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A First Approach to Solar Aviation with the Use of Axiomatic Design

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Abstract

In most of the existing solar airplanes, solar cells are allocated on the top of the wing. From the perspective of Axiomatic Design, this configuration creates a dependency between power, endurance and wing load. The aim of this article is to present a preliminary formulation of the design problem with quantitative transfer functions to which Suh's principles are then applied. The formulation of the design problem results in a set of two main functional requirements (FR), endurance and excess of specific mechanical power, which the laws of physics relate to their (DP) through the transfer functions. This formulation is exposed in a way that facilitates its use as an example for teaching Axiomatic Design. The analysis of the DP tendencies according to Axiomatic Design principles depicts the technological limits of solar aviation, focusing on how the aforementioned dependencies constraint the basic topology of solar aircraft to a large range of potential missions in terms of endurance and excess of specific mechanical power.

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Keywords: Solar Aviation, Axiomatic Design, Design Limits.

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1. Nomenclature

a_n	Normal acceleration
A_{SC}	Area covered by solar modules
b	Wingspan
C_D	Drag coefficient
C_{D0}	Drag coefficient when $C_L = 0$
C_L	Lift coefficient of the aircraft
D	Aerodynamic drag
e	Oswald efficiency
E_{batts}	Stored energy in the batteries
$E_{battsmax}$	Maximum storage of energy in the batteries
g	Gravity acceleration at sea-level
$I(t)$	Sun irradiation at an instant t
K_1	Induced drag constant
$k_{controller}$	Mass constant of the controller
$k_{gearbox}$	Mass constant of the gearbox
k_{motor}	Mass constant of the motor
k_{MPPT}	Mass constant of the MPPT
$k_{propeller}$	Mass constant of the propeller
$k_{structure}$	Mass constant of the structure model
L	Lift
M	Total weight of the aircraft
M_{1batt}	Weight of one battery cell
$M_{structure}$	Aircraft structure mass
$M_{propulsive}$	Mass of all the elements contributing to the propulsion of the aircraft
$M_{nonpropulsive}$	Mass of auxiliary equipment and payload
n	Load factor
n_batts	Number of batteries
$P_{mechanical}$	Excess of specific mechanical power
q	Dynamic pressure
S	Wing area
T	Propulsive force
t_1*	Initial time of batteries charge
t_2*	Final time of batteries charge
t_3*	Initial time of batteries charge for new solar cycle
$t_{discharge}$	Time while the batteries are discharged

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$T_{endurance}$	Estimated endurance of the aircraft
t_{solar}	Time flown with solar energy
V	Flight speed
V_a	Ascension speed
$\dot{W}_{propulsive}$	Power dedicated to motors and propellers
$\dot{W}_{non-propulsive}$	Additional power needed by the aircraft
$\dot{W}_{solar}$	Power input in the MPPT from solar power
λ	Aspect ratio
θ	Speed pitch angle
φ	Angle between propulsive force and X_{body} axis
ψ	Angle of attack
ρ	Air density
$\eta_{weather}$	Weather efficiency
η_{sc}	Efficiency of solar modules
η_{BEC}	Step-down converter efficiency
η_{camber}	Efficiency due to camber
η_{charge}	Efficiency of the batteries charge process
$\eta_{controller}$	Motor controller efficiency
$\eta_{discharge}$	Efficiency of the batteries discharge process
$\eta_{gearbox}$	Gearbox efficiency
η_{motor}	Motor efficiency
η_{MPPT}	Efficiency of the MPPT
$\eta_{propeller}$	Propeller efficiency
$\eta_{propulsive}$	Total efficiency of propulsive system
η_{solar}	Total efficiency related to solar cells

2. Introduction / State of the art

When studying a relatively new segment as solar aviation represents for aerospace industry, Axiomatic Design can be valuable in order to analyze the critical dependencies that depict the technological limits of new products [1]. As far as the design process in solar aviation is concerned, Irving [2] conducted one of the first studies on feasibility of solar aircrafts. Different models have been suggested with typical payloads and mission profiles [3][4], [5], [6] and significant information about the manufacturing process and performance of solar airplanes has been studied [7], [8].

