architecture of the MDO problem.**

The first sub problem of optimization is a nonlinear constrained minimization of a fitness function of two variables using genetic algorithm. The fitness function f_1 is a power-to-weight (P/W) ratio presented in Landolfo (2008) for the Maximum Load condition given in Eq. 1. The MATLAB function depends on the vector $\mathbf{x}_1$ composed of two project variables: wingspan (b) and chord (c). It is necessary also to define the fixed parameters vector $\mathbf{p}_1$. The constraint is the lift (L) required for the airplane given in Eq. (7).

The first optimization sub problem is defined as:

$$\begin{aligned} \min \quad & f_1(\mathbf{x}_1, \mathbf{p}_1) = \frac{P}{W} \; for\ Maximum\ Load \\ s.t. \quad & \\ & L - nW = 0; \\ & b_{Min} < b \leq b_{Max}; \\ & c_{Min} < c \leq c_{Max}. \end{aligned} \tag{19}$$

The vector of design variables $\mathbf{x}_1$ and fixed parameters $\mathbf{p}_1$ are:

$$\begin{cases} \mathbf{x}_1 = \{b \quad c\}^T \\ \mathbf{p}_1 = \{\eta_p \quad e \quad n \quad C_{D0} \quad C_{L_{Max}} \quad d_{t0} \quad v \quad m\}^T. \end{cases} \tag{20}$$

The purpose of the second sub problem is to minimize the spar volume in order to obtain the lowest possible mass. Thus, the design variables for this problem are a_1, a_2, t_1 and t_2. The constraints are the maximum bending and shear stresses allowed in the spar. The second sub problem of optimization is:

$$\begin{aligned} \min \quad & f_2(\mathbf{x}_2, \mathbf{p}_2) = V_{spar} = [(a_1\, a_2) - (a_2 - 2\, t_1)(a_1 - 2\, t_2)]\, b/2 \\ s.t. \quad & \\ & \sigma_{adm} \leq (\sigma_{esc}/SF); \\ & \tau_{spar} \leq 0.75\sigma_{esc}; \\ & a_{Min} < a_1 \leq 0.2\, c; \\ & a_{Min} < a_2 \leq 0.12\, c; \\ & t_{Min} < t_1 \leq 0.12\, c/2; \\ & t_{Min} < t_2 \leq 0.2\, c/2. \end{aligned} \tag{21}$$

Here, the vector of design variables $\mathbf{x}_2$ and fixed parameters $\mathbf{p}_2$ are:

$$\begin{cases} \mathbf{x}_2 = \{a_1 \quad a_2 \quad t_1 \quad t_2\}^T, \\ \mathbf{p}_2 = \{m \quad b \quad v \quad n \quad \rho \quad \sigma_{esc} \quad SF\}^T. \end{cases} \tag{22}$$

Another optimization sub problem is the tail mass minimization. The design variable is the distance l_h between aircraft CG and the aerodynamic center of the horizontal / vertical tail surface.

The quarter chord of the tail may be placed 3 chord lengths after the aerodynamic center of the wing according to Loureiro (1979). The upper and lower bounds of the design variable are estimated based on this information.

$$\begin{array}{ll} \min & f_3(\mathbf{x}_3, \mathbf{p}_3) = M_T = (\rho_1\, l_h\, \frac{\pi}{4}(d_{out}^2 - d_{in}^2) + \rho_2\, S_2\, b_h + \rho_3\, S_3\, b_v) \\ s.t. & \\ & F_{zw}\, x_w + F_{xw}\, z_w + M_{ac} - F_{zt}\, l_t = 0; \\ & 2c < l_h \leq 4c. \end{array} \tag{23}$$

For this problem, the vector of design variables $\mathbf{x}_3$ and fixed parameters $\mathbf{p}_3$ are:

$$\begin{cases} \mathbf{x}_3 = \{l_h\}^T, \\ \mathbf{p}_3 = \{c \quad b \quad \rho_1 \quad d_{out} \quad d_{in} \quad \rho_2 \quad S_2 \quad A.R._h \quad V_h \quad \rho_3 \quad S_3 \quad A.R._v \quad V_v\}^T. \end{cases} \tag{24}$$

4. NUMERICAL STUDIES

The aircraft conceptual sketch, shown in Fig. 3, is established a priori, as one of the first steps in the conceptual design. In the present case, it was defined as an aircraft with high cantilever wing, tubular tail cone and rectangular plan form for all surfaces. The constraint diagram mentioned in section 2.1 is shown in Fig. 4. The values of the variables used are presented in Tab. 1.

Most of the data shown in Tabs. 1, 2 and 3 are based on the values used by Landolfo (2008) due to similarity of work. The wing, rudder and elevator airfoil is chosen based on manufacturing skills. The material selected for the spar is balsa wood. The tail cone material is carbon fiber, estimating external and internal diameters. The C_L and C_D are given in NACA 0012 data for Reynolds number equal to 500 000. The rudder and elevator material is a generic foam, disregarding spars and ribs. It may be compensated with a density factor in future works.

