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Feasibility Study of a Hybrid Business Turboprop

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Abstract
The present study concentrates on the hybridising feasibility of a business turboprop aircraft. Consequently, the market considered is that of 6-10 passengers flying short distances, around 1000 to 1500 km. The propulsion is fully electric, including electrically driven propellers at the wing tips, but a meaningful fraction of the electrical energy is produced by a turbine gas generator. The fundamental issues considered are the degree of hybridization, and the influence of range and battery technology level in the feasibility of the concept.

1. Introduction. Green aviation and electric aircraft

Aviation is one of the main pillars of our economy and of our way of life (Goldberg et al 2018). It boosts commerce, tourism, cultural exchanges… Together with ICT it has shaped the global village concept we live in for its capability of connecting any places in the world, no matter how distant they may be, in astounding short time. Despite being relatively young, both key technological contributors have become indispensable. However, this does not mean that their development may continue without troubles.

Thus, civil aviation must face tremendous changes in the midterm future. Within 20 years, the envisaged growth could be unsurmountable, because passengers will double until 7 billion by 2034 (Gonzalez 2017). It is not only the congestion of major hubs or the airways; much more than that it is the growing awareness and endeavour to protect the environment. Pollution and noise appear among the issues to seriously be considered and solved.

In 2015, worldwide civil aircraft produced 780 million tons of CO2 and this figure will certainly triple by 2050, unless innovative technologies avoid that (Gonzalez 2017). During the last decades, air traffic evolved at high pace, somehow more moderate in North America or Europe, but with remarkable pressure in emerging economies such as China, India, or South America. Obviously, this growth suffered downturns with economic or health crisis, like the one recently produced by the Covid-19 pandemic (Yin et al 2020), but past experiences show that the recovery is quite fast.

For many decades, since its inception, civil aviation had safety as primary driver, distantly followed by performance. In recent times, environmental protection has taken the second place regarding air traffic management and aircraft operation (Zaporozhets et al 2020). Consequently, a significant effort will be done to encompass aviation evolution, i.e., design, manufacturing, and operation of aircraft, with this new scenario where terms such as sustainability play primary roles.

Although it does not produce as high emissions as other sectors, such as land use and forestry, building light and heat, road vehicles, etc (Mrazova 2014), aviation accounts for between 2,5 and 3,5% of all manmade CO2. Interestingly, various sources provide a diversity of figures of such impact, which could indicate that the studies and assessments around emissions are not fully clear (Noh et al 2015), (Kleiner 2007), (Mrazova 2014). Anyway, aviation appears always as one of the important sources for impact on the climate system, since the fuel burn produces 3,16 kg of CO2 per each kilogram of fuel used by aircraft (Mrazova 2014).

Atmospheric chemistry shows that CO2 exhibits a long lifetime, but there are other aircraft pollutants (NOx, aerosols, etc) whose effects depend on local conditions: geographical location, altitude, local weather and so on; and are more difficult to account for (Agarwal et al 2016), (Yin 2020).

Coherently with the above picture, all main authorities at European Union or United Nations level have elaborated various plans to reduce greenhouse gases and noise, from all sources. A large financial effort is foreseen that will change aeronautics to a non-classical market pull – innovation push scenario. In Europe, crucial initiatives are ACARE,

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Clean Sky and Horizon 2020 (EASA et al 2019) (Nicolosi et al 2021), (Janovec et al 2022). In the United States, it is the Aeronautics Strategic Implementation Plan, issued by NASA (Janovec 2022).

Specifically for aviation, the objectives include:

- 80% reduction in NOx emissions for landing/take-off cycles,
- 80% reduction in NOx emissions for cruise,
- 60% reduction in fuel burn,
- 52dB reduction in noise.

All above figures are established with respect to the 2005 best in class (Pornet and Isikveren 2015), (Liou et al 2017), (Palaia et al 2021).

Various approaches are envisaged to achieve these goals: evolutionary, revolutionary, and disruptive (Liou et al 2017), (Rohacs and Rohacs 2018). The results obtained by many researchers all over the world show that the former objectives are impossible with the evolutionary approach (Zaporozhets et al 2020) and much more radical solutions are required (Rohacs and Rohacs 2018).

Despite the massive financial support, civil aviation industry faces extremely challenging goals in mid to long term timeframe. Therefore, to achieve the prescribed reductions all key technologies should be involved: aerodynamic drag reduction, noise sources, green propulsion, more electrical aircraft, new generation flight management system, green materials, new manufacturing process and technologies… (Ling 2013), (Poernet and Isikveren 2015). However, as indicated earlier, mere evolutionary approaches will be insufficient to overcome the challenges (Agarwal et al 2016), (Zaporozhets 2020).

