Generalised Methodology for Sizing of Air Vehicles with Hybrid-Electric Propulsion

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ABSTRACT

In the context of increasing attention given to aircraft propulsive system electrification by the aeronautical community, Department of Aerodynamics, Energetics and Propulsion at ISAE-SUPAERO is undertaking an effort to develop a preliminary sizing tool for aeroplanes with propulsive architectures ranging between conventional gas turbine and different hybrid-electric solutions. The baseline used to initiate this work is semi-empirical handbook sizing method by Jan Roskam. The method is firstly extended by introducing a generic propulsive power architecture space, described parametrically by an array of power hybridisation parameters. Furthermore, a proposal of modified Breguet range equation is given for fuel weight iterations including batteries. With these upgrades, a trial sizing run was performed on a Pilatus PC-12 test case to verify the functioning of the developed tool. A more comprehensive study of an equivalent aeroplane powered with various hybrid-electric solutions is then presented, along with an associated parametric study. Mission performance results of all the hybrid architectures are inferior to the ones for the fuel-based baseline, with the most promising solutions indicated by the results being series and parallel hybrid architectures. While the tool produces qualitatively coherent results, the quantitative validity thereof is yet to be ascertained. Notably, the hybrid range equation needs to be further placed under scrutiny and its applicability for mission sizing of all the hybrid architectures of interest is to be validated. In the long run, the work will look into other current limitations such as lack of possibility to consider battery recharge in the mission calculations or lack of capability to perform sensitivity studies.

1.0 INTRODUCTION

1.1 Context

Rise in interest of the aeronautical community for aircraft electrification is observed by the day. The phenomenon can be broken down into two major axes: electrification of nonpropulsive systems (e.g. environmental control system (Sinnett, 2007)), or electrification of the propulsive system (NASEM, 2016). The latter is of particular interest for the potential it has for performance gain on the whole aircraft level, e.g. by enabling distributed propulsion concepts (Kim, 2010) which could improve propulsive efficiency relative to the state of the art and thus reduce mission fuel burn and environmental impact. An increasing number of technology demonstrators for passenger-class aeroplanes announced for flight testing (e.g. (Airbus, 2018) and (Sampson, 2018)) speaks for how keen the community is to pursue this trend and to mature the necessary technologies. Several architectural options exist for propulsion electrification, ranging from addition of electrical power to the gas turbine engine shaft, through different hybrid solutions, all the way to fully electrically (battery) powered configurations. (NASEM, 2016) While there are noteworthy challenges proper to electrical systems (e.g. inferior battery power densities compared to kerosene or system-level thermal management issues) that will stand in the way of extensive aeronautical applications for years to come, it is nevertheless of interest to develop tools and methods for sizing and design aircraft with electrified propulsion. With correct sizing methodologies it will be possible to lay groundwork for feasibility assessment of aircraft prone to electrification – from transport aeroplanes with distributed propulsion to drones or drone swarms running on battery power. (Gohardani, 2013) In recent years, Department of Aerodynamics, Energetics and Propulsion (DAEP) at ISAE-SUPAERO has gotten engaged in this domain though activities on distributed propulsor aerodynamic modelling (Lagha 2019) and on hybrid-electric propulsive system integration on the whole aircraft level. The latter activity, notably supported by “AEGIS” research grant to the department by SAFRAN Group, is the subject of this paper.

1.2 Previous Work and Knowledge Gap

In contrast to the concept space of possible (hybrid-)electric propulsive system architectures being well-defined (NASEM, 2016), the concept space or complete air vehicles with hybrid-electric propulsion is scattered and heterogeneous, being comprised of plethora of discrete and unique solutions. ((Bijewitz et al., 2016), (Brelje and Martins, 2019)) This stands as a counterpoint to the practical uniqueness of the traditional aeroplane concept space, since a vast majority of these have historically been configured as a “Tube and Wing” airframe with podded gas turbine propulsors. Historically it has been possible to think of a comprehensive aeroplane preliminary sizing and design method no matter how “dispersed” the concept space, which is not surprising given the virtual uniqueness of traditional aeroplane configuration. The preliminary design philosophy outlined in Roskam (1985) is of particular interest for introducing a distinction between a preliminary sizing, which defines a set of macroscopic dimensional parameters to meet desired aircraft mission requirements, and a subsequent preliminary design, where a selected concept is further elaborated within the scope narrowed by the preliminary sizing.

Several authors have given significant contribution to development of such method, aiming to size and design hybrid-electric propulsive architecture on the whole vehicle level. Accumulated work by Bauhaus Luftfahrt (Seitz et al. 2012) and SAFRAN Tech (Isikveren et al., 2018) presents valuable results on performance of different hybrid-electric powered aircraft. Moreover, new metrics based on energy and power such as Energy-Specific Air Range (ESAR) or Thrust Specific Power Consumption (TSPC) are introduced as means of bridging the gap in performance assessment based on fuel-based engine paradigm and concepts where alternative energy sources (electro-chemical or electrical) and propulsive architectures could be used. (Seitz et al., 2012) A relevant application of these methods and figures of merit for development of a new preliminary sizing method was undertaken in Pornet (2018), for a narrow-body aeroplane with hybrid-electric propulsion application, which inspired the current work to an important extent.

