



Feasibility studies on regional aircraft retrofitted with hybrid-electric powertrains

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ABSTRACT

Hybrid-electric powertrains are considered a promising technology for mitigating noxious emissions and promoting aviation decarbonization, particularly in regional segments with limited flight energy requirements. However, assessing their potential quantitatively proves challenging owing to system complexity and enlarged dimensions of the design space. Traditional aircraft design methods may not guarantee optimal integration of electrical systems, in particular considering batteries current low technology readiness. Therefore, a multidisciplinary optimization framework has been developed and employed to quantitatively assess the potentials of an ATR42-600 regional aircraft retrofitted with hybrid-electric powertrains. The new framework has been conceived to extend the open-source program OpenConcept available design capability to regional turboprop aircraft retrofitted with different hybrid-electric powertrain and thermal management system architectures. Different discipline modules were improved and adapted to commercial aviation by integrating CS/FAR 25 requirements and airworthiness constraints. Optimization campaigns assessed the performance of both serial-hybrid and parallel-hybrid configurations on reference and short-range missions, considering different future battery technology energy densities. Retrofitted solutions were assessed in terms of fossil fuel consumption and CO₂ emissions per kg of payload and km of range. Significant emission reductions can only be obtained for future battery technologies providing a specific energy of 1500 Wh/kg for a reduced payload of 3 t. Parallel-hybrid powertrains offer a more integrated solution for regional retrofitted aircraft. However, when CO₂ emissions are normalized by a variable payload, the advantages of the retrofit are only observed for ranges smaller than 600 NM. Analysis of the optimization results has yielded valuable insights on the optimal battery use, which must be allowed along all the mission including diversion to minimize CO₂ emissions. Hybridization strategies, using battery as booster during the most energetically-demeaning flight phases, such as take-off and climb, are not efficient for retrofitted aircraft, as optimal performance requires a combustion engine redesign.

1. Introduction

The air transportation industry is rebounding swiftly after a temporary reduction due to the COVID-19 pandemic, with forecasts suggesting an annual growth rate of 3% to 5% [1], [2]. However, this growth is accompanied by increased fossil fuel consumption and Greenhouse Gas (GHG) emissions, particularly carbon dioxide (CO₂), which currently contributes 2% to 4% of total atmospheric emissions and is projected to triple by 2050. Given global concerns about climate change, significant efforts are currently made to enhance the sustainability of the aviation sector through cleaner propulsion technologies. Various governmental institutions have set targets and milestones to expedite aviation decarbonization, with initiatives such as the Paris Agreement aiming to

reduce emissions by up to 55% by 2030 and achieve climate neutrality by 2050 [3].

The potential of fully electric propulsion to virtually eliminate gaseous emissions in contrast to conventional combustion engines is acknowledged; however, the existing constraints in battery technology impede its practicality for large-scale airliners [4]. Conversely, hybrid-electric propulsion systems, combining fossil fuels with battery power, have earned considerable attention, particularly for short-range aircraft [5,6].

Novel design methodologies based on physics and digital tools are crucial for integrating hybrid-electric propulsion into certifiable aircraft efficiently [7]. Existing design simulation tools for hybrid and electric aircraft lack flexibility and completeness [8]. Proprietary tools like

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GT-HEAT and LEAPS offer comprehensive analysis, while open-source platforms like SUAVE and OpenConcept provide flexibility for unconventional designs, though usually tailored for general aviation [9–12]. A gap still exists for multidisciplinary platforms specific to CS/FAR 25 regional aircraft [13].

Numerous studies have sought to assess the feasibility of hybrid regional aircraft. Conceptual designs of hybrid-electric aircraft are often challenging due to the low Technology Readiness Level and the absence of dedicated airworthiness regulations. The treatment of the hybridization factor, denoted as Φ , is critical to obtain scientifically sound results and evaluate the potential of hybrid propulsion [14]. As defined in Equation (1), Φ represents the fraction of total power, which is the sum of electric power (P_{electric}) and thermal power (P_{thermal}), generated by the electric source.

$$\Phi = \frac{P_{\text{electric}}}{P_{\text{electric}} + P_{\text{thermal}}} \quad (1)$$

In some early research activities Φ was assumed to remain constant throughout the entire mission. Subsequent research introduced more detailed constraints, considering different constant hybridization factors across mission ranges and technology levels [15–17]. Additionally, alternative sizing methods were proposed based on matching diagrams for different segments within desired hybridization ranges [18,19]. Marciello et al. [20] assessed the impact of multiple Φ values on the design efficiency of a hybrid-electric ATR42, which included distributed electric propulsion. Furthermore, Palaia et al. [21] hybridized an ATR42, using a parallel hybrid-electric powertrain and maintaining a constant Φ during the nominal trip while operating the diversion mission solely on fuel ($\Phi = 0$).

An alternative approach involves sizing the thermal engine for continuous power efficiency during cruise or throughout the entire mission, with the electric system providing additional power as needed. This strategy has been pursued by several authors [22–27]. A conceptual tool enabling the selection between constant thermal power fractions and boosting electric engine power for hybrid-electric turbofan aircraft design was presented in [28].

Studies have explored optimizing Φ for general aviation mission segments [4,29]. Riboldi studied motor-glider optimization, implicitly treating Φ as a variable [30,31]. Spinelli et al. used probabilistic logic to determine Φ variation [32]. Concerning regional aviation, Palaia and Salem considered Φ as a design variable (DV) [33,21]. However, these studies did not consider thermal management and airworthiness.

Previous investigations on hybrid-electric aircraft design fall short of providing a complete feasibility analysis for the regional aircraft. Hybridization performance has been treated from a very simplistic perspective in most of the research activities, lacking a consistent optimization to systematically address limits and potential of the propulsion technology. Moreover, not enough attention is drawn to the novel configuration feasibility and viability as operative and certification constraints are not sufficiently addressed from the first steps of the design [34]. Rapid industrialisation of aircraft electrification can be realised from a complete and multidisciplinary point of view, only fulfilling safety and viability constraints.

This research aims at assessing retrofitting the regional ATR42-600 turboprop with hybrid-electric propulsion in terms of performance, environmental, and economic impact. The assessment is performed using a developed hybrid-electric aircraft design and optimization framework, born as fork from OpenConcept [12], enabling conceptual design and optimization of CS/FAR25 regional hybrid-electric aircraft while ensuring airworthiness compliance. The tool includes thermal management system assessment and treats Φ as a design variable for mission performance optimization. The framework improves OpenConcept, adapting it for regional aircraft and complying with airworthiness regulations. It introduces novel modules and structures and assesses optimization against ATR42-600 retrofitting with serial and parallel-hybrid propulsion systems, considering various design approaches, Φ optimization

strategies, and component maturity levels. Results compare fuel burn and emission-optimized configurations across different mission ranges, powertrain architectures, and battery technology levels.

2. Aircraft multidisciplinary design and optimization platform

The in-house optimization software, initially derived from OpenConcept and built on OpenMDAO, is a Python-based tool for multidisciplinary design, analysis, and optimization [35]. Leveraging on OpenMDAO, it emphasizes gradient-based optimization and total derivatives computation for complex models, using an object-oriented programming approach.

Efficiency and robustness in Multidisciplinary Analysis (MDA) convergence make it suitable for complex system design, such as hybrid-electric aircraft [36]. It accommodates various hybrid-electric architectures and aircraft categories (FAR/CS 23 and 25), considering airworthiness requirements.

Enhancements beyond this paper scope include life cycle assessment (LCA) responses, design variables for mission parameters, and support for multi-objective optimization. Current capabilities allow modeling complex architectures integrating batteries, fuel cells, and hydrogen-burning thermal units.

The following sections provide a brief overview of the software main capabilities. A detailed description is available in reference [36].

2.1. Mission module

The mission module defines the nominal and diversion segments of the aircraft mission, ensuring compliance with CS/FAR 25 regulations for airworthiness requirements. The nominal mission encompasses phases like taxi, take-off, climb, cruise, descent, holding, and landing, as shown in Fig. 1. Additionally, the aircraft must occasionally perform an extended mission, specifying extra fuel or energy to be carried to comply with airworthiness requirements. The diversion mission involves climb, cruise, descent, and a holding phase, including a 30-minute loiter at an altitude of 1500 ft [37].

The system of equations governing the quasi-steady airborne mission phases is defined by Equations (2),

$$\begin{cases} L = mg \cos \gamma \\ T - D = mg \sin \gamma \end{cases} \quad (2)$$

where m represents the aircraft current mass, T is the thrust, D is the drag, g is the gravity acceleration, and γ is the climb angle. Thrust is assumed to be aligned with the velocity in all mission segments. For the ground phases, Equation (3) describes the aircraft dynamic behavior:

$$m \frac{dV}{dt} = T - D - \mu(mg - L) \quad (3)$$

where V indicates the airplane velocity, and μ is the friction coefficient with the runway. This equation describes the dynamic behavior during ground operations. System of Equation (2) is solved differently within the airborne mission segments. For certain flight conditions, such as climb and cruise, the software iteratively adjusts a control parameter, the throttle, to reach the desired climb angle while driving to zero the unbalance of thrust/drag. This iterative process is numerically set-up using OpenMDAO BalanceComp component to drive the residual to zero [38]. For other flight conditions, such as go-around, the throttle is set to a fixed value, and the software calculates the corresponding climb angle.

In the ground mission segments, the throttle setting remains fixed. Equation (3) is integrated to compute the aircraft speed and calculate the take-off and landing distances. To achieve this, the software uses the Integrator component provided by OpenConcept, which is instantiated twice for this purpose. This helps in simulating and calculating the required distances for both take-off and landing.

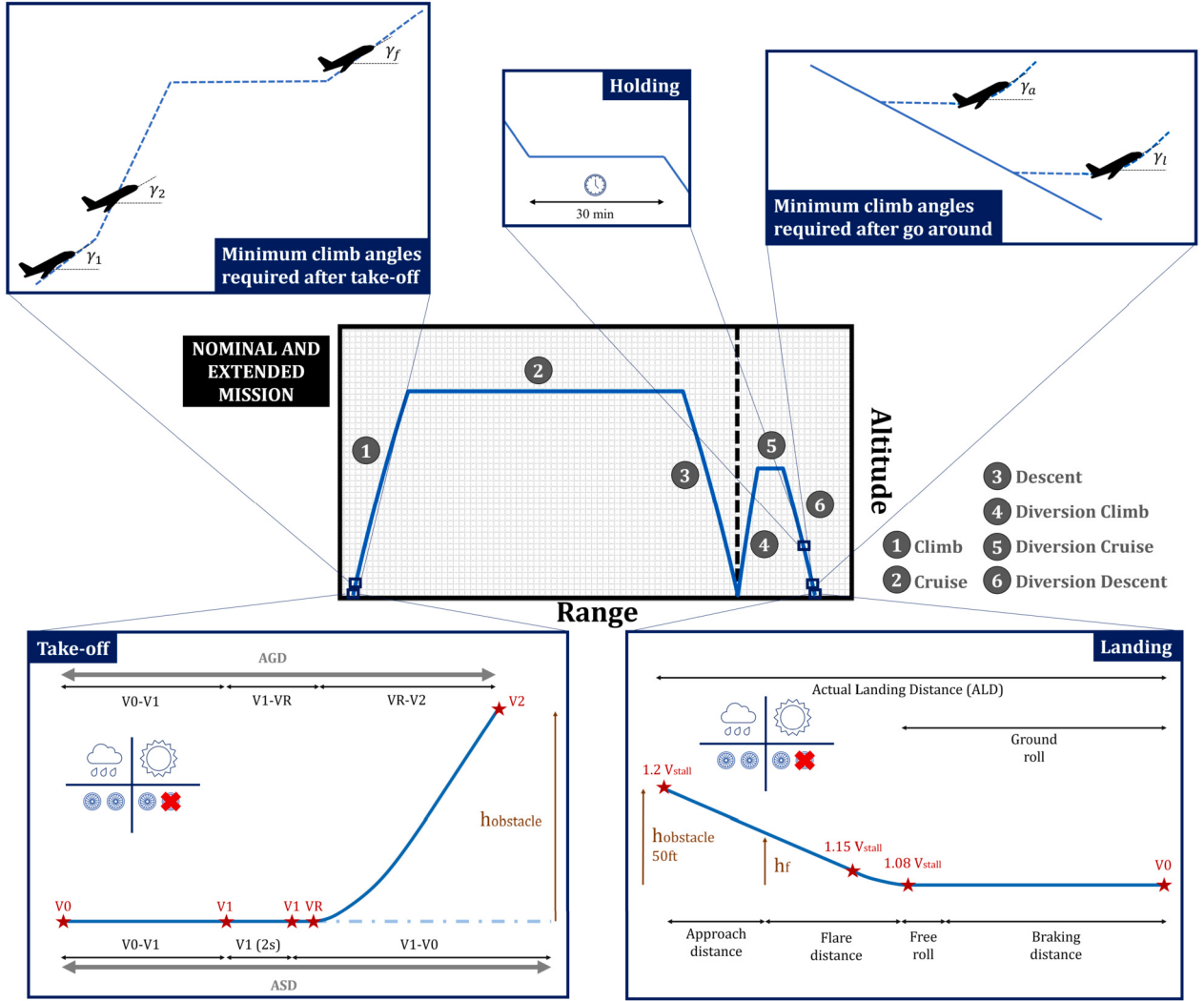


Fig. 1. Mission and segments.