Noth [8] carried out one of the most complete studies in the field, developing an analytical methodology for the conceptual design of solar airplanes that can fly for more than 24 hours.

In the framework of Prontas project, the achievement of an unmanned solar airplane with unlimited autonomy was studied [9], [10], [11], while Leutenegger et al. [5] studied parameters such as the influence of the latitude on the maximum constant altitude for sustained flight and the influence of season and latitude on the endurance of a small radio-controlled UAV. Spangelo et al. [12] analysed path planning for closed ground tracks, and Shiau et al. [13] used genetic algorithms for determining the optimal sizing and cruise speed of solar UAVs.

Concerning the topology of solar aircrafts, many have been developed at a prototype scale for particular design points [14][15]. According to the specificities of their mission, different configurations are considered to meet the mission profile requirements. Concerning long endurance and high altitude airplanes (HALE), Brandt et al. developed a methodology by adapting classic methods to solar aircrafts based on a specific mission profile [4], NASA developed HALE-UAV ‘Pathfinder’ for stratospheric flights [16], and SolarShip Inc. [17] have considered other hybrid designs.

The scope of this paper is to analyze from Axiomatic Design perspective the main dependencies of wing based solar airplanes, manned or unmanned, that may restrict the achievement of unlimited endurance with manoeuvring capabilities.

To accomplish this design problem, the following methodological steps will be followed: first the design problem will be formulated in terms of the minimum set of independent FRs and the main input and system constraints. Second, the available DPs will be identified by reformulating the FRs in terms of transfer functions. Finally the independence of FRs will be analyzed.

For this purpose, this article is structured as follows: a brief historical review is presented first, then the design problem is formulated to finally complete Axiomatic Design analysis.

3. Brief historical review

Solar aviation, understood as flown in the atmosphere, was born in 1974, when the Sunrise I UAV flew in California (USA) [18]. It completed twenty-eight successful flights before being destroyed by turbulence. The first manned power aircraft exclusively by solar energy was the Gossamer Penguin in 1980 [7]. One year later, Paul MacCready flew across the English Channel on board of the Solar Challenger. Eric Raymond crossed the United States with the Sunseeker in 1990 over 21 flights with a total duration of 121 flying hours. Solar aircrafts, mainly UAVs, thrived in the 90s as the technologies and efficiencies of the solar cells improved. NASA studied new configurations in the late 90s with Pathfinder, Centurion and Helios HALE UAVs, but they were not a total success and fragility problems lead to the in-flight destruction of the Helios [16]. SoLong UAV [15] and the manned Solar Impulse [6], [19] flew at night in 2005 and 2010 respectively relying only on the stored solar energy; the latter recently crossed the United States and it has achieved many records of endurance and distances.

4. Formulation of the design problem

The objective of this Section is to formulate the design problem by means of the challenge definition, the minimum set of independent functional requirements (FR) and the main constraints acting in the design problem.

4.1. Challenge definition

As a first approach, a solar airplane capable of achieving long endurances under cruise conditions with an excess of mechanical power for succeeding unexpected manoeuvres at any time should be able to complete a large variety of missions. Therefore, the challenge definition can be enounced as: study the basic topology and the main dependencies of wing-based solar airplanes that can maximize endurance and ensure excess of mechanical power at any time.

4.2. Selection of the minimum number of independent FR in a neutral solution environment

According to Axiomatic Design, FRs and their constraints are the minimum set of independent elements that characterizes the complete design problem for each level of the design hierarchy [21]. In the case in study, the minimum set of FRs for the first level of the design hierarchy to ensure the maximum number of mission profiles can be settled by two functional requirements: 1) endurance during cruise conditions $T_{endurance}$ and 2) excess of specific mechanical power $P_{mechanical}$ for performing unexpected manoeuvres. Note that to ensure the independence of the two FRs, endurance is enounced for cruise conditions where $P_{mechanical} \approx 0$.