These works demonstrate that a preliminary sizing method capable of encompassing aircraft with both fuel- and electric-driven propulsive systems is necessary for providing a comprehensive methodological starting point for hybrid-electric air vehicles design space exploration and further disciplinary studies on aero-propulsive physics. For this reason, DAEP has initiated an in-house development of one such method, of which a very first version was presented in Elmousadik et al. (2018).

1.3 Objectives

The objective of the work presented in this paper is to set up a preliminary sizing methodology that would allow an efficient preliminary exploration of hybrid-electric-powered air vehicle design space. In the first place, a general description of the design space is set forth as the main goal. The user shall be able to choose an architecture of interest and to evaluate its performance for given top level requirements and mission profile. Secondly, an application case shall be provided, in order to provide a first insight into the feasibility of the developed process and preliminary assessment of the methodology.

2.0 METHODOLOGY OUTLINE

The methodology has its starting point in definition of a generic concept space for propulsive power architectures. Once such a description is provided, the next step is to propose a range equation which would take into account possibility of embarking electric energy source on board. With power and range estimation methods in hand, a mass-performance loop (Roskam, 1985) is set up for a typical mission profile, to yield constraint diagram and quantitative results for architectures of interest.

2.1 Generic Power Architecture

This first step followed a need to find adequate means for analytical description of propulsive architecture design space ranging between conventional gas turbine propulsion, across various hybrid-electric solutions and all the way to all electric propulsive solutions. (NASEM, 2016) Inspired by schematic/analytical solutions for series- and parallel-hybrid architectures previously presented in Pornet (2018), a first schematic description of generic power architecture was made. (Fig. 2.1-1) With two sources of power (fuel and batteries), across three power branches (*a*, *b*, and *c*) that can be intertwined to yield hybrid solutions, the architecture presents the complete array of power (P_i) transformations, efficiencies (η_i) and power lapses (β_i) to be taken into account when estimating the useful power. The fuel branch *c* covers conventional production of motive power by means of a gas turbine engine; branch *a* covers production of electrical motive power by means of a turboshaft engine; finally, branch *b* covers purely electrical motive power provided by batteries. Factors ξ_i on the *b* branch are used for defining power split if the battery power is split among branches. The reader should note that battery recharge possibility (either by generators or propeller windmill) is indicated by the dashed red line, but no implementation of this capability has been made at this point. The choice of elementary power parameters (supplied, installed and transmitted) and equivalent efficiencies to take into account is conforming to guidelines given by (AIAA, 2019), as previously defined in works such as (Seitz et al., 2012) or (Pornet, 2018). (Eqn. 1)

$$\eta_{EC} = \frac{P_{ins}}{P_{sup}};\ \eta_{TR} = \frac{P_{TR}}{P_{ins}};\ \eta_{PR} = \frac{P_{use}}{P_{TR}} \tag{1}$$

In order to enable parametric description of this architectural space, three power hybridisation factors are proposed, inspired by degree of hybridisation for power parameter, as defined in Isikveren et al. (2014), representing ratios between installed power on one of the respective branches and the total installed power. (Eqn. 2)

$$\boldsymbol{\varepsilon} = \frac{\boldsymbol{P_{ins,b}}}{\boldsymbol{P_{ins,tot}}};\ \boldsymbol{\phi} = \frac{\boldsymbol{P_{ins,a}}}{\boldsymbol{P_{ins,tot}}};\ \boldsymbol{\theta} = \frac{\boldsymbol{P_{ins,c}}}{\boldsymbol{P_{ins,tot}}} \tag{2}$$

$$(\text{with } \boldsymbol{P_{ins,tot}} = \boldsymbol{P_{ins,a}} + \boldsymbol{P_{ins,b}} + \boldsymbol{P_{ins,c}})$$

The power hybridisation factors are chosen with objective to allow consistent algebraic modelling of relations between installed and useful powers along with desired power division between electrical and chemical energy sources. They adhere to the following rule: algebraic sum of the chosen power hybridisation factors must be equal to 1, with all of the three factor values having [0,1] range. This provides necessary conditions for coherent description of all the desired architectural possibilities (Table 2.1-1).

Table 2.1-1: Parametric description of propulsive architecture space.