Regarding propulsion, two major fields of research are defined: propulsion itself and fuels. On the one side, focus is aimed at new fuels. Biodiesel and biobutanol could be burned in conventional turbofan engines (Kleiner 2007), (Noh et al 2015) and provide a significant reduction in CO2 balance. The real zero-emissions scenario would require the use of hydrogen, which is not at all an easy matter, in terms of production, distribution and safety of operation (Kleiner 2007).

On the propulsion side, again, two divergent lines of research are found: more or less revolutionary turbofan designs and electric propulsion. Some concepts look promising within the jet engine family: propfans, ultra-high-bypass ratio turbofans, etc. But it seems that the most powerful solution to achieve the goals in emission reduction is electric propulsion, since it completely eliminates direct emissions.

However, the development of electric propelled aircraft requires relevant changes in some key issues. The most restricting one is the specific energy of batteries. Fuels used in aviation, Jet A1 kerosene, and AVGAS 100 aviation gasoline, provide around 12 kWh/kg (Janovec 2022). Most used batteries, Li-ion and the like, offer 0,25 to 0,35 kWh/kg, which is around two orders of magnitude below common fuels (Palaia et al 2021), (Rohacs and Rohacs 2018), (Ma et al 2015), (Janovec et al 2022). To be competitive in the regional aviation market a minimum of some 0,80 kWh/kg should be reached (Palaia et al 2021). This means that even short-range aviation is still far from becoming fully electrified. Another relevant variable is the specific power or power density of batteries, of secondary importance in hybrid aircraft, particularly when the hybridising degree is rather low and most of the power in take-off comes from a gas turbine, as it occurs with potential concepts for mid-term future like the one considered in the present paper.

Due to the enormous energy and power requirements of large aircraft, electric propulsion is not viable in such aircraft, neither today nor in a near future. Interestingly, wide bodies and single-aisle airplanes, with more than one hundred seats, are responsible of 90% of fuel burn (Yin et al 2020), (Zaporozhets et al 2020). The trend for more electric aviation is on its way but marching quite slowly. So far, only limited applications have been developed to replace other energy sources onboard aircraft. It must be recalled that in commercial airplanes engines have two distinct functions: propulsion as a primary role, and power source for hydraulics, cabin climatization, onboard electricity, etc (Wroblewski and Ansell 2019), (Rohacs et al 2022).

As indicated above, short-range aircraft appear as promising platforms to assess and develop new possibilities of airplane design which would include fully electric or hybrid propulsion layouts (Soares and Voskuijil 2020). Some technologies and procedures could later be transferred to airliners (Soares and Voskuijil 2020). That is why the authors of the present paper have addressed the preliminary feasibility study of a hybrid turboprop, more appropriate for current and emerging battery capabilities (Riboldi 2018), (Wroblewski and Ansell 2019), (Soares and Voskuijil 2020), (Zaporozhets et al 2020).

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Electric and hybrid propulsion are quite appealing for various reasons: first, emissions are sharply reduced, thus following the predominant philosophy to protect the environment; second, owners and operators will be less dependent of oil prices and availability; last, electric motors are much more reliable and very economical from investment, operation and maintenance points of view (Riboldi and Gualdoni 2016), (Riboldi 2018), (Hashemi et al 2021).

The airplane design process includes various phases, commonly arranged in an iterative way, to define a sound configuration, to make a suitable sizing of all aircraft parts, and to confirm expected performances. The configuration of a hybrid turboprop may be quite similar, in external appearance, to a conventional design. However, the main problem to solve when dealing with fully new concepts, such as the hybrid propulsion airplane is that most sizing methods, developed for conventional configurations, are no longer applicable (Kamal and Ramirez-Serrano 2019), (De Vires et al 2020), (Sgueglia et al 2020), and new approaches and expressions must be elaborated.

The key variable, in all design cases, is to find a figure for the maximum take-off weight with a relatively small error, say 3 to 5% (Kamal and Ramirez-Serrano 2019). In a hybrid airplane this variable may be expressed as

$$W_{to} = W_{ice} + W_f + W_m + W_{bat} + W_e + W_{pl} \tag{1}$$

Where the subscripts correspond to internal combustion engine, fuel, electric motor, batteries, empty weight (structure and equipment), and payload, respectively (Riboldi and Gualdoni 2016), (Riboldi 2018). W_{ice} and W_m depend on the power required and the power architecture.

Once MTOW has been estimated, new mathematical expressions must be developed for main performances: take-off, climb, cruise, landing, etc. With these formulae a design space must be explored to find the best combination of power to weight ratio and wing loading (Torenbeek 1982), (Roskam 1985), (Raymer 2018). Then, the designer makes suitable choices within the available design space and finds the best value for wing area and total power, which must be distributed among the number of engines and propellers.

It must be recalled that the airplane has two different levels of power demand: one for take-off, very high, to meet ground roll or total take-off distance requirements; and a second one for cruise, with a more moderate level. The airplane must satisfy both, in the most efficient way (Bataller-Planes et al 2008).