4.2.1. Endurance ($T_{endurance}$)

With independence of the mission profile, every mission must be accomplished in a certain time. When all the mission profiles are considered together, a requirement for fulfilling the maximum number of mission profiles is to maximize endurance. In the particular case of solar aviation, for endurances larger than 24 hours a new cycle of Sun irradiation begins. Depending on the energy level of the batteries, weather conditions, geographical location and date, once an endurance of 24 hours is achieved, perpetual endurance could be achieved too. To ensure FR independence, endurance is considered under cruise conditions.

4.2.2. Excess of specific mechanical power ($P_{mechanical}$)

For any mission considered, performances about ascending, accelerating or manoeuvring are required. When all the missions' profiles are considered together, a requirement for fulfilling the maximum number of operational profiles is to have an excess of specific power in every instant of the mission for increasing the ability of the aircraft to modify its flight situation.

4.3. Physical and technological laws: system & input constraints

4.3.1. Flight physics

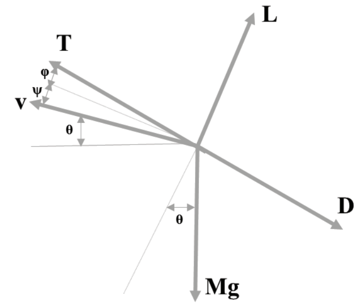

Figure 1. Force equilibrium for general flight situation.

Figure 1 illustrates the force equilibrium for a general flight situation. The force equilibrium equations in the X and Z axes lead respectively to Eq. 1 and Eq. 2 (x-direction is parallel to the relative airspeed and positive to the nose, and z-direction is positive upwards). This model assumes that neither lateral wind component nor lateral forces exist.

$$M\frac{dV}{dt} = Tcos(\psi+\varphi) - D - Wsin(\theta) \tag{1}$$

$$Ma_n = L + T\,sin(\psi+\varphi) - Mgcos(\theta) \tag{2}$$

Normal acceleration in Eq. 2 takes into account whether the aircraft is manoeuvring. This equation can be redefined without any loss of generality by means of the load factor *n*. Assuming $\psi + \varphi \ll 1$, Eq. 2 can be written as:

$$n = \frac{L}{Mg} = \frac{a_n}{g} + cos(\theta) - \frac{T}{Mg}(\psi+\varphi) \tag{3}$$

And from Eq. 1, the specific mechanical power can be expressed:

$$P_{mechanical} = Vsin(\theta) + \frac{d}{dt}\left(\frac{V^2}{2g}\right) = V\frac{T-D}{Mg} \tag{4}$$

Aerodynamic forces are stated in Eq. 5 and 6, where a parabolic polar curve is assumed.

$$C_L = \frac{L}{qS} = \frac{nMg}{qS} \tag{5}$$

$$C_D = \frac{D}{qS} = C_{D0} + K_1 C_L^2 \tag{6}$$

Combining Eq. 5 and 6 with Eq. 4 $P_{mechanical}$ can be re-expressed:

$$P_{mechanical} = \frac{V}{Mg}\left[T - C_{D0}qS - K_1\frac{(nMg)^2}{qS}\right] \tag{7}$$

4.3.2. Mass prediction and structural model

The total mass of the airplane M can be decomposed into the following terms:

$$M = M_{propulsive} + M_{non-propulsive} + M_{structure} \tag{8}$$

$M_{propulsive}$ refers to the mass of all the elements contributing to the propulsion of the aircraft: solar cells, batteries, maximum power point tracker (MPPT), controller, motor, gearbox and propeller. $M_{nonpropulsive}$ refers to the mass due to the auxiliary equipment and payload. Finally, $M_{structure}$ includes the mass due to the structure of the aircraft.