	ε	ϕ	θ	power branch
Conventional	0	0	1	c
Turboelectric	0	1	0	a
All Electric	1	0	0	b
Series Hybrid	$0 < \varepsilon < 1$	$0 < \phi < 1$	0	a, b
Parallel Hybrid	$0 < \varepsilon < 1$	0	$0 < \theta < 1$	b, c
Part. Turboelectric	0	$0 < \phi < 1$	$0 < \theta < 1$	a, c
S/P Part. Hybrid	$0 < \varepsilon < 1$	$0 < \phi < 1$	$0 < \theta < 1$	a, b, c

The schematic relationship can easily be translated into algebraic relationship between total useful power and the power supplied by the sources, across all three branches. Under conditions implied by choice of [ε, ϕ, θ] the equation will be reduced to parts that cover individual architectures. (Eqns. 3-6)

$$P_{use} = P_{use,\,a+b} + P_{use,\,b} + P_{use,\,c+b} \tag{3}$$

$$P_{use,a+b} = \eta_{PR,\,a+b} * (\beta_a P_{ins,a} \eta_{TR,a} + \xi_{ba} \beta_b P_{ins,b} \eta_{TR,ba}) \tag{4}$$

$$P_{use,b} = \eta_{PR,\,b} \xi_{bb} \beta_b P_{ins,b} \eta_{TR,bb} \tag{5}$$

$$P_{use,c+b} = \eta_{PR,\,c+b} * (\beta_c P_{ins,c} \eta_{TR,c} + \xi_{bc} \beta_b P_{ins,b} \eta_{TR,bc}) \tag{6}$$

With all the data (P_i, η_i, β_i, ξ_i) combined into one common hybridisation parameter H, it is possible to write expressions for installed power necessary on each branch, for provided necessary useful power and hybridisation ratios. (Eqns. 7-9)

$$P_{ins,a} = \frac{\phi}{H(\varepsilon, \phi, \theta)} * P_{use} \tag{7}$$

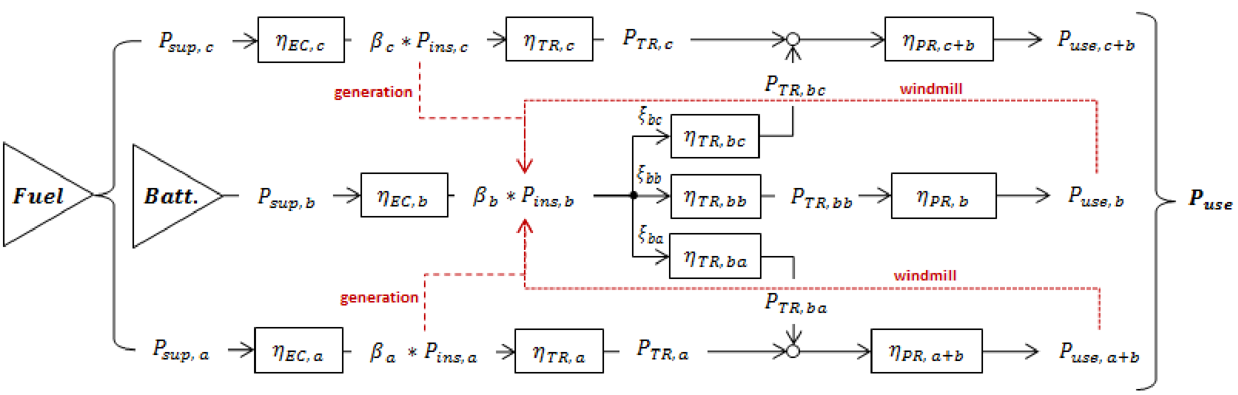

Figure 2.1-1: Generic hybrid-electric propulsive system power architecture.

$$P_{ins,b} = \frac{\varepsilon}{H(\varepsilon,\phi,\theta)} * P_{use} \tag{8}$$

$$P_{ins,c} = \frac{\theta}{H(\varepsilon,\phi,\theta)} * P_{use} \tag{9}$$

With:

$$H = \eta_{PR,\,a+b} * \left(\phi\beta_a\eta_{TR,a} + \eta_{TR,\,ba}\xi_{ba}\varepsilon\beta_b\right) + \eta_{PRb}\,\eta_{TR,\,bb}\xi_{bb}\varepsilon\beta_b + \eta_{PR,\,c+b} * (\theta\beta_c\eta_{TR,c} + \eta_{TR,\,bc}\xi_{bc}\varepsilon\beta_b) \tag{10}$$

This generic power relationship which takes into account elementary power transformations and losses associated to processes and altitude effects is integrated into the mass-performance sizing loop which will be presented in section 2.3.

While the ambition of such representation is to build a bridge between analytical quantitative analysis and discrete architectural solution space, it is clear that quantitative nature of the equation is empirical, i.e. the model is highly data-dependent for all that concerns parameters such as e.g. component efficiencies, weights or battery power densities.

2.2 Range Equation

Constant battery mass as it discharges energy throughout mission will bear repercussions on range and fuel weight estimation capabilities for hybrid-electric vehicles. To remedy this, a first attempt was made to extend the conventional (i.e. for aeroplanes with fuel-based propulsion) Breguet range equation (Eqn. 11).

$$R_{conv} = \frac{1}{g * PSFC} * \frac{L}{D} * ln\left(\frac{W_0}{W_1}\right) \tag{11}$$

Where *PSFC* is the power-specific fuel consumption, and W_0 and W_1 are weights at start and end of cruise, respectively. Using the same rationale as for derivation of the conventional range equation (i.e. cruise conditions under hypothesis of constant lift-to-drag ratio and flight speed) an equivalent equation can be derived for a battery-powered vehicle (Eqn. 12).