2.1.1. Take-off phase

The Take-off (TO) phase is divided into multiple segments for improved modeling and the introduction of safety constraints, as shown in Fig. 1. These segments are identified by the speeds V_0 (0 m/s), V_R ($1.1 V_{stall}$), and V_2 ($1.2 V_{stall}$), where V_{stall} represents the stall speed in the take-off configuration. The take-off decision speed, V_1 , is a state variable that is determined by the program as detailed below. To compute the Take-off Field Length (TOFL) for a twin-engine configuration, two specific cases are considered: All Engines Operative (AEO, N) and One Engine Inoperative (OEI, N-1), under both wet and dry runway conditions. An iterative procedure concerning V_1 is established to derive the Balanced Field Length (BFL). The BFL is defined as the distance traveled when the Accelerate-Go Distance (AGD_{N-1}) matches the Accelerate-Stop Distance (ASD) in case of an engine failure, see Equation (4). The necessary field lengths for take-off under various scenarios are obtained through the integration of Equation (3), using specific parameters tailored to each combination of the number of operative engines and field conditions [39]. This meticulous process ensures safety during take-off and provides accurate field length requirements for a range of scenarios.

$$BFL \rightarrow AGD_{N-1} = ASD \quad (4)$$

ASD and AGD are determined based on the maximum values obtained across all combinations following the calculation of distances in each condition (Equations (5) and (6)):

$$ASD = \max\{ASD_{N-1 \text{ dry}}, ASD_{N-1 \text{ wet}}, ASD_{N \text{ dry}}, ASD_{N \text{ wet}}\} \quad (5)$$

$$AGD_{N-1} = \max\{AGD_{N-1 \text{ dry}}, AGD_{N-1 \text{ wet}}\} \quad (6)$$

The TOFL is determined as the greater value between the BFL and 115% of the AGD with AEO on a dry runway, i.e.,

$$TOFL = \max\{AGD_{N \text{ dry}} \cdot 1.15, BFL\} \quad (7)$$

The TOFL value is typically subjected to an upper bound by the optimizer.

Under nominal conditions, $AGD_{N \text{ dry}}$ is a byproduct of the aforementioned procedure for calculating TOFL. Furthermore, the integration of Equation (3) allows the calculation of the energy consumption, from both fuel and battery usage. This comprehensive approach ensures that both take-off performance and energy requirements are considered during mission planning.

2.1.2. Landing

The landing phase comprises three primary segments: approach and flare, free roll, and braking phases (Fig. 1). The landing distance is determined by integrating Equation (2), tailored to the specific conditions of each phase. When modeling the nominal or extended mission, the actual weight of the aircraft and a dry landing field are considered. However, to evaluate the Landing Field Length (LFL) in compliance with airworthiness regulations, the analysis uses the aircraft Maximum

Table 1
Climb gradient requirements according to [40].

	Take-off			Go-around	
	1 ^o segment	2 ^o segment	3 ^o segment	Approach	Landing
γ_{min}	0%	2.4%	1.2%	2.1%	3.2%
h_{ref}	35 [ft]	400 [ft]	1500 [ft]	-	-
Engine conf.	OEI	OEI	OEI	OEI	AEO
Power set	RTO	RTO	MCT	GA	-
Landing gear	Retraction	Retracted	Retracted	Retracted	Extended
Ref. speed	V_R	V_2	$1.45 V_{stall}$	-	-
Ref. weight	W. after $V_R - V_2$	W. after $V_R - V_2$	Segment end	MLW	MLW
Ground effect	Without	Without	Without	-	-
Flaps conf.	TO	TO	Clean	Approach	Landing

Landing Weight (MLW), a dry landing field surface, and One Engine Inoperative (OEI) conditions.

2.1.3. Climb

To ensure compliance with airworthiness regulations [40], minimum climb gradients must be met under specific conditions. Five additional segments are introduced to include these constraints in the optimization process. In each segment, Equation (2) is tailored to the specified conditions, and climb gradients are calculated. Table 1 lists climb conditions, configuration settings and target climb gradients for different segments. Engine power settings are specified as well, where RTO stands for Reserve Take-Off configuration, MCT for Maximum Continuous Thrust, and GA for Go-Around power. The take-off climb, following an engine failure, imposes minimum climb gradient requirements for the initial, second, and final segments. Additionally, specific performance criteria must be satisfied during a go-around (aborted landing) for both AEO and OEI conditions.

2.2. Aerodynamics and weight estimation

The aerodynamic module employs classic drag polar method to calculate the drag force, lift, and stall speed [39]:

$$C_D = C_{D0} + K C_L^2 \quad \left(K = \frac{1}{\pi A R e} \right) \quad (8)$$

Here, C_{D0} represents the parasitic drag coefficient (or zero lift drag coefficient), K is the induced drag factor, AR is the wing aspect ratio, and e is the Oswald factor. The Oswald factor for the clean configuration is computed using Obert's method [41], while parasite drag is estimated using the equivalent skin-friction method described in reference [39]. To perform such calculations, the framework determines the wet area of each aircraft substructure and the total wet area based on the expressions provided in reference [42].

The aerodynamic module incorporates corrections to the drag polar to take into account variations in take-off and landing configurations. These corrections include incremental parameters extracted from BADA [43] for different flap configurations and the extended or retracted landing gear. The Oswald factor is also adjusted following the suggestions from [42]. Another source of incremental drag is introduced in the event of propeller failure and a stopped propeller [42]:

$$\Delta C_D = 0.00125 \cdot n_p \cdot \frac{D_p^2}{S} \quad (9)$$

Here, n_p represents the number of blades per propeller, D_p is the propeller diameter, and S is the aircraft wing surface area. Ground effect is considered in the relevant segments by adjusting the value of K as proposed in [39]:

$$\frac{K_{effective}}{K} = \frac{33 \cdot (h/b)^{1.5}}{1 + 33 \cdot (h/b)^{1.5}} \quad (10)$$

being h the wing height above ground and b the wingspan.

The mass estimation module uses semi-empirical relations from various sources, including [39,44,42,45], to predict the Operative Empty Weight (OEI) of the aircraft. These relationships are adapted and

specialized to characteristics and requirements of regional turboprop aircraft, ensuring accurate mass estimations for this particular case. Verification and calibration have been carried out against ATR42-600 published data, as detailed in next sections. The OEI is determined using a weight breakdown approach, expressed by the equation:

$$OEI = W_{structure} + W_{power\ plant} + W_{systems} + W_{furnishing} + W_{oper\ items} \quad (11)$$

Equation (11) breaks down the OEI into contributions from different components, including the structure ($W_{structure}$), power plant ($W_{power\ plant}$), systems ($W_{systems}$), furnishing ($W_{furnishing}$), and operational items ($W_{oper\ items}$). It is worth noticing that when hybrid-electric powertrains are introduced, the weights of new components, such as electric motors and inverters, must be added to the OEI calculation. However, the weight of the battery packs (W_{batt}) as flight energy source, is not considered as part of the OEI. However, the weight of the battery packs (W_{batt}) as flight energy source is not considered as part of the OEI. That entails that multifunctional or structural batteries are not contemplated in this study.

2.3. Hybrid-electric powertrain module

This module models both propulsion and the Thermal Management System (TMS) of the aircraft. Its flexible and modular design allows the simulation of various powertrain architectures, such as serial- and parallel-hybrid configurations discussed in this paper. The module employs conceptual-level models for electrical power system components, each accepting inputs such as maximum power, efficiency, and providing outputs related to weight and costs [36]. Additionally, it calculates battery state of charge throughout the mission. Operating power and waste heat generation based on efficiency are computed for each powertrain components at every flight point.

The thermal gas turbine-based powertrain is modeled with simple formula relating the operating power to the fuel burnt via Power Specific Fuel Consumption (PSFC) and estimating weight as function of the maximum power.

Propeller performance is modeled using the Hamilton Standard method for three and four-bladed cases [46]. For modern six-bladed propellers as for the reference aircraft, ATR42-600, accurate performance maps were not available from manufacturers, so extrapolations were made based on recommended values and practices from reference [44].

The powertrain modules also simulate auxiliary systems supporting power generation and ensuring optimal working conditions. For electric power generation, the main auxiliary system is the TMS, essential for preventing damage from heat accumulation. This sub-module is closely inspired by the concepts presented in reference [47] and offers a set of analysis routines for simulating the thermal management of electrical components. It provides users with the flexibility to design air or liquid cooling system architectures and select between quasi-steady or time-accurate temperature monitoring. Moreover, the module includes a model for a ducted heat exchanger which calculates additional internal flow drag using either an incompressible or compressible ap-

Table 2
CO₂ emissions rates for hybrid/electric aircraft [50,51].

CO ₂ emissions due to fuel combustion	3.14 kg CO ₂ / kg fuel	[50,51]
CO ₂ emissions due to fuel generation	0.61 kg CO ₂ / kg fuel	[50,51]
CO ₂ emissions due to electricity generation	0.42 kg CO ₂ / kWh electricity	[50]

proximation. It also accounts for heat exchange between the crossflow plate-fin channels.

2.4. Environmental impact indicators

For a conventional powertrain, the amount of fuel required for a safe journey along the planned route is calculated according to EU-OPS 121 regulations [48]. This calculation encompasses nominal and extended mission fuel, and includes contingency fuel (equivalent to 5% of the fuel consumed during the nominal trip). The primary focus of employing hybrid-electric powertrains is to reduce the environmental impact of aviation, particularly in terms of carbon dioxide (CO₂) emissions. While non-CO₂ emissions are gaining increasing attention, this study focuses on CO₂ emissions as pollution metric [49]. All CO₂ emissions arising from in-service fuel consumption (tank-to-wake), the industrial production of both required fuel (well-to-tank), and electricity to recharge batteries are taken into account. Table 2 lists the CO₂ emission rates from different sources. The fuel combustion emission rate is based on stoichiometric calculations, while the parameters related to energy generation vary significantly among different countries according to the predominant production method. Average values were therefore employed.

2.5. The optimization problem

An optimization problem is formally defined as:

$$\begin{cases} \underset{\mathbf{x}}{\text{minimize}} & f(\mathbf{x}) \\ \text{subject to} & \mathbf{c}(\mathbf{x}) \leq \mathbf{0} \end{cases} \quad (12)$$

being $f(\mathbf{x})$ the objective function, $\mathbf{x}$ the Design Variables (DVs) vector, and $\mathbf{c}(\mathbf{x})$ equality and inequality constraints. The effective realization of the Multidisciplinary Optimization (MDO) problem hinges on choosing a suitable architecture [52] that governs the interaction between the optimizer and the MDA processes, responsible for delivering the desired responses. While the framework offers flexibility and supports various monolithic architectures, this study employs a hybrid approach that combines MDF (Multidisciplinary Feasible) and SAND (Simultaneous Analysis and Design) methods.

2.5.1. The MDA

In the MDA process, multiple levels of iterations are employed. At the highest level, an iterative process is used to estimate the time required for flying specific mission segments to meet the prescribed total range by implicitly integrating the range equation. Moving to a lower level, certain mission segments involve an additional iterative process to obtain the proper powertrain settings. In these cases, a state variable representing the power (throttle, δ) for power generation components of the propulsion system is iteratively adjusted to minimize the residual of Equation (2). For other mission segments, such as take-off and landing, the throttle setting is fixed, and the same equation is integrated, as discussed in section 2.1.1. In case of hybrid-electric aircraft, the hybridization factor Φ is selected and fixed during the MDA. Consequently, after iteratively setting δ , the power requirements from the electric and non-electric components can be directly determined. However, one or more nested iterative procedures may be established to implicitly solve the nonlinear governing equations accommodating the input and output interfaces of the implemented classes [12].