Because the purpose of this article is not to present a detailed modelled for each term, a few basic considerations are going to be made. In order to predict $M_{propulsive}$, Noth [8] presents a linear model that relates the power of a propulsive element with its mass, $M_{propulsive} = \sum_{i}^{n} k_i \dot{W}_i$ $M_{nonpropulsive}$ considers the mass of those aircraft systems that contribute to neither propulsion nor structure, such as avionics or payload. $M_{structure}$ relates the structure sizing with its weight in order to predict the relations between structural behaviour, size and mass. Due to geometrical non-similarity when fuselage and wingspan sizes increase, the validity of existing models is restricted to a range of wingspans, weights or type of aircraft. As a consequence, no accurate model exists for all the possible airplane configurations. For instance, for small wingspans (less than 4'5m), Noth obtained $M_{structure} = k_{structure} S^{1.55} \lambda^{1.30}$ which becomes too pessimistic as the wingspan increases [8].

4.3.3. Solar irradiance model

A simplified sinusoidal model for solar irradiance [8] which defines the variation of solar irradiance intensity during a 24 hours period for a given geographical location has been considered. Main two parameters involved in this model are the maximum irradiance and daytime, both of them specific for each date of the year and location. Eq. 9 represents this model considering noon as the time origin:

$$I(t) = I_{max} \left| cos\left(\frac{\pi t}{T_{day}}\right) \right| \tag{9}$$

4.3.4. Energy balance

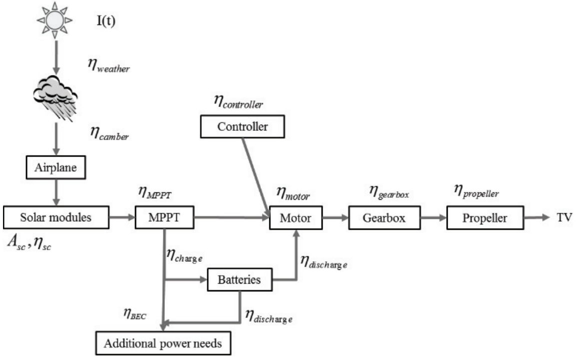

Figure 2. Propulsive system considered.

The block diagram in Figure 2 illustrates the propulsive system of a solar aircraft considered in this study. During daytime conditions, the Sun irradiation may be reduced by atmospheric conditions, which is reflected by an efficiency $\eta_{weather}$. The resultant irradiation strikes the surface A_{SC} of the solar cells that generate electric power with a total efficiency η_{sc}. Each solar cell sees a different incidence angle from the Sun's rays due to the camber of the aerofoil, and this is reflected by a camber efficiency η_{camber}. To ensure that the solar modules work at the maximum power point of their voltage-current curves, a Maximum Power Point Tracker (MPPT) is used with an efficiency η_{MPPT}.

The aircraft power needs are to power the motors and propellers using a total amount $\dot{W}_{propulsive}$ which includes losses, and also to power auxiliary systems such as the avionics, navigation instruments or the payload $\dot{W}_{nonpropulsive}$. In the latter case, they may need a reduction in the voltage, which leads to the use of a step-down converter with an efficiency η_{BEC}.

Regarding the recharge process of the batteries, it is done with an efficiency η_{charge}. When the energy obtained from the MPPT is not enough to power the aircraft, those batteries discharge with an efficiency $\eta_{discharge}$.

The functioning of the motors is monitored by an electronic controller or converter. The torque power delivered is transmitted through a gearbox to the propellers, which give the propulsive force conditioned by their own efficiency (η_{motor}, $\eta_{controller}$, $\eta_{gearbox}$, $\eta_{propeller}$).

