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New Preliminary Sizing Methodology for a Commuter Airplane with Hybrid-Electric Distributed Propulsion

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Keywords— Hybrid-electric, dual-energy, aircraft design, preliminary sizing, integrated aircraft, batteries, propulsion systems.

Abstract

Following the latest aggressive emission and external noise reduction targets set by the aviation industry, there now exists an increasing amount of research and emphasis on changing the propulsion and power systems of aircraft. With the aim of expanding upon the current repository of studies focusing on tightly-coupled and integrated hybrid-electric propulsion systems, a new approach to the preliminary sizing of such systems is introduced herein. The purpose of this approach is to establish a methodology for the fully integrated preliminary sizing of hybrid-electric airplanes. The specificity of the established methodology is its ability to cover a variety of implementations on different hybrid-electric airplanes. Furthermore, a sensitivity study is performed to assess the impact of hybridization ratio on airplane max takeoff weight. Finally, in order to assess its accuracy and validity, due to the lack of data on hybrid-electric airplanes the sizing method was applied to the aircraft chosen as baseline for this study, Pilatus PC-12. The obtained results were compared to the existing set of data on this airplane, and the match between the two was found to be satisfactory.

Nomenclature

α	Loss due to flight conditions
η	Battery efficiency
η_b	Battery efficiency
η_{EC}	Energy Conversion efficiency
η_{em}	Electric motor efficiency
η_{gas}	Turboshaft efficiency
η_i	Inverted efficiency

η_{PR}	Propulsion efficiency
η_p	Propeller efficiency
η_{TR}	Transmission efficiency
η_w	Wire transmission efficiency
c_b	Battery specific energy
c_p	Thermal engine specific fuel consumption
g	Acceleration of gravity
L/D	Aircraft glide ratio
P_{elec}	Electric power
P_f	Fuel power
P_{ins}	Installed power
P_p	Propeller power
P_{sup}	Supplied power
P_{use}	Useful power
Soc	State of Charge
W_{batt}	Battery weight
W_E	Empty weight
W_F	Fuel Weight
W_{PL}	Payload weight
W_{TO}	Take-off weight
EIS	Entry into service
MTOW	Max Take-Off Weight
PPS	Propulsion and Power Systems
TOFL	Take-Off Field Length

1. Introduction

Air transport industry has been steadily growing over the years and is projected to maintain a significant growth in the future [1]. This increase in flights is also expected to translate into a considerable environmental impact, both locally (polluting emissions [2], noise [3]) and globally (greenhouse gas emissions [4]). As a result of the rising environmental awareness, there are increasing calls for regulations on emissions caused by the aviation industry. Aircraft manufacturers are expected to drastically reduce fuel consumption and emissions for the next generation aircraft. It is within this context that the AEGIS (Aero EnGine Innovative Studies) research partnership was formed between ISAE-SUPAERO and SAFRAN Group. AEGIS provided the framework for the project presented in this paper: development of a sizing methodology for hybrid-electric airplanes. This will provide a first step towards development of novel design methods for air-

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planes that meet the next generation air transportation vision.

Hybrid-electric propulsion systems are an attractive solution to the issues outlined above, for the simple reason that electrification not only offers the capability to reduce emissions (both in terms of gaseous emissions and noise), but could also unlock the potential for more energy-efficient aircraft. However requirements for high energy density, high re-charging speeds and long battery life cycles are currently the major limiting factor in development of fully electric aircraft for commercial use. For this reason, hybridization of short to medium range airplanes is considered instead as a more feasible alternative for the time being. The methodology presented herein is projected for a 2035 entry into service airplanes, and uses extrapolated information given in the studies published so far, which provide the maturity level of current technologies. In order to develop such innovative aircraft that meets a predefined set of requirements corresponding to a design mission, there needs to be a preliminary sizing phase where an initial set of basic aircraft parameters are derived. This initial step is in line with design steps outlined in aircraft design references ([5], [6], [7], [8] and [9]). The aim of this paper is to propose certain modifications and complements to the classical preliminary sizing methodologies typically applied to standard aircraft systems, and then apply them to concepts whose fully integrated performance needs to be assessed. The method, situated at a low level of design, considers coupling between major constituents within the aircraft such as the airframe, primary energy source(s), power generation, main systems, propulsion, the management of thermal and electrical power. The work presented in the paper is divided into five sections. The first section reviews the classical steps encountered when designing an aircraft and positions the methodology presented herein within those steps. The second section establishes a generic form of hybrid-electric architecture that combines the classical hybrid architectures that are encountered in literature. The derived novel preliminary sizing methodology, along with the application and verification, is presented in the third, fourth and fifth section. The sixth section provides a sensitivity study that assesses the impact of the hybridization ratio on airplane maximum takeoff weight. The final chapter presents conclusions and future perspectives on this topic.

2. Aircraft design methodology overview

The aircraft design process involves several distinct phases; they differ in terminology from one aircraft design book to another, but the overall process remains the same. The most commonly encountered design steps taken from [5], [5] and [5] are summarized in figure 1.

