



ASME Accepted Manuscript Repository

Institutional Repository Cover Sheet

Cranfield Collection of E-Research - CERES

ASME Paper

Title: On the Design of Integrated Thermal Management Systems for eVTOL Aircraft

Authors: Sangkeun Kang, Chana Saias, Ioannis Roumeliotis, Olivier Broca

ASME Conf Title: ASME Turbo Expo 2023: Turbomachinery Technical Conference and Exposition

Volume/Issue: Volume 5: Cycle Innovations Date of Publication (VOR* Online): 28th September 2023

ASME Digital Collection URL: <https://asmedigitalcollection.asme.org/GT/proceedings-abstract/GT2023/86984/V005T06A023/1168019>

DOI: 10.1115/GT2023-103097

*VOR (version of record)

On the Design of Integrated Thermal Management Systems for eVTOL Aircraft

Sangkeun Kang

Researcher
Centre for Propulsion Engineering,
Cranfield University,
Bedfordshire MK43 0AL, UK
e-mail: sangkeun.kang@cranfield.ac.uk

Chana Saias

Research fellow
Centre for Propulsion Engineering,
Cranfield University,
Bedfordshire MK43 0AL, UK
e-mail: chana-anna.saias@cranfield.ac.uk

Ioannis Roumeliotis

Lecturer
Centre for Propulsion Engineering,
Cranfield University,
Bedfordshire MK43 0AL, UK
e-mail: i.roumeliotis@cranfield.ac.uk

Olivier Broca

Siemens Industry Software SAS
La Cité Internationale
84, quai Charles de Gaulle
69006 Lyon, France
e-mail: olivier.broca@siemens.com

ABSTRACT

Vertical take-off and landing aircraft equipped with hybrid or fully electric propulsion systems have gained interest for urban air mobility applications. Besides the low energy density of state of art electric systems, their strict operating temperature requirement and low-grade heat highlight the importance of effective thermal management system design. This paper aims to establish a design process for the thermal management system for a parallel hybrid electric tiltrotor and showcase it for a system consisting of direct ram-air cooling cold plate and phase change materials. The cold plate is designed for cruise conditions and used as heat acquisition and cooling system, whereas the phase change materials is used as a heat storage system when ram air is limited. Three sizing conditions are considered comprising the baseline design mission, a certification condition for One Engine Inoperative (OEI) and an off-design mission. The effects of thermal management system integration on overall performance are quantified and discussed. The results indicate that designing the system for the baseline sizing mission results in a 5% increase in fuel burn per payload relative to the equivalent counterpart without thermal management consideration. This penalty increases to 14% when certification requirements are accounted. It is highlighted that for shorter mission thermal management is the limiting factor for battery utilization, hence the expected benefits are significantly reduced. As electric components technology increases thermal management becomes less sensitive to the sizing conditions.

Keywords: rotorcraft, hybrid-electric propulsion, thermal management system, cooling system, heat loads

INTRODUCTION

Due to the steep population density increase in urban areas, traveling time waste and emissions caused by traffic congestion have become a concern both in terms of public convenience and environmental sustainability. Urban air mobility (UAM) is a new application in the aviation industry that is being considered as a promising solution that can offer a new mode of transportation in such urban areas [1], [2].

This application is envisioned to carry up to 15 passengers (PAX) for short and frequent flights that range of 50 nautical miles or less, utilizing the vertical take-off and landing capability of rotorcraft configurations. Hybrid or fully electric propulsion systems depending on size and mission are considered for this application targeting to minimize the environmental footprint [3]. Many studies and industrial demonstrators have shown the large variability in UAM architectures, but due to the low power and energy density of state of art electric systems, fully electric configurations are limited to small passenger capacities and low endurance [1]–[3].

In conventional gas turbine engines, the waste heat is rejected through the exhaust of the engine. In hybrid or fully-electric configurations on the other hand, the electric components generate low-grade waste heat that cannot be rejected in such a way. Moreover, the operating temperature requirements of the electric components are strict. In this context effective Thermal Management Systems (TMS) to ensure operation within appropriate temperature limits are necessary, for both hybrid and fully electric systems [4].

