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Preliminary Design of Integrated Power and Thermal Management Systems for Hybrid Electric VTOL Aircraft Architecture

Hybrid and fully electric vertical takeoff and landing (VTOL) aircraft present promising solutions to improve urban traffic congestion but face challenges such as limited power and energy density and stringent thermal constraints, requiring effective thermal management. Optimizing the available onboard electrical energy in hybrid-electric aircraft is crucial, and this can be achieved through the implementation of an effective power management strategy. This paper presents a design methodology for an integrated power and thermal management system (IPTMS) using a parallel hybrid-electric civil tilt-rotor aircraft modeled after XV-15, as a case study. A multidisciplinary optimization platform is developed, integrating IPTMS optimization with rotor aerodynamics, flight dynamics, gas turbine performance, mission analysis, and electric powertrain performance models. The study evaluates both direct air-cooling and liquid-cooling options, incorporating Phase Change Materials (PCMs) for heat storage to identify the most effective thermal management solution. A design space exploration is conducted across various degrees of hybridization (DoH) to assess performance impacts both with and without the integration of the thermal management system (TMS). The results indicate that lower DoH with air-cooling TMS result in energy efficiency and emission improvement, while higher DoH configurations encounter thermal load and payload constraints. For shorter-range, double-leg missions, air-cooling with PCMs proved beneficial, achieving up to 8.76% improvement in energy efficiency and emission reductions of 12.95% for CO₂ and 1.66% for NO_x. Although electrification optimizes energy use and emissions, conventional aircraft still outperform when the maximum payload constraint is lifted through enhanced payload allocation. This work provides a comprehensive framework for IPTMS design in hybrid-electric VTOL aircraft, balancing power and thermal management for efficient and sustainable operation.

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1 Introduction

The continuous rise in urban population density has led to increasing challenges, such as traffic congestion, resulting in wasted time and increased emissions. These issues disrupt public convenience and pose significant threats to environmental sustainability. One promising solution gaining attention within the aviation industry is urban air mobility (UAM), a novel concept designed to alleviate urban congestion through short-range air transportation for up to 15 passengers over distances of 50 nautical miles or less [1,2]. By employing vertical takeoff and landing (VTOL) rotorcraft configurations, UAM offers flexibility for dense urban environments. Depending on the specific mission and aircraft size, hybrid or fully electric propulsion systems can be adopted to minimize environmental impact [3]. UAM has the potential to drastically

reduce travel time within cities while also contributing to lower greenhouse gas emissions.

Despite the potential benefits of UAM, several limitations hinder its full-scale implementation, particularly regarding electric propulsion systems. While electric powertrains offer an environmentally friendly alternative, their current energy and power density remain insufficient for larger passenger capacities or extended flight durations, restricting fully electric configurations to smaller aircraft with shorter endurance [1–3]. In addition, battery technology has not yet reached the maturity needed to support the long-range capabilities of larger UAM vehicles. The energy density limitations mean frequent recharging or battery swaps may be necessary, adding operational complexity.

Insufficient energy and power density inevitably make the propulsion system heavier, and the need for effective thermal management further exacerbates this challenge for hybrid or fully electric UAM vehicles. In conventional gas turbine engines, most heat loads are released through the engine exhaust. However, electric powertrains produce low-grade waste heat that must be carefully managed to avoid overheating. This has prompted the need

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for advanced thermal management systems (TMS) to ensure electric components operate within safe temperature limits [4].

Conventional thermal management approaches, such as ram-air or liquid-cooling systems, are frequently used due to their relative simplicity and effectiveness, as highlighted in multiple studies [4–11]. Chapman et al. [4] developed a liquid-cooled TMS for three electrified aircraft propulsion concepts: a turbo-electric aircraft with an aft boundary layer propulsor designed for 154 passenger regional-range mission, a turbo-electric tiltwing VTOL aircraft for 15 passenger UAM mission, and a fully electric 48 passenger aircraft for short range mission. The study introduced general sizing relationships for such systems. Later research expanded on this by optimizing a liquid-cooled TMS for a turbo-electric tilt-wing aircraft and conducting transient analysis under specific mission scenarios, such as a four-minute hover on a hot day [5]. Similarly, the design of a liquid-cooling TMS for a six-passenger parallel hybrid electric (PHE) quadrotor was explored by employing performance maps for heat exchanger design. However, this was limited to evaluation at specific mission points without considering dynamic full-mission analyses [6].

Other notable TMS designs include Kellermann et al.'s [8] ram-air-based TMS for a 180-passenger short-range partial-turbo-electric aircraft, which integrated air-coolant heat exchangers and liquid-cooled cold plates. The study highlighted the importance of the designed TMS in performing effectively under off-design conditions (takeoff and climb), specifically regarding the puller fan's ability to handle the most demanding thermal management scenarios. Although multipoint simulations were conducted to evaluate the TMS at the aircraft level, overall performance across a complete mission was not thoroughly examined. Rheaume et al. [9–11] explored liquid-cooling TMS methodologies for PHE commercial single-aisle aircraft, focusing on optimizing fuel burn across multiple mission points. However, these studies primarily evaluated the systems under steady-state conditions, leaving dynamic mission-level analyses unexplored.

A notable gap in the literature is the limited assessments of the integration penalties associated with TMS design. For instance, Zhao et al. [12] optimized liquid-cooling TMS by focusing on minimizing weight and power consumption, but their study was confined to battery TMS. Additionally, current research lacks a thorough investigation of how power management strategies (PMS) affect TMS design despite PMS directly shaping the heat load profile throughout a mission. Ouyang et al. [13] underscored the importance of the integrated power and thermal management system (IPTMS) in their study, offering a comprehensive review of recent research on the topic. Their work highlights the increasing significance of integrating power and thermal management to enhance system performance and efficiency in advanced aircraft designs. Furthermore, comprehensive comparisons between various cooling technologies and their effectiveness in rotorcraft applications remain largely unexplored.

A design methodology for both liquid and air-cooling TMS for a PHE tilt-rotor aircraft, modeled after the NASA XV-15, is discussed in the works of the authors [14,15]. The air-cooling TMS, which incorporates a direct air-cooled cold plate and phase change materials (PCMs) for heat storage during flight phases with limited ram-air availability, showed limitations under high heat load conditions (hover and climb), particularly in cases with a high degree of hybridization (DoH) [14]. To address this, a liquid-cooling TMS was developed for the same scenario, proving to be a more efficient solution compared to the air-cooling TMS. Furthermore, designing the TMS for nominal conditions (Cruise) rather than peak conditions (Hover) resulted in a more optimized design. However, the studies focused only on specific DoH, optimizing the PMS without considering the TMS integration impact. The results revealed that TMS integration has a significant impact, emphasizing the need for a multidisciplinary optimization platform for IPTMS design. Such platform would facilitate IPTMS optimization by addressing TMS integration penalties early in the design process [15].

This study builds upon a previously developed methodology by incorporating an IPTMS into the design and optimization framework for novel civil rotorcraft applications [16]. The expanded approach aims to offer a more comprehensive preliminary design and tradeoff analysis for a PHE rotorcraft configuration, utilizing a multidisciplinary framework. Models for rotor aerodynamics and flight dynamics are adopted from the previously developed methodology. At the same time, mission analysis, gas turbine performance, and electric powertrain performance are extended, updated where necessary, and validated to incorporate the thermal management of electric powertrains. This extension ensures integrated operation and optimized performance synergy across the overall system.

The analysis focuses on exploring the design space of PHE rotorcraft architecture using a multidisciplinary framework, aiming to optimize both power and thermal management strategies. This study offers new insights and guidance for the preliminary design of IPTMS for PHE tilt-rotor aircraft.

2 Methodology

The developed multidisciplinary framework with IPTMS is outlined in Fig. 1. For a given mission profile, the aircraft system with a hybrid-electric propulsion system (HEPS) is designed based on the DoH, which includes the power ratio (H_P) and energy ratio (H_E) as defined by Schmitz and Hornung [17]. The gas turbine cycle is scaled down according to the power ratio

$$H_P = \frac{P_{\text{electric}}}{P_{\text{total}}} \quad (1)$$

$$H_E = \frac{E_{\text{electric}}}{E_{\text{total}}} \quad (2)$$

The heat load profile of the electric powertrain is determined based on the mission PMS used. The TMS is then designed at a specified design point, and its power penalty—comprising induced drag power and power off-take from auxiliary components (puller fan, pump, etc.)—is calculated. The penalty is then factored into the gas turbine performance, resulting in increased fuel consumption. This is to ensure the convergence of the electric powertrain heat load profile. The overall Take-off Weight (TOW) of the aircraft is subsequently determined, as well as the performance output for the given mission.

The modeling approach for the rotorcraft, HEPS, and TMS has been thoroughly covered by the authors in Refs. [14–16]. Following this established methodology, a brief overview is provided in Secs. 2.1–2.3, along with explanations of the modifications made to respective disciplines.