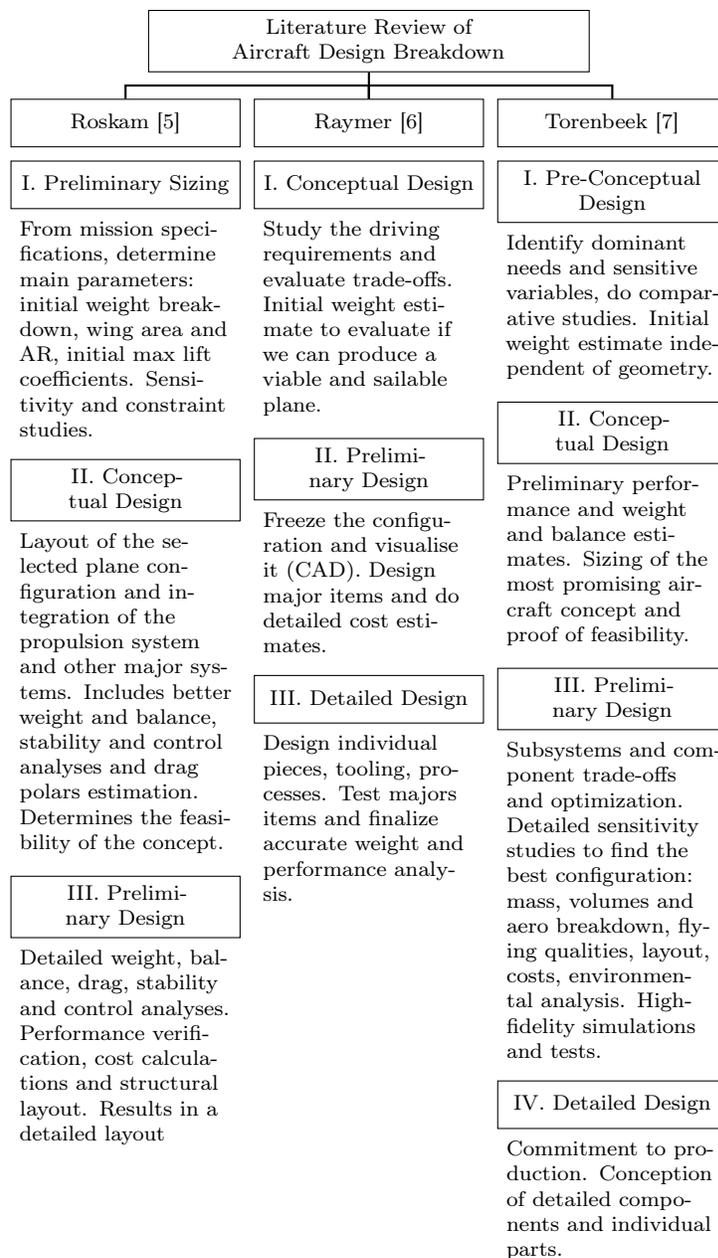


Figure 1: Literary review of aircraft design methodologies

In this work, the design steps are used as presented in [5], notably: Preliminary Sizing, Conceptual Design and Preliminary Design. The first crucial phase, the Preliminary Sizing answers basic questions on the airplane size, weight, and performance, i.e. the macroscopic size and performance parameters. This phase specifies whether an affordable and feasible aircraft can be designed and built to meet the requirements given in the mission and certification specifications. Once this initial step is validated and the results are deemed satisfactory, the Conceptual Design phase can begin. This phase provides

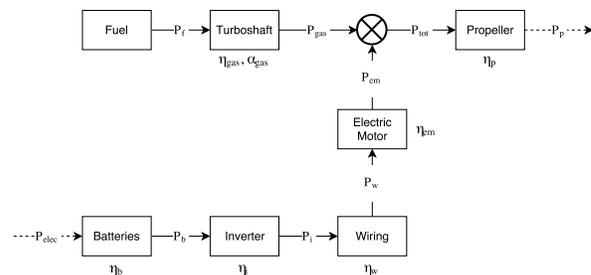
the first iteration on outline and details of the airplane components and systems (avionics, materials, engines...), requirements for occupant comfort (pressurization, galleys, lavatories...), ergonomics, detailed aerodynamics and flight mechanics features (wing size, high lift devices, etc.), are fully specified. The conclusion of this step gives an initial loft where major changes and design reiterations have been finalized, which allows to proceed with further detailed design of the aircraft. The last design phase, the Preliminary Design, also known as the full scale development phase, takes the major parts designed in the previous step and breaks them down into individual components that are to be designed and analyzed separately, after which specifications regarding fabrications, assembly and maintenance are determined. The end of this phase, once the design gets a favorable approval for production launch, is later followed by strict and extensive testing. The outcome of these steps, if all successful, is an aircraft fully ready to operate. The methodology presented in this paper regards the very first phase of aircraft design, the Preliminary Sizing; the remaining steps (Conceptual Design and Preliminary Design) fall out of the scope of the paper.

3. Generalization of hybrid electric architectures

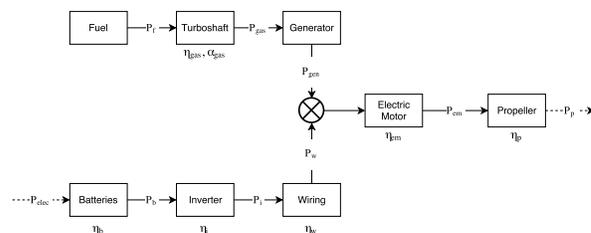
The term hybrid-electric is used for encompassing all electrically based Propulsion and Power Systems (PPS) solutions. An All-Electric PPS relies solely on batteries to provide both propulsive and non-propulsive energy for all modes of aircraft operation. Hybrid-Electric PPS utilizes thermal engines in tandem with batteries. The utilization of batteries during and after in flight phases varies: they can either be replaced during the turn-around, recharged during the turn-around, recharged in-flight via generators coupled to the thermal engine and/or recharged through another form of energy recovery. In the literature ([9], [10], [11], [12] and [13]) hybrid-electric architectures can be classified under two main categories: parallel hybrid systems and series hybrid systems. Additionally, a combination of these two can be considered as the third category.

Parallel hybrid systems have two energy sources linked mechanically to the propellers (figure 2a), which would mean in the current case that both the turboshaft and the electric motor are connected to the propulsive shaft, linked to the propeller. On the other hand, the series hybrid system (figure 2b) has two energy sources linked electrically, with a turboshaft generating electricity through a generator. Both this generated power and the power from the batteries are used for powering the electric motors directly linked to the propellers.

Taking ref. [9] as the starting point for properly defining efficiencies and power parameters related to series and parallel hybrid systems, a generic architecture



(a) Parallel hybrid-electric architecture, adapted from [9]



(b) Series hybrid-electric architecture, adapted from [9]

Figure 2: The two main types of hybrid-electric architectures

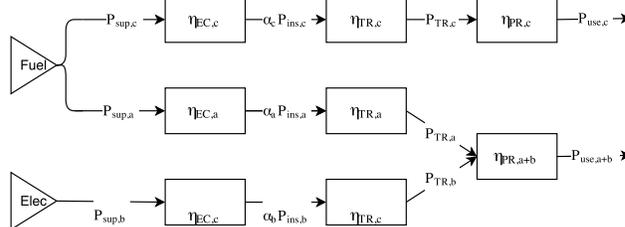


Figure 3: Generic hybrid architecture

is defined, which combines these two solutions into a single hybrid-electric propulsor design space. On top of the hybrid-electric architecture represented by branches (a) and (b), a third fully conventional branch (c) is added to the architecture to encompass the case where the aircraft would additionally use traditional kerosene-powered engines. The schematic overview of this generic configuration is given in figure 3.