$$R_{elec} = \frac{c_b}{g} * \eta * \frac{L}{D} * \frac{W_{bat}}{W} \tag{12}$$

Where c_b is the battery specific energy, η is the overall efficiency of the propulsion system, and W_{bat} and W are battery weight and aircraft weight respectively. A hybrid energy system equation development initiated through the W_0/W_1 term accounting. The idea is to see how much weight can be lost to the battery presence, and then to add that effect at a later stage in order to obtain a complete weight accounting capability. It is done by weighing the weight fraction by respective battery and fuel energy flows (Eqn. 13), which enables a simple distribution of power between purely fuel-produced power ($\dot{m}_e$ = 0) and purely battery-produced power ($\dot{m}_f$= 0).

$$\frac{W_0}{W_1} \Rightarrow \frac{\left(\dot{m}_e + \dot{m}_f\right) * W_0}{\dot{m}_e * W_0 + \dot{m}_f * W_1} \tag{13}$$

In this expression, $\dot{m}_f$ is the fuel mass flow and $\dot{m}_e$ is an equivalent fictitious parameter (Eqn. 14) which deals with electric energy flow.

$$\dot{m}_f = PSFC * P_{use,f} \; ; \; \dot{m}_e = \frac{P_{use,e}}{c_b} \tag{14}$$

Replacing these back into Eqn. 13, and expressing respective useful powers along the fuel-based and battery-based branches with the relations given by Eqns. 4 to 10, the following expression is obtained for the conventional range:

$$R_{conv} = \frac{1}{g}\left(\frac{L}{D}\right) ln\left[\frac{\left(X_f + X_e\right)W_0}{W_0 X_e + X_f W_1}\right] \tag{15}$$

With X_f and X_e (Eqn. 16) being grouped efficiency, power hybridisation, specific energy and power lapse terms.

$$X_f = \frac{1}{H} PSFC\left[\theta\beta_c \eta_{TR_c} \eta_{PR_{c+b}} + \phi\beta_a \eta_{TR_a} \eta_{PR_{a+b}}\right]$$

$$X_e = \frac{1}{c_b}\frac{\varepsilon}{H}\beta_b\left[\xi_{ba}\eta_{TR_{ba}}\eta_{PR_{a+b}} + \xi_{bb}\eta_{TR_{bb}}\eta_{PR_b} + \xi_{bc}\eta_{TR_{bc}}\eta_{PR_{c+b}}\right] \tag{16}$$

With Eqn. 15 in hand, the user has a possibility to account for conventional range “lost” to the embarked batteries and their constant mass. In turn, to account for how much range can be gained by the power provided by the batteries, the same development is undertaken, this time starting with Eqn. 12, which yields an expression for the range provided by the batteries (Eqn. 17).

$$R_{elec} = \frac{1}{g}\eta\frac{c_b X_e (W_0 - W_1)}{W_0 X_e + X_f W_1} \tag{17}$$

Adding Eqn. 17 to Eqn. 15 yields the final range equation (Eqn. 18).

$$R_{hyb} = \frac{1}{g}\left(\frac{L}{D}\right)\left[\frac{ln\left(\frac{\left(X_f + X_e\right)W_0}{W_0 X_e + X_f W_1}\right)}{PSFC} + \eta\frac{c_b X_e (W_0 - W_1)}{W_0 X_e + X_f W_1}\right] \tag{18}$$

While a hypothesis of linear addition of two contributions is rather simplistic and probably not representative of performance to be expected in a real-life application, precedence was given to constructing a first working version of the sizing tool, so this version of the equation was retained until further refinement.