After this model the following variables can be defined:

$$\eta_{propulsive} = \eta_{propeller} \eta_{gearbox} \eta_{controller} \eta_{motor} \tag{10}$$

$$\eta_{solar\ energy} = \eta_{MPPT} \eta_{SC} \eta_{weather} \eta_{camber} \tag{11}$$

Hence, the solar power available to power the aircraft, the propulsive power as a function of the torque and the non-propulsive power at a certain instant t can be defined as:

$$\dot{W}_{solar} = I(t) A_{SC} \eta_{solar\ energy} \tag{12}$$

$$\dot{W}_{propulsive}(t) = \frac{TV}{\eta_{propulsive}} \left(\chi + \frac{1-\chi}{\eta_{charge} \eta_{discharge}} \right) \tag{13}$$

$$\dot{W}_{non\text{-}propulsive}(t) = \frac{\dot{W}_{equipments}}{\eta_{BEC}} \left(\chi' + \frac{1-\chi'}{\eta_{charge} \eta_{discharge}} \right) \tag{14}$$

Where $\chi \in [0,1]$ and $\chi' \in [0,1]$ are a function of time that takes into account that the power used for propulsion or for auxiliary units can come from the MPPT or from the batteries, which implies two different efficiencies in the conversion of the solar power to propulsive power.

From previous equations, the variation of stored energy in the batteries is given by Eq. 15:

$$\frac{dE_{batteries}}{dt} = \dot{W}_{solar} - \dot{W}_{propulsive} - \dot{W}_{nonpropulsive} \tag{15}$$

And combining Eq. 12, 13, 14 and 15, the energy balance for the solar airplane during the time of the mission [t_0, $t_{0+}t_{endurance}$] can be written:

$$\Delta E_{batt}(t) = \int_{t_0}^{t_0+t_{endurance}} \left(\begin{array}{l} I(t) A_{SC} \eta_{solar\ energy} - \dfrac{TV}{\eta_{propulsive}} \left(\chi + \dfrac{1-\chi}{\eta_{charge} \eta_{discharge}} \right) \\ - \dfrac{\dot{W}_{equipments}}{\eta_{BEC}} \left(\chi' + \dfrac{1-\chi'}{\eta_{charge} \eta_{discharge}} \right) \end{array} \right) dt \tag{16}$$

4.3.5. Propulsive efficiencies and batteries characteristics

A typical range of values for the η_i efficiencies can be found in Noth [8], which strongly depend on state of the art of the considered technologies.

Complex models exist for Li-Ion [20] and Li-Po batteries [21]. For the purposes of this study, in order to preliminary analyze the contribution of the batteries' weight to the total weight of the aircraft, a simplified model is considered, where the energy stored in the batteries is written as a function of the energetic density of the batteries k_{batts} (J/kg) the number of batteries and their mass:

$$E_{max\,batt} = n_{batt} k_{batt} M_{1batt} \geq E_{batt}(t) \tag{17}$$

5. Axiomatic Design Analysis

In this section the two FRs will be quantitatively formulated from the law of physics obtained in section 4.3, deriving the transfer functions that relate them with the available DPs and approaching their analysis from Axiomatic Design perspective.

5.1.1. Quantitative reformulation of functional requirements

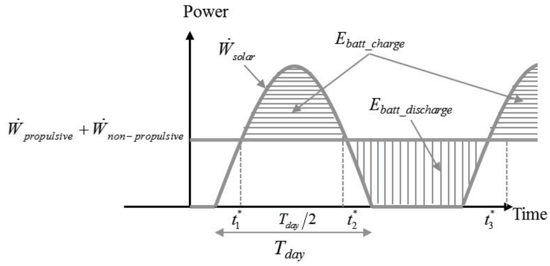

Figure 3. Simplified energy balance model.