Three efficiency factors characterize this generic architecture:

- η_{EC} the energy conversion efficiency,
- η_{TR} the transmission efficiency,
- η_{PR} the propulsion efficiency.

As seen in table 1, these factors take different values depending on the choice of the configuration (series or parallel). The following nomenclature is employed: η_{gas} is defined as the turboshaft efficiency, η_b as battery efficiency, η_i as inverter efficiency, η_w as wire transmission efficiency, η_{em} as electric motor efficiency and finally η_p as propeller efficiency.

Next, the appropriate powers are defined as:

- P_{sup} is the supplied power such that $P_{sup,a} = P_f$ is the fuel power and $P_{sup,b} = P_{elec}$ the electric power,
- P_{ins} is the installed power such that $P_{ins,a} = P_{gas}$ is the turboshaft power and $P_{ins,b} = P_b$ is the battery power,
- P_{use} is the useful power such that $P_{use} = P_p$ is the propeller output power.

The supplied powers can be toggled to represent emergency cases. Following this broad approach helps to maintain the generic nature of the preliminary sizing method by representing a potential engine failure in terms of delivered power percentage. This way, the design space can be properly confined at preliminary sizing without requiring a detailed configuration description.

Table 1: Generic efficiencies

Efficiencies	Parallel	Series
$\eta_{EC,a}$	η_{gas}	η_{gas}
$\eta_{EC,b}$	η_b	η_b
$\eta_{TR,a}$	1	η_{gas}
$\eta_{TR,b}$	$\eta_i \eta_w \eta_{em}$	$\eta_i \eta_w$
η_{PR}	η_p	$\eta_{em} \eta_p$

The coefficient α models the power loss due to flight conditions (altitude and Mach number effects). To further characterize this generic architecture, two hybridization ratios are defined.

$$\left\{ \begin{array}{l} \varepsilon = \frac{P_{ins,b}}{P_{ins,b} + P_{ins,a}} \\ \phi = \frac{P_{ins,a}}{P_{ins,c} + P_{ins,a}} \end{array} \right. \quad (1a)$$

$$\left\{ \begin{array}{l} \varepsilon = \frac{P_{ins,b}}{P_{ins,b} + P_{ins,a}} \\ \phi = \frac{P_{ins,a}}{P_{ins,c} + P_{ins,a}} \end{array} \right. \quad (1b)$$

The hybridization ratio ε compares the installed electrical power to the total installed power provided by the two energy sources, where $\varepsilon = 0$ stands for a fully kerosene based propulsion, and $\varepsilon = 1$ represents the case of a fully electrical aircraft. As for the hybridization ratio ϕ , it serves to compare the installed thermal energy which powers the hybrid branch to the total installed thermal energy in case the traditional kerosene-powered engines are used. Where $\phi = 1$ represents the case where the traditional kerosene-powered engines are not part of the architecture ($P_{sup,c} = 0$). The hybridization ratios can be defined differently for each flight segment depending on the target aircraft architecture and mission.

To size the aircraft, the installed power is deduced from the useful power which is necessary for flight. It is known that:

$$P_{use} = P_{use,a+b} + P_{use,c} \quad (2a)$$

$$P_{use,a+b} = \eta_{PR,a+b}(P_{TR,b} + P_{TR,a}) \quad (2b)$$

Equation (1a) can be rearranged and simplified using figure 3 and equation (2b) so that:

$$\begin{aligned} \varepsilon &= \frac{\frac{P_{TR,b}}{\alpha_b \eta_{TR,b}}}{\frac{P_{TR,b}}{\alpha_b \eta_{TR,b}} + \frac{P_{TR,a}}{\alpha_a \eta_{TR,a}}} \\ &= \frac{\frac{P_{TR,b}}{\alpha_b \eta_{TR,b}}}{\frac{\alpha_a \eta_{TR,a} P_{TR,b} + \alpha_b \eta_{TR,b} P_{TR,a}}{\alpha_b \alpha_a \eta_{TR,b} \eta_{TR,a}}} \\ \varepsilon &= \frac{\alpha_b \alpha_a \eta_{TR,b} \eta_{TR,a} P_{TR,b}}{(\alpha_a \eta_{TR,a} - \alpha_b \eta_{TR,b}) P_{TR,b} + \frac{\alpha_b \eta_{TR,b}}{\eta_{PR,a+b}} P_{use,a+b}} \\ &\times \frac{1}{\alpha_b \eta_{TR,b}} \end{aligned} \quad (3)$$

Furthermore, the required installed power necessary for flight for each energy branch, can be computed, such that:

$$\begin{aligned} P_{ins,a} &= \frac{P_{TR,a}}{\alpha_a \eta_{TR,a}} \\ &= \frac{P_{use,a+b}}{\alpha_a \eta_{TR,a} \eta_{PR,a+b} \left[\frac{\alpha_b \eta_{TR,b} \varepsilon}{\alpha_a \eta_{TR,a} (1-\varepsilon)} + 1 \right]} \end{aligned} \quad (4)$$

and respectively:

$$\begin{aligned} P_{ins,b} &= \frac{P_{TR,b}}{\alpha_b \eta_{TR,b}} \\ &= \frac{P_{use,a+b}}{\alpha_b \eta_{TR,b} \eta_{PR,a+b} \left[\frac{\alpha_a \eta_{TR,a} (1-\varepsilon)}{\alpha_b \eta_{TR,b} \varepsilon} + 1 \right]} \end{aligned} \quad (5)$$

To add the effect of branch (c), from the equation system (2) it can be deduced that:

$$P_{ins,c} = \frac{1}{\alpha_c \eta_{PR,c} \eta_{TR,c}} P_{use,c} \quad (6a)$$

$$P_{ins,a}(1-\phi) = \frac{\phi}{\alpha_c \eta_{PR,c} \eta_{TR,c} P_{use,c}} \quad (6b)$$