By continuously monitoring the power output of powertrain components, fuel and electric energy consumption can be calculated. Integrating these values over the course of the mission yields the amount of

fuel consumed and the electric energy used from the batteries [36]. When TMS is integrated into the model, it allows component temperature calculations. These temperatures are determined by the heat balance, considering both the heat generated by machine operation and the heat removed by the cooling system, [47]. Iterative processes are carried out using *OpenMDAO* Newton solver [12]. The *OpenMDAO* analytic derivative capability greatly simplifies the process by computing total derivatives of the problem. That is an essential feature for efficient gradient-based optimization.

It is worth noticing that, due to the employed structure of the framework, the MDA does not itself yield a “feasible” aircraft, as explained in more details in section 2.5.2.

2.5.2. The MDO

Fuel consumption-based metrics have been chosen as objective functions in this research. Numerous parameters describing aircraft components can be considered as optimization design variables. Since this paper focuses on retrofitting an existing aircraft, most aircraft design parameters are held constant, and only the power ranges of the powertrain components are treated as DVs. The platform seeks the optimal allocation of electric energy along the mission by treating the Φ (hybridization factor) of each segment as DVs. Alternatively, Φ values can be set by user-defined inputs.

Payload can also be considered a DV, with an upper bound set by the aircraft maximum payload. To address aircraft Take Off Weight (TOW) variability during the MDO, two strategies are employed. In the first strategy, the TOW is treated as a DV, with an upper bound equal to the Maximum Take Off Weight (MTOW), and a constraint is added to ensure the consistency of the weight breakdown (Table 7). The second strategy allows for the introduction of a ballast, setting TOW = MTOW, and adding a constraint to ensure non-negative ballast.

To size the propulsion system and limit power regimes of the powertrain components, their maximum powers are treated as DVs, changing during the optimization. Constraints are added at the optimization level to ensure the relative use of powertrain components along the mission, as expressed by the state variables (throttles). Similarly, the size of the battery pack is a DV, driven by a constraint on the final State of Charge (SoC).

Other constraints include the TOFL, LFL and climb gradients, as specified in Table 1. If thermal management system is included in the model, maximum allowable temperatures for electric components are set as constraints, which influence the sizing of the cooling system in terms of coolant reservoir mass and heat exchanger dimensions [47].

2.6. Specialisation to twin serial/parallel hybrid-electric aircraft

The propulsion architectures for a twin serial and parallel hybrid-electric aircraft are modeled as shown in Fig. 2. In the serial-hybrid setup, each power line consists of a combustion engine (CE) driving a mechanically-coupled electrical generator, forming a turbogenerator. This produces alternating electrical current, which is then converted to Direct Current (DC) and combined with the battery one at the DC distribution bus. The combined power is then converted back to alternating electrical power feeding the AC Electric Motor (EM) driving the propeller. The main state variable for thrust control is the EM throttle (δ_{EM}), adjusted iteratively along with the CE throttle (δ_{CE}) to meet the target hybridization factor. Power losses are considered defining electric component efficiencies. In the parallel-hybrid architecture, each power line has a combustion engine and an electric motor powered by batteries, integrated via a Combining GearBox (CGB) to drive the

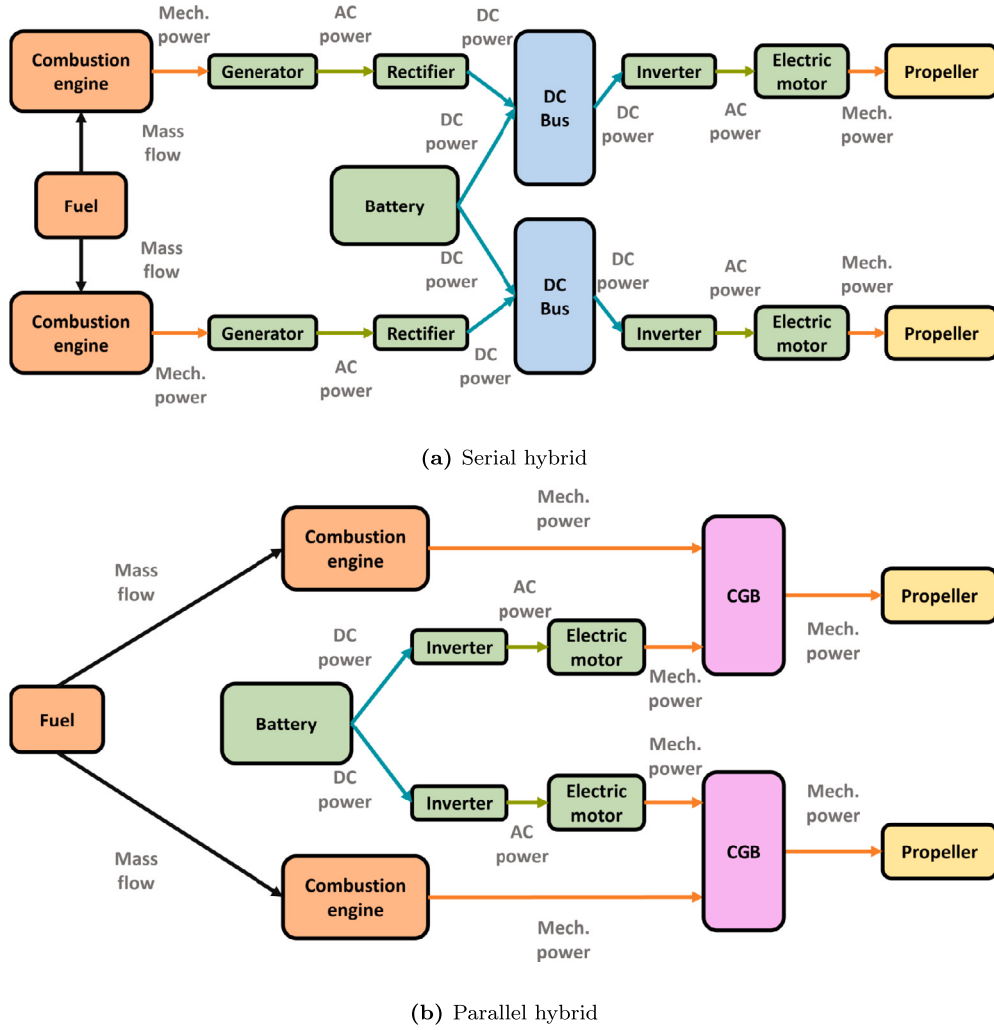


Fig. 2. Propulsion architectures.

propeller. The MDA adjusts the gearbox power fractionally (δ_{CGB}) to achieve desired thrust levels, with iterative adjustments to engine and motor powers for consistency in modeling [36].

The hybridization factors for serial- and parallel-hybrid architectures, Φ_{serial} and $\Phi_{parallel}$ are defined as follows:

$$\Phi_{serial} = \frac{P_{battery\ out}}{P_{inverter\ in}} = \frac{P_{battery\ out}}{P_{battery\ out} + P_{rectifier\ out}} \quad (13)$$

$$\Phi_{parallel} = \frac{P_{EM\ out}}{P_{CGB\ in}} = \frac{P_{EM\ out}}{P_{CE\ out} + P_{EM\ out}} \quad (14)$$

Φ can vary among different mission segments, but it is kept constant within one segment. Fig. 3 shows the conceptual arrangement of the MDA workflow.

The procedure described in section 2.3 was applied to calculate thrust by generating the propeller maps presented in Fig. 4. Fig. 4a and Fig. 4b contain data used to estimate the thrust coefficient (C_T) and propeller efficiency based on power coefficient (C_P) and advance ratio (J), respectively. These maps were created in compliance with established standards [44,46] and were generated for an Activity Factor $AF = 180$ and an integrated design lift coefficient (C_L) of 0.5. Both a constant and variable PSFC over the range of throttle settings were assumed for turbo-shafts and turboprops. The case of constant PSFC prevents the possibility of minimizing the aircraft fuel consumption by selecting optimal engine throttle; in this case the PSFC value was set at $8.33 \cdot 10^{-5}$ kg/(kW·s) [36]. The case of variable PSFC has been dis-

cussed in section 4.1.2 to suggest the need for gas turbines designed to perform more efficiently at non-standard throttle levels.

2.6.1. The TMS

Liquid-cooled TMS is employed for the reference ATR42-600 retrofit (Fig. 5). This system uses a closed hydraulic circuit to circulate a liquid coolant, which collects waste heat from electrical components through the use of cold plates [47]. Subsequently, a ducted heat exchanger cools the hot coolant releasing the accumulated heat to the air. To enhance thermal inertia, an additional reservoir is included in the model, and its weight is duly considered. The liquid employed for cooling the electrical components is a mixture of propylene glycol and water, possessing a specific heat of 3801 J/(kg·K). The specific heat of aerospace-grade aluminum, is used to determine the thermal mass of the electric components, and it is approximately 921 J/(kg·K). The geometric properties of the offset strip fin heat exchanger [47] and the design specifications of the cold plates for all electrical components (Fig. 5), their initial temperatures, and the mass flow rate of the liquid coolant have been configured as follows:

- initial temperature of all electric components: 15 °C;
- coolant mass flow: 0.5 kg/s;
- geometry of the cold plates (for all the electric components): 0.2 m of longitude (L_{cp}) with 150 microchannels set in parallel with 1 mm of width (w_{cp}) and 20 mm in height (h_{cp}) (Fig. 5);

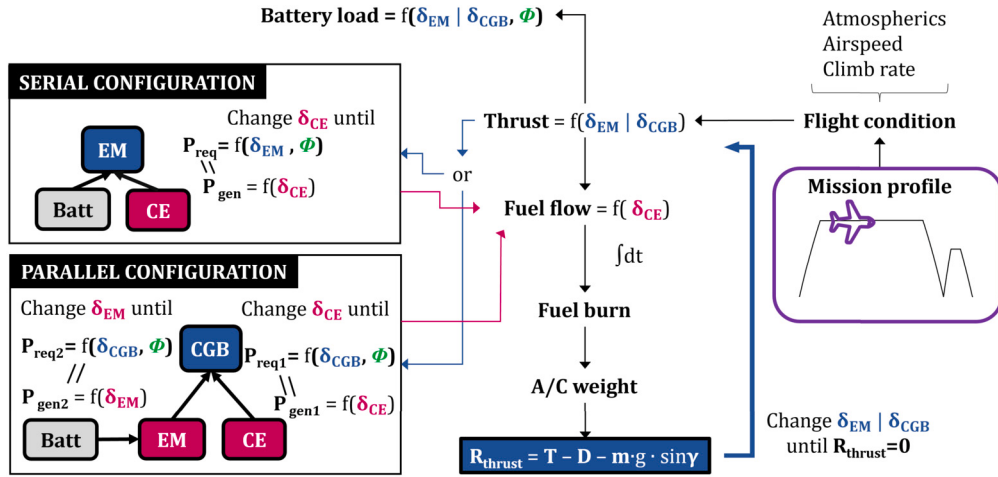


Fig. 3. Analysis workflow during mission segments, with highlighted interactions between control parameters.

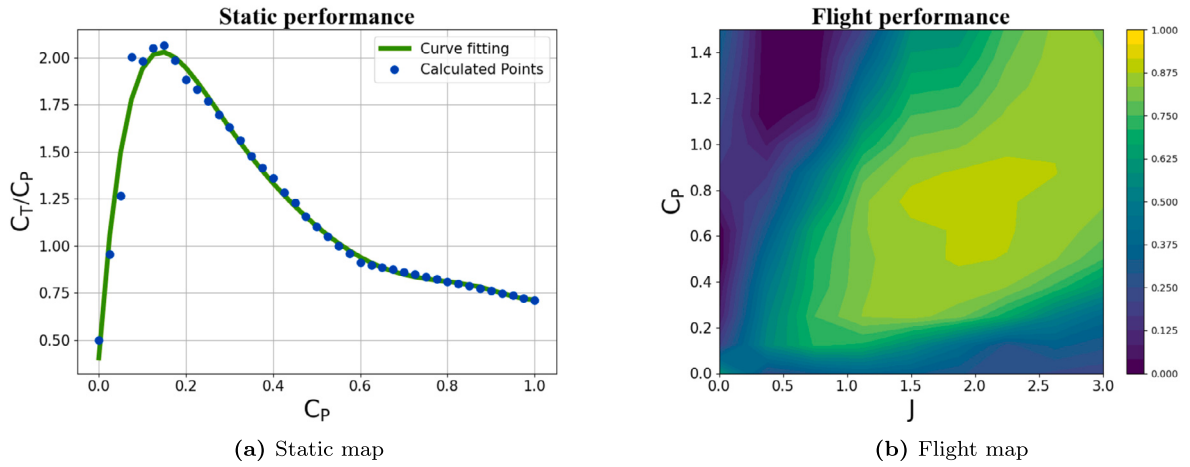


Fig. 4. Static and flight propeller maps for six blades with $AF = 180$ and $C_{T_e} = 0.5$.