The most important performance for transport aircraft, irrespectively of its size, small turboprop, single-aisle jet or large widebody airliner, is the link between range and all key variables of the airplane operation. For internal combustion engines, this expression, known as the Breguet range equation, provides a formal relation between the distance to be covered by the aircraft, on one side, and main airplane weights (W_{to}, W_f, etc), flying conditions (speed, lift over drag ratio), and propulsion system behaviour (specific fuel consumption, propeller efficiency), o the other. In turboprop airplanes, the equation reads

$$R = \frac{\eta}{c_p}\frac{L}{D} ln \frac{W_i}{W_f} \tag{2}$$

Where R is range, η the propeller efficiency, c_p the specific fuel consumption of the engine during the cruise phase of the trip. Additional fuel is burn during take-off, climb, descent, and landing phases. The total amount of fuel burn in non-cruise segments can be accurately computed with various methods (Roskam 1985), (Raymer 2018).

Obviously, this equation does not hold for electric, or hybrid airplanes and new expressions must be derived. In the case of a fully electric propulsion system the equivalent expression is (Traub 2011), (De Vries et al 2020)

$$R = \eta_{eg}\eta_p e_{bat} \frac{L}{D} \frac{W_{bat}}{(W_{oe}+W_{pl}+W_{bat})} \tag{3}$$

Where η_{eg} and η_p are the electrical generating and propeller efficiency, respectively, e_{bat} the specific energy of batteries, and W_{oe} the operating empty weight (airframe, equipment, power system, etc). It must be recalled that η_{eg} includes the performance effects of batteries, wires, inverter, controller, and electric motor.

For hybrid arrangements, there are several proposals for the range equation, one of which will be presented in a later chapter dedicated to the mathematical formulation of the hybrid aircraft design.

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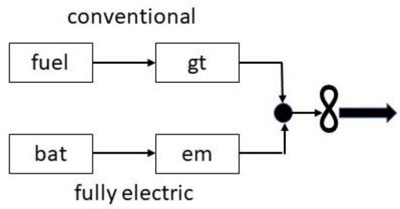
Recent research has shown that hybrid propulsion concept, in one of its many possibilities, is much closer in terms of feasibility than a full electric aircraft and, therefore, the present paper is centred in that idea. Hybrid electric technologies have been used in vehicles for many years and proved to be safe and highly efficient in lowering fuel consumption and emissions. Then, the step to introduce it in aviation looks quite appropriate. Furthermore, as indicated before, a moderate size turboprop with short range appears to be a good segment to start with.

Needless-to-say, the economic viability of hybrid electric aircraft is unknown (Wroblewski and Andell 2019), and many problems must be solved before the concept becomes fully operative, but at least initial studies look very promising. Among these problems, one key issue is the power management strategy, i.e., the suitable utilisation of full electrical propulsion or a hybrid arrangement (in adequate degrees) in the various phases of flight: take-off, climb, cruise, descent, and landing (Yin et al 2020). Other aspects to be considered include the development of models and procedures to check the state of health of the batteries (Hashemi et al 2021) or to provide appropriate colling and thermal management to cope with thermal instability of battery rows (Rohacs and Rohacs 2018).

The last point to be dealt with in these introductory paragraphs is the hybrid architecture itself, for there are many ways of defining it and many ways to arranging the various components of such propulsion system.

Figure 1 depicts the two main arrangements for the hybrid aircraft (Riboldi 2019), (De Vries et al 2020), (Soares and Vosjuijil 2020), although several variants exist. The choice of one or another depends on many parameters and decisions about the general configuration of the aircraft, since airplane design is always achieved as a compromise among the many disciplines involved: aerodynamics, structures, flight mechanics, control, propulsion, economics, etc (Torenbeek 2013), (Nicolai 2021).



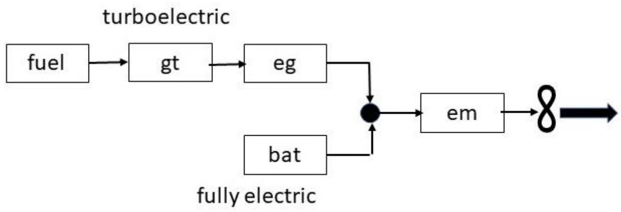


Figure 1. Schematic representation of the two main hybrid architecture layouts: parallel arrangement with mechanical node (top) and serial architecture with electrical node between energy sources (bottom). gt stands for gas turbine, em is electrical motor and eg is electrical generator.

2. Aircraft configuration and preliminary sizing

The turboprop market is divided into four segments: general aviation, commuters, business airplanes and commercial aircraft; each one with its specific owners and users. The present paper is centred in just the business airplane case. General aviation airplanes are too small and light to offer any possibility for hybrid aircraft due to the burden imposed by batteries. On the opposite side, commercial aircraft are too large and energy density and power density are far from the figures required by such vehicles. Business turboprops look promising for their innovative character exhibited along the last decades.