By defining H as:

$$H = \frac{1-\phi}{\phi} \frac{\alpha_c \eta_{PR,c} \eta_{TR,c}}{\alpha_a \eta_{TR,a} \eta_{PR,a+b} \left[\frac{\alpha_b \eta_{TR,b} \varepsilon}{\alpha_a \eta_{TR,a} (1-\varepsilon)} + 1 \right]} \quad (7)$$

The useful power of branch (c) as a function of the hybrid useful power is obtained, along with the hybrid useful power as a function of the total power used for flight:

$$P_{use,c} = H P_{use,a+b} \quad (8a)$$

$$P_{use,a+b} = \frac{P_{use}}{1+H} \quad (8b)$$

Equations (4) through (8) can be used for sizing the installed powers from a given required flight power and given hybridization ratios.

The previously derived ratio H cannot be used directly. In fact, as ϕ approaches 0 (which is the case of a fully electric aircraft), H goes to infinity. In order to avoid this singularity, the proposed sizing procedure is as follows:

Find P_{use} the power required for flight,

Calculate $P_{use,a+b}$ using equation (9), which does not result in the same singularity as H alone,

Calculate $P_{use,c}$ using equation (2a),

Calculate $P_{ins,a}$, $P_{ins,b}$ and $P_{ins,c}$ using equations (11), (10) and (6a).

To eliminate singularities, write equation (8b) as:

$$P_{use,a+b} = \frac{1}{1 + \Pi(\varepsilon - 1 + \phi - \varepsilon\phi)} \quad (9)$$

Such as:

$$\Pi = \frac{\alpha_c \eta_{PR,c} \eta_{TR,c}}{\phi \eta_{PR,a+b} (\alpha_a \eta_{TR,a} (\varepsilon - 1) - \varepsilon \alpha_b \eta_{TR,b})} \quad (10)$$

The same can be done for equations (4) and (5):

$$P_{ins,b} = \frac{\varepsilon P_{use,a+b}}{\alpha_a \eta_{PR,a+b} \eta_{TR,a} (1 - \varepsilon) + \alpha_b \eta_{PR,a+b} \eta_{TR,b} \varepsilon} \quad (11)$$

$$P_{ins,a} = \frac{(\varepsilon - 1) P_{use,a+b}}{\alpha_a \eta_{TR,a} (\varepsilon - 1) - \alpha_b \eta_{TR,b} \varepsilon} \quad (12)$$

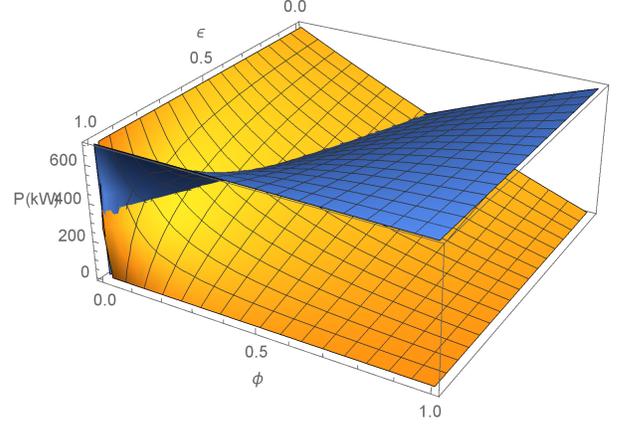
Power evolution as a function of hybridization ratios ε and ϕ can be seen in figure 4. For a given ϕ different than 0, as ε increases, the closer to a fully electric aircraft the architecture is, $P_{ins,b}$ increases, whilst $P_{ins,a}$ decreases, i.e more installed power is required from the electric source than from the thermal source. Inversely when ε decreases more installed thermal power is needed. When branch (c) is taken into account, when conventional thermal energy is part of the architecture, the effect of varying ϕ can be seen, such as when $\phi = 1$ both branch (c) useful and subsequently installed power are null. This figures allows to have a complete overview of both the useful and the installed powers significance and contribution as a function of the two hybridization ratios.

4. Estimating the range of a hybrid aircraft

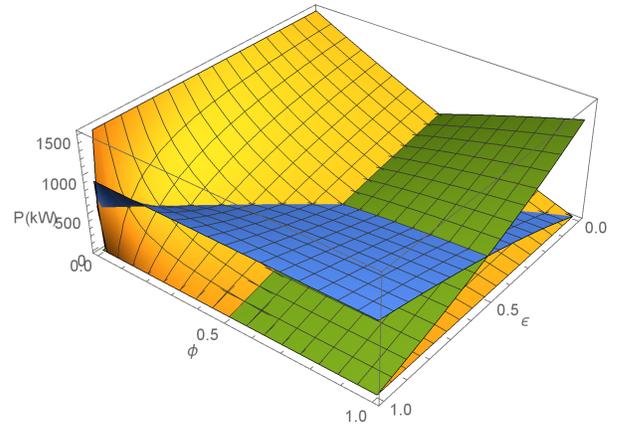
The well-known Breguet range equations [14] for both conventional and fully electric aircraft are taken as the starting point:

$$R_{conv} = \frac{1}{g \times c_p} \frac{L}{D} \ln \frac{W_0}{W_1} \quad (13a)$$

$$R_{elec} = \frac{c_b \eta}{g} \frac{L}{D} \frac{W_{batt}}{W_0} \quad (13b)$$



(a) $P_{use,a+b}$ (blue) and $P_{use,c}$ (orange)



(b) $P_{ins,a}$ (green), $P_{ins,b}$ (blue) and $P_{ins,c}$ (orange)

Figure 4: Evolution of various powers as a function of ϕ and ε

where g is the acceleration of gravity, c_p the thermal engine specific fuel consumption (SFC), $\frac{L}{D}$ the aircraft glide ratio, W_0 the initial mass, W_1 the end mass, c_b the battery specific energy, and η the battery efficiency and W_{batt} the battery mass. Finding a convincing solution for the range of an hybrid-electric aircraft with power and energy hybridization is not simple. Equations (12a) and (12b) cannot merely be linearly added together (a strategy used by some authors [15]) because the equations are necessarily coupled when both sources of energy are used concurrently.