Conventionally, a ram-air-based or liquid cooling system is often used for the TMS of the Electric Propulsion System (EPS) due to its simplicity, as noted in previous studies (e.g. [4]–[9]). The direct ram air cooling method is chosen due to its simplicity, ease of integration, and lack of requirement for a secondary medium for cooling. However, the impact of TMS integration within the aircraft system on fuel, energy, and emission performance at both aircraft and mission levels has not been quantified or discussed.

For VTOL aircraft, the hover phase is considered the most challenging flight phase for thermal management, not only due to the peak propulsion power requirement but also because the air inlet flow is limited. Nevertheless, the TMS is recommended to be optimized at cruise while complying with the hovering cooling requirement to avoid overdesign issuing and the relevant weight/drag and power consumption penalties [10]. Thus, the importance of thermal management for hover and transition flight conditions has been emphasized in many studies, as eVTOL aircraft must be capable to deal with transient heat spikes during such conditions [10], [11].

The literature currently available on integrated TMS design and assessment for PHE (Parallel Hybrid Electric) rotorcraft reveals a knowledge gap. This paper aims to design the TMS for a PHE tiltrotor aircraft modelled after the NASA XV-15. The objective of this work is to develop an integrated methodology for the design and optimal implementation of hybrid electric civil rotorcraft. Three different sizing conditions are considered in this study: a 700 km sizing mission, a certification condition for One Engine Inoperative (OEI), and an off-design mission consisting of two consecutive 150 km missions without any recharging in between. A direct ram-air cooling cold plate is selected due to its simplicity, and Phase Change Materials (PCMs) are used for the conditions where ram-air is limited. Current state of the art and advanced technology levels are considered for the electrical components in the study to identify the expected benefits from electrification today and in the future. The power management strategy (PMS) is optimized with respect to the flight mission phases. The energy efficiency and emission penalty associated with the TMS integration are quantified and discussed. The effect of accounting for additional sizing conditions other than the design mission is quantified.

METHODOLOGY

The following section describes the approach used for the design and assessment of the integrated TMS PHE tiltrotor aircraft. The overall workflow of the study is shown as a flowchart in Figure 1. The PHE propulsion system is sized for the design mission applying the method presented by Saias et al. [12]. The TMS is then designed and optimized based on the requirements of a nominal design mission and a failure-mode scenario. The designed PHE – TMS integrated in the tiltrotor aircraft is assessed for an off-design mission to ensure operation within temperature performance limits. The PMS is optimized with respect to the mission phases for each mission considered, by targeting to achieve minimum energy consumption. The drag and weight impact of the TMS are taken into account, as well as the restrictions posed by its dimensions, more intricate installation challenges are not considered.

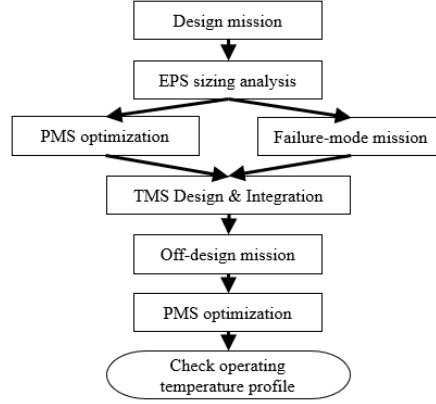


Figure 1. Overall workflow chart

Rotorcraft model

An integrated framework for preliminary design and performance assessment of rotorcraft architectures [12], [13] is utilized in this paper. The modelling approach comprises a set of comprehensive modules, each applicable to the different aspects of rotorcraft performance. The integrated modelling approach incorporates the features of the first level of Padfield’s hierarchical paradigm, appropriate for performance studies [14].

The rotorcraft module includes aerodynamic models for the different aircraft components. The rotor model utilizes steady-state non-linear blade element theory [15] coupled with different inflow models for axial propeller flight and flight with an edgewise flow component. In hovering flight, the rotor aerodynamics model is coupled with Bhagwat’s inflow model [16]. The method caters to the inclusion of the swirl induced velocity component. In conversion mode, the Mangler and Squire inflow model, a potential flow based analytical model that has found extensive application, is employed [17]. Look-up tables describing the steady-state aerodynamic characteristics for the airfoil sections are deployed. Experimental look-up tables expressed as functions of incidence and side-slip angles are utilized for the prediction of fuselage and wing-pylon forces and moments [18].