Based on market needs and the current state of batteries, it seems that in the short term the most viable option is a medium-sized propeller-driven aircraft with a series hybrid architecture. The motors are all electric and can receive power from both the batteries and from a gas generator located in the tail cone of the aircraft. Four electric-driven propellers are used in the present configuration, as shown in Fig. 2.

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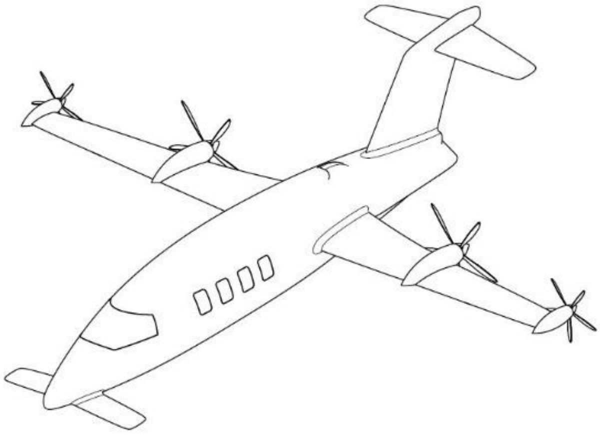

Figure 2. Airplane configuration with three lifting surfaces and four electrically driven propellers.

In addition to the typical mid-wing position of conventional props, two smaller electric motors are added at the wing tip. All four propellers are of the pusher type. This arrangement has many advantages, such as allowing more laminar flow on the wing, increasing the area of the wing influenced by the propeller suction and sharply decreasing the intensity of the wing tip vortex, thus reducing the vortex induced drag of the aircraft (Snyder and Zumwalt 1969). Since the power transmission to electric motors is achieved by mean of cables, they can be placed at any strategic position required. Another key feature of this arrangement is that the wing tip mounted propellers dispose of half the power of those situated at mid span. That is to avoid a too large yawing moment in case of failure of such electric motor, which would require a too large vertical tail and rudder, with the corresponding increase in drag and weight. The modifications of the conventional fuselage tail to accommodate the turbine also imply some difficulties, but they are of lesser level as compared to the overall architecture change.

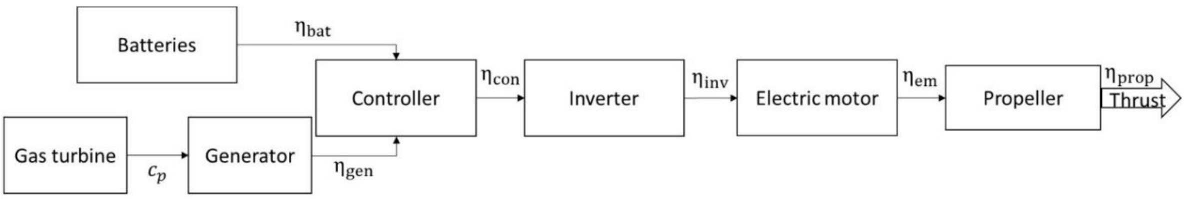

Figure 3. Schematic representation of power production and distribution.

The hybrid-electric architecture is presented in Figure 3. There are two possible paths from energy storage to thrust generation, one coming from the batteries and the other coming from the gas turbine and an electric generator, this last turning fuel into electric power. Each system has its corresponding efficiency (Table 2). Due to the existence of two paths for the creation of thrust, two types of total efficiency are distinguished, depending on power origin: one coming from the batteries (η_{tb}) and another starting at the gas turbine (η_{tg}).

$$\eta_{\mathrm{tb}} = \eta_{\mathrm{bat}}\eta_{\mathrm{con}}\eta_{\mathrm{inv}}\eta_{\mathrm{em}} \tag{4}$$

$$\eta_{\mathrm{tg}} = \eta_{\mathrm{gen}}\eta_{\mathrm{con}}\eta_{\mathrm{inv}}\eta_{\mathrm{em}} \tag{5}$$

Based on different estimates of battery specifications for the next few years, two scenarios are proposed, one in which the specific energy of the batteries (e_s) is 350 Wh/kg and another more optimistic one with 500 Wh/kg. Both scenarios contemplate a specific power value of 500 W/kg. The geometry is obtained from similar business turboprop aircraft with three lifting surfaces, such as the P180 Avanti or the Beechcraft Starship. Wing data are used for polar drag estimation and are collected in Table 1:

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Table 1: Wing geometry and battery parameters

Geometry	
Aspect ratio (AR)	11
Taper ratio (λ)	0.34
Batteries	
Specific energy (e_s)	350, 500 Wh/kg
Power density (p_s)	500 W/kg

Table 2: Efficiency of the diverse systems

ystem	Efficiency	Value
Batteries	η_{bat}	95%
Generator	η_{gen}	95%
Controller	η_{con}	98%
Inverter	η_{inv}	95%
Electric motor	η_{em}	90%

Once the aircraft configuration and the main design parameters are selected, the iterative process whose objective is the conceptual design of the aircraft is presented below. The MTOW of the aircraft is expressed in terms of the total energy through the degree-of-hybridization. Then, the total energy required to satisfy the range plus reserves is calculated, which in turn is a function of the weight of the aircraft.