2.3 Workflow

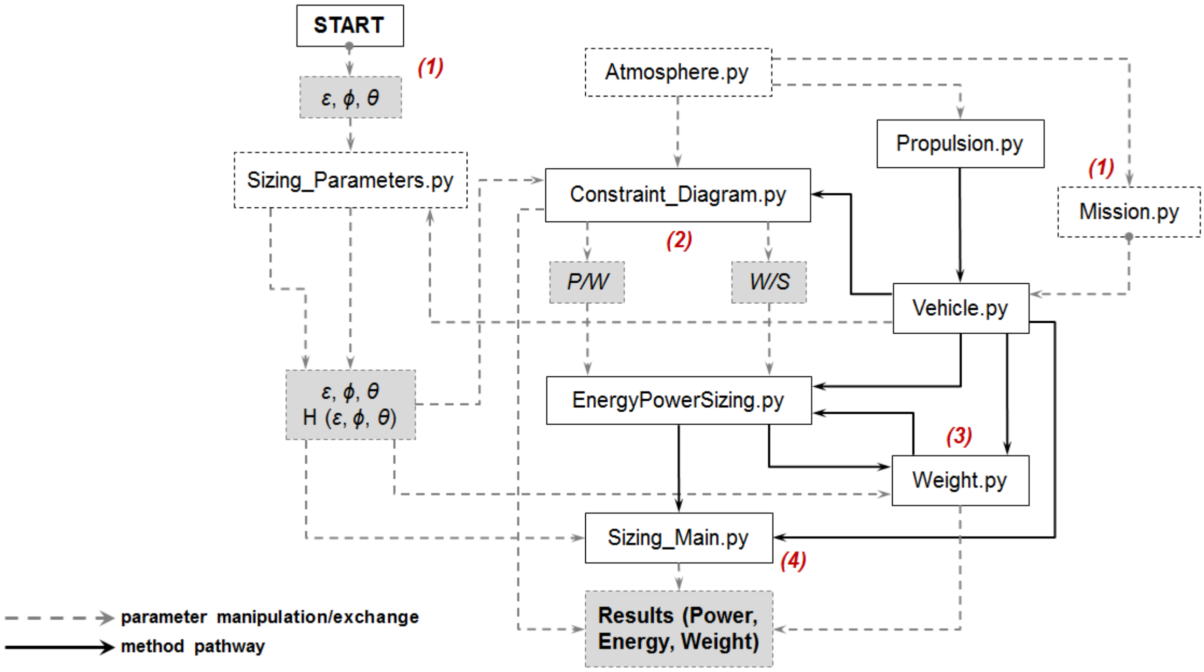

Figure 2.3-1: Sizing code structure and workflow.

The method is structured as a standard mission mass-performance loop calculation; the complete workflow with associated process enumeration, illustrated with the sizing tool constituent modules, is presented in Figure 2.3-1. It is initiated by introduction of top level requirements and mission parameters, along with the propulsive architecture to be studied by selection of appropriate values of the power hybridisation ratios (1). This information is then transferred to the constraint diagram calculation where the matching chart us set up, which plots power over weight ratio as a function of the wing loading (2). Having H as input value, it is possible to determine the necessary power for flight for each flight phase and in turn the installed power for all three power branches (a,b,c). In the design space now bound by the matching chart, an optimum is sought by looking for the point of minimum power over weight ratio and maximal wing loading. The parameters from the constraint diagram are power loading, wing loading, V_{stall} and C_{Lmax}. At the time of the writing of the paper, the equations developed for constraint diagram cover the basic mission flight phases: takeoff, climb, cruise and landing. Power loss cases are also implemented in the matching chart process in order to allow taking into account potential propeller failure cases when constraining a distributed propulsion (i.e. multi-propeller) configuration sizing space, but it represents nothing more than a simple percentage of lost power.

The optimal power to weight ratio values of the takeoff, climb and cruise segment are then given as input to the weight estimation (3), together with previously provided data on architecture and mission requirements. Starting from an initial guess value of the maximum takeoff weight, a weight sizing loop calculates the empty weight from the takeoff weight guess, the payload, the fuel weight and the battery weight that had been determined a step further ahead. After completing the weight iteration process main characteristic weight values are identified. In a final step, the iterated weight is used to calculate the absolute power and energy values and to plot the results. (4)

Table 3.1-2: Verification study results.

	Unit	Pilatus PC-12	Simulated A/C	Difference to ref.
Aircraft				
Wing loading	$[N/m^2]$	1801	1571	- 12.7%
Max. Power to weight	$[kW/kg]$	0.189	0.190	+ 0.5%
Weights				
Max. Takeoff Weight	$[kg]$	4740	4854	+ 2%
Empty Weight	$[kg]$	2891	3054	+ 5%
Fuel Weight	$[kg]$	744	880	+15.5%
Payload Weight	$[kg]$	920	920	0%
Gas Turbine Weight	$[kg]$	205	238	+ 14 %
Power				
Max. Shaft Power	$[kW]$	895	924	+ 3%

3.0 CASE STUDY

3.1 Verification Study

For a first verification of the methodology, a preliminary sizing study was carried out on a case that corresponds to a Pilatus PC-12 aeroplane. While this choice does not take into account any electrification effects, it was meant to serve as the first demonstration of the tool operability. Top level requirements for the test case are summarised in Table 3.1-1, and the results compared to the reference in Table 3.1-2. The reference data were either retrieved from the publicly available manufacturer data sheets (Pilatus, 2013) or from aircraft preliminary sizing textbooks like Roskam (1985).

The comparison of the maximum takeoff weight and the maximum shaft power values show a satisfying discrepancy of about 2% and 3% respectively. Although the power and weight values are matching, results for wing loading, fuel weight and gas turbine weight are showing non-negligible difference. A major uncertainty remains, which might contribute importantly to the error, regarding the empirical values used for component efficiencies and weight models e.g. scaling engine weight with power, which need further refinement. Nevertheless, the developed tool was operational and capable of producing results for provided input, so an example study was undertaken to assess its capability to work with the complete developed power and range equations.

3.2 Hybrid-Electric Aeroplane Sizing

The study was extended on a hypothetical PC-12 equivalent aeroplane, propelled by one of the six hybrid-electric architectures as outlined in NACEM (2016), each defined by hybridisation ratio array $[\varepsilon, \phi, \theta]$ conforming to the definitions provided in section 2.1. A summary of the results is provided in Table 3.3-1. Starting with the maximum takeoff weight, it can be seen that except of the all-electric case the weight increase of the remaining architectures is within 30 % for an optimistic battery specific energy of 800 Wh/kg and a maximum battery usage of 20 % of the total power supply. Particularly the turboelectric and partial turboelectric architecture seem to have the same takeoff weight, which is due to the fact that the same internal combustion engine simulation model has been used both for turboshaft engine and for the traditional aircraft gas

Table 3.1-1: PC-12 top level requirements.