The tilt-rotor is a laterally symmetric vehicle that allows to de-couple the roll-yaw degrees-of-freedom. A Newton-Raphson approach is applied to obtain the control angles required to achieve an equilibrium trim state for a specified operating condition. The process converges to a set of control inputs to reach equilibrium. Details on the development and validation of the individual methods are presented by Saias et al. [12], [13].

The framework for rotorcraft performance is then coupled with a weight estimation module and with Simcenter Amesim software for propulsion system steady-state and transient analysis. The different simulation models are integrated within a mission analysis module enabling analyses at aircraft and mission levels.

Hybrid-Electric Propulsion system

In order to assess the hybrid configurations for rotorcraft propulsion, relevant dynamic models are built in Simcenter Amesim. As this study is more focused on the TMS design and assessment, a brief description of the propulsion system components is given herein. The designed EPS model is built in Simcenter Amesim software by utilizing components from the “Gas Turbine”, “Electric Motor and Drives” and “Electric Storage” libraries [19], [20]. A PHE propulsion system is then built by integrating the EPS model with the gas turbine engine model in the Simcenter Amesim simulation.

Additionally, a thermal capacity model of the electric components is developed to identify the operating temperature profile throughout the flight mission, by assessing the waste heat profile of each component to the developed thermal capacity models. The developed thermal capacity models are built in Simcenter Amesim software by using components from the “Thermal” library.

Gas turbine model

To model the gas turbine, Simcenter Amesim Gas Turbine library is used [21]. The gas turbine library set of components can be combined with a plethora of components of other libraries such as electrical, control, and thermal, allowing the modeling of conventional and unconventional engines, the analysis of integrated systems, and the application of different control strategies. A simple cycle engine model adapted on publicly available experimental data for T700-GE-700 is used as the baseline [21]. It is broadly used in helicopter applications.

The core gas turbine can be scaled down to gain performance and weight benefits as discussed in [12], [22], as the propulsion power requirement from the core gas turbine system is reduced with respect to the degree of hybridization (DOH). In the study, the core gas turbine is scaled down by using a simple correlation with the design mass flow rate per Equation (1). It is reasonable to assume that the design mass flow rate is directly proportional to the power output of the gas turbine at the design point. Note that this equation does not take into account the turbomachinery components efficiency penalty by down scaling, and smaller engines stand to benefit as a result. The compressor and turbine maps are scaled accordingly.

$$m_{GT} \propto (\dot{m}_{design})^{1.45} \quad (1)$$

Electric Power Train

Weight and specific heat capacity are the parameters required for the development of a thermal capacity model of the electric components. The temperature profile of the component with respect to the applied heat load can be analyzed if specific heat capacity and mass of the component are known. Note that the electric components are assumed to be homogeneous bodies for the specific heat capacity calculation. The electric component weight is estimated based on the methodologies described in [22].

Electric motor

A permanent magnet motor is known to have the highest power density to date (around 8kW/kg), and copper is the baseline material for its winding conductor [23]. Hence, the specific heat capacity of copper is used in the electric motor thermal capacity model for the current technology level.

A fully superconducting electric motor is considered for the advanced technology level. The power density at the advanced technology level is assumed to be 10kW/kg [24]. Magnesium diboride is assumed as the baseline material of the superconducting motor, as it is considered as one of the simplest and most cost-effective superconducting materials [25]. Hence, its specific heat capacity is used in the thermal capacity model for advanced technology.

Inverter

Silicon-based power electronics are widely used today, and Silicon-Carbide power electronics are enabling technology expected in the future for MW power level [24]. Thus, the specific heat capacity of each material is used in the inverter thermal capacity model. Note that the same power density level as the electric motor is assumed both at the current and advanced technology levels [24], [26].

Battery

In this study, lithium-ion battery type is considered for both current and advanced technology cases. The system level energy density of 250Wh/kg and 600Wh/kg is assumed in the study for the current and advanced technology level respectively [10], [13], [24]. The specific heat capacity of the lithium-ion battery pack given in [27] is used in the thermal capacity model.

Boundary conditions

The heat load of the EPS is determined by the power requirement and component efficiencies throughout the flight mission. The assumed electric component design efficiencies and operating temperature range based on current and future technology are shown in Table 1 [4]. Note that the motor and inverter efficiencies are assumed to be constant throughout the mission, whereas the battery efficiency is variable throughout the flight mission according to the operating condition of the battery, such as the SOC and voltage variation [27]. A peak battery operating temperature limit is included in parentheses representing the value allowed for a short duration up to 10 minutes [4].