The weight fraction $\ln\left(\frac{W_0}{W_1}\right)$ from equation (13a) is discussed in the following. W_1 is defined as the end-of-cruise weight and W_0 the start-of-cruise weight such that $W_0 = W_1 + W_\Sigma = f_e W_0 + W_{pay} + W_\Sigma$. W_{pay} is the payload weight and W_Σ the total energy weight (fuel and batteries), the empty weight being a fraction f_e of W_0 .

According to equation (13b), the final weight for a fully-electric cruise still equals W_0 since no fuel is con-

sumed. For our hybrid scenario, the ratio of the final weights is weighted by the fuel flows (fossil: \dot{m}_f , electric: \dot{m}_e). Note that the fuel flow for the batteries does not have physical sense, but it will be replaced with more consistent variables.

$$\frac{W_0}{W_1} \Rightarrow \frac{(\dot{m}_f + \dot{m}_e)W_0}{\dot{m}_e W_0 + \dot{m}_f W_1} = \frac{(\dot{m}_f + \dot{m}_e)W_0}{\dot{m}_e W_0 + \dot{m}_f (W_{pay} + f_e W_0)} \quad (14)$$

By definition, the power specific fuel consumption equals the fuel flow divided by the power. Hence, with c_p the kerosene specific fuel consumption, SFC_e the electricity 'specific fuel consumption', P_f the power from fossil fuel and P_e the power from the batteries:

$$\begin{aligned} \frac{W_0}{W_1} &\Rightarrow \frac{(c_p P_f + SFC_e P_e)W_0}{SFC_e P_e W_0 + c_p P_f (W_{pay} + f_e W_0)} \\ &= \frac{(c_p \frac{P_f}{P_{tot}} + SFC_e \frac{P_e}{P_{tot}})W_0}{SFC_e \frac{P_e}{P_{tot}} W_0 + c_p \frac{P_f}{P_{tot}} (W_{pay} + f_e W_0)} \end{aligned} \quad (15)$$

According to the two previously defined hybridization ratios, ε and ϕ , and with $c_b = \frac{1}{SFC_e}$ being the battery specific energy, the power ratios in equation (15) can be simplified such that:

$$\frac{W_0}{W_1} \Rightarrow \frac{(c_p c_b (1 - \varepsilon) + \varepsilon)W_0}{\varepsilon W_0 + c_p c_b (1 - \varepsilon)(W_{pay} + f_e W_0)} \quad (16)$$

Looking at equation (13b), the same is to be done to the fraction $\frac{W_{batt}}{W_0} c_b$. Where W_{batt} is generalized to get W_Σ , since $W_\Sigma = W_{batt}$ if $\varepsilon = 1$. Along with $P_e = \dot{m}_e c_b$ and using the same reasoning as before, it can be found that:

$$\frac{W_\Sigma}{W_0} c_b \Rightarrow \frac{\varepsilon[(1 - f_e)W_0 - W_{pay}]}{\varepsilon W_0 + c_p c_b (1 - \varepsilon)(W_{pay} + f_e W_0)} \quad (17)$$

Replacing those ratios in equations (13a) and (13b) and adding those equations together with the right efficiencies, a range equation for hybrid-electric aircraft can be obtained:

$$\begin{aligned} R = & \frac{\eta_{elec} L}{g D} \left[\frac{\eta_g}{c_p} \ln \left(\frac{(\varepsilon + (1 - \varepsilon)c_b c_p) W_0}{\varepsilon W_0 + (1 - \varepsilon)c_b c_p (W_{pay} + f_e W_0)} \right) \right. \\ & \left. + \frac{\varepsilon c_b ((1 - f_e)W_0 - W_{pay})}{\varepsilon W_0 + (1 - \varepsilon)c_b c_p (W_{pay} + f_e W_0)} \right] \end{aligned} \quad (18)$$

When considering that the gas turbine also splits its power between the generator and a propeller (branch (c) in the previously defined generic architecture), ϕ can be included in the formula by impacting the gas turbine specific fuel consumption c_p such that:

$$c_{p,hybrid} = c_{p,a}\phi + c_{p,c}(1 - \phi) \quad (19)$$

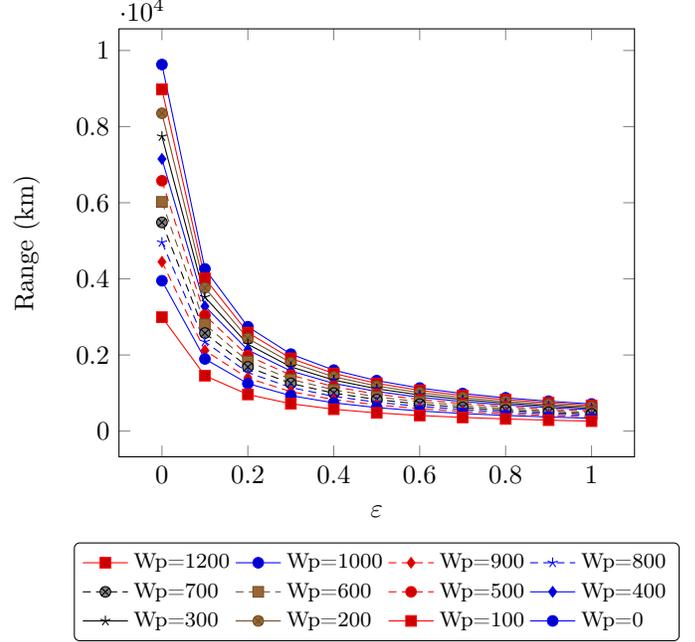


Figure 5: Effect of payload weight on range ($c_b = 300\text{Wh/kg}$)