2.1 Tilt-Rotor Performance. The integrated modeling approach incorporates the first level of Padfield's hierarchical paradigm for rotorcraft performance studies, utilizing aerodynamic models for components such as rotor, fuselage, empennage, and wings, alongside a nonlinear trim model [18]. The rotor model applies steady-state nonlinear blade element theory with inflow models specific to flight conditions, including Bhagwat's hovering or high-speed cruise model and Mangler and Squire's model for conversion mode [19–21]. Experimental look-up tables are used to predict forces and moments for fuselage and wing-pylon, while a Newton–Raphson method determines control inputs for trim equilibrium [22]. Further information is available in Ref. [16].

2.2 Hybrid-Electric Propulsion System. The HEPS model is developed using SIMCENTER AMESIM software as the primary modeling platform [23]. The model integrates dynamic components from the software's "GAS TURBINE," "ELECTRIC MOTOR AND DRIVES," and "ELECTRIC STORAGE" libraries. This allows for transient simulations across the entire flight mission [24,25].

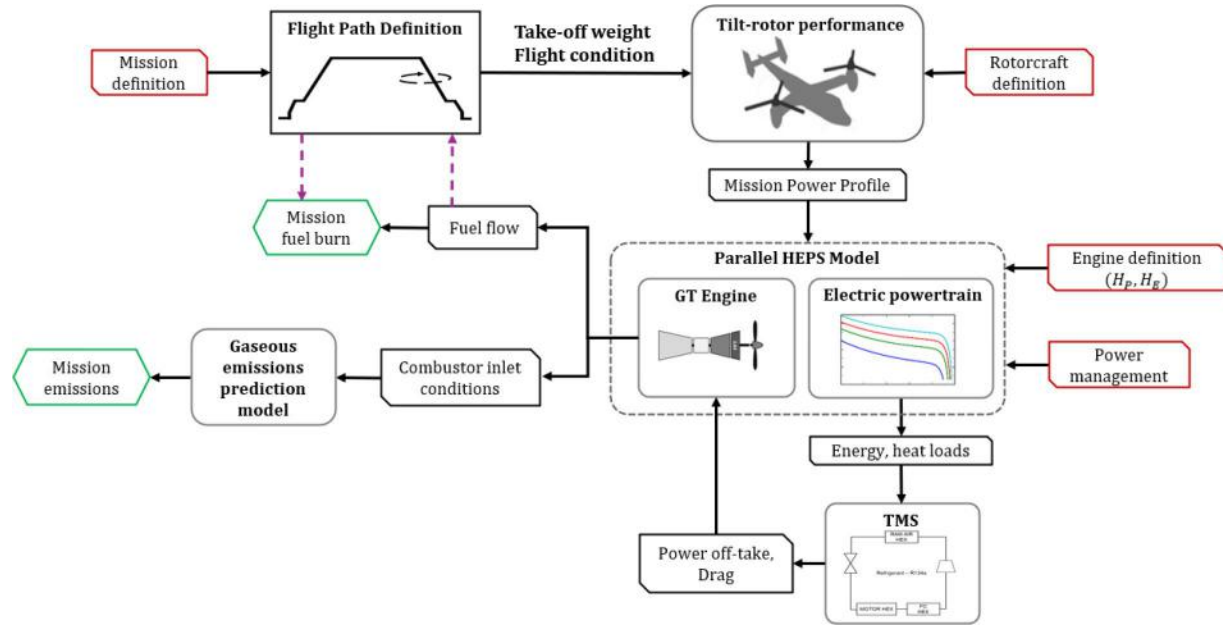


Fig. 1 Multidisciplinary PHE rotorcraft framework with IPTMS

2.2.1 Gas Turbine Model. The gas turbine engine model is built on a simple cycle configuration, adapted from experimental data specific to the General Electric T700 engine [26]. This gas turbine engine variant is widely used in rotorcraft applications.

Scaling down the core gas turbine for a specified DoH offers performance and weight benefits, as outlined in Refs. [16] and [27]. This downscaling is achievable due to the decreased propulsive power demand on the gas turbine engine, resulting from the addition of the electric powertrain. However, a tradeoff analysis should also examine the potential lifing benefits for the gas turbine due to the reduced power demand. In line with the method used in Ref. [14], the scaling of the gas turbine engine is achieved by applying a simple correlation with the design mass flowrate, as shown in the following equation [28]:

$$m_{GT} \propto (W_{\text{design}})^{1.45} \quad (3)$$

However, reducing the size of the gas turbine engine would result in a decline in the efficiency of its components. This decrease in efficiency is estimated using a simple correlation with the reduction of the design mass flowrate [29].

2.2.2 Electric Powertrain. The key performance indicators (KPIs) for the components that constitute the electric powertrain are summarized in Table 1 [4,30–33]. A simple homogeneous body is assumed when developing the thermal capacity model for the electric powertrain components to evaluate temperature variations throughout the mission profile.

The electric machine (EM) component models both the motor and inverter using an equivalent static energetic approach. The motor's performance is represented through a map that is scaled for each configuration considered in this study [27]. A design point efficiency of 0.94 is assumed, reflecting the combined efficiency of the

reference motor at 0.96 and inverter at 0.98, though this efficiency will vary based on the scaled performance map of the machine [4]. The combined losses from both the motor and inverter are generated according to the given operating condition. To enhance power density through integrated design, the motor and inverter typically share the same cooling system [34]. Consequently, the thermal capacity model applies to the average specific heat capacity of the primary materials in both components. At the same time, the inverter's operating temperature limit is assumed for the EM. The weights of the motor and inverter are estimated using the method described in Ref. [27].

The battery presizing tool in SIMCENTER AMESIM software is used to determine the weight and performance of the battery pack. This tool can generate the characteristics of a fictitious Li-ion cell that meets the specifications of the battery system in a vehicle using commercial Li-ion cell data [25]. The equivalent circuit model of the battery pack simulates both battery performance and heat generated during discharging. The peak battery operating temperature limit, shown in parentheses, indicates the maximum allowable value for short periods of up to 10 min [4].

2.3 Thermal Management System Design and Integration With the Platform. The TMS for the tilt-rotor model incorporates both air-cooling and liquid-cooling techniques. Further details on the air-cooling TMS and liquid-cooling TMS are discussed in Secs. 2.3.1 and 2.3.2, respectively, with a more detailed overview in Refs. [14] and [15]. In each system, heat from electric components is collected by cold plates and transferred to air, which acts as the final heat sink. The air-cooling system uses air directly for heat dissipation, eliminating the need for additional heat transfer devices. In contrast, the liquid-cooling system requires an air-coolant heat exchanger to remove the collected heat.

Table 1 KPIs of the electric powertrain [4,30–33]

Components	Type/ C_p determination	Power/energy density (kW/kg)/(Wh/kg)	Operating temperature limit (°C)
Motor	Permanent magnet motor/copper as baseline material (winding conductor)	8 (power)	80
Inverter	Silicon-based power electronics/silicon as baseline material	8 (power)	60
Battery	Lithium-ion battery/provided in SIMCENTER AMESIM software	250 (energy)	45 (up to 60 for 10 mins operation)

Table 2 PCM properties [35–37]

Component	Battery	EM
Paraffins	$C_{22}H_{46}$	$C_{26}H_{54}$
T_{melting} (°C)	44	56.3
LHF (kJ/kg)	249	256
ρ (kg/m ³)	778	800
$c_{p,\text{solid}}$ (kJ/kg/K)	1.51	1.80
$c_{p,\text{liquid}}$ (kJ/kg/K)	2.38	2.37

PCMs are also evaluated for specific test cases as a heat storage solution. Various paraffins, selected for their wide temperature range, safety, reliability, and cost-effectiveness, are summarized in Table 2 [35–37]. PCMs are suitable for heat storage due to their noncorrosive properties and predictable performance [35]. The design of the phase change material (PCM) heat storage system is based on two key assumptions: effective heat transfer during cooling phases and thermal insulation when not in use.

The weight of the PCM is calculated using its latent heat of fusion (LHF) and heat capacity up to the melting temperature, as shown in Eq. (4). An additional factor of 20% is included to account for auxiliary systems that may be required to enhance heat transfer to the heat storage system

$$m_{\text{PCM}} = 1.2 \times \frac{Q_{\text{store}} - m_{\text{PCM}} \cdot c_{p,\text{solid}}(T_{\text{melting}} - T_{\text{initial}})}{\text{LHF}} \quad (4)$$

Aircraft dimensions were carefully considered when integrating the designed TMS into the NASA XV-15 (Moffett Field, CA) tilt-rotor platform. While optimization favors a wider cold plate to minimize weight and pressure drop, the width is constrained to 1.5 m, matching the fuselage width of 1.52 m, as listed in Table 3 [38]. Similarly, the heat exchanger design favors a longer coolant side length. It is also constrained to 1.5 m to align with the fuselage width, with a maximum length of 3.5 m to match the fuselage length [38].