The maximum take-off weight (MTOW) is divided into several contributions:

$$\mathrm{MTOW} = \mathrm{OEW}_0 + 1{,}25\mathrm{BW} + \mathrm{FW} + \mathrm{PL} = \mathrm{OEW}_0 + 1.25\frac{E_{tot}\psi}{\eta_{\mathrm{tb}} e_s} + \frac{E_{tot}(1-\psi)}{\eta_{\mathrm{tg}}} c_p + PL \tag{6}$$

Where OEW_0 is defined as the base operational empty weight (3000 kg) to which the weight of the batteries (BW) is added together with a 25% weight penalization in structural reinforcement (that increases OEW). The fuel weight (FW) and the payload (PL) complete MTOW. The payload is estimated as 100 kg per passenger, giving a total of 1000 kg. The weight of the batteries can be expressed in terms of total energy by multiplying by the degree-of-hybridization and divided by the specific energy and the overall efficiency from batteries to propeller. Similarly, the fuel weight is expressed in terms of total energy provided by the gas generator and the corresponding efficiency.

Chapter 1 showed how the Breguet equation for the range is modified when the propulsion is fully electric. However, in a hybrid-electric configuration an expression for the range can be derived as a function of the degree-of-hybridization. The range is expressed in integral form as

$$R = \int_{t_i}^{t_f} V dt \tag{7}$$

The variation of energy per unit of time available to power the aircraft in cruise conditions, independently of its origin, fuel or batteries, is expressed below.

$$\frac{dE}{dt} = -\frac{P_{cru}}{\eta_{\mathrm{me}}\eta_{\mathrm{inv}}\eta_{\mathrm{con}}} \tag{8}$$

Where the available cruise power for uniform, stationary and levelled flight can be expressed in terms of the flight conditions and the aerodynamic efficiency as

$$P_{cru} = \frac{WV}{\eta_{\mathrm{prop}}(L/D)_{cru}} \tag{9}$$

Introducing Eqs. 8 and 9 in Eq. 7 and splitting the weight into its various contributions as indicated in Eq. 6, the following expression is obtained:

$$R = \int_{E_i}^{E_f} \eta_{\mathrm{con}}\eta_{\mathrm{inv}}\eta_{\mathrm{em}}\eta_{\mathrm{prop}}(L/D)_{\mathrm{cr}} \frac{dE}{\mathrm{OEW}_0 + 1.25\frac{E\psi g}{\eta_{tb} e_s} + \frac{E(1-\psi)g}{\eta_{tg}} c_p + \mathrm{PL}} = \frac{\eta_{\mathrm{con}}\eta_{\mathrm{inv}}\eta_{\mathrm{em}}\eta_{\mathrm{prop}}(L/D)_{\mathrm{cr}}}{1.25\frac{\psi g}{\eta_{tb} e_s} + \frac{(1-\psi)g}{\eta_{tg}} c_p} \ln\left(\frac{1.25\frac{E_i\psi g}{\eta_{tb} e_s} + \frac{E_i(1-\psi)g}{\eta_{tg}} c_p}{1.25\frac{E_f\psi g}{\eta_{tb} e_s} + \frac{E_f(1-\psi)g}{\eta_{tg}} c_p}\right) \tag{10}$$

From this range expression for a hybrid-electric aircraft it can be concluded that to maximize range the aircraft should fly at maximum lift-to-drag ratio, as in conventional turboprops.

This flight condition is defined by the drag polar of the aircraft, which is estimated from the geometry of the Piaggio P180 Avanti assuming an increase in the span efficiency parameter because of the interaction between the wingtip propeller slipstream and the wingtip vortex (Sinnige et al. 2019). In the current research, the drag polar is modelled as (Torenbeek 1976, Roskam 1987)

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$$c_D = 0.0180 + 0.0276c_L^2 \tag{11}$$

Once the aircraft drag polar is obtained, and a cruise velocity of 300 knots (556 km/h) is selected as a reasonable velocity for a fast turboprop aircraft on short ranges, the flight altitude can be obtained from the lift equation.

$$\mathrm{c_{L_{cr}}} = \mathrm{c_{L_{(L/D)max}}} = \sqrt{\frac{\mathrm{c_{D_0}}}{\mathrm{k}}} = 0.808 \tag{12}$$

$$\rho_{\mathrm{cr}} = \frac{W_{\mathrm{cr}}}{1/2V_{cr}^2 S c_{Lcr}} \rightarrow\ h_{cr} \tag{13}$$

Where W_{cr} is estimated as 0.97 MTOW since fuel weight is a modest fraction of take-off weight.