Table 1. Electric components boundary condition [4]

		Motor	Inverter	Battery
Efficiency	Current	96%	98%	variable
	Advanced	98.5%	99.5%	variable
Operating temperature limit [°C]	Current	80	60	45 (up to 60 for 10 mins operation)
	Advanced	68	60	45 (up to 60 for 10 mins operation)

PMS optimization

The propulsion power requirement from the EPS is defined by the PMS for a specific range of missions, which is segmentally optimized for flight mission phases. The PMS optimization is conducted using the optimization tool of the Simcenter Amesim software, targeting minimization of the overall energy consumption. At the same time, a battery minimum State of Charge (SOC) of 21~22% is used as a constraint in the optimization. This discharge margin is set to avoid irreparable damage to the battery electrodes, caused by depth discharges [28]. The optimization technique is based on the Nonlinear Programming Quadratic Line search (NLPQL) algorithm [27].

TMS Design & Integration with the Platform

The TMS design and integration with the tilt rotorcraft model process is discussed herein. The schematic of the TMS and TMS design process are depicted in Figure 2 and Figure 3 respectively. The battery is cooled in parallel due to the stringent operating temperature limit. A direct ram air-cooled cold plate with rectangular fin is used as the TMS at cruise conditions, as it is relatively simple to design, and the cold ambient air at high altitudes can be utilized as heat sink. The design methodology described in [29] is used to design the direct ram air-cooled cold plate. The weight of the direct ram air-cooled cold plate is calculated by using volume (V_{cp}) and cold plate material density (ρ_{cp}) correlation, where volume is a function of width of the cold plate (w), length of the cold plate (l), number of fins (n_f), height of the fin (h_f), thickness of the base plate (t_{base}), and thickness of the fin (t_{fin}), as shown in Equation (2) [29].

$$W_{coldplate} = V_{cp} \cdot \rho_{cp}$$

$$\text{where } V_{cp} = f(w, l, n_f, h_f, t_{base}, t_{fin}, \rho_{cp}) \quad (2)$$

An optimization algorithm is then used to calculate the minimum cold plate weight while complying with the operating temperature and pressure limits [30]. The cold plate geometries are used as variables in the optimization while keeping constant fin and base plate thickness. Optimizing the geometry of a cold plate for minimum weight depends heavily on the air mass flow rate during operation. Through sweep analysis, the air mass flow rate that results in the minimum total weight of the cold plate is determined, as well as the

total pressure drop across the cold plate and the overall heat transfer coefficient. Once the optimal air mass flow rate is identified, a ducting system is designed to meet the required pressure drop at the design conditions. The designed cold plate and ducting system model is built in Simcenter Amesim software by using “Gas Mixture” library.

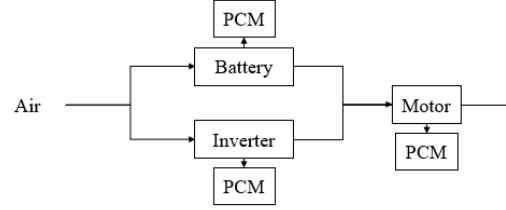


Figure 2. TMS design schematic

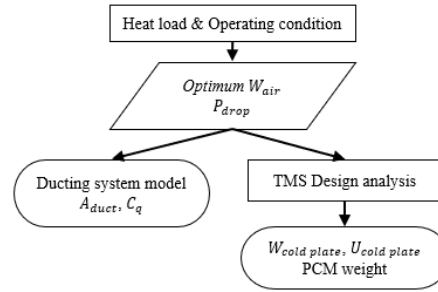


Figure 3. TMS design process

PCMs are used as heat storage system for the flight phase where ram air is limited. As the operating temperature limit is different for each electric component, various types of PCMs are considered, with their properties shown in Table 2 [31]–[33]. Two assumptions are made in the PCM heat storage system: sufficient heat transfer rate is guaranteed to the PCMs while cooling and the PCMs are considered thermally insulated when the heat storage system is inactive. The weight of the PCM is calculated by using latent heat of fusion (LHF) and its heat capacity from ambient temperature to the melting temperature, as shown in Equation (3).