In addition to weight and power consumption penalties, drag power induced by the air duct system is also considered. The air duct is assumed to have minimal exposure to the external environment, and the effect of additional wetted area is excluded. Spillage drag at the diffuser inlet is calculated using Eq. (5) from Kellerman et al. [8], while the drag at the nozzle is calculated using Eq. (6)

$$P_{\text{spill}} = K_{\text{spill}} [W_{\text{air}}(v_1 - v_0) + A_1(p_1 - p_0)] \times v_0 \quad (5)$$

$$P_{\text{nozzle}} = W_{\text{air}}(v_{\text{exit}} - v_0) \times v_0 \quad (6)$$

2.3.1 Air-Cooling Thermal Management System. The air-cooled cold plate is designed using a semi-empirical method from Ref. [39]. This method evaluates cooling fin performance and calculates fin temperatures based on a given air temperature and heat load. A plain fin geometry is selected to minimize pressure drop, as no puller fan is included. Differential evolution (DE) algorithm [40] minimizes the cold plate's weight while ensuring surface temperature and pressure drop limits. In this approach, cold plate dimensions are variable, with fin and base plate thickness kept constant.

The dimensions of the cooling duct are determined by the maximum size of the cold plates, with additional margins applied to

Table 3 Fuselage cabin dimensions of XV-15 aircraft [38]

Interior dimensions:	
Length (m)	3.99
Width, maximum (m)	1.52
Width, at floor level (m)	1.22
Height, maximum (m)	1.52
Height, underwing (m)	1.37

account for potential wall effects. The design features an 8-deg diffuser angle, a 30-deg nozzle angle, and a fixed area ratio of 2 to ensure efficient airflow [8,41]. The connecting duct is 0.1 m in length. The weight of the entire duct system is calculated, assuming it is made of aluminum. A series configuration is used to minimize the overall size of the duct system, with the battery cold plate placed at the front due to its more stringent temperature requirements. The air-cooling TMS design schematic is presented in Fig. 2.

The air-cooling TMS design process is shown in Fig. 3. For a given heat load and operating condition, the particle swarm optimization (PSO) algorithm is used to determine the required air mass flowrate to minimize the overall fuel burn penalty. This penalty accounts for both the added weight and the induced drag power caused by the air duct [40]. A sensitivity analysis of the gas-turbine engine model is conducted to establish the correlation between these penalties by assessing their impact on propulsive power requirements and the resulting increase in total aircraft fuel consumption. By expressing both weight and induced drag power penalties in terms of their contribution to fuel burn, the optimization ensures a balanced tradeoff between these factors [15].

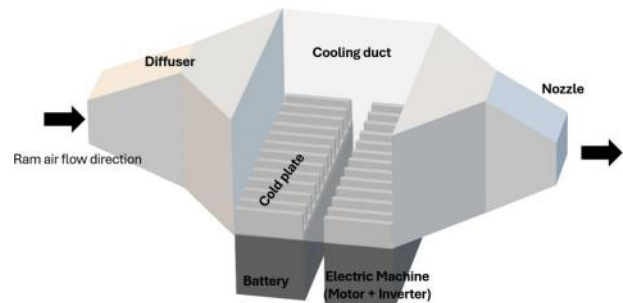
2.3.2 Liquid-Cooling Thermal Management System. Figure 4 shows the design schematic of the liquid-cooling TMS. Liquid-cooled cold plates are used to absorb heat from the electric components, employing a thermal-hydraulic and ϵ -NTU model to balance heat transfer, volume, and pressure drop [42]. An offset-strip fin geometry (1/9–22.68) is chosen for its high heat transfer efficiency, and an ethylene glycol-water mixture is selected as the coolant [43,44].

The air-coolant heat exchanger is a compact plate-fin design optimized for integration despite crossflow designs having lower effectiveness than counterflow configurations [45]. The same offset-strip fin geometry is used for both air and liquid coolant layers. The air-duct system is designed to fit the heat exchanger, while the coolant line is not covered in this study. A puller fan and coolant pump are included to circulate air and coolant. The weight of these components is estimated using the displacement-weight correlation from Ref. [5], and their power consumption is calculated using Eqs. (7) and (8) [5]. The puller fan's isentropic efficiency is assumed to be 75%, while the pump is considered ideal

$$P_{\text{pump}} = \frac{W_{\text{coolant}} \Delta P_{\text{coolant}}}{\rho_{\text{coolant}}} \quad (7)$$

$$P_{\text{fan}} = W_{\text{air}} \times c_{p,\text{air}} \times (T_{\text{out,fan}} - T_{\text{in,fan}}) / \eta_{\text{fan}} \quad (8)$$

Figure 5 illustrates the design process for the liquid-cooling TMS. For a specified heat load and operating condition, the PSO algorithm is applied to optimize key variables [40]. These include liquid coolant flowrate, air mass flowrate, and pinch temperature. The pinch temperature is defined as the temperature difference between the cold side inlet (air) and the hot side outlet (liquid coolant). The objective is to minimize the overall fuel burn penalty by accounting for both added weight and power penalties, with their relationship determined through a sensitivity analysis of the gas turbine engine

**Fig. 2 The air-cooling TMS design schematic**

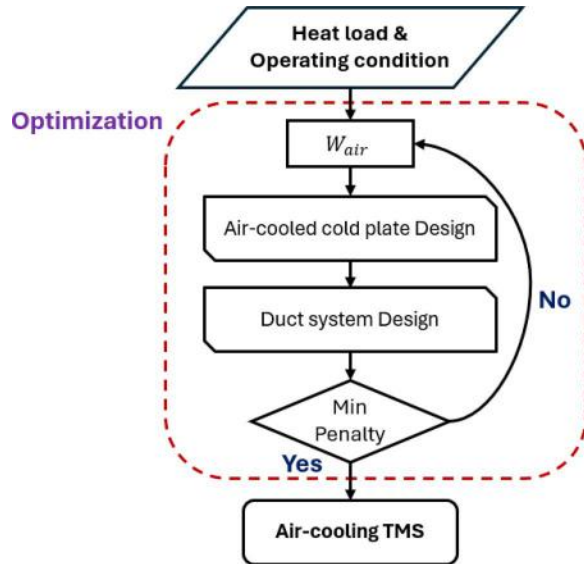


Fig. 3 Air-cooling TMS design process

model, as discussed earlier. Power consumption includes the puller fan, coolant pump, and drag induced by the air duct. The primary design parameters of the cold plate and heat exchanger are geometrical parameters such as width and length, and these are optimized to achieve minimum weight while maintaining an acceptable pressure drop.

A series configuration, as shown in Fig. 4, is assessed to align with the conclusion from the previous study that it results in a more efficient design compared to the parallel configuration at the same design point [15].

2.4 Integrated Power and Thermal Management System Optimization and Reduced Order Modeling. Exploring the design space and identifying optimal solutions for both power and thermal management strategies in PHE aircraft is computationally intensive. This complexity arises from the coupling between the PMS, which dictates the thermal management requirements, and the TMS, which impacts the overall performance of the aircraft. To reduce computational cost, assumptions and heuristics from previous studies [14,15] are applied to simplify the TMS design process, providing a more efficient starting point. These assumptions streamline the exploration of design options, but further efficiency is required for detailed optimization. As a result, reduced-order modeling becomes crucial for accelerating the evaluation of

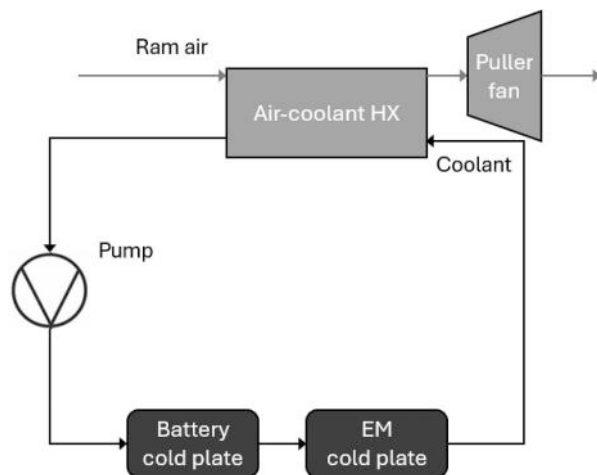


Fig. 4 The liquid-cooling TMS design schematic

each discipline while maintaining the accuracy needed for system-level interactions.

2.4.1 Thermal Management System Design Assumption & Requirement. The design point (DP) for the TMS is selected based on findings from previous studies [14,15], which identified cruise as the optimal phase for TMS configuration. Cruise is chosen because it constitutes a major part of the flight mission, leading to a more efficient overall design. Previous analyses demonstrated that focusing on cruise conditions results in improved system performance.