- geometry of the heat exchanger: two side-channels, one for the hot side and one for the cold side, each measuring 1 mm in width (w), 12 mm in height (h), and 6 mm in length. The thickness between these channels is 0.102 mm (t_f), and the thickness between the cold and hot stacks is 0.2 mm (t_p) (Fig. 5).

The liquid cooling system arrangement (Fig. 5) is design taking into account the location of electric components into the airframe, and assigns a higher priority in the cooling sequence to the most critical components based on their heat release or vulnerability to temperature [36].

The implicit drag penalty involved in such a cooling process is estimated following the method presented in [47]. An incompressible approach is taken. Freestream pressure and static pressure at the outlet are assumed to be equal. The resulting force is computed using Equations (15) and (16). Equation (15) calculates the mass flow ($\dot{m}$) inside the duct based on the cross-section in the nozzle (A_{nozzle}), the pressure loss along the exchanger (Δp_{hex}), the static pressure loss (ξ_p , set to 0.15), and the freestream speed and density (U_∞ , ρ). Additionally, the gross thrust loss factor (C_{fg} , set to 0.98), accounting for duct losses, is introduced in the drag estimation (D_{duct}) in Equation (16). This force increment is directly included with the aircraft total drag estimated in Section 2.2.

$$\dot{m} = A_{nozzle} \cdot \sqrt{\frac{\rho^2 \cdot U_\infty^2 + 2 \cdot \rho \cdot \Delta p_{hex}}{1 + \xi_p}} \quad (15)$$

$$D_{duct} = \dot{m} \cdot (U_\infty - \frac{C_{fg} \cdot \dot{m}}{A_{nozzle} \cdot \rho}) \quad (16)$$

2.7. Failure cases and certification

The reference aircraft type is certified according to CS/FAR 25 regulation, however, no established certification framework or specific allowed means of compliance are currently available for potential hybrid-electric versions of the aircraft. Current CS/FAR 25 airworthiness regulation was interpreted and adapted to the ATR42-600 retrofitted design as it would happen for a new hybrid-electric aircraft seeking type certification. One of the key addressed aspects is the adaptation of the OEI condition. While OEI for a traditional twin-engine aircraft involves one engine being completely out of service, the hybrid-electric configuration, showing two branches per line, presents additional redundancy. To adopt a conservative approach, the authors have considered OEI as being equivalent to losing the whole power and thrust at one of the two propellers. For the serial-hybrid case, this translates to a simultaneous interruption of both battery and turbo generator electric branch upstream of the bus in one of the two powertrains or the failure of one line downstream (Fig. 2a). For parallel-hybrid case, it involves a simultaneous failure of one electric and one non-electric branch upstream of the gearbox of one powertrain or the failure of the gearbox/propeller of one line (Fig. 2b). For the sake of consistency, the battery pack is assumed divided into two separate sections, each supplying one powertrain of the aircraft. This compartmentalization helps ensure redundancy in the event of a power system failure.

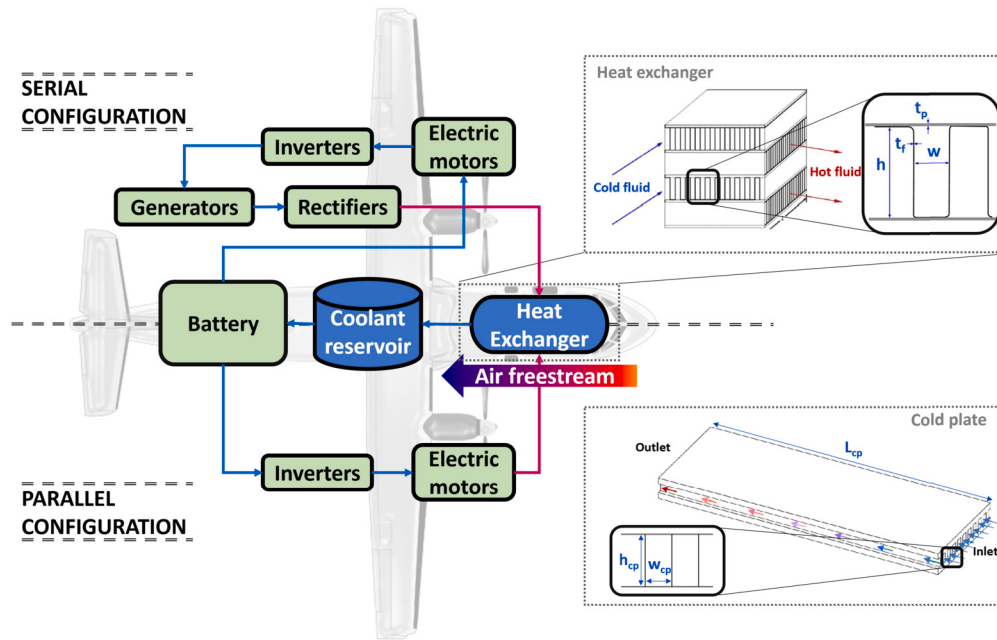


Fig. 5. Liquid cooling system scheme for serial (left-top) and parallel (left-bottom) hybrid propulsion architectures. Right-top/bottom images rearranged from [53,54,12].

Table 3

Mission defined by altitude, estimated airspeed (EAS) and vertical speed (VS) in the different phases.

Phase	Altitude [ft]	EAS [knots]	VS [ft/min]	Range [NM]
Take-off	[0-35]	$\geq V_2$	-	
Climb	[0-23000]	160	1000	
Cruise	23000	202.5	0.0	685
Descent	[23000-0]	240	-1500	
Diversion climb	[0-1400]	160	1600	
Diversion cruise	14000	165	0.0	100
Diversion descent	[14000-0]	240	-1500	
Holding	1500	130	0.0	
Landing	[50-0]	$\leq 1.2V_{\text{stall}}$	-	

Table 4

Comparison of predicted and publicly available ATR42-600 data [55,56].

	Published	Predicted	Deviation
TOFL (MTOW) [m]	1165	1110.4	-4.7%
LFL [m]	1126	1087.9	-3.4%
OEW [kg]	11700	11700	0%
Trip fuel [kg]	1165	1110.4	-4.7%
Total fuel [kg]	2342	2207	-5.8%
Trip time [min]	152.5	148.7	-2.5%
Wetted surface [m ²]	339.1	334	-1%

3. Case study and design application

3.1. Verification against conventional aircraft

The ATR42-600 has been selected as the reference aircraft for electrification retrofit. Initially, the platform is executed in a MDA mode to assess the propulsion, aerodynamic, and weight modules, as well as the analysis routines prediction capability against conventional ATR42-600 data [55,56]. During this phase, a slight deviation between the published and predicted OEWs of the ATR 42-600 was identified. To address this inconsistency, a correction factor of 0.903 is introduced for the weight of structural components. The discrepancy is attributed to the extensive application of composite materials in the aircraft (approximately 19% of weight), whereas the handbook formulae used for predictions are based on metallic aircraft. Following this calibration, the MDA is re-executed with a design payload of 4560 kg on a reference mission of 685 nautical miles (NM), characterized by the parameters listed in Table 3. Table 4 shows the MDA result validation against published data [55,56]. The data demonstrates a strong correlation, indicating that the platform and models are deemed suitable to serve as a foundation for the hybrid-electric retrofit of the aircraft.

3.2. Hybrid-electric retrofit campaign

3.2.1. Multidisciplinary analyses

Both serial and parallel hybrid powertrain architectures (Fig. 2) are integrated in the ATR42-600 airframe. Fig. 6 shows the Input and Output (I/O) interface of the MDA. The most relevant MDA inputs, are:

- Mission specifications: range, altitude, rate of climb, speed or duration for all the segments.
- Geometry parameters: wing planform and geometry, fuselage dimensions, tail geometry.
- Aerodynamic parameters: $C_{L_{TO}}^{\max}$ and $C_{L_{LA}}^{\max}$, i.e., the maximum lift coefficient in TO and landing configurations, respectively.
- Operative specifications: MTOW, MLW, maximum number of passengers, max payload, number of crew members (needed for the estimation of OEW) and dive speed (needed for fuselage structural weight estimation).
- Propulsive specifications: power rating and technological level of the powertrain components; battery weight; propeller diameter; PSFC of the combustion engine; hybridization factor for different mission segments.
- TMS specifications: thermal properties of the liquid coolant, geometry and material characteristics of ducted heat exchangers and cold plates, coolant flow rate and initial temperatures [47].

The analysis computes the following output:

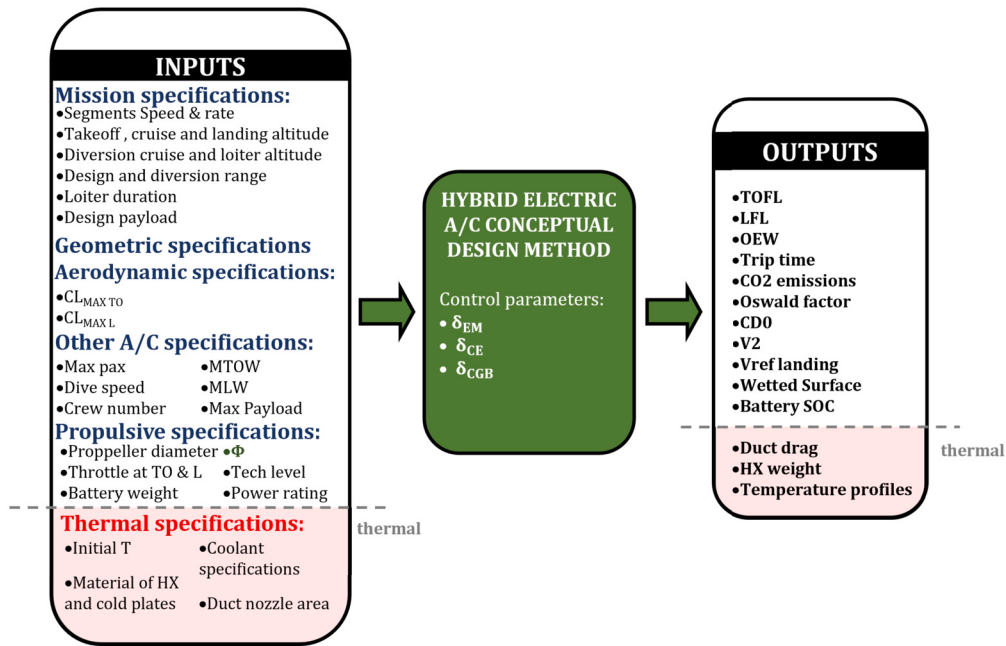


Fig. 6. Relevant inputs and outputs of the MDA.

Table 5
Components technical parameters [13,45,58–62].

		Ψ_P [kW/kg]	Ψ_E [Wh/kg]	η	T_{limit} [°C]	Reference
Electric machines	Battery	1	500-1500	0.95	60	[58,59,13]
	Converters	19	-	0.99	150	[60,59,13]
	Electric motors & generators	13	-	0.96	180	[60,61,13]
Thermal/mechanic machines	CE	3.79	-	-	-	[45]
	CGB	Eq. 17	-	0.98	-	[62]

- Mission performances: TOFL, LFL, Mission duration and relevant speeds (e.g., V_1 , V_2 for the TO phase); climb angles in the certification segments.
- Aerodynamics: the polar parasitic drag C_{D0} and Oswald factor, e , as they are calculated from the aircraft geometry.
- Weights: the OEW and the fuel burn W_f along the mission.
- Batteries: SoC of the battery during the mission.
- For the cases with TMS: temperature profiles of the electric components along the mission, the drag induced by the duct design and the weight of the heat exchanger.

When performing the MDO, combination of inputs and outputs can be used as DV and constraints, respectively.

One of the primary challenges for viable and feasible aircraft electrification is the limited specific energy of current batteries. Projections indicate a substantial enhancement of battery performance in the coming years, and the same trend is expected for other electrical machines as well [57]. All these specifications defining performance associated with a certain technology can be referred to as the *technology level* of a component, at a specific time frame. The year 2035 is frequently regarded as a foreseen time horizon, when electric components can reach a level of maturity supporting the widespread adoption of powertrain electrification in regional aircraft [4]. This study aligns the technological level of the components with the forecasts for the year 2035. Table 5 lists the specific power (Ψ_P), specific energy (Ψ_E), and efficiency (η) values for the electric machines considered in the analysis. Maximum temperature limits of each electric component which are technology-level dependent are shown in Table 5 as well.