In Figure 3 a simplified model of the energy balance is considered for this section purposes. Additionally, the following assumptions will be made:

- It will be considered that the energy to power the airplane comes directly from the solar cells during day flight or from the batteries during night flight, which implies:
  $$\chi(t) = \chi'(t) = 1 \text{ for } t \in \left[t_1^*, t_2^*\right]$$
  $$\chi(t) = \chi'(t) = 0 \text{ for } t \notin \left[t_1^*, t_2^*\right]$$
- As a first approach, the average of power needs $\dot{W}_{propulsive}$ and $\dot{W}_{nonpropulsive}$ will be considered constant and equal at day and night flight. Consequently the excess of $P_{mechanical}$ would be used for punctual changes of flight situation.

5.1.2. Excess of specific mechanical power

From flight physics and energy balance in section 4.3, $P_{mechanical}$ for day and night flight can be written.

$$\begin{gathered}
P_{mechanical}\big|_{day} = \frac{\eta_{propulsive}\left(\eta_{solar\ energy} A_{SC} I(t) - \dfrac{1}{\eta_{charge}}\dfrac{dE_{batt}}{dt} - \dfrac{\dot{W}_{non-propulsive}}{\eta_{BEC}}\right)}{Mg} \\
-\frac{\left(C_{D0} + K_1 C_L^2\right)}{C_L^{3/2}} n \sqrt{\frac{2Mg}{\rho S}} \\
P_{mechanical}\big|_{night} = \frac{\eta_{propulsive}\left(\eta_{discharge}\dfrac{dE_{batt}}{dt} - \dfrac{\dot{W}_{non-propulsive}}{\eta_{BEC}}\right)}{Mg} \\
-\frac{\left(C_{D0} + K_1 C_L^2\right)}{C_L^{3/2}} n \sqrt{\frac{2Mg}{\rho S}} \\
P_{mechanical\ average}\big|_{day} = P_{mechanical\ average}\big|_{night}
\end{gathered} \tag{18}$$

Where the condition $P_{mechanical} > 0$ should be satisfied at any time of the mission.

5.1.3. Endurance

In order to complete endurances larger than 24 hours, a well-dimensioned aircraft in terms of power needs should be able to reach the next solar cycle.

To calculate endurance, the additional assumptions are done:

- Endurance can be written:

$$T_{endurance} = t_{solar} + t_{discharge} \tag{19}$$

- Where $t_{solar} = t_2^* - t_1^*$ and $t_{discharge} = t_3^* - t_2^*$
- Load factor $n=1$.
- To keep the independence of FR definition, endurance is calculated for $P_{mechanical} = 0$ which implies $T = D$ and V and h constant.

Under these assumptions, t_{solar} can be calculated from Eq. 15 making $dE_{batt}/dt = 0$ which yields to:

$$t_{solar} = \frac{2T_{day}}{\pi} \arccos\left(\frac{\dfrac{\sqrt{\left(\dfrac{2Mg}{\rho S}\right)}\dfrac{\left(C_{D0} + K_1 C_L^2\right)}{C_L^{3/2}}}{\eta_{propulsive}} Mg + \dfrac{\dot{W}_{non-propulsive}}{\eta_{BEC}}}{\eta_{solar\ energy} A_{SC} I_{max}} \right) \tag{20}$$

To calculate $t_{discharge}$, energy balance is now apply between night time $\left[t_2^*, t_3^*\right]$:

$$\begin{gathered}
\int_{t_2^*}^{t_3^*} \left(\dot{W}_{propulsive} + \dot{W}_{non-propulsive}\right) dt \leq \eta_{discharge} \Delta E_{batts}\big|_{charged} \\
\Delta E_{batts}\big|_{charged} = \eta_{charge} \int_{t_1^*}^{t_2^*} \left[\dot{W}_{solar} - \left(\dot{W}_{propulsive} + \dot{W}_{non-propulsive}\right)\right] dt
\end{gathered} \tag{21}$$