Identifying design heat load for the TMS is challenging due to the continuously changing aircraft weight as fuel burns and the varying state of charge (SOC), which directly impacts battery heat generation. To ensure a conservative estimate, the aircraft weight variation throughout the mission is not accounted for, assuming the aircraft weight is kept constant as the TOW. As SOC decreases during the mission, the battery heat load increases, requiring the TMS to adapt to rising thermal demands. This relationship between SOC and battery heat load is key to understanding how the TMS adapts during various flight phases, as shown in Fig. 6.

For the DP at cruise, a SOC of 25% is assumed to provide a conservative estimate of the battery heat load. Battery heat generation is calculated by accounting for both irreversible and reversible heat mechanisms. Irreversible heat arises from discharge resistance (R_{dis}), represented by the Joule heating term $I^2 R_{dis}$, where I is the discharge current. Reversible heat is associated with entropy changes during electrochemical reactions in the battery, represented by the entropic coefficient $\frac{dU}{dT}$, which describes how voltage varies with temperature. The total heat generation in the battery is expressed by the following equation [23,46,47]:

$$Q_{batt} = I^2 R_{dis} - I \cdot T \cdot \frac{dU}{dT} \quad (9)$$

Although the TMS is designed for cruise as the DP, it must also manage off-design (OD) conditions, particularly during hover, which represents a peak thermal load phase. In a previous study using the XV-15-like aircraft as a test case, the liquid-cooling TMS, which was designed for cruise, was found to be insufficient under OD conditions, even with increased air or coolant mass flow rates [15]. This issue is especially critical for the EM cooling systems, which experience significantly higher heat loads during hover. In contrast, the battery cooling system avoids this issue due to its higher

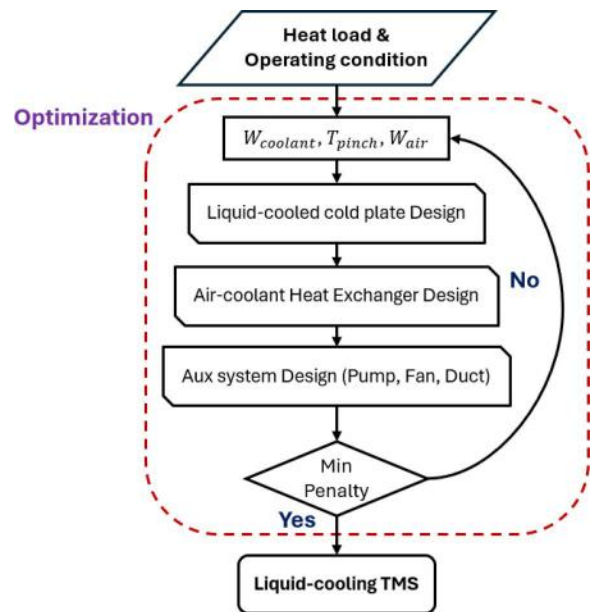


Fig. 5 Liquid-cooling TMS design process

operating temperature limits during short OD phases (Table 1). To address the thermal management challenges for the EM under OD conditions, two design approaches were proposed [15]:

- (1) Liquid-cooling A: uses PCMs as heat storage
- (2) Liquid-cooling B: increases EM heat load for hover

The heat storage system, using PCMs, is designed based on the thermal demands during critical flight phases. To ensure effective thermal management, heat loads are calculated for both hover and climb phases, which are considered peak conditions. A SOC of 25% is assumed for hover, representing a conservative estimate for managing the highest heat loads. For the climb, a SOC of 90% is assumed, as this is the expected SOC level during this flight phase.

2.4.2 Reduced Order Modeling. Reduced order modeling is applied to reduce the computational burden of full-scale multi-disciplinary optimization while preserving accuracy. By simplifying each discipline, reduced order model (ROM) captures key system behaviors in lower-dimensional models, enabling faster evaluations and more efficient optimization of power and thermal management interactions. This time efficiency is crucial, as the full-scale TMS design optimization can take several minutes, whereas ROM reduces this to less than a second.

Table 4 summarizes the developed ROMs for each discipline, including the model type, training inputs and responses, and model accuracy. The ROM for the HEPS provides key performance responses for various DoHs, focusing on power and energy ratios. Since the accuracy root-mean-square error (RMSE) varies depending on the specific DoH values, the table does not provide a fixed RMSE. Instead, an expected accuracy level is included to represent the varying accuracy across different DoH scenarios.

The TMS ROM shows lower accuracy compared to models in other disciplines due to the complexity of relationships between its variables. The training data for the TMS design is generated through an optimization process aimed at minimizing both weight and power off-take penalties. This approach ensures good accuracy for the combined penalty. However, the TMS ROM struggles to accurately predict individual responses, such as TMS weight and power consumption. Increasing the size of training datasets could improve the accuracy of these specific responses and is recommended for future work.

Note that the surrogate modeling toolbox and ROM builder from SIMCENTER AMESIM software were used to facilitate the reduced order modeling process [23,48].

2.4.3 Integrated Power and Thermal Management System Optimization. Figure 7 depicts the IPTMS optimization process. It begins by defining the design mission, including the DoH of the aircraft (H_P and H_E), which determines the design of the HEPS. This study conducts a design exploration by sweeping power and energy ratios in the range of 0.1 to 0.3 sweep, using a step size of 0.01. Due to the low energy density of the battery, a DoH higher than 0.3 is

deemed infeasible [14–16]. In their study, Danis et al. [32] employed a power ratio of 0.16 for the same aircraft.

The aircraft is then initialized with its initial fuel, estimated TMS weight, and maximum payload weight. A maximum payload of 1,000 kg is assumed, representing a capacity for 10 passengers. The available volume also limits the payload as the passenger seat configuration restricts the maximum payload that can be accommodated [14,49]. In each design iteration, the payload is recalculated so that the aircraft never exceeds maximum take-off weight (MTOW). When the available payload drops below a minimum threshold of 100 kg (equivalent to 1 passenger), the design process is terminated, as the configuration is considered infeasible for the given DoH.

Once the aircraft is initialized, the TMS design process begins based on the given PMS. Using design assumptions and requirements, three TMS configurations are designed to meet the specified thermal management requirements. For each configuration, the aircraft weight is updated iteratively until convergence, with the target of stabilizing near the MTOW to maximize payload capacity. Following weight convergence, the mission is simulated iteratively and simultaneously across all three TMS configurations until fuel burn convergence. The power off-take from the designed TMS is reflected in the gas turbine performance, ensuring the heat load of the electric powertrain remains unaffected.

The PMS is optimized using the particle swarm optimization method to minimize the energy-to-revenue work (ETRW), a metric used to assess the energy efficiency of hybrid-electric systems [40,50,51]. ETWR is defined as shown in the following equation:

$$\text{ETRW} = \frac{m_{\text{fuel}} \cdot \text{LHV} + E_{\text{electric}}}{m_{\text{payload}} \cdot g \cdot \text{Range}} \quad (10)$$

The optimization includes constraints to maintain the battery SOC within a defined range. The minimum SOC of 20% prevents damage from deep discharges, while the upper SOC limit ensures optimal utilization of the stored electrical energy [52]. Ultimately, the TMS configuration achieving the minimum ETWR is selected as the final design.

3 Definition of Test Case

The aircraft model used in this study is based on NASA's XV-15, as outlined in Ref. [22] and validated in Ref. [16]. This civil rotorcraft combines high-speed cruise capability with vertical takeoff and landing, making it ideal for UAM operations. The baseline configuration includes two turboshaft engines mounted on nacelles, with detailed specifications provided in Refs. [22] and [38]. While the aircraft's MTOW is 6803 kg, an MTOW of 6500 kg is assumed in this study to ensure VTOL capability for UAM applications. Hot day conditions (international standard atmosphere (ISA) + 25 °C) are used by default to verify the TMS performance under extreme conditions. Sections 3.1–3.3 outline the test case missions assessed in this study.

3.1 Certification Point Condition. A 5-minute contingency hover is applied to the TMS design to ensure the aircraft system can manage engine failures in line with certification point condition requirements [33]. During this mission, the PHE tilt-rotor operates at its maximum power rating, utilizing the full capacity of the electric powertrain power [16]. For this certification point, the PCM heat storage system is used to meet the thermal management requirements.

3.2 Single-Leg, Long-Range Mission. A single-leg mission with a 700 km range is selected to evaluate the IPTMS of the PHE tilt-rotor aircraft, as it closely aligns with the design range of the aircraft. Figure 8 illustrates the altitude and power profile for the baseline, unelectrified aircraft configuration. The key design results for the baseline aircraft are summarized in Table 5.