The weight-to-power correlation for the thermal-mechanical elements, i.e., the combustion engine (CE), is derived from the general expression provided in [45]. This expression is corrected to match the

dry weight of the PW127M engine, published in literature [63]. The weight of the CGB is estimated using an exponential correlation with the gear box maximum power as shown in Equation (15). This correlation was derived fitting literature data [64].

$$W_{CGB} = 10^{0.5847 \cdot \log(P_{CGB} [\text{hp}]) + 0.3486} [\text{lb}] \quad (17)$$

3.2.2. Optimization campaign

To explore the feasibility of the hybrid-electric retrofit, a design of experiment (DoE) was conducted. Each point within the DoE represents an optimization process. The top-level parameters considered for the DoE are:

- Range of the nominal mission (test case mission detailed in Table 3).
- Serial- or parallel-hybrid architecture (and conventional powertrain for reference purposes).
- Specific energy of the battery (Ψ_E).
- Integration of a TMS.

Each case within the DoE is identified by a code, as explained in Table 6. This nomenclature is used to distinguish between different scenarios, encompassing serial or parallel hybrid powertrain architectures, variable or set Φ values throughout the mission ($\Phi_{nom} \in x$, $\Phi_{ext} = 0$), and specific energy values for the batteries. For example, the case “S1D0” identifies an optimization aimed at minimizing fuel consumption with a serial hybrid configuration, where Φ is a design variable only for the nominal part of the mission, a fixed payload of 3 tons is assumed, and the battery system has a specific energy of 500 Wh/kg (Ψ_E). The cases denoted as C1 and C2, optimizing the conventional aircraft, serve as reference performances for the hybrid-electric retrofit.

Table 6
Test campaign identifier.

Powertrain	Case	Optimization details
Conventional	C1	$f(\mathbf{x}) = 1/\text{Payload}$
	C2	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \text{Payload} = 3 \text{ [t]};$
Serial	S1	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi \in \mathbf{x}; \text{Payload} = 3 \text{ [t]}; \Psi_E = 500 \text{ [Wh/kg]}$
	S2	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi \in \mathbf{x}; \text{Payload} = 3 \text{ [t]}; \Psi_E = 1500 \text{ [Wh/kg]}$
	S1D0	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi_{\text{nom}} \in \mathbf{x}, \Phi_{\text{ext}} = 0; \text{Payload} = 3 \text{ [t]}; \Psi_E = 500 \text{ [Wh/kg]}$
	S2D0	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi_{\text{nom}} \in \mathbf{x}, \Phi_{\text{ext}} = 0; \text{Payload} = 3 \text{ [t]}; \Psi_E = 1500 \text{ [Wh/kg]}$
Parallel	P1	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi \in \mathbf{x}; \text{Payload} = 3 \text{ [t]}; \Psi_E = 500 \text{ [Wh/kg]}$
	P2	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi \in \mathbf{x}; \text{Payload} = 3 \text{ [t]}; \Psi_E = 1500 \text{ [Wh/kg]}$
	P1D0	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi_{\text{nom}} \in \mathbf{x}, \Phi_{\text{ext}} = 0; \text{Payload} = 3 \text{ [t]}; \Psi_E = 500 \text{ [Wh/kg]}$
	P2D0	$f(\mathbf{x}) = W_{\text{fuel}}(\text{nom.}); \Phi_{\text{nom}} \in \mathbf{x}, \Phi_{\text{ext}} = 0; \text{Payload} = 3 \text{ [t]}; \Psi_E = 1500 \text{ [Wh/kg]}$

Table 7
DV's and constrains of the optimizations.

Design variables
W_{battery}
$P_{\text{elect. machines rating}}$
$P_{\text{mech. machines rating}}$
Φ
Thermal
Duct area nozzle
Coolant reservoir mass
Number of HX channels
Scalar constrains
$R_{\text{TOW}} = \text{TOW} - W_{\text{fuel}} - \text{OEW} - \text{Payload} - W_{\text{batt}} \geq 0$
$\text{TOFL} \leq 1110.4 \text{ m}$
$\text{LFL} \leq 1087.9 \text{ m}$
γ : Table 1
$\text{SoC}_{\min} \geq 0.2$
Vector constrains
$P_{\text{OEI TO}} \leq 1.0 \cdot P_{\text{rating}}$
$P_{\text{AEO TO, climb, div. climb}} \leq 0.9 \cdot P_{\text{rating}}$
$P_{\text{rest of segments}} \leq 0.89 \cdot P_{\text{rating}}$
Thermal
$T_{\text{elect. machines}} \leq T_{\text{limit}}$ Table 5

Table 7 lists the design variables and constraints for the optimization cases. TOFL and LFL values are based on the performance data for the traditional ART42-600 configuration, as indicated in Table 4. The climb angles (γ) conform to the requirements detailed in Table 1. As shown, there is a consistency check on the TOW (R_{TOW}).

Energy and power constraints have been conservatively defined for the no-failure conditions ($P_{\text{AEO TO, climb, div. climb}}$ and $P_{\text{rest of segments}}$), with these components not allowed to operate at more than 89% or 90% of their maximum power. Only in case of one engine inoperative the powertrain components can operate at their maximum power ($P_{\text{OEI TO}}$). Similarly, for maximizing durability, battery SoC should not fall below 20% at the end of the mission, which implies a maximum discharge of 80%.

When considering the TMS, additional constraints have been imposed on the temperatures of all electrical components, as detailed in Table 5.

All optimizations were conducted using the Sequential Least Squares Programming (SLSQP) algorithm, which is implemented in *OpenMDAO ScipyOptimizeDriver* and is based on the *Scipy* library. Typical used tolerance for the optimization was set to 10^{-6} . The nonlinear solver employed in *OpenMDAO* uses the Newton Solver with backtracking line search, enhancing robustness in the optimization process; tolerance for convergence was set to 10^{-7} . The linear systems were solved with *OpenMDAO* direct solver.

4. Results and discussion

This section describes and discusses the findings from the optimization campaigns. To enhance the discussion and facilitate the comparisons of different campaign results, two supplementary metric parameters have been defined. With reference to Equation (18), the fuel burn

and the emitted CO_2 (along the nominal part of the missions), normalized by kg of payload and km of range, provide a mean of comparison to assess the environmental impact of retrofitted hybrid/electric regional aircraft against the service viability (kg of passengers and goods transported per km).

$$\theta_{W_{\text{fn}}} = \frac{\text{Trip fuel [g]}}{\text{Payload [kg]} \cdot \text{Range [km]}} \quad \theta_{W_{\text{CO}_2\text{n}}} = \frac{\text{Trip CO}_2 \text{ [g]}}{\text{Payload [kg]} \cdot \text{Range [km]}} \quad (18)$$

Two primary optimization campaigns have been performed. The first campaign entails the selection of a reference mission with a range of 685 NM. In the second campaign, the mission range is varied within 200 to 500 NM, with the aim of exploring the retrofit potential across a spectrum of feasible missions.

4.1. Hybrid-electric retrofit performing the reference mission

Among the substantial volume of data generated during the initial optimization campaign, Table 8 lists mass breakdown, including the OEW – which encompasses the powertrain systems excluding batteries – battery weight, payload and the weight of fuel and global emitted CO_2 over the nominal mission (excluding the diversion mission). Additionally, the metric parameters defined in Equation (18) are presented. Table 9 shows a comprehensive breakdown of the powertrain system weight for different hybrid-electric retrofits. Cases C1 and C2 represent the optimizations for the non-retrofitted aircraft when payload is maximized (4723 kg) and fixed at 3000 kg, respectively. The selection of the 3000 kg of payload for Case C2 represents a compromise between economic viability and the optimizer intention to reduce payload for the hybrid-electric cases. Case C1, maximizing the payload, results in a reduction of θ_W indicators. The respective C1 and C2 optimization outcomes serve as a term of comparison for the hybrid-electric retrofit solutions. Data of Table 8 and Table 9 illustrate that even if less fuel is burned, if the electric technology lacks weight efficiency as for the case P1, a payload reduction is needed to accomplish a particular mission, resulting in an overall worsened performance of the specified indicators in Equation (18). Case C2 scales the TOW for a fixed payload of 3000 kg and the reference mission. This approach results in an inefficient use of the airframe, as the TOW is smaller than the MTOW (18600 kg). Nonetheless, it provides a benchmark for hybrid-electric retrofits, embarking the same payload.

Fig. 7 shows the distribution of the hybridization factor Φ throughout the mission as function of different powertrain configuration and battery technologies. These plots display the optimal fuel and electricity usage within each flight segment of the extended mission. The observed trends reflect the decisions made by the optimizer for each powertrain architecture, based on the relative efficiency of fuel versus batteries and the feasibility of allocating electric energy during diversion missions.

Upon preliminary examination, flights characterized by higher values of battery specific energy Ψ_E are linked to a more pronounced degree of electrification (higher Φ), resulting in diminished levels of fuel consumption and emissions (Table 8).

Table 8

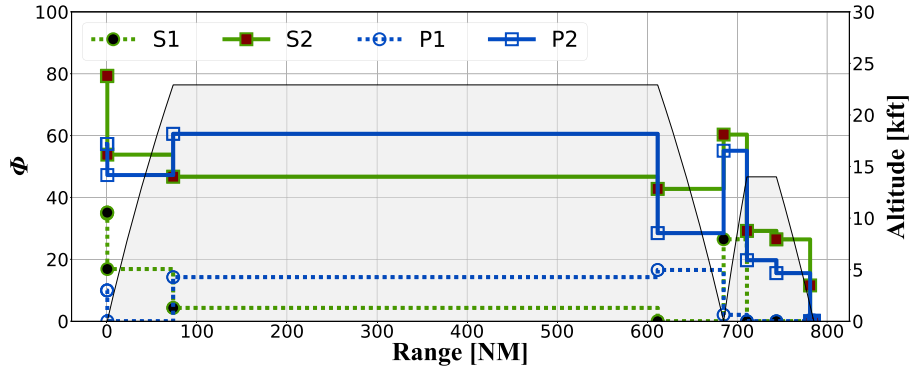
Results of the optimizations for the reference mission. No TMS considered.

	TOW [kg]	W_{batt} [kg]	OEW [kg]	Payload [kg]	Fuel W_{fn} [kg]	$\theta_{W_{\text{fn}}}$ [°]	CO ₂ W_{CO2n} [kg]	$\theta_{W_{\text{CO2n}}}$ [°]
C1	18600	-	11700	4723	1627	0.271	6118	1.02
C2	16803	-	11700	3000	1582	0.416	5948	1.56
S1	18600	1197	12176	3000	1670	0.439	6430	1.69
S2	18600	2663	11584	3000	934	0.245	4652	1.22
P1	18600	1742	11842	3000	1465	0.385	5797	1.52
P2	18600	3022	11415	3000	733	0.193	4134	1.09

*Units: $\frac{\text{g}}{\text{kg} \cdot \text{km}}$.**Table 9**

Powertrain weight breakdown.

[kg]	W_{batt}	W_{CEs}	W_{nc}	W_{EMs}	$W_{\text{Rectifiers}}$	$W_{\text{Inverters}}$	$W_{\text{Generators}}$	W_{CGBs}
S1	1197	707	820	287	133	198	202	-
S2	2663	395	672	287	74	198	113	-
S1D0	649	960	939	287	180	198	274	-
S2D0	889	971	944	287	182	198	277	-
P1	1742	885	243	29	-	20	-	194
P2	3022	420	478	168	-	116	-	194
P1D0	1721	903	236	25	-	17	-	194
P2D0	2120	926	348	91	-	63	-	194

**Fig. 7.** Φ throughout the reference mission.**Table 10**

Selected results for design range 685 NM with TMS. The differences when excluding the TMS are enclosed in parentheses.

	W_{batt} [kg]	OEW [kg]	Payload [kg]	Fuel W_{fn} [kg]	$\theta_{W_{\text{fn}}}$ [°]	CO ₂ W_{CO2n} [kg]	$\theta_{W_{\text{CO2n}}}$ [°]
S1	656 (-45.2%)	12629 (+3.72%)	3000	1728 (+3.47%)	0.4540	6590 (+2.48%)	1.731
S2	1023 (-61.58%)	12564 (+8.46%)	3000	1438 (+53.9%)	0.378	5898 (+26.7%)	1.550
P1	1713 (-1.65%)	11867 (+0.21%)	3000	1467 (+0.13%)	0.385	5804 (+0.12%)	1.525
P2	2283 (-24.45%)	11887 (+4.13%)	3000	921 (+25.6%)	0.242	4572 (+10.6%)	1.201

*Units: $\frac{\text{g}}{\text{kg} \cdot \text{km}}$.