That substituting with propulsive, non-propulsive and solar power expressions and solving the integral results in:

$$t_{discharge} \leq \frac{\eta_{charge}\eta_{discharge}\eta_{solar}A_{sc}I_{max}\dfrac{2T_{day}}{\pi}sen\left(\dfrac{\pi\dfrac{t_{solar}}{2}}{T_{day}}\right)}{\dfrac{\sqrt{\left(\dfrac{2\mathrm{Mg}}{\rho\,\mathrm{S}}\right)}\dfrac{\left(C_{D0}+K_1C_L^2\right)}{\mathrm{C_L}^{3/2}}}{\eta_{propulsive}}Mg+\dfrac{\dot{W}_{non-propulsive}}{\eta_{BEC}}} \quad (22)$$

$$-\eta_{charge}\eta_{discharge}t_{solar}$$

Finally, combining Eq. 19, Eq. 20 and Eq. 22 and considering $sin(arccos(x))=\left(1-x^2\right)^{1/2}$ $T_{endurance}$ results:

$$T_{endurance}=\left(1-\eta_{charge}\eta_{discharge}\right)t_{solar}+$$

$$\frac{2T_{day}}{\pi}\left[\left(\frac{\eta_{solar\ energy}A_{SC}I_{max}}{\dfrac{\sqrt{\left(\dfrac{2\mathrm{Mg}}{\rho\,\mathrm{S}}\right)}\dfrac{\left(C_{D0}+K_1C_L^2\right)}{\mathrm{C_L}^{3/2}}}{\eta_{propulsive}}Mg+\dfrac{\dot{W}_{non-propulsive}}{\eta_{BEC}}}\right)^2-1\right]^{1/2} \quad (23)$$

5.2. Available design parameters

Two equations have been found that relate the two functional requirements with the following parameters:

- Aerodynamic parameters: C_L, C_{D0}, k_1,
- Mission profile parameters: $\rho(h)$, n , $I(t)$, T_{day}
- State of the art and technological constraints: $\eta_{propulsive}$, $\eta_{solar\ energy}$, η_{BEC}, k_i (i=batts, structure, solar cells, propulsive elements), $\dot{W}_{nonpropulsive}$
- Wing load design parameters: M, S, A_{sc}, n_{batts}

5.3. Parametric analysis and consequences on solar airplanes basic topology

As it is derived from the design equations, the number of DPs is bigger than the number of FRs. According to Axiomatic Design theory, the problem results into a coupled design [21].

The purpose of this paper is to obtain preliminary conclusions on the basic topology of solar aircraft and their potential range of missions. For that reason, the analysis will be focused on the wing load design parameters and their dependencies with FRs. Naturally, the aerodynamic and technological parameters depend on the state of the art (described by the efficiencies, coefficients and mass constants) and their expected tendencies should tend to maximize airplane's efficiency which occurs for a particular value of C_L/C_{D0} and η_i the closest to 1. Finally, the mission profile parameters define the ariplane's operational point and to study the potential range of mission should not be frozen and this level of the design hierachy. To maximize FRs probability of success means in terms of Axiomatic Design to minimize the information content by selecting the design specifications according to the system range (common range = 1). Consequently, FRs acceptance interval should be as big as possible [1], [21].

In order to minimize the dependencies generated in both FRs when achieving $P_{mechanical}>0$ for each instant of the mission and $T_{\mathrm{endurance}}$ ensuring 24 hours cycle, the following tendencies for the geometric DPs can be identified:

- Wing load because of structure and solar cells should diminish, i.e: $Mg/S\downarrow$ and $Mg/A_{sc}\downarrow$ which is a widely known and referenced conclusion that can be extensively completed in Noth, Philipps or Leutenegger et al. studies [5], [8], [3], where the limits for sustained flight are traced.
- Additionally, because of its influence in $P_{mechanical}$ and in the total weight of the airplane (Mg), n_{batts} should be kept to the minimum to ensure night flight with available $P_{mechanical}$.
- As an additional consideration, it is derived that the power consumed by non-propulsive elements should be kept at minimum and the maximum payload (M_{PL}) admitted will be strongly restricted.
- Figure 4 summarizes the aforementioned tendencies.