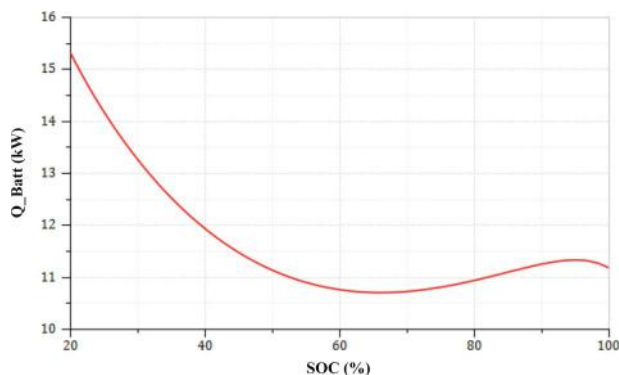


Fig. 6 Battery heat load variation against its SOC

Table 4 KPIs of ROM for respective disciplines

Discipline	ROM type	Training inputs	Trained response	RMSE	R^2
<i>Tilt-rotor performance</i>	Gaussian process	Altitude, velocity, vertical speed, weight	P_{hover} (kW) P_{cruise} (kW)	0.87 3.51	1.000 1.000
<i>HEPS</i> <i>GT</i>	Static neural network (Levenberg–Marquardt)	Altitude, Mach, power, ISA deviation, H_P	W_{fuel} (kg/s)	10^{-6}	1.000
			P_3 (Pa)	10^{-4}	1.000
			T_3 (K)	10^{-2}	1.000
			TET (K)	10^{-1}	1.000
			Q_{Batt} (kW)	10^{-2}	1.000
<i>Electric powertrain</i>	Static Neural Network (Levenberg–Marquardt)	Power, T_{Batt} , SOC, H_P , H_E	Q_{EM} (kW)	10^{-2}	1.000
<i>TMS</i> <i>Air-cooling</i>	Gaussian process	Q_{Batt} , Q_{EM}	Combined penalty (kg)	11.81	0.992
			m_{TMS} (kg)	10.50	0.987
	Gaussian process	Q_{Batt} , Q_{EM}	$P_{\text{offtake, DP}}$ (kW)	2.08	0.942
			Combined penalty (kg)	11.81	0.995
			m_{TMS} (kg)	6.67	0.975
			$P_{\text{offtake, DP}}$ (kW)	1.79	0.994
			$P_{\text{offtake, OD}}$ (kW)	0.74	0.977

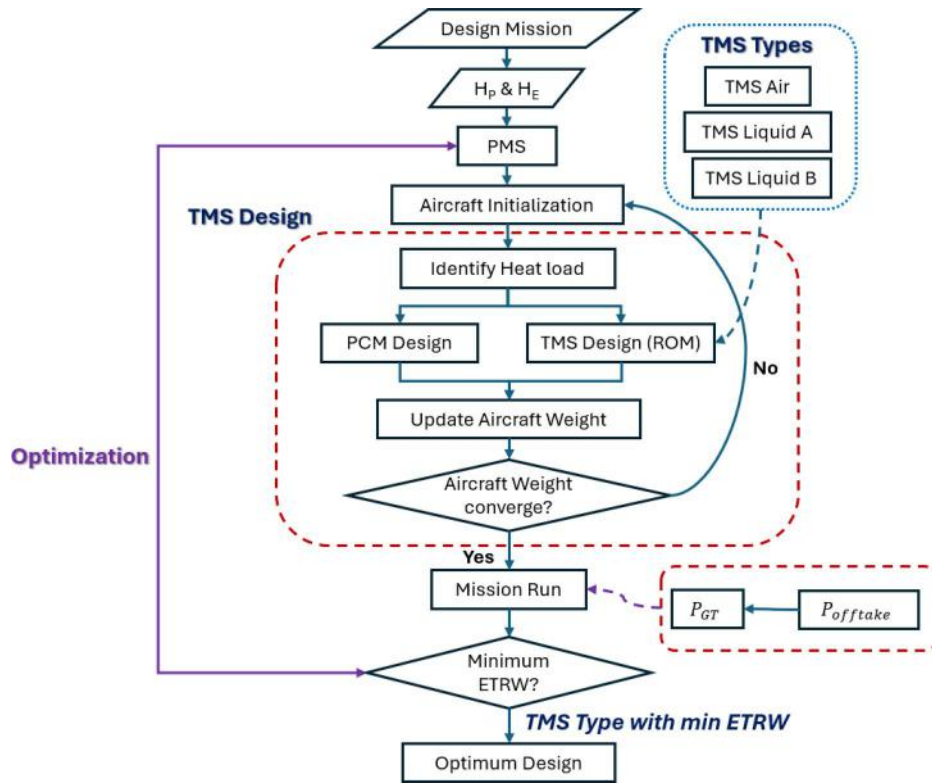


Fig. 7 IPTMS optimization process

3.3 Double-Leg, Short-Range Mission. The double-leg mission consists of two consecutive 150 km legs. It is particularly relevant for UAM applications, simulating typical operations that require two trips without recharging. This setup reflects the shorter-range requirements of UAM services. Figure 9 presents the altitude and power profile for the baseline aircraft configuration. Key design results for this mission are detailed in Table 6.

4 Results

The developed multidisciplinary optimization platform performs a sweep analysis over a DoH range of 0.1 to 0.3 for power ratio (H_P) and energy ratio (H_E) with a step of 0.01, enabling comprehensive design exploration. Two cases are considered: one optimizing only the PMS without TMS integration, and another incorporating TMS in IPTMS optimization. The comparative assessment between two cases highlights the impact of TMS integration on hybrid-electric aircraft design.

Additionally, both single-leg and double-leg mission scenarios are analyzed to assess the impact of operating requirements on design results. The single-leg mission is used as the design point for comparison. The H_E is based on the energy consumption of a 6500 kg TOW aircraft for the single-leg mission (3.6 MJ), while the H_P is defined using the GE-T700 engine's maximum power rating (1210 kW).

4.1 Design Exploration With Power Management Strategies Optimization

4.1.1 Single-Leg, Long-Range Mission. Figure 10 illustrates the HEPS weight results across the specified design space, defined as the sum of the gas turbine and electric powertrain weights. The results indicate that HEPS weight is primarily influenced by the H_E due to the battery energy density. Additionally, a mismatch in the power-to-energy ratio leads to heavier battery system weight, as this ratio directly impacts the battery pack design, including the number

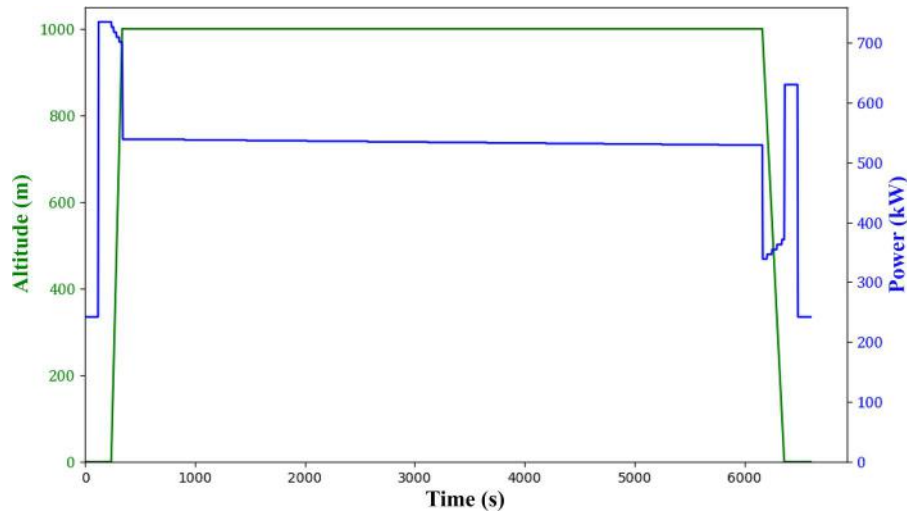


Fig. 8 Altitude and power profile—single-leg mission

of cells in series and parallel [53]. The hatched area represents regions where the design is infeasible, mainly due to exceeding the MTOW limit.

The ETRW results for each assessed design case are depicted in Fig. 11, with the red dashed line marking areas of improvement over the baseline aircraft. The maximum percentage improvement observed is 8.7%, achieved under design assumptions that limit maximum payload capacity. While hybrid-electric aircraft cases operate near the MTOW limit, the baseline aircraft's TOW remains comfortably below this threshold, as shown in Table 5. This suggests that energy efficiency could be further enhanced by utilizing the unused weight for additional payload. However, this additional payload capacity necessitates enhanced passenger seat allocation or other mitigation strategies. A more in-depth analysis of the baseline aircraft's potential with increased payload is discussed in Sec. 4.2.1.

4.1.2 Double-Leg, Short-Range Mission. Figure 12 illustrates the HEPS weight across the design space for the double-leg mission scenario. Compared to the single-leg mission, the feasible design space is significantly expanded due to the shorter mission range, allowing more electric powertrain capacity. However, a distinct less optimum region appears in the lower H_P area, where design constraints impose limitations. This arises from the design constraints that restrict the battery SOC to ensure the onboard electric energy is fully utilized. Within this low H_P region, the electric power capability is insufficient to fully utilize the available battery energy, leading to less optimum integrated design which represented as empty area in Fig. 12.