4.1.1. Optimization considering TMS

The impact of the TMS, which is necessary to maintain temperatures of the electric components below critical limits, is presented in Table 10 for the cases S1, S2, P1, and P2. Relative differences from analogous optimization results, neglecting the TMS, are quantified as percentages enclosed in parentheses. As anticipated, integrating TMS entails a notable increase in fuel consumption and emissions, ascribed to the additional weight and heightened aerodynamic drag introduced by the system. The penalties are more pronounced for the serial-hybrid architecture, while increasing the battery specific energy, results in a heavier TMS and notably related higher fuel consumption and CO₂ emissions than corresponding cases without TMS. Nevertheless, the presence of the TMS nudges the optimal solution toward reduced reliance on electric energy in favor of fuel further highlighting electric powertrain current technology limits (Table 10).

Fig. 8 depicts operative temperature trends along the mission for different powertrain components including limit temperatures (red line).

Maximum temperature is reached by almost each electric component at the end of climbing phases being the TMS sizing conditions (Fig. 8). Whereas the rectifier is the critical component sizing the TMS for serial powertrains, batteries are driving the TMS weight penalty for parallel configurations.

4.1.2. Reference mission campaign discussion

The optimization process seeks the ideal usage of electric energy, striking a balance between the inherent inefficiencies in using the battery as a source of energy instead of fossil fuel and reducing the aircraft CO₂ emissions. Concurrently, it optimizes the powertrain sizing to fulfill performance constraints.

Serial- vs parallel-hybrid In all the proposed scenarios, the parallel-hybrid architecture consistently outperforms the serial-hybrid counterpart (Table 8). The performance difference tends to be more pronounced, in cases with higher battery specific energy. This result is consistent with previous studies such as [13,7], which also noted su-

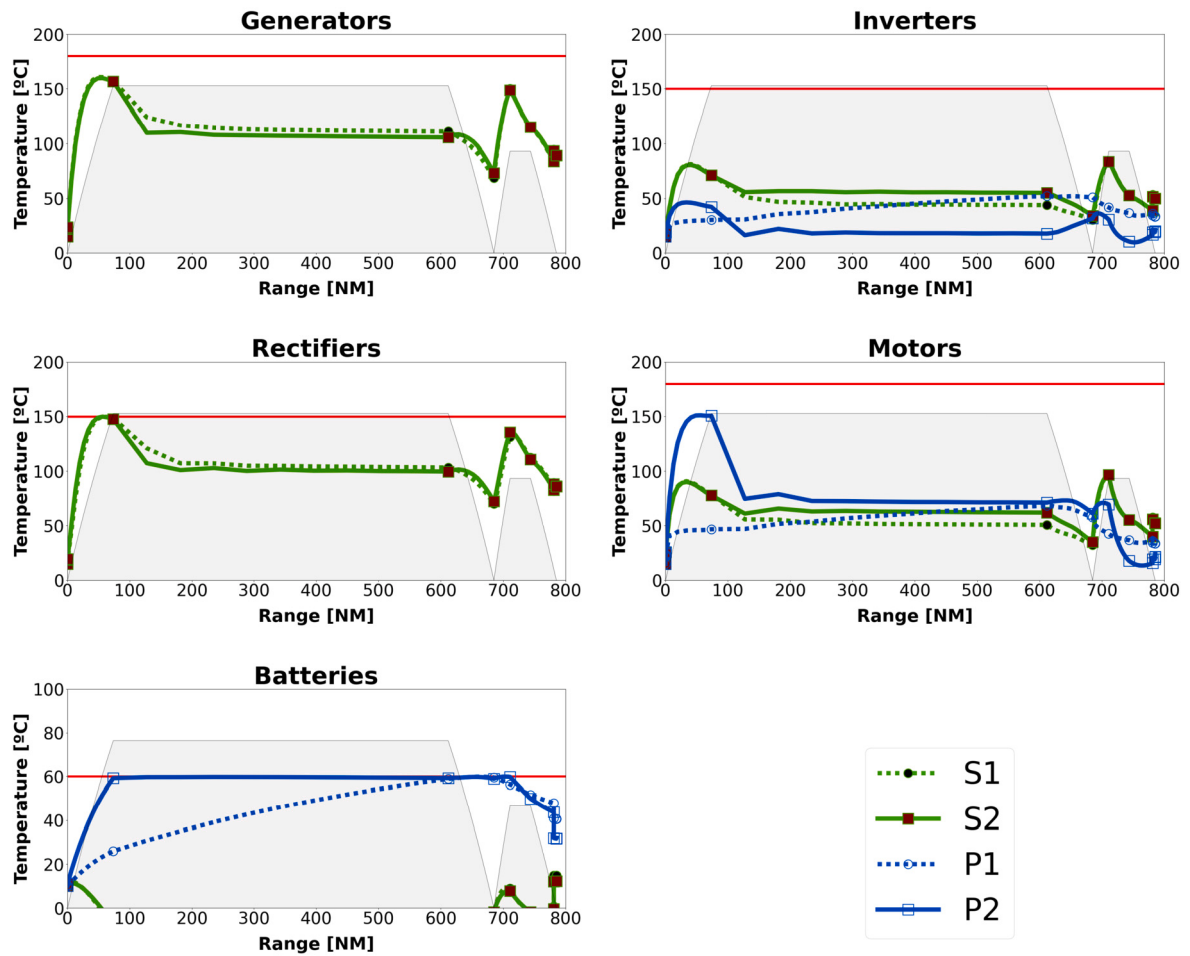


Fig. 8. Temperature profile for powertrain components (685 NM range case).

perior performance exhibited by the parallel-hybrid architecture across a broad spectrum of applications.

In comparison to Case C2, the serial-hybrid retrofit S1, which assumes the least optimistic specific energy projection by the year 2035, is unlikely to yield reductions in fuel consumption and CO₂ emissions, even without accounting for the penalties associated with the TMS. Only with the most optimistic battery technology projection, S2, the serial-hybrid architecture is expected to achieve a reduction in fuel consumption (Table 8). Conversely, the parallel-hybrid retrofit exhibits potential for reducing fuel consumption and CO₂ emissions, even when the lowest battery specific energy is considered. The reduced efficiency of the serial-hybrid architecture becomes evident when examining the mass breakdown listed in Table 9. A comparison between serial and parallel counterparts, such as S1 and P1, reveals that the weight of the electric portion of the powertrain (excluding batteries), denoted as W_{nc} , is lower for the parallel-hybrid architecture. This can be partially ascribed to a reduced number of components. It is interesting to note that, for the serial-hybrid retrofit, lower values of W_{nc} are observed with higher-technology batteries (S2 vs. S1), while the opposite holds true for the parallel-hybrid case (P2 vs. P1). This phenomenon can be attributed to the distinctive arrangement of powertrain components in the two proposed architectures, as illustrated in Fig. 2. In the serial-hybrid case, the weight of the electric motors remains unchanged since they are responsible for driving the propellers and the flight power requirements depend on the fixed TOW. Several components connect the combustion engine to the inverter (referred to as the thermal line). As battery efficiency increases, smaller energy is required from the thermal line resulting in lighter components driving the overall powertrain weight reduction. Conversely, in the parallel-hybrid case,

the weight of the gear-box remains unaltered, as it is primarily dependent on the total power needed to be transmitted to the propellers. However, larger inverters and electric motors are required for more efficient batteries leading to heavier electric lines and smaller thermal units.

Active constraints Modeling several airworthiness constraints with a reasonable accuracy proves to be paramount for the effective optimization application as the TOFL is an active constraint for all cases except C2. For all hybrid-electric solutions, the final SoC of the battery is an active constraint, expressing the intrinsic need to use all available electricity to minimize fuel consumption and CO₂ emissions. The climb gradients are not active constraints in this initial optimization campaign. Concerning powertrain components, the optimizer is oriented towards weight-efficiency, achieved by operating them at their maximum allowed power for the specific segment through a synergistic selection of the hybridization factor and component sizing. The behavior is a consequence of the electric component models characterised by a constant efficiency and a linear correlation between max power and weight. The complexity of the problem is further heightened by the objective function, which endeavors to increase the usage (and weight) of electric components. Consequently, altering the hybridization factor for a single segment has a notable impact on the overall optimal sizing of the powertrain.

Gradient-based optimization and local minima It is well known that gradient-based optimization is inherently prone to find local minima that would possibly invalidate some of the conclusions. Preliminary global search investigations, taking into account the impact of numer-

Table 11

Results of the additional optimizations for the reference mission. No TMS considered.

	TOW [kg]	W_{batt} [kg]	OEW [kg]	Payload [kg]	Fuel W_{in} [kg]	$\theta_{W_{in}}$ [°]	CO ₂ W_{CO2n} [kg]	$\theta_{W_{CO2n}}$ [°]
S1D0	18600	649	12656	3000	1701	0.447	6503	1.71
S2D0	18600	889	12678	3000	1447	0.380	5888	1.55
S1_{tocl}	18600	1124	12238	3000	1675	0.440	6445	1.69
S2_{tocl}	18600	1176	12239	3000	1645	0.432	6370	1.67
P1D0	18600	1721	11861	3000	1465	0.385	5799	1.52
P2D0	18600	2120	12005	3000	941	0.247	4605	1.21
P1_{tocl}	18600	1840	11751	3000	1485	0.390	5849	1.54
P2_{tocl}	18600	1877	11785	3000	1461	0.384	4605	1.21

*Units: $\frac{g}{kg \cdot km}$.

ical errors and the effect of the initial conditions, indicate that for the considered optimization cases and relative design spaces, the problem seems to exhibit convexity.

Convenience of the hybrid retrofit The feasibility and viability of a hybrid retrofit cannot be solely assessed based on fuel consumption reductions, especially if such reductions are achieved at the expense of a reduced payload. If a fixed demand for air mobility is considered, aircraft can either operate a higher number of flights carrying a smaller payload or a lower number of flights with a larger payload to transport the same total number of passengers. The primary objective is to minimize the net environmental impact in terms of CO₂ emissions. This highlights the fact that the metrics in Equation (18) are closely correlated with the environmental impact of a viable aviation service, as they normalize fuel consumption and CO₂ emissions against the transported payload and the flown range. Observing the last column of Table 8, listing $\theta_{W_{CO2n}}$, it becomes evident that none of the provided retrofitted solutions achieves a specific pollution index lower than that of the aircraft C1, a conventional powertrain aircraft with the maximum payload for the selected range. In other words, if the same number of passengers must be transported, all hybrid-electric retrofitted solutions result in higher CO₂ emissions. Only the parallel-hybrid one, boarding battery technology with 1500 Wh/kg of specific energy, manages to deliver similar performance. Comparing these results with the specific fuel burn, $\theta_{W_{in}}$, it seems that, with the highest level of battery technology, the parallel-hybrid retrofit can outperform the traditional powertrain case C1. This apparent contradiction is explainable by reminding that CO₂ emissions take into account not only emissions from fuel combustion but also those generated by the production of fuel and the electricity to recharge the batteries, as discussed in Section 2.4.

Battery technology advances must exceed even the most optimistic specific energy predictions (1500 Wh/kg) for a hybrid-electric retrofit of an ATR42-600 flying the reference mission to have a sensible reduction on CO₂ emissions.

Effects of the TMS The conclusions drawn thus far have been obtained without accounting for the weight penalties imposed by the TMS. These penalties are more substantial for the serial-hybrid retrofit, as it inherently incorporates more electric components in the powertrain line. Overall, the parallel-hybrid architecture exhibits superior performance. As the degree of hybridization increases due to more advanced batteries, the impact of the TMS becomes more pronounced, owing to the elevated heat generation requiring dissipation. For the serial-hybrid configuration, the critical component in term of heat generation is the rectifier (Fig. 8), which must be cooled before the liquid is refreshed in the heat exchanger. The highest temperature is reached at the top-of-climb. In the parallel-hybrid configuration, the critical component is the battery pack, even though it is the first component to be cooled. In the case P2, which relies heavily on electric energy, the entire cruise and the diversion climb are conducted at the highest acceptable temperatures (Fig. 8).