Design range	550 nm
Capacity	9 PAX + 1 Pilot
Cruise speed and altitude	270 KTAS, 21000 ft
Takeoff field length (MTOW, SL, ISA)	≤ 1000 m
Approach speed	< 120 KTAS
Landing field length (MLW, SL, ISA)	≤ 1000 m

turbine. The table also lists further weight information of mechanical and electrical transmission systems such as generator or gearbox which should be taken into account when analysing the takeoff weight. The reader is reminded that the results are presented as a result of the demonstration of the functionality of the developed tool; the methods employed will be placed under further scrutiny in upcoming work.

3.3 Parametric Sizing Space Exploration

A preliminary parametric study was then performed on a series hybrid case, where the hybridisation factor ε influence on range was observed for different battery specific energies. The graphs in Figure 3.3-1 illustrate the evolution of payload-range diagram for three different values for battery specific energy (300 Wh/kg, 700 Wh/kg, 1000 Wh/kg). Note that for all settings the take-off weight was kept constant for

Table 3.3-1: Summary of preliminary study results for various architecture sizing cases.

	Unit	Series	Parallel	Series/ Parallel	All Electric	Turbo-elec.	Partial Turboelec.
Aircraft							
Parameter Setting $[\varepsilon, \phi, \theta]$	$[-]$	[0.2,0.8,0]	[0.2,0,0.8]	[0.1, 0.3, 0.6]	[1,0,0]	[0,1,0]	[0,0.4,0.6]
Wing loading	$[N/m^2]$	1514	1571	1571	1343	1571	1571
T/O Power to weight	$[kW/kg]$	0.199	0.193	0.198	0.168	0.206	0.198
Range	$[Nmi]$	550	550	550	550	550	550
Weights							
Max. Take-off Weight	$[kg]$	6529	6300	5520	56713	4860	4860
Empty Weight	$[kg]$	4047	3912	3450	31561	3059	3059
Fuel Weight	$[kg]$	900	875	878	0	881	881
Payload Weight	$[kg]$	920	920	920	920	920	920
Gas Turbine Weight	$[kg]$	325	305	274	0	259	249
Battery Weight	$[kg]$	662	593	272	24233	/	/
E-motor Weight	$[kg]$	181	28	63	1129	119	46
Generator Weight	$[kg]$	182	0	72	0	195	75
Power (Cruise)							
Power from Turboshaft	$[kW]$	1059	0	326	0	1037	398
Power from Batteries	$[kW]$	265	237	109	9693	0	0
Power from Turboprop	$[kW]$	0	948	652	0	0	597
Total Cruise Power	$[kW]$	1324	1185	1087	9693	1037	995
Technology							
Battery Specific Energy	$[Wh/kg]$	800	800	800	1300	/	/
E-motor Specific Power	$[kW/kg]$	5	5	5	5	5	5
Generator Specific Power	$[kW/kg]$	5	5	5	5	5	5

simplification reasons, and that the fuel volume constraint has not been taken into account at this time. The results indicate that increasing values of battery specific energy improves the potential to achieve long ranges compared to the current state of the art of battery technology. While the observation is trivial, it is reassuring with respect to the verification of the method operation and its robustness for a broader range of conditions. For a more precise insight, further studies on different architectures will be conducted.