Similarly, cases with improved ETRW over the baseline aircraft expand considerably in this mission scenario, as depicted in Fig. 13, with a maximum improvement of 11.8%. This expansion underscores the potential benefits of electrification for shorter operating ranges, where higher DoH becomes viable. The results indicate that as mission range decreases, the benefits of electrification become more pronounced, supporting a strategic shift toward hybrid-electric

configurations that optimize energy usage in short-range applications.

4.2 Design Exploration With Integrated Power and Thermal Management System Optimization

4.2.1 Single-Leg, Long-Range Mission. The design exploration with IPTMS optimization is conducted for the single-leg mission. As discussed, the TMS design type achieving optimal ETRW is selected for each design case. The resulting optimized TMS configurations across the design space are presented in Fig. 14. The trends on the optimum configurations are hereafter explained. Notably, the liquid-A option (using PCMs as heat storage system) is not selected in any part of the design space. This outcome aligns with findings from the previous study, which indicated that liquid-A is less efficient compared to liquid-B (increasing EM heat load for hover).

Compared to the exploration results obtained with PMS optimization without TMS, the feasible design space is noticeably reduced, highlighting the impact of TMS integration on overall performance. This reduction is represented by the empty area bounded by the dashed line, with the maximum feasible H_E is reduced from 0.24 to 0.22.

In lower H_E regions, an air-cooling TMS is preferred due to limited onboard electrical energy, resulting in lower electric power usage and a correspondingly lower thermal load. Under these conditions, air-cooling is the most efficient choice. Previous studies indicate that the weight of the air-cooling TMS increases significantly as thermal loads become more demanding. However, below a certain heat load threshold, the air-cooling TMS outperforms liquid-cooling, as it relies on air as both the coolant and the final heat sink, naturally resulting in a simpler, lighter design [14,15]. As H_E rises, however, onboard electric energy and subsequent thermal load increase, making liquid-cooling TMS more effective for managing higher heat loads. For cases with high H_P , electric power usage is concentrated during hover and climb to reduce the thermal load in cruise, allowing air-cooling to stay efficient. However, the benefits of this approach diminish as PCMs increase in weight, drastically limiting payload capacity.

The designed TMS weight results are presented in Fig. 15, which presents the weight results for the designed TMS, with the transition area from air-cooling to liquid-cooling highlighted by an arrow. This marks the design cases where liquid-cooling TMS begins to offer advantages. However, including components requiring auxiliary power, such as the coolant pump and puller fan, increases total fuel consumption for the mission, as shown in Fig. 16.

However, the trends for TMS weight and fuel consumption across the design space are less distinct, as these outcomes are not direct

Table 5 Design results of the baseline aircraft—single-leg mission

TOW (kg)	5785.48
Fuel (kg)	669.13
Payload (kg)	1000
CO ₂ emission (kg)	2099.44
NO _x emission (kg)	6.58
ETRW	4.191
Fuel per payload	0.669

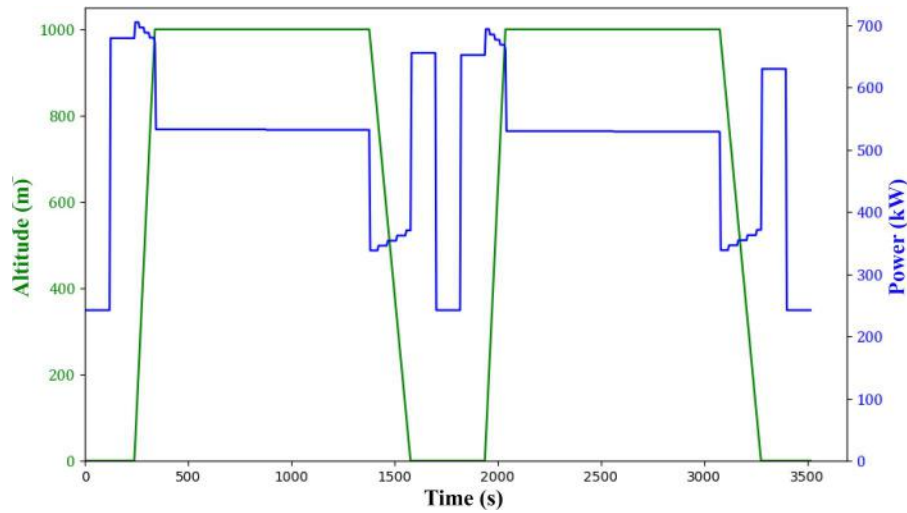


Fig. 9 Altitude and power profile—double-leg mission

Table 6 Design results of the baseline aircraft—double-leg mission

TOW (kg)	5468.09
Fuel (kg)	351.66
Payload (kg)	1000
CO ₂ emission (kg)	1103.37
NO _x emission (kg)	3.47
ETRW	5.140
Fuel per payload	0.352

objectives of the optimization criteria. Additionally, fluctuations in the TMS weight trend may result from the relatively low accuracy of the TMS design ROMs. In contrast, the onboard payload variation across the design space (Fig. 17) shows a relatively clear trend. This is because the payload has a greater impact on the ETRW results.

Figure 18 presents the ETRW results across the design space for the single-leg mission. The dashed line within contour plot indicates design cases that achieve improved ETRW over the baseline aircraft when TMS design and integration are not considered. With IPTMS optimization, however, the number of design cases surpassing the ETRW of the baseline is significantly reduced. This again emphasizes the impact of TMS design and integration.

Design cases with $H_E = 0.1$ and an air-cooling TMS option demonstrate improved ETRW, with the $H_p = 0.16$ case having been identified as the optimal configuration. Table 7 provides the

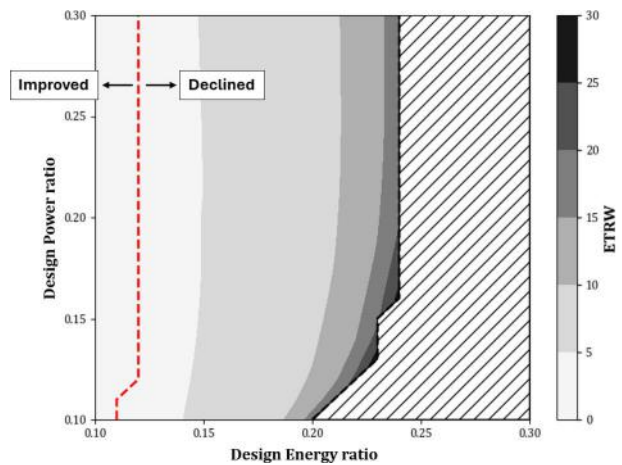


Fig. 11 ETRW—single-leg mission; PMS optimization

summarized result for this optimal case, including the optimized PMS. The ETRW is enhanced by 4.55%, and electrification further reduces emissions, lowering CO₂ by 6.56% and NO_x by 1.03%. Emission calculations are based on the P3T3 method and the kerosene combustion equation [14]. Figure 19 illustrates the