Misconceptions on energy allocation The approach of not constraining a priori the design space with predetermined hybridization factor se-

lection but allowing for an extensive exploration of the potential of a hybrid-electric retrofit has yielded valuable insights, some of which may initially appear counterintuitive. The first insight pertains to an unexpected optimal distribution of electric energy throughout the mission. It is of practical relevance to minimize fuel consumption only on the nominal segments of the mission since diversion missions are infrequent. Therefore, the temptation to avoid allocating electric energy in the diversion phase arises investigated by the cases SXD0 and PXD0. However, as evident from Tables 8 and 11, such a choice leads to higher fuel consumption (measured over the nominal part of the mission). Fig. 9a) show the hybridization factor trends throughout the mission when using electric energy only on the nominal mission. It is evident, from a comparison with Fig. 7, that the advantage of using electric energy during the diversion phases increases the hybridization factor during the nominal phases of the mission. Among the possible explanations are the potential to operate the diversion part of the mission with a lighter aircraft (resulting in reduced fuel consumption) and the use of electric power in phases where it is more efficient, such as during the diversion climb. This statement is substantiated by observing the active constraint of the optimization. As general trend, not using electric energy in the diversion mission, the diversion climb is the real sizing mission segment for the combustion engine, as it is the only one where it works at its maximum allowable regime (90%). With the need of carrying such a larger combustion engine, the optimizer has less room to increase battery weight and, hence, electric energy use.

A second lesson learned challenges the idea that the most efficient approach to design hybrid-electric aircraft is to use batteries exclusively during take-off and climb phases. A series of parallel optimizations has been conducted, allowing changes in the hybridization factor only during the take-off and climb phases in both nominal and diversion segments (SX_{tocl} and PX_{tocl}). The results are summarized in Table 11, and the distribution of the hybridization factor is shown in Fig. 9b. For both architectures and battery technology levels, the optimal strategy involves allocating a portion of the battery energy during the cruise phase. The drawbacks of allocating battery energy solely during take-off and climb phases become particularly pronounced in case of the highest level of battery specific energy. In fact, a large battery weight is required to comply with the energy demand of take-off and climb phase which acts as ballast for the other mission phases. The optimization campaign results prove that the option of using batteries exclusively during take-off and climb phases is not feasible for retrofitted aircraft.

The need for alternative gas turbines design The results presented so far have implicitly assumed that the engine operates at all power regimes maintaining an almost constant value of the PSFC averaged on the gas turbine operational regime. However, this assumption is non-conservative and tends to lead to optimistic solutions. In traditional aircraft, gas turbines are sized to provide the maximum thrust required and are optimized to offer the lowest PSFC under cruise reference conditions or specific throttle settings [65]. Operating far from the optimal throttle setting typically results in increased PSFC. This argument challenges the misconception that designing a hybrid-electric aircraft involves using available commercial smaller thermal units that work

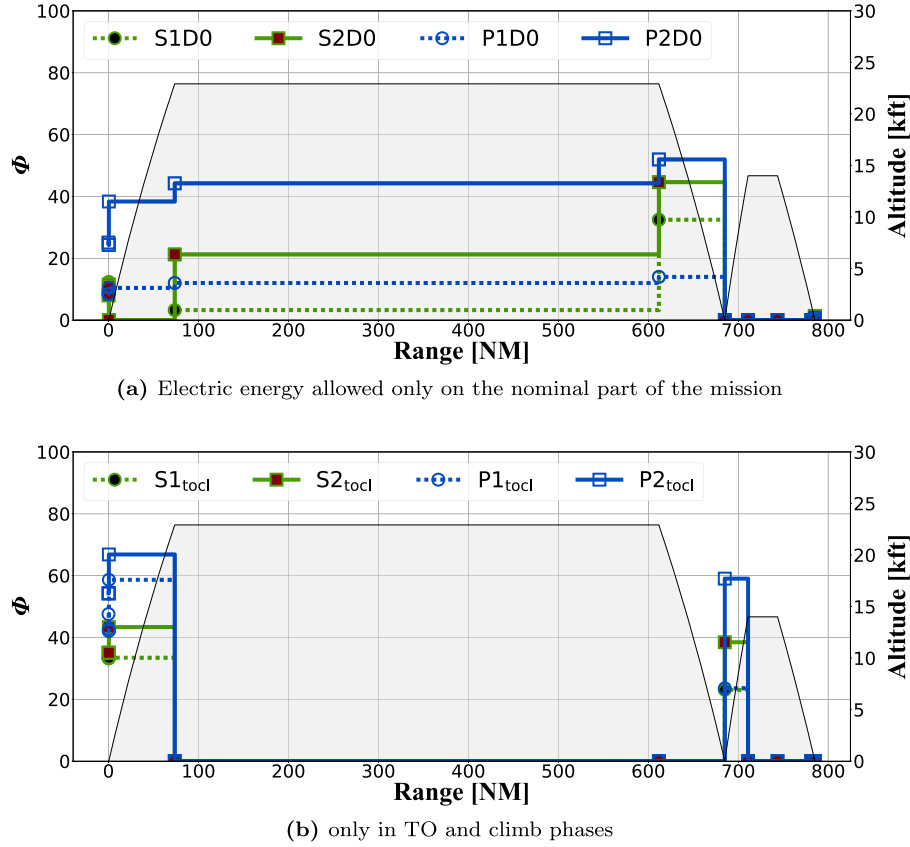


Fig. 9. Φ throughout the reference mission. Electric energy allowed only on the nominal part of the mission (top) or only in TO and climb phases (bottom).

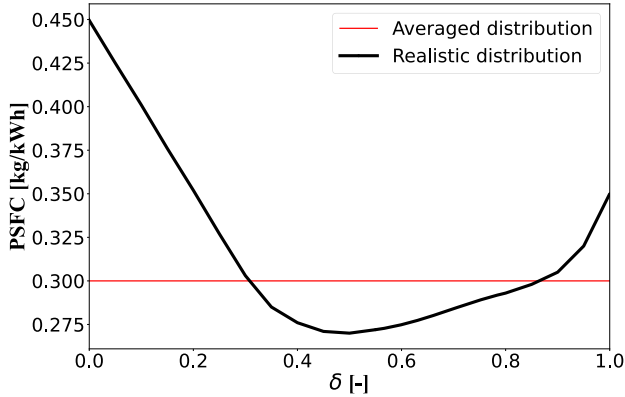


Fig. 10. PW127 gas turbine PSFC vs throttle (realistic distribution data from [65]).

consistently at their maximum power, with any additional power needs being supplied by the batteries. To address this issue, additional optimizations were conducted considering a simplified PSFC dependency from the throttle settings, characteristics of the considered PW120 turboprop as described in reference [65] (see Fig. 10). This engine exhibits the lowest PSFC at approximately 50% of the throttle. Optimization results for variable PSFC are summarized in Table 12, indicated with the subscript *psfc*. Fig. 11 shows the gas turbine throttle settings throughout the mission for both constant and variable PSFC cases. Notably, in case of the variable PSFC, the throttle settings in cruise are closer to the values yielding the lowest PSFC. Instead, for constant PSFC, the throttle settings are notably higher. The optimal strategy seeks to strike a balance between various competing factors, including: (a) smaller thermal units are lighter, but operating them at higher power settings would

result in higher PSFC; (b) lower battery efficiencies necessitate greater reliance on fuel; (c) the hybridization factor for TOFL (and generally, climb gradient assessments) determines the relative sizes of the thermal units. Finding the right trade-off among these factors is critical for the overall efficiency and performance of the hybrid-electric aircraft. It is plausible to argue that, in the context of hybrid-electric aircraft, the adoption of an innovative thermal unit design that achieves the lowest PSFC at higher throttle settings would yield significant advantages.

4.2. Effects of the design range

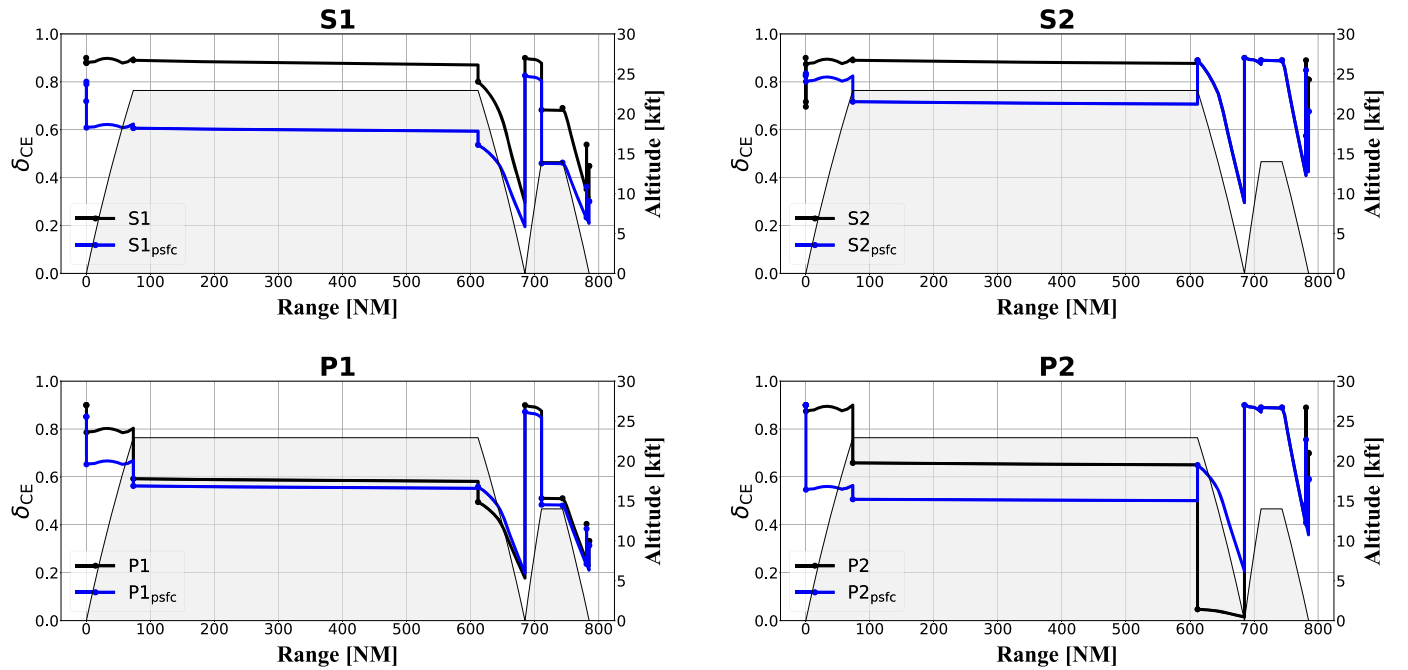
The optimization campaign for the reference mission range of 685 NM yielded discouraging results. Nevertheless, the retrofit with a hybrid-electric solution has the potential of reducing the environmental impact for shorter ranges. To investigate these potential improvements, the optimization campaign has been reiterated for a series of shorter ranges. Table 13 lists the outcomes pertaining 200 NM, 300 NM and 500 NM ranges, for the reference aircraft (C), and serial (S1/S2) and parallel (P1/P2) hybrid retrofitted cases. For the sake of comparison, the TOW was maintained at the MTOW, thus yielding, in certain conditions, a payload exceeding the selected reference of 3 tons. For the two non-hybrid reference cases, C1 achieved the maximum payload, albeit with a TOW marginally below the MTOW for the range of 300 and 200 NM, whereas C2, characterized by a fixed payload of 3 tons, exhibited a considerably lower TOW.

Observing the fuel burn trends, it becomes evident that the two battery technology levels yield distinct scenarios. In Scenario 1, representing a pessimistic outlook for battery technological advancements, even for the shortest range of 200 NM, the optimizer fails to identify a fully electric solution. Consequently, the optimizer achieves a reduction in fuel consumption by maintaining the TOW at the MTOW value and prioritising battery use. At a range of 500 NM, the serial-hybrid option consumes approximately the same amount of fuel as a conven-

Table 12

Results of the optimizations with a variable PSFC - power curve. No TMS considered.

	TOW [kg]	W_{batt} [kg]	OEW [kg]	Payload [kg]	Fuel $\overline{W}_{f_n}$ [kg]	$\theta_{W_{f_n}}$ [°]	CO ₂ $\overline{W}_{CO2n}$ [kg]	$\theta_{W_{CO2n}}$ [°]
S1_{psfc}	18600	682	12790	3000	1561	0.410	5985	1.57
S2_{psfc}	18600	2511	11724	3000	896	0.235	4506	1.18
P1_{psfc}	18600	1828	11917	3000	1331	0.350	5312	1.40
P2_{psfc}	18600	3032	11485	3000	635	0.167	3814	1.00

*Units: $\frac{g}{kg \cdot km}$.**Fig. 11.** Gas turbine throttle along the mission.**Table 13**

Optimization results for different mission ranges. No TMS considered.