To determine the total energy required for the flight (take-off, cruise, descent and landing) plus the reserves needed for the flight to an alternate airport, a block time of $t(h) = R/Vcr + 1$ hour is estimated, taking into account that we must add half an hour of flight to the alternate plus the extras of the second climb, a certain loiter and approach manoeuvres to the latter. On the power consumption side, it is necessary to count with some 5 kW of power required by the onboard systems and equipment. Thence, with Eqs. 9 and 13, a final expression for the total energy is used in the flight is

$$E_{tot} = (P_{cr}+5)t(h) = \left[\frac{V_{cr}}{\eta_{\mathrm{prop}}}\frac{0.97\left(\mathrm{OEW_0}+1.25\frac{E_{tot}\psi g}{\eta_{tb}e_s}+\frac{E_{tot}(1-\psi)g}{\eta_{tg}}c_p+\mathrm{PL}\right)}{(L/D)_{\mathrm{cr}}}+5\right]\left(\frac{R}{V_{\mathrm{cr}}}+1\right) \tag{14}$$

The process is, obviously, iterative but convergence is easily obtained. The results presented in section 4 are a good proof of that.

3. Operational requirements

To determine the wing loading and power-to-weight ratio that will define the aircraft design point, the relevant aircraft performances are expressed in terms of these variables to ensure that the selected configuration meets all appropriate requirements. These manoeuvres are take-off, second segment climb, and landing. Additionally, take-off and cruise available maximum powers are calculated to determine the upper limit for the power-to-weight ratio, due to limitations imposed by the power density of batteries.

Take-off
An energy method (Roskam 1985) is used to link the main design variables to P/W as

$$\frac{P}{W} \geq K_{TO}\left(g\frac{W}{S_w}\frac{1}{C_{L_{maxTO}}}\right)^{1.5}\frac{1}{S_{TO}} \tag{15}$$

The power-to-weight ratio must be sufficient to perform a take-off in around 1200 m (S_{TO}). In the former expression $K_{TO} = 0.294$ and $C_{L_{maxTO}}$ relies in the range 2-2.4. These values represent state-of-the-art figures for business turboprops.

Second segment
The aircraft must be able to climb along a straight slope at constant speed (safety speed of take-off), after the failure of one of the engines, which is represented by the fraction $N_e/(N_e\text{-}1)$. According to Torenbeek (1976 and Roskam (1985)

$$\frac{P}{W} \geq \frac{1}{\eta_{prop}}\frac{N_e}{N_e-1}\sqrt{\frac{2.88g}{\rho_{SL}C_{L_{maxTO}}}\frac{W}{S_w}}\left(\gamma+\frac{1}{\left(\frac{C_L}{C_D}\right)_{2nd}}\right) \tag{16}$$

In the former expression the lift-over-drag ratio is smaller than in cruise, because the aircraft keeps deployed the high lift system it used during the take-off manoeuvre. The value of the climbing slope, γ, is established by the airworthiness authorities, and depends on the number of engines.

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Landing

In an analogous way to what has been indicated for the take-off performance, the aircraft wing loading must not exceed the one required to land in a 1200 m runway (Roskam 1985).

$$\frac{W}{S_w} \le \frac{W_{TO}}{W_L} K_L S_L \frac{C_{L_{maxL}}}{g} \tag{17}$$

Where $K_L = 1.045$, $C_{L_{maxL}} = 2.8$ and the ratio between MTOW and MLW, ($\frac{W_{TO}}{W_L}$), depends on the hybridising ratio. In the present work it is established at 1.02.

Take-off available maximum power

The power required during take-off must not exceed the one provided by the batteries and the gas generator (see Table 1).

$$\frac{P}{W} \le \frac{P_{maxTP}\eta_{tg} + p_S BW \eta_{tb}}{MTOW} \tag{18}$$

Cruise available maximum power

A similar expression must be derived for the required cruise power.

$$\frac{P}{W} \le \frac{P_{maxTP}\eta_{tg}\frac{\rho_{cru}}{\rho_{SL}} + p_S BW \eta_{tb}}{W_{cru} g} \tag{19}$$

4. Results

Following the iterative process sketched in section 2, values for MTOW, BW, FW are obtained as a function of the degree of hybridization, ψ, for three relevant ranges of the short-range turboprop market: 500, 1000 and 1200 km. This process is first carried out at achievable battery technology level, 350 Wh/kg, and then repeated for a not distant future (500 Wh/kg) as presented in the Introduction.