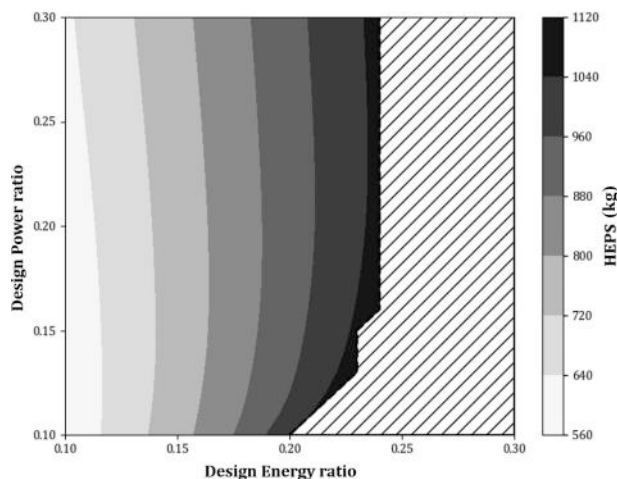


Fig. 10 HEPS weight—single-leg mission; PMS optimization

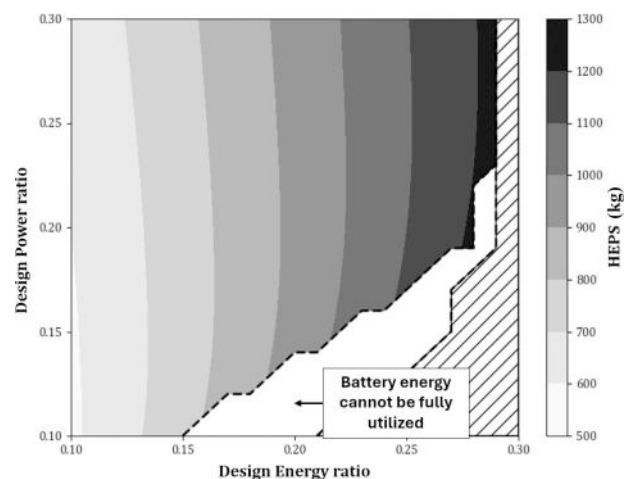


Fig. 12 HEPS weight—double-leg mission; PMS optimization

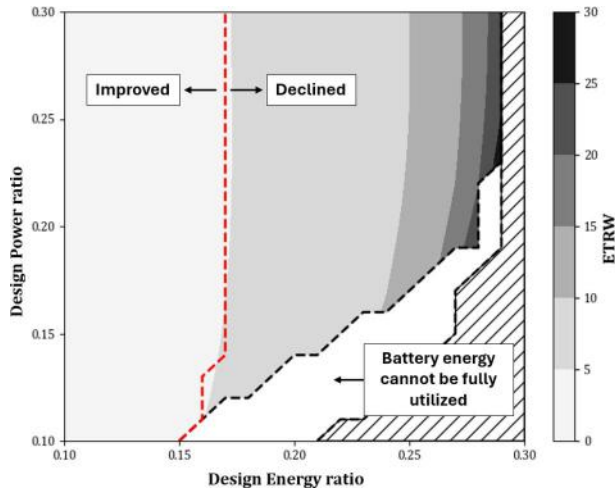


Fig. 13 ETRW—double-leg mission; PMS optimization

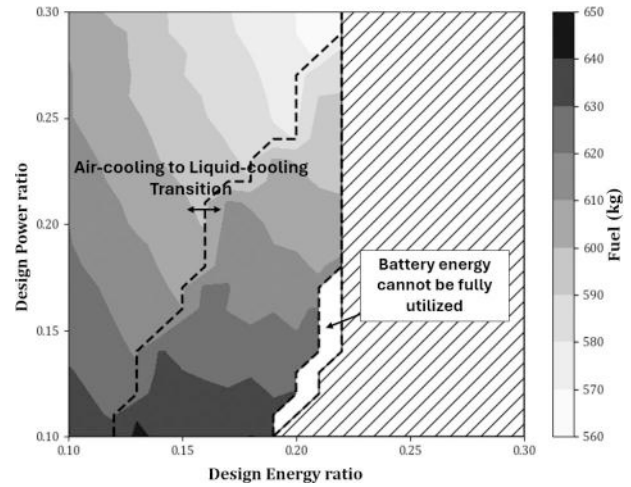


Fig. 16 Fuel consumption—single-leg mission; IPTMS optimization

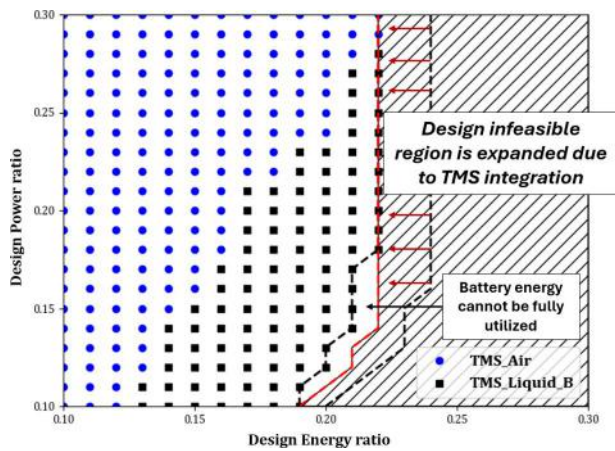


Fig. 14 TMS design type—single-leg mission; IPTMS optimization

operational highlights of the optimal case, detailing the cooling strategy and PMS for each phase of the mission. During transient heat spikes, especially when ram air is limited, the PCM is employed as a heat storage system for thermal management.

The TOW of the designed hybrid-electric aircraft is already constrained to the maximum, while the baseline aircraft retains

additional weight capacity that could accommodate more payload. However, this potential is currently limited by maximum payload assumptions related to passenger capacity. If this payload constraint were lifted through enhanced passenger seat allocation or other strategies, the ETRW could improve by 40% compared to the optimum hybrid-electric case, as indicated in Table 8. This highlights the need for a more detailed integration study to assess the potential benefits of electrification.

4.2.2 Double-Leg, Short-Range Mission. Similarly, the design exploration for the double-leg mission is conducted with the TMS design types across the design space shown in Fig. 20. The infeasible region resulting from TMS design and integration impacts is significantly expanded compared to the single-leg mission. The shorter operating range in the double-leg mission allows for a higher DoH capability, which in turn increases thermal management requirements. Consequently, the impact of TMS integration becomes more pronounced.

Interestingly, liquid-A is selected for several design cases. A detailed analysis reveals that liquid-A, which uses PCMs for transient heat storage, can outperform liquid-B under specific conditions, as liquid-B increases the design heat load for the EM. When the added weight of PCMs does not impact payload capacity, liquid-A becomes advantageous. Unlike liquid-B, which relies on continuous liquid-cooling throughout all mission phases, liquid-A activates auxiliary components only as needed, reducing power off-takes and enhancing efficiency. A comparative assessment of the

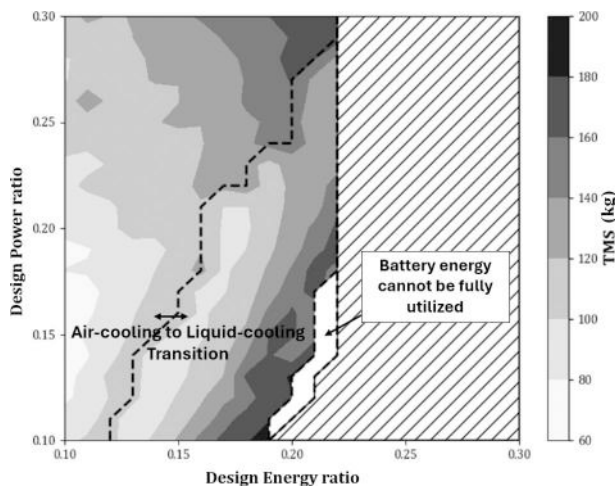


Fig. 15 TMS weight—single-leg mission; IPTMS optimization

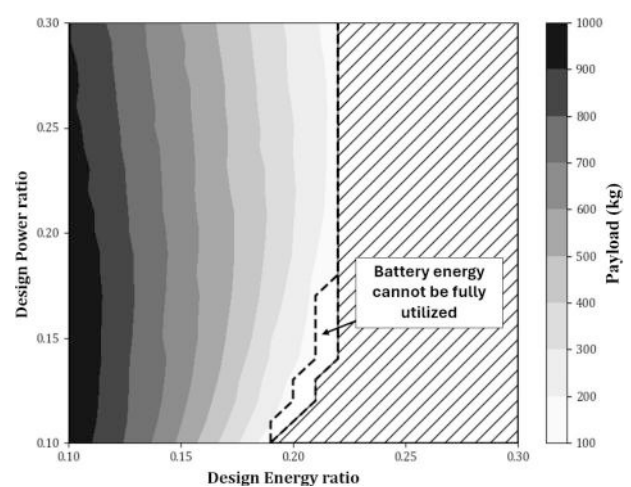


Fig. 17 Payload—single-leg mission; IPTMS optimization

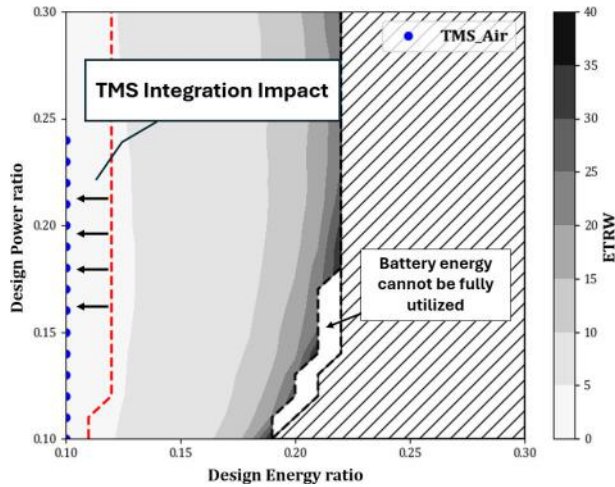


Fig. 18 ETRW results—single-leg mission; IPTMS optimization

Table 7 Optimal design case result—single-leg mission

H_E	0.1			
H_p	0.16			
TOW (kg)	6500			
HEPS design				
Per engine	Gas turbine (kg)	191.10		
	Battery (kg)	350.56		
	Motor (kg)	24.20		
	Inverter (kg)	24.20		
TMS (air-cooling) (kg)	70.41			
Payload (kg)	1000			
Fuel (kg)	625.25			
CO ₂ emission (kg)	1961.78			
NO _x emission (kg)	6.512			
ETRW	4.000			
Fuel per payload	0.6253			
PMS				
Phase	Hover	Climb	Cruise	Descent
Power split	0.953	0.920	0.932	0.939

two TMS configurations for the relevant design case is provided in Table 9.