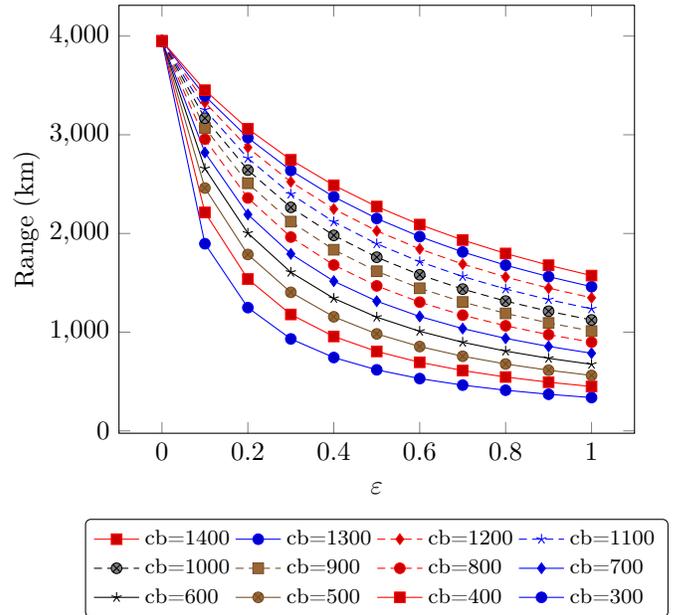


Figure 6: Effect of battery specific energy on range ($W_p = 1000\text{kg}$)

As shown in figures 5 and 6, even slightly increasing ε away from 0 (that is, moving from a conventional architecture to a slightly hybridized architecture) drastically reduces the range due to the much lower specific energy of batteries compared to that of fuel.

5. Investigation

5.1. Requirements and objectives

In order to evaluate the derived methodology, it was decided it would be applied to a small commuter aircraft similar to the Pilatus PC-12. Since batteries tend to limit aircraft to short ranges, hybrid-electric configurations may become competitive in the thin haul market. Table 2 details the top-level requirements for the airplane to be designed:

Table 2: Top-Level Aircraft Requirements

EIS	2035
Range	350NM + 100NM diversion
Capacity	9 PAX
Altitude	21000ft
TOFL	approx. 1000m
Approach speed	<120KTAS
MTOW	4-5000kg

The mission profile consists of a taxi-out, followed by a take-off, climb and cruise, before the descent and taxi-in. Additionally, a 100NM diversion and a 30 minutes contingency hold were assumed. The technology levels shown in table 3 were assumed for the electric powerchain:

Table 3: Technology levels [16]

Tech	2017	2035	2050
Motor Peak Spec. Pwr (kW/kg)	6.6	9.9	20.6
Motor Nominal Spec. Pwr (kW/kg)	4.9	7.4	15.4
Controller Spec. Pwr (kW/kg)	0.08	0.08	0.08
Batt. Spec. Energy <5C (Whr/kg)	200	600	1200
Batt. Spec. Energy >5C, <20C (Whr/kg)	150	450	900
Batt. Spec. Energy >20C, <60C (Whr/kg)	100	300	600
Motor Peak Efficiency	0.95	0.97	0.98
Gearbox Efficiency	0.97	0.98	0.99
Controller Efficiency	0.98	0.99	0.99
Battery Efficiency	0.98	0.98	0.99
Powerchain Efficiency	0.89	0.92	0.95

5.2. General Sizing Process

The developed general methodology is in great part inspired by the work presented in ref. [9]; the sizing process flow chart is presented in figure 7.

The power sizing for the hybrid electric aircraft, for every flight segment, will start from the constraint diagram as shown in figure 7. This diagram allows to represent each segment of flight in terms of power loading (power to weight ratio) as a function of wing loading (weight to wing surface ratio). In the case of hybrid electric aircraft, it is common practice to use the power loading on the y-axis instead of thrust loading, which is traditionally used for jet airplanes. The equations have

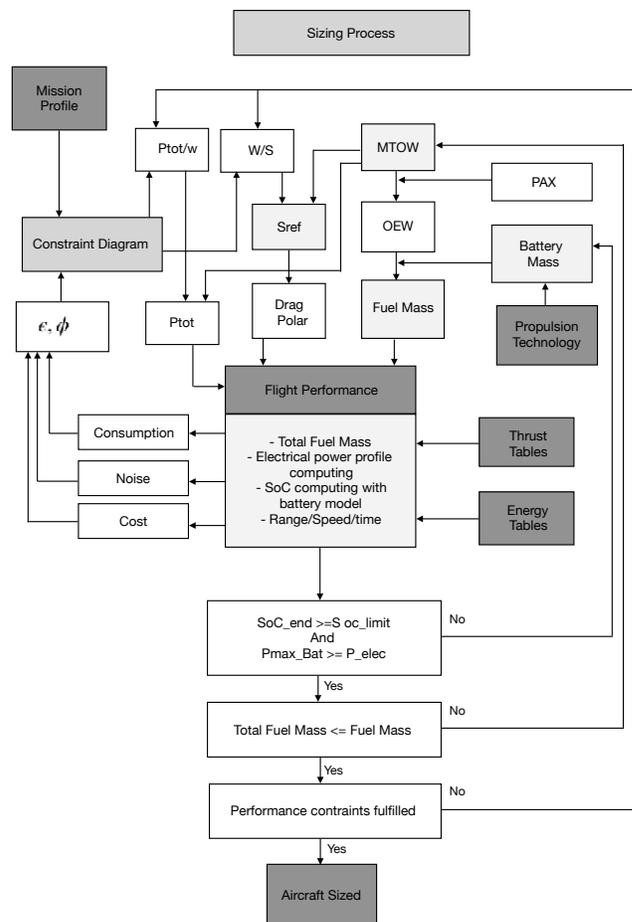


Figure 7: Hybrid aircraft sizing process, adapted from [9].

been deduced from the document of Gudmundsson [8] in order to obtain the minimum power loading and the maximum wing loading and to deduce the optimal sizing in Weight for at the current sizing phase. However, in order to be succinct, only one segment of flight will be developed in this paper, which is the take-off, typically the most energy demanding flight mission segment for a commercial aircraft. The other flight segments can be analyzed in a similar manner.