$$W_{PCM} = \frac{Q_{store} - W_{PCM} \cdot c_{p,solid}(T_{melting} - T_{initial})}{LHF} \quad (3)$$

Table 2. PCM properties [31]–[33]

Component	Battery	Inverter	Motor	
			Current	Advanced
Paraffins	$C_{22}H_{46}$	$C_{26}H_{54}$	$C_{34}H_{70}$	$C_{30}H_{62}$
$T_{melting}$ [°C]	44	56.3	75.9	65.4
LHF [kJ/kg]	249	256	269	251

ρ [kg/m ³]	778	800	800	810
$c_{p,solid}$ [kJ/kg/K]	1.51	1.80	1.85	1.91
$c_{p,liquid}$ [kJ/kg/K]	2.38	2.37	2.40	2.17

To consider the integration of the designed TMS to the platform, the XV-15 aircraft dimension data are taken into account. As the cold plate design tends to have a higher value of width for weight minimization, the width in the optimization process is constrained to a maximum value of 1.5m in accordance with the fuselage width (1.52m, [34]).

Additionally, for the advanced technology assessment, different aircraft components are scaled accordingly to reflect to the expected technology improvements. The anticipated weight scaling factor for structure, auxiliary system, and turboshaft engine is 0.93, while the specific fuel consumption is expected to scale down with a factor of 0.91 at advanced technology level [10]. The designed TMS consists of cold plate, PCMs and ducting system, and respective weight is considered as weight penalty to the overall PHE tilt rotorcraft system. Apart from the weight penalty the drag power induced at the diffuser inlet and nozzle outlet should be considered, as the ram air duct system is assumed to be integrated with negligible exposure to the surroundings. Note that the impact due to the additional wetted area is not considered in the study. The spillage drag power induced at the diffuser inlet and the drag power induced at the nozzle outlet are calculated according to Equation (4) [6] and Equation (5). In the ducting system model, a diffuser angle of 8 degrees and nozzle angle of 30 degrees is applied [6], [35]. The length of the diffuser and nozzle is calculated by assuming fixed contraction. The ducting system weight is calculated accordingly by assuming the length of each coolant line as 1m.

$$P_{spill} = K_{spill}[W_{air}(v_1 - v_0) + A_1(p_1 - p_0)] \times v_0 \quad (4)$$

$$P_{nozzle} = W_{air}(v_{exit} - v_0) \times v_0 \quad (5)$$

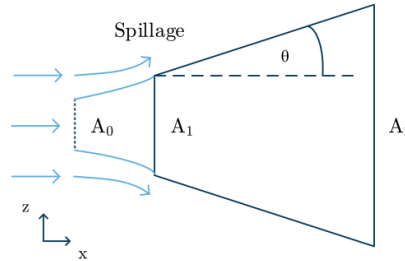


Figure 4. Diffuser model [6]

Finally, the induced drag power due to the TMS integration is calculated and added to the propulsion power requirement in the mission.

DEFINITION OF TEST CASE

The configuration selected for this study is a civil rotorcraft inspired by the NASA XV-15 [18] developed and validated in [12]. This configuration combines high-speed cruise and vertical take-off and landing capabilities, which are key factors for air taxi operations. The baseline configuration is powered by two nacelle-mounted turboshaft engines and has been extensively documented in [18], [36]. An engine based on the General Electric T700-GE700 suitable for this application is used as the baseline [21]. The PHE tiltrotor variant is selected based on [12] and the top-level parameters are presented in Table 3. The sizing conditions selected for the design and size the TMS of PHE VTOL aircraft system are described herein. Note that the hot day condition (ISA 24.85) is considered as default in the study to ensure the capability of the designed TMS in the extreme condition. The selected PHE tiltrotor propulsion system has been optimized for energy efficiency without any TMS consideration, and the optimum power ratio and energy ratio given in Table 3 are the obtained results [12]. As seen the optimum power ratio is 0.79, while Danis et al. [10] suggested a value of 0.84 in their work for this aircraft.

$$Power\ ratio = \frac{P_{GT}}{P_{total}} \quad (6)$$

$$Energy\ ratio = \frac{E_{GT}}{E_{total}} \quad (7)$$

Table 3. PHE powertrain sizing

Maximum power rating [kW]	1,210	
Optimum power ratio [-]	0.79	
Maximum motor power [kW]	254.1	
	Current	Advanced
Motor weight [kg]	31.8	25.5
Inverter weight [kg]	31.8	25.4
Battery sizing result		
Energy consumption [kWh]	1005.6	
Optimum energy ratio	0.85	
Battery energy [kWh]	150.8	
	Current	Advanced
Battery weight [kg]	488.3	203.5

Design mission

A 700km range and 6,500kg GTOW mission is selected as the design mission in accordance with the design range of the XV-15 aircraft. The required propulsion power profile and altitude profiles for the mission are shown in Figure 5.