Unlike the single-leg mission results, liquid-cooling TMS is predominantly selected across the design space for the double-leg mission. Due to the shorter mission range, electric power usage is generally higher, resulting in an increased thermal load in most

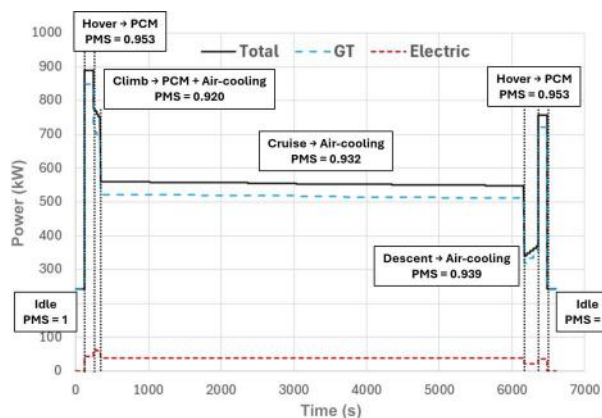


Fig. 19 Optimal design case operation highlights—single-leg mission

Table 8 Maximum payload performance of the baseline aircraft—single-leg mission

TOW (kg)	6500
Fuel (kg)	684.3
Payload (kg)	1699.3
CO ₂ emission (kg)	2147.11
NO _x emission (kg)	6.86
ETRW	2.522
Fuel per payload	0.403

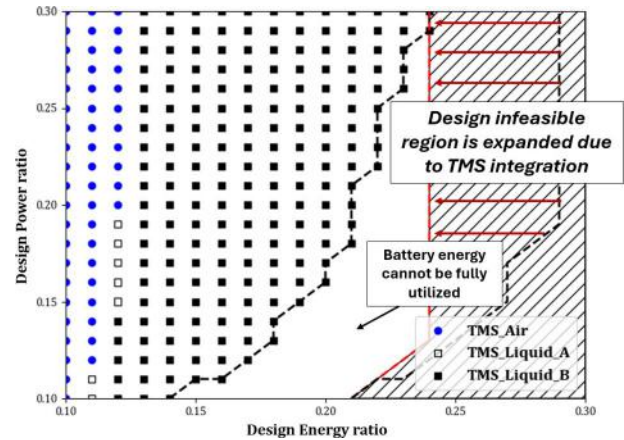


Fig. 20 TMS design type—double-leg mission; IPTMS optimization

design cases. As a result, the heat load threshold at which air-cooling TMS remains advantageous is surpassed, making liquid-cooling the more effective option.

Figure 21 presents the ETRW results across the design space for the double-leg mission. Similarly, the area between the two dashed lines represents design cases that become less energy efficient than the baseline aircraft when TMS integration impact is considered. As discussed, the impact of TMS integration is more pronounced compared to the single-leg mission. Nevertheless, the number of design cases with improved ETRW is increased compared to the single-leg mission, highlighting the suitability of hybrid-electric configurations for enhancing energy efficiency in short-range missions.

Relatively low DoH design cases are identified as those that benefit most from electrification. Due to the energy density limitations of electric components, high DoH designs show impractical configurations by significantly diminishing the payload capacity of the aircraft, thus compromising the fundamental purpose of transportation.

Table 10 summarizes the results of the most optimal design case, including optimized PMS. For this case, electrification improves the ETWR by 8.76% and achieves emission reductions of 12.95% for CO₂ and 1.66% for NO_x.

In the optimal case for a double-leg mission, the TMS weight is approximately twice that of the single-leg mission. This increase is

Table 9 Liquid-cooling TMS design comparison

$H_E = 0.12$		
$H_P = 0.19$		
	TMS liquid-A	TMS liquid-B
TOW (kg)	6491.07	6434.56
Payload (kg)	1000	1000
TMS (kg)	229.99	170.48
Fuel (kg)	320.93	323.93
ETRW	4.926	4.969
Fuel per payload	0.321	0.324

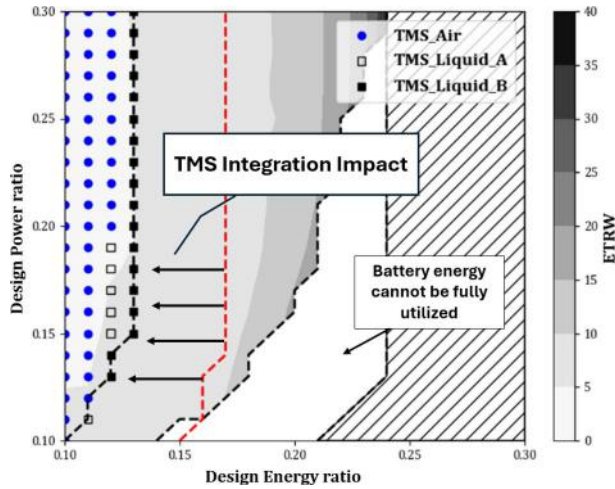


Fig. 21 ETRW results—double-leg mission; IPTMS optimization

Table 10 Optimal design case results—double-leg mission

H_E					0.11
H_p					0.3
TOW (kg)					6,460.51
HEPS Design					
Per engine	Gas turbine (kg)				146.71
	Battery (kg)				419.92
	Motor (kg)				45.38
	Inverter (kg)				45.38
TMS (air-cooling) (kg)					215.41
Payload (kg)					1000
Fuel (kg)					306.14
CO ₂ emission (kg)					960.52
NO _x emission (kg)					3.412
ETRW					4.690
Fuel per payload					0.3061
PMS					
Phase	Hover	Climb	Cruise	Descent	
Power split	0.794	0.804	0.895	0.843	

driven by the higher design heat load, which results from the increased DoH, as reflected in the optimized PMS.

5 Conclusion

This study proposed a multidisciplinary optimization framework incorporating IPTMS for hybrid-electric tilt-rotor aircraft applications. While rotor aerodynamics and flight dynamics models were adopted from a previously developed methodology [16], mission analysis, gas turbine performance, and electric powertrain performance models were extended, updated where necessary, and validated to include thermal management of electric powertrains.

Design exploration was conducted for both a single-leg, long-range mission and a double-leg, shorter-range mission across a range of DoH values (H_E and H_P between 0.1 and 0.3). The results revealed that lower DoH designs, particularly at $H_P = 0.16$ with air-cooling TMS for the single-leg mission, improved ETRW by 4.55% and achieved emission reductions of 6.56% for CO₂ and 1.03% for NO_x. For the double-leg mission, designs at $H_E = 0.11$ and $H_P = 0.3$ demonstrated even greater benefits, with electrification improving the ETRW by 8.76% and reducing CO₂ and NO_x emissions by 12.95% and 1.66%, respectively.

These findings highlight the effectiveness of hybrid-electric configurations in enhancing energy efficiency for short-range missions. However, while electrification enhances energy efficiency and reduces emissions in these optimal cases, the baseline aircraft (without electrification) still demonstrates significant potential for

improved energy efficiency when additional payload capacity is utilized through enhanced passenger seat allocation or other strategies. This potential improvement diminishes the comparative benefits of electrification, highlighting the need for a detailed integration study to assess the advantages of electrification.

Higher DoH configurations faced challenges due to increased thermal loads and payload constraints. Air-cooling TMS was found to be optimal in lower DoH regions, whereas liquid-cooling options proved more effective for higher thermal load conditions. Additionally, liquid-cooling with transient heat storage using PCMs proved advantageous when payload capacity was not compromised.

The developed multidisciplinary IPTMS optimization platform effectively guided the preliminary design of hybrid-electric VTOL aircraft, supporting advancements in sustainable and efficient rotorcraft technology.

Nomenclature

A	= area (m ²)
c_p	= specific heat capacity (kJ/kg·K)
CO ₂	= carbon dioxide
DE	= differential evolution
DOH	= degree of hybridization
DP	= design point
EM	= electric machine
ETRW	= energy-to-revenue work ratio
H_E	= energy ratio
H_P	= power ratio
HEPS	= hybrid electric propulsion system
I	= current (A)
IPTMS	= integrated power and thermal management system
ISA	= international standard atmosphere
K_{spill}	= empirical spillage coefficient
KPI	= key performance indicator
LHF	= latent heat of fusion (kJ/kg)
m	= mass (kg)
MTOW	= maximum take-off weight
NO _x	= nitrogen oxides
OD	= off-design
PMS	= power management strategy
PSO	= particle-swarm optimization
R	= resistance (Ohm)
RMSE	= root-mean-square error
ROM	= reduced order model
SOC	= state of charge (%)
T	= temperature (°C)
TMS	= thermal management system
TOW	= take-off weight
UAM	= urban air mobility
v	= velocity (m/s)
VTOL	= vertical take-off and landing
W	= mass flow rate (kg/s)
ε	= effectiveness
ρ	= density (kg/m ³)

Subscripts

dis	= discharge
f	= fuel
GT	= gas turbine

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