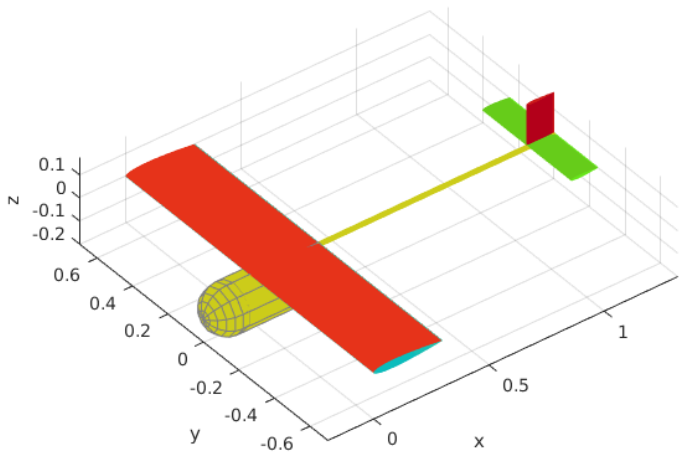

Figure 3: **Conceptual sketch.**

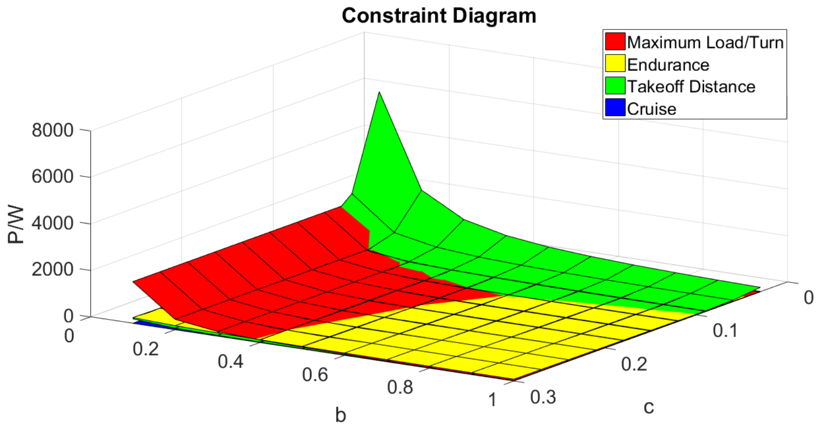

Figure 4: **Design space defined by Eqs. (1) to (4) based on design variables (chord and span).**

The optimization sub problem 1 is achieved through genetic algorithm (MATLAB). The values of the variables used are shown in Tab. 1. Optimization terminated when average change in the fitness value was less than 10^{-8} and constraint violation was less than 10^{-5}. The number of generations was 15. The number of function evaluations was 47350. The optimum design variables were b = 0.8817 m and c = 0.2904 m, when b_{Max} = 1 m and c_{Max} = 0.3 m. The fitness function value was 24.4136 W/N.

The optimization sub problem 2 is achieved through the *fmincon* MATLAB function (MathWorks, 2018). The purpose of such a function is to find the minimum value of a nonlinear function with multiple variables, starting from an initially estimated value. The variables values used are shown in Tab. 3. The wing chord and wingspan measurements used will be those obtained in the optimization of power-to-weight ratio. The results are a_1 = 0.0581 m, a_2 = 0.0349 m, t_1 = 0.0010 m and t_2 = 0.0010, when $0.001 < a_1 \leq 0.2\ c$, $0.001 < a_2 \leq 0.12\ c$, $0.001 < t_1 \leq 0.12\ c/2$ and $0.001 < t_2 \leq 0.2\ c/2$.

The optimization sub problem 3 is also achieved through genetic algorithm (MATLAB). The values of the variables used are presented in Tab. 2. The optimum design variable was l_h = 0.5809 m, when $2\ c < l_h \leq 4\ c$. The fitness function value was 0.477752 kg. Then, it was possible to estimate c_h = 0.1697 m, c_v = 0.1151 m, b_h = 0.5092 and b_v = 0.1266 m.