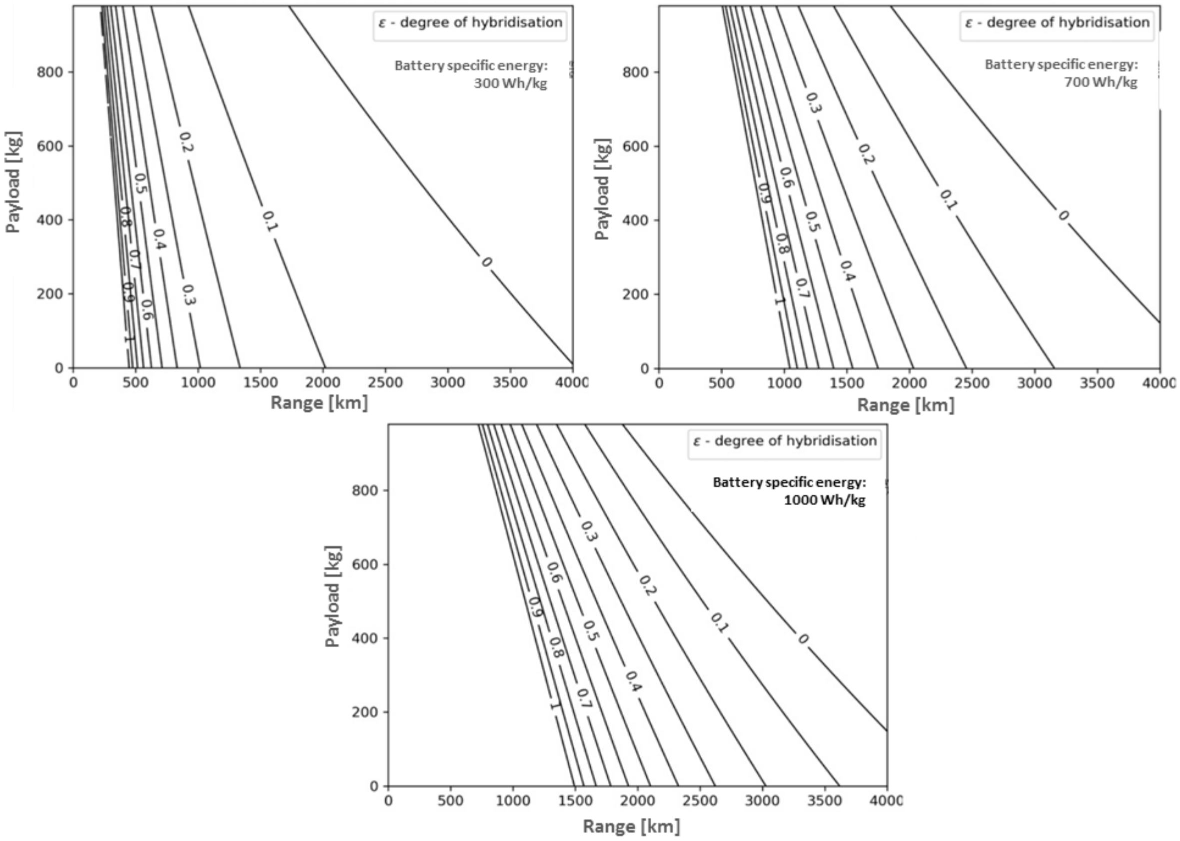

Figure 3.3-1: Payload-Range analysis for a hypothetical PC-12 equivalent with series hybrid propulsion, for three values of battery specific energy.

4.0 FUTURE WORK AND PERSPECTIVES

While the first consolidated methodology has been laid out, and with first results produced, it represents but the groundwork; significant improvement and subsequent validation yet remain to be carried out:

- Due to numerical problems with the new equation that could not have been solved in time for this paper, the presented range equation is only used directly in the parametric study part of the tool while the range calculation in the primary sizing part still relies on the traditional fuel fraction method; correcting for this drawback is the first priority for the further work.

- Regardless of the mentioned numerical problem, it is of equal importance to assess the validity assessment and potentially even give a different definition of the range equation, since the traditional Breguet formulation is quite difficult to maintain coherently for this wide array of architectural possibilities. Exergy-based range equation proposals exist in literature, which could be of interest to exploit in a tool which unifies a technologically heterogeneous design space using energy/power as the common denominator.

- Including recharging capability for the batteries, in order to enable more versatile mission profiles.
- Including nonpropulsive power off-takes.
- Developing sensitivity analysis capabilities for the tool.
- In the long run, developing optimisation capabilities for the tool, as for the time being the architectural choice is uniquely a matter of user input.

It is worth mentioning that the tool is also conceived for tackling Boundary Layer Ingestion propulsion related problems, which was judged to be beyond the scope of this paper. Further work equally includes consolidation and upgrade of these aspects.

5.0 CONCLUSION

The paper presents a first attempt at creating a preliminary sizing tool for aeroplanes with propulsive architectures ranging between typical gas turbines and different (hybrid) electric options, based on the classic Roskam semi-empirical mass-performance loop method. To this purpose, a generic power architecture is outlined, which in turn yields generic equation for determination of installed power needed on the different power branches. The architecture is analytically described by an array of three power hybridisation ratios, one for each propulsive power production scenario: mechanical (gas turbine engine), electrical (battery driven propellers and electrical to mechanical (electrical power produced by a turboshaft). A simple composite range equation is then given, in order to enable taking into account electrical energy sources (batteries) along with fuel-based power. These corrections are introduced into conventional matching chart method directly for the parametric studies, and for the time being indirectly for the basic sizing study, through separate calculations of weight fractions to be used in the traditional Breguet equation.

Preliminary verification case shows decent match of obtained results to the reference Pilatus PC-12 data, except for the fuel weight and wing loading; this discrepancy is strongly to be due to the nature of the developed range equation. Results for different hybrid-electric architectures, the same mission and same payload and range indicate the most performing architectures to be the parallel hybrid and series hybrid, while all electric architecture underperforms by far, even with increased battery specific energy. First preliminary studies on series hybrid architecture payload-range relationship show an expected degrading effect of hybridisation relative to all-fuel alternative, which can be somewhat offset with higher battery specific energy. While it yet remains to significantly improve the methodology, as well as the parameter databases behind its semi-empirical models, its potential to enable quick and efficient design space exploration at preliminary sizing level is already tangible.