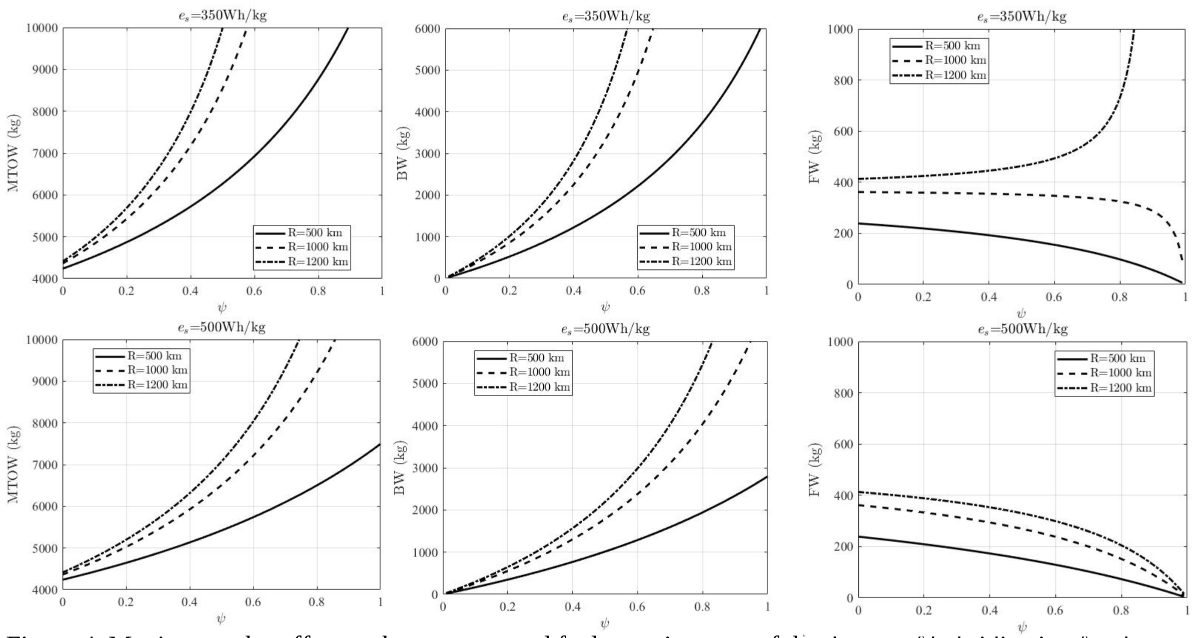

Figure 4. Maximum take-off mass, battery mas and fuel mass in terms of the degree of hybridization for three selected ranges.

The above pictures show how MTOW is greatly influenced by BW, and both increase with the degree-of-hybridization and range. The increase is rather sharp for hybridization level close to full electric aircraft. The fuel weight (FW) tends to zero with increasing degree-of-hybridization. Nevertheless, for the 350 Wh/kg case it can be observed how FW tends to infinite, since the mission cannot be accomplished. A higher value of the specific energy, 500 Wh/kg, not yet

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available, allows to design and operate hybrid turboprops with a reasonable all up weight, and even permits a full-electric aircraft for the shorter range. Since it is desired to increase hybridization as much as possible without compromising the weight, a degree-of-hybridization of 40% and 60% are selected for the 350 and 500 Wh/kg specific energies, respectively. A rounded range value of 1000 km is selected as representative of key routes: London-Milan, Madrid-Paris, Tokyo-Seoul, Singapore-Jakarta, Chicago-Washington, Los Angeles-Salt Lake City, Atlanta-Miami, Bogota-Caracas, Sao Paulo-Brasilia, and many others. The corresponding operating point leads to the following specifications.

Table 3. Main results for the 1000 km range case, with two battery technology level alternatives.

Specific energy	ψ	MTOW	BW	FW	Cruise altitude
350 Wh/kg	0.4	7172 kg	2254 kg	355 kg	43000 ft
500 Wh/kg	0.6	7212 kg	2380 kg	238 kg	43300 ft

Interestingly, the cruise altitude obtained from Eq. 13 is higher than that of conventional flights, which entails a side benefit in terms of market and operation: such hybrid aircraft do not have to compete for airspace with conventional turboprop nor turbofan airplanes, as turboelectric aircraft fly at higher flight levels. Another relevant aspect is that since the total weight is less than 8600 kg and the number of passengers is less than 19, the aircraft can be certified under CS-23/FAR-23 standards, which are less demanding than the CS-25/FAR-25.

Once the aircraft weight is selected, the design point, defined by the wing loading and power-to-weight ratio, as indicated in section 3, can be obtained to fulfil all requirements. The plots shown in Fig. 5 correspond to two different battery specific energy: current top class, and achievable in near future; both are represented for the 1000 km range case, although the performance limits are the same except for the upper power limits due to the different battery weight.