Assuming that the thrust, the drag and the lift are constant during take-off, the ground roll can be expressed as follows:

$$s_{TOG} = \frac{0.5W_{TO}(V_{LOF} - V_w)^2}{F_{TO} - D_{TO} - \mu(W_{TOG} - L_{TO}) - W_{TOG} \sin \theta} \quad (20)$$

By definition:

$$V_{TO} = 1.2V_{stall} \quad (21)$$

From equation (20) and (21), it can be inferred that:

$$s_{TO} = \frac{0.5 \times 1.44V_{stall}^2 W_0}{g(T - D + \mu(L - W_0))} \quad (22)$$

with:

$$D = 0.5\rho S(0.84V_{stall})^2 \left(C_{D0} + k\left(\frac{C_{Lmax}}{0.71}\right)^2 \right) \quad (23)$$

and:

$$L = 0.5\rho S(0.84V_{stall})^2 \frac{C_{Lmax}}{0.71} \quad (24)$$

And from (22) the power loading needed for the take-off can be deduced:

$$\frac{T_{TO}}{W_0} = \frac{\beta}{\alpha} \left(\frac{0.84^2 q_{stall}}{\beta W_0 / S} \left(C_{D0} - \mu \frac{C_{Lmax}}{0.71} + k \left(\frac{C_{Lmax}}{0.71} \right)^2 \right) \right) + \frac{\beta}{\alpha} \left(\mu + \frac{1.44 V_{stall}^2}{0.5 g_{STO}} \right) \quad (25)$$

with

$$T = T_{sl} \times \alpha$$

being the adjusted thrust to barometric altitude.

For each flight segment, the power loading needed is developed for different flight conditions. The application of the derived equations for all flight segments, yields the diagram presented in figure 8.

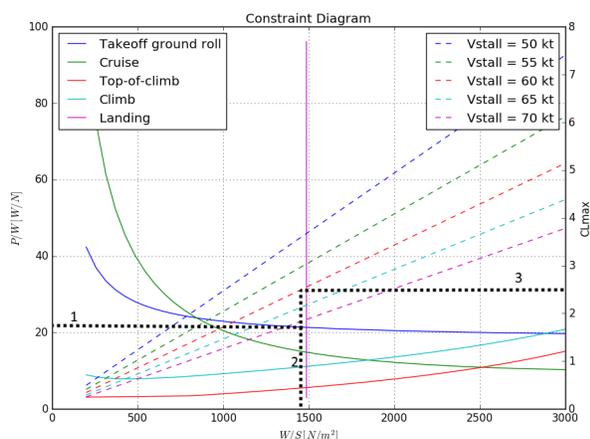


Figure 8: Constraint diagram ($\varepsilon = 0.1$, $\phi = 1$)

5.3. Weight Sizing

For a given mission specification, this section presents a method for estimating:

- Take-off gross weight, W_{TO}
- Empty weight, W_E
- Mission Fuel Weight, W_F
- Mission Battery Weight, W_{Batt}

The process of estimating values for W_{TO} , W_E consists of the following steps:

- Step 1: Determine the mission payload weight W_{PL} ,

- Step 2: Guess an initial value of take-off weight $W_{TO_{Guess}}$ using pre-existing regression lines for aircraft type,
- Step 3: Calculate a tentative value for W_{OE} ,
- Step 4: Determine a guessed mission battery weight $W_{Batt_{Guess}}$,
- Step 5: Calculate the mission fuel weight W_F ,
- Step 6: Deduce the flight performance with the weights estimated previously in addition to the drag polar,
- Step 7: Compute the state of charge (SoC) of the batteries with the specified battery model, and the electrical power profile,
- Step 8: Compare the SoC of the batteries at the end of the mission to the SoC limit given by the batteries specifications, as well as the maximum power delivered by the batteries and the electrical power needed. If the previous conditions are not validated, a reiteration on the battery weight must be performed,
- Step 9: If the previous conditions are validated, compare the total fuel mass computed to the estimated fuel mass. If the later is smaller, a reiteration on the $W_{TO_{Guess}}$ is performed, steps 2 to 9 will be repeated until an agreement is reached and all conditions are satisfied within a pre-selected tolerance.

Mission payload weight: Mission Payload Weight W_{PL} is imposed in the mission specification. This payload weight for commercial airplanes consists of passengers and baggage as well as cargo. Because FAR 23 certified airplanes are frequently operated by owner/pilot it is not unusual to define the crew weight as part of the payload in these cases. For passengers in a commercial airplane an average weight of 175 lbs per person and 30 lbs of baggage is a reasonable assumption for short to medium flights. The crew consists of the cockpit crew and the cabin crew. The number of people in each crew depends on the airplane and its mission and the total number of passengers carried. The minimum number of cabin crew members required is specified by certification rules FAR 91.215, reference 8. For crew members an average weight of 175 lbs plus 30 lbs of baggage is a reasonable assumption.

$$W_{PL} = W_{crew} + W_{Pass} + W_{cargo}$$

Determining the battery system weight: The weight of the battery system is determined from the constraint diagram (once the P_{elec} and E_{elec} are specified) and the battery specifications.

Determining the fuel weight: Rather than using the analytical Bréguet equation to calculate the fuel weight used during the cruise, a numerical method from the results of the constraint diagram is used. Since

the developed hybrid Bréguet equation is very complex and is implicitly defined, an analytical solution cannot be found simply. To counter this obstacle, a direct numerical approach was employed. At each step of the weight sizing loop, the power outputs from the batteries and the thermal engine are derived using the constraint diagram and the MTOW estimation. The cruise range is discretized and the fuel consumption in addition to the battery energy usage at each step of the cruise are computed. This yields the total fuel weight and the required energy from the batteries.

5.4. Verification and application of the methodology

In order to verify the derived methodology, the application on the Pilatus PC-12 requires to set the degree of hybridization ϵ to zero. The comparison between the parameters resulting from applying the methodology and those in the aircraft data sheet was found to be satisfactory. With a total MTOW of 4816kg, compared to 4700kg the results are fairly comparable. The authors would like to precise that one parameter from the original PC-12 mission requirements was modified, since a feasible electrically powered vehicle could not be sized while conforming to realistic energy density levels. In particular, the PC-12 range requirement was reduced from the original data sheet value of 1500NM to 350 NM. A study of the influence this disparity has on the results validity will be performed at a later stage; it is advised to take this into account when evaluating the current comparison.