	TOW [kg]	W_{batt} [kg]	OEW [kg]	Payload [kg]	Fuel $\overline{W}_{f_n}$ [kg]	$\theta_{W_{f_n}}$ [°]	CO ₂ $\overline{W}_{CO2n}$ [kg]	$\theta_{W_{CO2n}}$ [°]
Range = 500 NM								
C1	18600	-	11700	5151	1214	0.255	4565	0.957
C2	16372	-	11700	3000	1172	0.422	4407	1.586
S1	18600	1863	12063	3000	1145	0.412	4554	1.639
S2	18600	4431	10964	3000	122	0.044	2071	0.746
P1	18600	2299	11786	3000	992	0.357	4094	1.474
P2	18600	3605	11478	3000	108	0.039	2077	0.748
Range = 300 NM								
C1	18271	-	11700	5300	757	0.257	2846	0.967
C2	15907	-	11700	3000	729	0.437	2741	1.644
S1	18600	2633	11906	3000	567	0.340	2489	1.493
S2	18600	2771	11477	4023	7.1	0.003	1127	0.504
P1	18600	2996	11649	3000	475	0.285	2232	1.339
P2	18600	2732	11440	4080	7.6	0.003	1161	0.512
Range = 200 NM								
C1	18029	-	11700	5300	527	0.268	1982	1.009
C2	15673	-	11700	3000	507	0.456	1906	1.715
S1	18600	2919	11918	3000	280	0.252	1457	1.311
S2	18600	2655	11253	4572	7.1	0.004	791	0.467
P1	18600	3395	11538	3000	217	0.195	1296	1.166
P2	18600	2727	11220	4537	7.6	0.005	815	0.485

*Units: $\frac{g}{kg \cdot km}$.

tional aircraft optimized for the same payload (C2 case). In contrast, the parallel-hybrid architecture already demonstrates a performance advantage in terms of fuel consumption and CO₂ emissions. As the range is shortened, the reduction in fuel burn and emissions compared to the C2 aircraft becomes more pronounced. In Scenario 2, representing an optimistic outlook for battery technological advancements, the optimized

hybrid-electric retrofit results in reduced fuel burn. At a range between 300 to 500 nautical miles, the optimizer advocates for all-electric solutions. Beneath this range, shorter distances would lead to lower TOW, with additional payload accommodated to maintain the MTOW. It is worth noting that, due to implementation limitations in modeling a hybrid-electric aircraft, it is not possible to achieve a solution that per-

Table 14

Selected results for design range 300 NM with TMS. The differences when excluding the TMS are enclosed in parentheses.

	W _{batt} [kg]	OEW [kg]	Payload [kg]	Fuel		CO ₂	
				W _{fn} [kg]	$\theta_{W_{fn}}$ [°]	W _{CO2n} [kg]	$\theta_{W_{CO2n}}$ [°]
S1	1770 (−32.8%)	12641 (+6.17%)	3000	641 (+13.1%)	0.3845	2683 (+7.79%)	1.609
P1	2786 (−7.01%)	11834 (+1.59%)	3000	492 (+3.57%)	0.2951	2269 (+1.66%)	1.361

*Units: $\frac{g}{kg \cdot km}$.

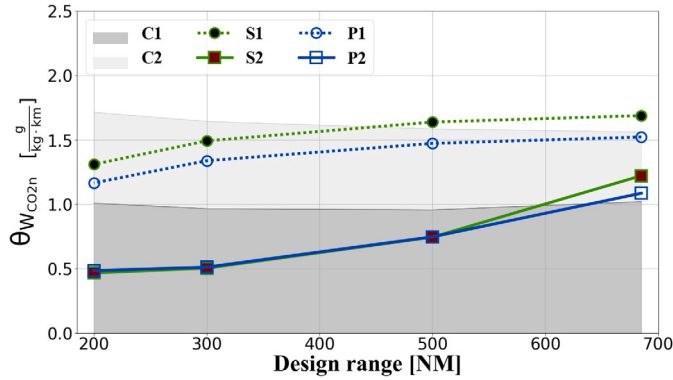


Fig. 12. Specific CO₂ emissions ($\theta_{W_{CO2n}}$) vs design ranges for different configurations.

fectly matches that of a fully electric aircraft. This explains the small amount of residual fuel burn for S2 and P2 cases in the ranges of 300 and 200 nautical miles, as shown in Table 13.

An additional perspective on this intricate scenario is given through the tracking of the metrics presented in Equation (18). The primary focus centers on the metric related to CO₂ emissions, denoted as $\theta_{W_{CO2}}$. The reference values for this metric are those corresponding to the configuration C1, representing the ATR42-600 with a conventional powertrain. Fig. 12 provides a graphical summary of the values found in the last row of Table 13.

In Scenario 1, it is noteworthy that for all ranges, both hybrid architectures exhibit poorer performance compared to non-electric aircraft. For missions spanning from 200 NM to the maximum range, around a 1 g of CO₂ is generated per unit of payload and range with the conventional powertrain designed around the reference airframe. Conversely, Scenario 2 presents a distinct outcome. In this case, both hybrid architectures lead to a reduction in CO₂ emissions per unit of payload and range compared to non-electric aircraft below a range threshold of approximately 600 NM for the serial-hybrid and 640 NM for the parallel-hybrid.

The inclusion of the TMS negatively impacts the performance of the hybrid-electric aircraft, as evidenced by the results presented in Table 14. These results pertain to a range of 300 NM and battery level according to Scenario 1. While shorter ranges typically involve lower heat generation, optimizing for shorter ranges leads to a greater use of the electric powertrain components. This second effect appears to dominate, resulting in larger OEW and related fuel consumption and CO₂ emission penalties. Fig. 13 shows the temperature trends for select powertrain components throughout the 300 NM range optimization. Once again the rectifier was the serial hybrid-electric TMS sizing component experiencing the maximum temperature at the end of the climb phase. Batteries drive the design on the TMS for parallel-hybrid powertrain but they reach their limit temperature at the end of the diversion climb segment.

5. Conclusions

The objective of this study was to assess the feasibility of retrofitting a regional aircraft with hybrid-electric powertrains using a multidisciplinary optimization platform capable of modeling the relevant physics

and exploring the ultimate potential of both serial- and parallel-hybrid solutions. Multiple optimization campaigns were conducted, which involved varying battery specific energy based on pessimistic and optimistic forecasts for the year 2035, considering different mission ranges, and accounting for the effects of the thermal management system. The results revealed that battery energy density relative to optimistic forecasts (1500 Wh/kg) can lead to hybrid-electric retrofit solutions with a up to 30% reduction impact on CO₂ emissions for a fixed payload of 3 t. However, when CO₂ emissions are normalized by a variable payload, the advantages of the retrofit are only observed for ranges smaller than 600 NM, i.e., shorter than the reference one. With battery specific energy relative to less optimistic forecasts (500 Wh/kg), the parallel-hybrid retrofit can reduce CO₂ emissions by 2.6% compared to a traditional powertrain optimized for the same mission and payload. However, for the serial-hybrid retrofit, the CO₂ emission reduction holds only for ranges below 500 NM. When CO₂ emissions are normalized by the variable payload, the traditional powertrain still produces the least pollution.

Modeling a thermal management system, essential for the safe operation of electric systems, has a negative impact on the performance of the hybrid-electric concept, resulting in extra mass and drag. TMS increases CO₂ emissions produced per kg of payload per km of range from 0.1% up to 27%. Accurate modeling of TMS is required to properly assess the feasibility of novel hybrid-electric powertrains.

Some misconceptions were refuted, and counterintuitive results provided valuable lessons learned. One of these lessons suggests that reduced CO₂ emissions can be achieved when the optimizer has the flexibility to allocate electric energy during the diversion phase of the mission, even though the metric being minimized (fuel consumption) is evaluated only for the nominal part of the mission. Likewise, a prevailing paradigm advocated by several authors, stating that smaller thermal units generally operating at their maximum power with additional power for take-off and climb phases provided by batteries are efficient for hybrid-electric aircraft, resulted not applicable for regional plane retrofit. More sustainable solutions can be found by reallocating battery energy to the cruise phase at the expense of the take-off and climb phases. Furthermore, considering realistic non-constant dependency of PSFC from relative power suggests the use of lower throttle settings as a synergistic measure to reduce fuel burn.

Future work in this area will involve the integration of Life Cycle Cost Analysis, local air quality and noise analysis, and improved system integration prediction modules. The goal is to improve the prediction capability of the current platform by providing higher level of accuracy while offering a more comprehensive assessment of the environmental impact of hybrid-electric or fully-electric aircraft. Additionally, given the uncertainties in technology development, robust optimization techniques can be valuable in providing a broader perspective and suggesting the potentials of adopting such powertrains.

CRedit authorship contribution statement

Raul Quiben Figueroa: Writing – review & editing, Writing – original draft, Visualization, Validation, Methodology, Investigation, Formal analysis. **Rauno Cavallaro:** Writing – review & editing, Writing – original draft, Validation, Supervision, Software, Project administration, Methodology, Formal analysis, Conceptualization. **Andrea Cini:** Writing – review & editing, Supervision, Project administration, Methodology, Conceptualization.

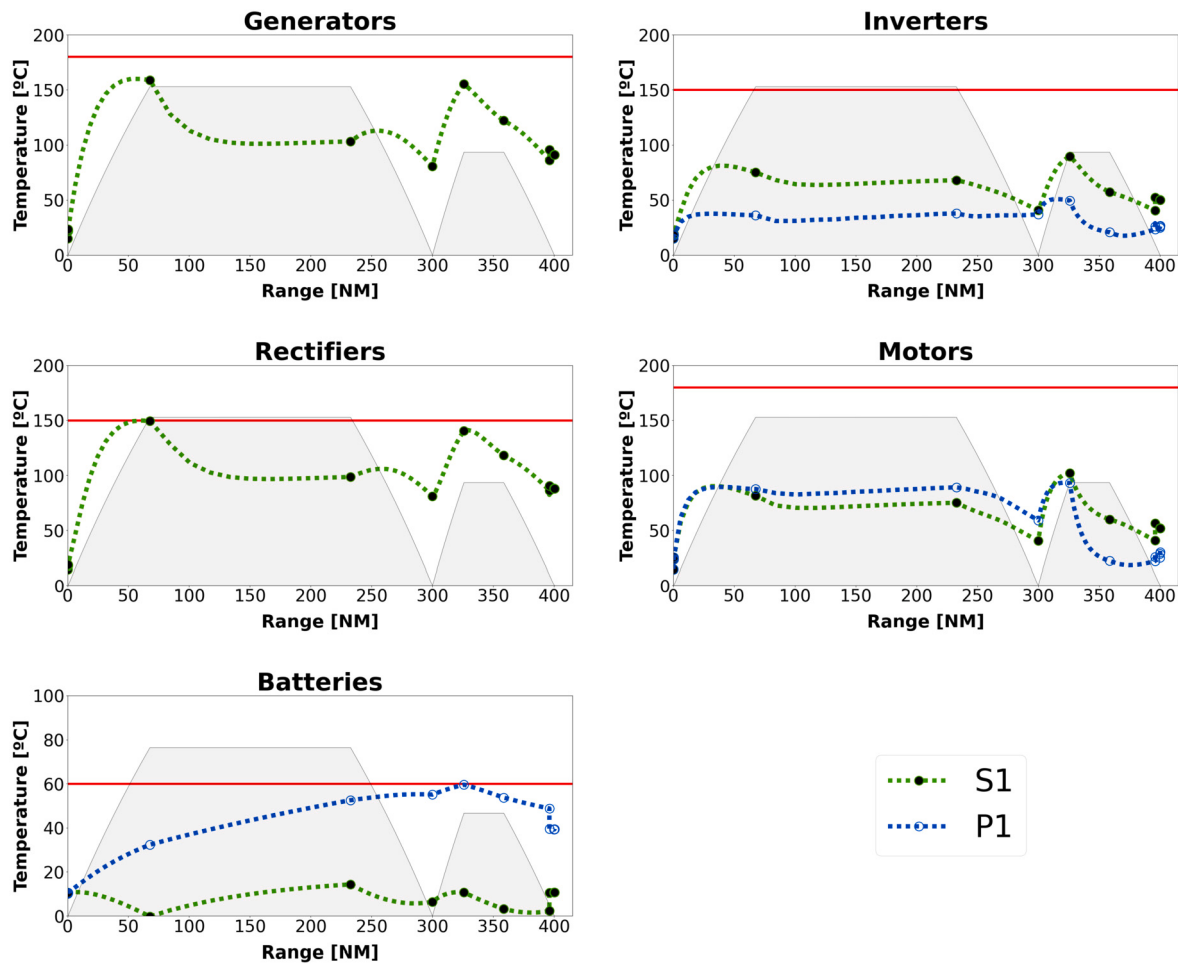


Fig. 13. Temperature profile for powertrain components (300 NM range case).

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

Data availability

The results of all shown optimization can be freely accessed at this link: <https://gitlab.com/uc3maslab/HEpaper>.

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