Table 1: **Preliminary design data.**

Flight altitude	150	m
m	2	kg
Max weight		N
ρ	1.204	kg/m^3
v	20	m/s
n	4.4	
α	5^o	
e	0.8	
η_p	0.7	
d_{t0}	14	m
b_{Min}	0.001	m
c_{Min}	0.001	m
Wing airfoil data		
$C_{L_{Max}}$	1.4	
C_{D0}	0.015	
C_M	-0.0125	when $\alpha = 5^o$

Table 2: **Tail design data.**

V_h	0.675	
V_v	0.0375	
$A.R._h$	3	
$A.R._v$	1.1	
ρ_1	1800	kg/m^3
d_{out}	0.03	m
d_{in}	0.02	m
ρ_2	100	kg/m^3
ρ_3	100	kg/m^3
x_w	5% $\bar{c}$	
z_w	0.06	m
Airfoil	NACA 0012	
C_L	0.6275	when $\alpha = 5^o$
C_D	0.01036	when $\alpha = 5^o$

Table 3: **Wing stress design data.**

σ_{esc}	1.3	MPa
SF	2	
a_{Min}	0.001	m
t_{Min}	0.001	m

Table 4: **Output data.**

b	0.8817	m	l_h	0.5809	m
c	0.2904	m	c_h	0.1697	m
a_1	0.0581	m	c_v	0.1151	m
a_2	0.0349	m	b_h	0.5092	m
t_1	0.0010	m	b_v	0.1266	m
t_2	0.0010	m			

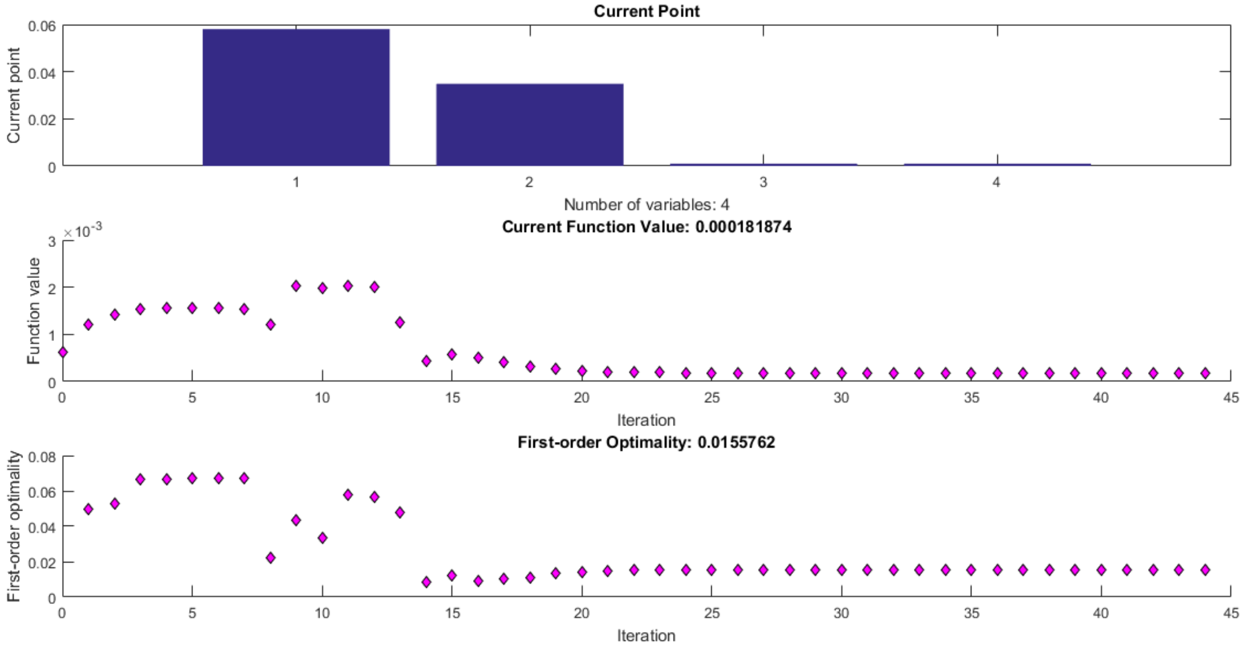

Figure 5: **Fitness value vs. Generations of the sub problem 2.**

5. CONCLUSION

In this paper, a procedure for preliminary design of RPAs was discussed. It is a tool that can already be used to obtain a feasible design of an aircraft allowing manufacture and operation. As in traditional aircraft design, the proposed procedure requires a large amount of inputs, i.e. wing, elevator and rudder airfoil characteristics and material, cruise and stall speed estimation, load factor, take-off distance, spar material characteristics, elevator and rudder aspect ratio, tail cone material and sizing. The final result depends on multiple information and estimation based on similar models. If more sub problems are added to the formulation, the solution gets closer to the optimum once more arguments are taken into account. In this case, more inputs would be necessary to get a coherent result. A positive aspect is the automation obtained for the search of an adequate combination of all parameters. As a design technique, it would be more reliable if others MDO architectures were tested, aiming to determine which one is more efficient for the problem in study. The most appropriate architecture often depends on the nature of the problem and on the designers experience.

This is an ongoing project, and results are being studied and the methodology is being refined. The optimization process is being improved to include more design tasks and better compliance with MDO standards. Meanwhile, a goal is to manufacture the aircraft design obtained with the methodology, and then evaluate their performance.

6. ACKNOWLEDGMENT

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