6. Take-Off Weight Sensitivity Study

Following the development and the verification of the methodology for the preliminary sizing, it is important to conduct sensitivity studies on the parameters which influence the results the most. The sensitivity study helps with finding which parameters are critical to the design and configuration choice. Most importantly in the case of hybrid-electric aircraft design, the sensitivity study offers a clear insight into the technological advancements that must be further pursued, which becomes even more critical in cases where unconventional missions are sought. To carry out this sensitivity study, analytical as well as numerical iterative methods for computing take-off weight sensitivities were used. The analytical derivations are derived from Roskam's [5] methods and adjusted to the hybrid-electric aircraft case. Starting from:

$$W_E = W_{TO} - W_F - W_{PL} - W_{batt} \quad (26)$$

using the regression formula introduced in Roskam's [5] method:

$$\log_{10}(W_E) = [\log_{10}(W_{TO}) - A]/B \quad (27)$$

to eliminate W_E from equation (26) yields:

$$\log_{10} W_{TO} = A + B \log_{10}(W_{TO} - C) \quad (28)$$

with A and B being the regression line constants, and:

$$C = W_F - W_{PL} - W_{batt} \quad (29)$$

To study the sensitivity of W_{TO} to any given parameter X , W_{TO} in equation (28) is partially differentiated. A and B being constants for a given aircraft type, their partial derivatives are both zero, which results in:

$$\frac{1}{W_{TO}} \frac{\partial W_{TO}}{\partial X} = \frac{B}{(W_{TO} - C)} \left(\frac{\partial W_{TO}}{\partial X} - \frac{\partial C}{\partial X} \right) \quad (30)$$

which can be rearranged to solve for $\partial W_{TO}/\partial X$:

$$\frac{\partial W_{TO}}{\partial X} = \left(\frac{B}{1 - B - C/W_{TO}} \right) \frac{\partial C}{\partial X} \quad (31)$$

The derivative $\partial W_{TO}/\partial X$ is called the aircraft growth factor due to X parameter. For example, in order to derive the sensitivity of the W_{TO} to payload weight, then let $X = W_{PL}$. The translation of $\partial W_{TO}/\partial X$ in this case expresses that - assuming mission performance remains the same- for each kilogram of payload added, the W_{TO} will have to be increased by $\partial W_{TO}/\partial X$ kilograms.

As mentioned previously, one of the most crucial parameters that significantly limit the design of hybrid-electric aircraft is the battery specific energy. In fact when studying the sensitivity of the W_{TO} to the battery specific energy, $X = c_b$ for different values of hybridization ratio ϵ , it is very easy to see from figure 9 that a slight increase in the "electrification" of the classic aircraft, yields high W_{TO} , which tends to go to infinity quite fast. The most advanced Li-Ion batteries of today are able to provide around 200Wh/kg [17], which hinders the potential for advanced hybridization. However, battery technology has been improving for many years and specific energy continues to grow at a rate of 5 to 8 percent each year [18]. At this rate, specific battery energy could reach values between 600Wh/kg and an optimistic 1000Wh/kg by the horizon of 2035, which would enable, for the same mission requirements and configuration set today, to reach very high degrees of hybridization.

7. Conclusion

A preliminary sizing methodology for a hybrid-electric aircraft was introduced in this paper. Firstly, a generic hybrid architecture is derived from series and parallel architectures presented in the literature. It contains a hybridized dual-energy (electric and fuel) propulsor associated with a conventional, single-energy (fuel) propulsor. With a judicious choice of efficiencies and the declaration of two non-dimensional variables (hybridization ratio and dual energy/single energy ratio), this description

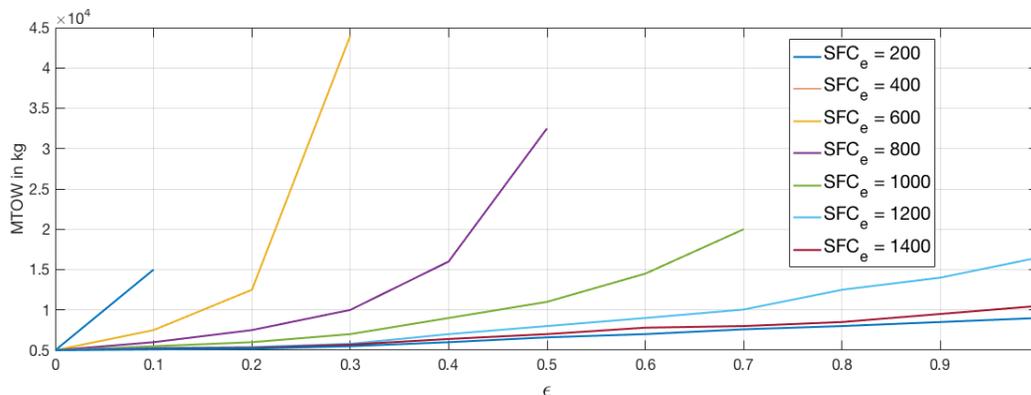


Figure 9: MTOW as a function of ϵ for different battery specific energy SFC_e

provides a generic design space to model multiple potential configurations at a high-level, with particularities of a coupled hybrid-electric propulsion system taken into account. The presented variables can be specified separately for different flight segments of the airplane mission. After establishing the power requirements for each mission flight segment, the power and energy split can then be computed and used to size the propulsion system. Although finally not performed in the course of this work - constraining the design space by representing hypothetical engine failure case in terms of percentage of power loss is also possible; it remains to be incorporated in the future work. A new coupled range equation is also derived, taking into account the combined use of two energy sources and their differing impact on fuel weight consumption during flight. In practice, the presented methodology is intended to be used in the early stages of aircraft design to provide first order estimates of major sizing parameters (aircraft weight, power and energy usage, range) as well as to enable trade studies, independently of any specific configuration the user might choose for the target aircraft. The preliminary sizing step of the hybrid-electric aircraft design process is completed and verified. The first stones are laid for the next step of the process: the conceptual design. The current developed methodology can help to shed some light on the technological shortfalls that need to be further developed and looked into to make such complex aircraft feasible for a 2035 entry into service.

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