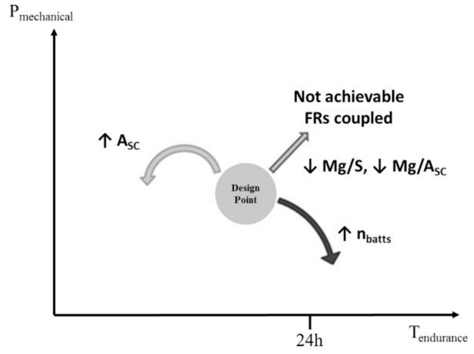

Figure 4. FRs map and DPs influence [24].

Indeed, $\Delta P_{mechanical}/\Delta A_{sc}$ and $\Delta P_{mechanical}/\Delta n_{\mathrm{batt}}$ starts being positive until reaching a maximum where the additional power does not compensate the increase in weight, which turns more critical the coupling with $T_{endurance}$. As a result, the bigger A_{sc} and n_{batt} for power and endurance purposes, the more the total weight of the aircraft arises FRs coupling, which points that technological efforts are conducted to increase specific mass constants and efficiencies for solar cells and batteries. Note that the purpose of this paper is not to find an optimum but the general tendency of DPs that minimize FRs coupling.

5.4. Brief discussion for pedagogical aspects

As stated for the purpose of this work, the objective is to show to students how the concept of independence [21] can be used in order to delimitated which are the design limits imposed by the law of physics to accomplish a particular challenge. In that sense, according to the audience and the objectives of the seminars, the following questions can be formulated to the students:

- Can you find another set of FRs for the first level of hierarchy maintaining the same design challenge?
- In that case, do the input and system constrains change?
- By which strategies can the Information Axiom be included when a lack of quantitative information exists?

- Do the DPs tendencies generate new FRs for the next level of the design hierarchy?
- How do the first level DPs tendencies condition the selection of the operational point of the airplane?
- Which could be the next level of the design hierarchy?
- How innovation could be used to invent or reformulate new DPs or new airplane configurations?

6. Conclusions

The aim of this article was to outline a solar airplane design problem in terms of Axiomatic Design, in order to serve as an example when teaching Suh's principles in their quantitative formulation to students and practitioners. The results obtained in terms of wing load and airplane performances are well known and referenced, so the suitability of this design problem to explicit Axiomatic Design principles.

As a result, the design problem in the first level of the design hierarchy has been formulated and a first approach to solar aviation dependencies completed. Two independent functional requirements were settled to represent a whole variety of missions: excess of specific mechanical power $P_{mechanical}$ to ensure manoeuvring capabilities and endurance during cruise conditions $T_{endurance}$ for ensuring day and night permanent flight.

The physical and technological laws behaving as input and system constraints were written in order to obtain the design equations relating the aforementioned FRs with DPs, resulting in a coupled design with the number of DPs greater than FRs. The obtained DPs were assembled into four groups: aerodynamic, mission profile, state of the art and technological constraints and wing load factors. With the aim of focusing the study on the main dependencies conditioning the basic topology of the aircraft independently from the mission profile, the available design parameters selected were: wing surface, solar cells surface, mass of the airplane (payload included) and number of batteries.

In order to maximize the number of achievable missions, both FRs acceptance intervals were forced to be maximum, and a parametric analysis focusing on main dependencies was completed. FRs coupling was characterized by a strong dependency between the two FRs and the total weight of the aircraft, embodied by the wing surface, the solar cells surface, the payload weight and the number of batteries. As a result, DPs tendencies small wing loads and limited payload minimize FRs coupling, which restricts for manned solar airplanes missions requiring perpetual flight with an excess of mechanical power.

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