Concerning potential military significance of the presented work, its flexibility with respect to mission profiles makes it suitable for civilian and military aircraft applications alike. However, any such divergence at this time would also require tempering with the equations, as well as significant performance database upgrade. For example at this level of the tool maturity, surveillance-type drones are feasible as target applications; for more different applications and mission than that, e.g. weapon-carrying vehicles, significant upgrades will be needed.

6.0 ACKNOWLEDGMENTS

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7.0 REFERENCES

Airbus (2018), "The future is electric", available at: https://www.airbus.com/newsroom/news/en/2018/07/the-future-is-electric.html (accessed on 1 July 2019)

Aircraft Electric Propulsion and Power (AEPP) Working Group (2019) Guidelines for Analysis of Hybrid Electric Aircraft System Studies: Nomenclature, Pictographic Representations, Standalone and Combined Properties and Attributes, Metrics, and Figures of Merit, AIAA, available at: https://www.aiaa.org/docs/default-source/uploadedfiles/publications/standards/hybrid-electric_properties_attributes.pdf?sfvrsn=c8eb8f11_0

Bijewitz, J., Seitz, A., Hornung, M. (2016),"A Review of Recent Aircraft Concepts Employing Synergistic Propulsion-Airframe Integration", paper presented at the Congress of the International Council of the Aeronautical Sciences (ICAS), 26-29 September, Daejeon, Korea.

Brelje, B. J., Martins, J. R. R. A. (2019), Electric, Hybrid, and Turboelectric Fixed-Wing Aircraft: A Review of Concepts, Models, and Design Approaches, Progress in Aerospace Sciences, 104, 1-19, doi:10.1016/j.paerosci.2018.06.004

C. Pornet (2018), Conceptual Design Methods for Sizing and Performance of Hybrid-Electric Transport Aircraft, PhD dissertation, Technische Universität München – Fakultät für Maschinenwesen, Munich, Germany, available at: https://mediatum.ub.tum.de/1399547

Elmousadik, S., Ridard, V., Sécrieru, N., Joksimovic, A., Maury, C., Carbonneau, X. (2018), "New Preliminary Sizing Methodology for a Commuter Airplane with Hybrid-Electric Distributed Propulsion", paper presented at Advanced Aircraft Efficiency in a Global Air Transport System (AEGATS) Conference, 23-25 October 2018, Toulouse, France.

Gohardani, A.S. (2013), A synergistic glance at the prospects of distributed propulsion technology and the electric aircraft concept for future unmanned air vehicles and commercial/military aviation, Progress in Aerospace Sciences 57, 25-70, https://doi.org/10.1016/j.paerosci.2012.08.001

Isikveren, A.T., Kaiser, S., Pornet, C., Vratny, P.C. (2014), Pre-design Strategies and Sizing Techniques for Dual-Energy Aircraft, Aircraft engineering and aerospace technology, 86(6), 525-542, doi:10.1108/AEAT-08-2014-0122

Isikveren, A.T., Fefermann, Y., Maury, C., Level, C., Zarati, K., (…) Thoraval, B. (2018), Pre-design of a Commuter Transport Utilising Voltaic-Joule/Brayton Motive Power Systems, Aeronautical Journal -New Series- 122(1248), 205-237. DOI: 10.1017/aer.2017.126

Kim, D.H. (2010), "Distributed Propulsion Vehicles", paper presented at the 27th Congress of the International Council of the Aeronautical Sciences (ICAS), 19-24 September, Nice, France.

Lagha M., Duplaa S., Binder N., Carbonneau X., Rodriguez B. (2019), "Similarity Study of Lightly Loaded Shrouded Rotors for Distributed Propulsion", paper presented at ASME Turbo Expo 2019, 17-21 June, Phoenix, AZ, USA.

National Academies of Sciences, Engineering, and Medicine (2016), "Commercial Aircraft Propulsion and Energy Systems Research: Reducing Global Carbon Emissions", National Academies Press, Washington, DC, USA. DOI: 10.17226/23490.

Pilatus (2013), "PC-12 NG Just the Facts", available at: http://www.synerjet.com/assets/pc-12-justthefacts.pdf (consulted 1 July 2019)

Roskam, J. (1985), Airplane Design, Part I, Roskam Aviation and Engineering Corporation, Ottawa, USA.

Sampson, B. (2018), "Zunum confirms flight testing schedule for hybrid-electric aircraft", available at: https://www.aerospacetestinginternational.com/news/electric-hybrid/zunum-confirms-flight-testing-schedule-for-hybrid-electric-aircraft.html (accessed on 1 July 2019)

Seitz, A., Schmitz, O., Isikveren, A.T., Hornung, M. (2012), "Electrically Powered Propulsion: Comparison and Contrast to Gas Turbines", paper presented at Deutscher Luft- und Raumfahrtkongress, 10-12 September, Berlin, Germany.

Sinnett, M. (2007), "787 No-Bleed Systems: Saving Fuel and Enhancing Operational Efficiencies", Boeing Aero Magazine Issue 28, The Boeing Company, available at: https://www.boeing.com/commercial/aeromagazine/articles/qtr_4_07/article_02_1.html (accessed on 1st of July 2019)