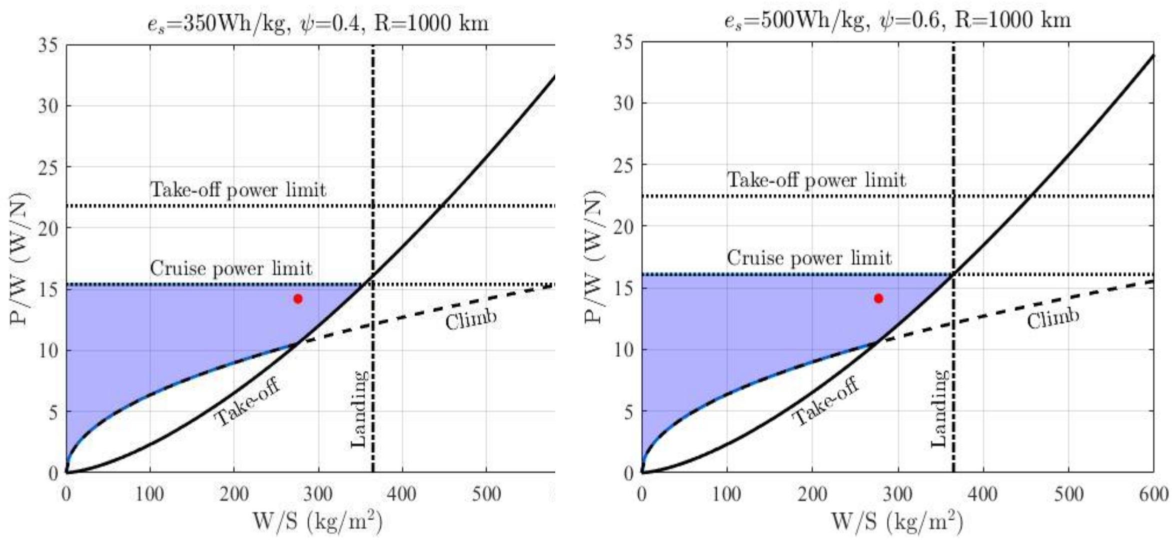

Figure 5. Design point plot representing the power to weight ratio requirements for various key performances in terms of the airplane wing loading, for the two specific energies studied. Red dots are the selected design points.

The coloured area corresponds to the set of wing loadings and power-to-weight ratio that satisfy all three performance conditions. The importance of the upper power limit must be noticed. This threshold will move upwards if the specific power of the batteries is increased, although it can also be shifted to higher values with a more powerful gas generator. Regarding the coloured area, low wing loadings must be discarded because that would imply fantastically large wings, i.e., far from an efficient airplane. The best design points are close to the upper right corner: with a reasonable wing geometry and size the aircraft meets all requirements. A certain clearance can be left to allow future, larger models with similar design specifications.

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Table 4. Design points for the 1000 km range case, for the two alternatives analysed.

Specific energy	ψ	W/S	P/W	S_w	P_{tot}
350 Wh/kg	0.4	275.8 kg/m^2	14.2 W/N	26 m^2	1000 kW
500 Wh/kg	0.6	277.4 kg/m^2	14.1 W/N	26 m^2	1000 kW

5. Conclusions

The research reported here has addressed and confirmed the feasibility of a hybrid business turboprop airplane as one of the relevant markets where electrical propulsion can be efficiently applied. The advantages regarding emissions, noise and operating cost have also been indicated. The main findings are stated in the following paragraphs.

With available technology, specifically battery performance, only medium size aircraft exhibit features compatible with battery-dependent hybrid electric propulsion. Very small aircraft are discarded for the burden imposed by the extra weight of batteries. Large aircraft are also discarded by battery power density and specific energy limits.

Electric propulsion offers higher flexibility than conventional turboprops, regarding the overall configuration and the location of propellers with respect to the airframe. In the present case, the best solution seems to be that with four propellers, two at their common mid-span location and two at the wing tips to improve the aerodynamics of the airplane, specifically C_{Lmax} and $(L/D)_{max}$.

Despite the completely different energy source, better range conditions are the same for fuel burning engines and for battery powered aircraft; namely, the aircraft must fly at a combination of wing loading and dynamic pressure that corresponds to the best lift-over-drag ratio. In the case of this business hybrid turboprop, a speed of 300 knots matches flight level around 43000 ft.

With current battery technology and a hybridization level of 0,4 it is possible to fly 1000 to 1200 km, with 1000 kg payload, while keeping the airplane all up weight under reasonable limits, and allow demanding certification under FAR/CS-23 airworthiness requirements. With a moderate improvement in battery performance, the hybridization level could increase up to 0,6, holding the aforementioned benefits. A big leap in battery performance would be needed to allow full electric longer, faster flights, or to design larger electric airplanes.

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