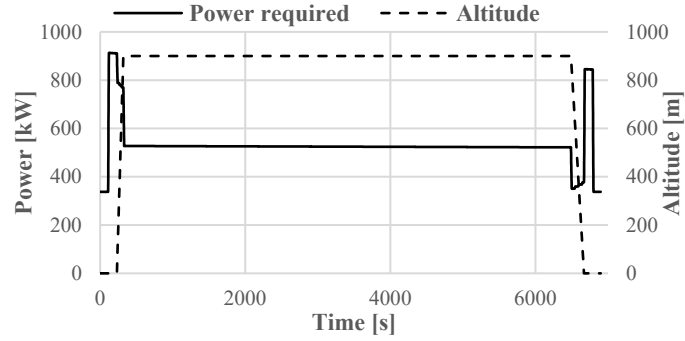


Figure 5. Design mission profile - 700km range & 6,500kg GTOW

Failure-mode mission

It is important to ensure the capability of the designed aircraft system for potential engine-out scenarios. Hence, 5 minutes OEI contingency hover condition is applied to design the TMS to comply with the failure-mode mission requirement [10]. The PHE tiltrotor is set to operate at the maximum power rating for the failure mode mission, where the maximum motor power available is applied [12].

Off-design mission

As discussed, the UAM application is designed to operate frequent short-range missions. Hence, two consecutive 150km missions without recharging in between is used as an off-design mission to assess the TMS performance for frequent short mission capability. Figure 6 represents the propulsion power requirement and altitude profile for the applied off-design mission.

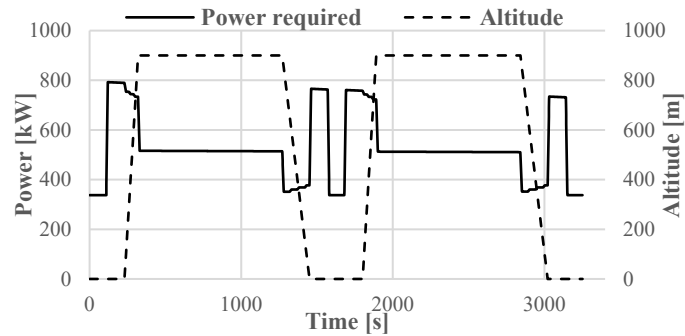


Figure 6. Off-design mission profile – 2 consecutive 150km range & 6,000kg GTOW

RESULTS

The obtained results from the defined test case are discussed herein. The results are divided into the sizing of the TMS for the design mission, certification condition and off-design mission. The impact of the TMS integration for the selected design conditions are evaluated and discussed. Next, the overall performance of the integrated PHE tiltrotor with the TMS are quantified and discussed at mission level.

DESIGN MISSION

Based on the defined battery energy capacity, the optimized PMS for each mission segment is presented in Table 4. Note that during the PMS optimization, the power ratio is constrained to not exceed the maximum electric motor power available which has been defined in the EPS sizing.

Table 4. Optimized PMS for the design mission

Mission Phase	Power ratio	
	Current	Advanced
Idle	1.000	1.000
Hover	0.826	0.816
Climb	0.891	0.818
Cruise	0.897	0.896
Descent	1.000	1.000
Hover	0.879	0.856
Idle	1.000	1.000

As seen in Figure 7, the waste heat produced at hover is significantly higher than the one produced during cruise. Sizing the ram air cold plate at hover would result in an excessively oversized TMS and hence, it is avoided. The ram air cold plate is designed at nominal conditions (Cruise), and in order to comply with the transient heat spike during hover and climb, where ram air inlet is limited, the PCMs are used as heat storage system. Also, the waste heat from the electric motor and inverter is reduced at the advanced technology level due to efficiency improvement.

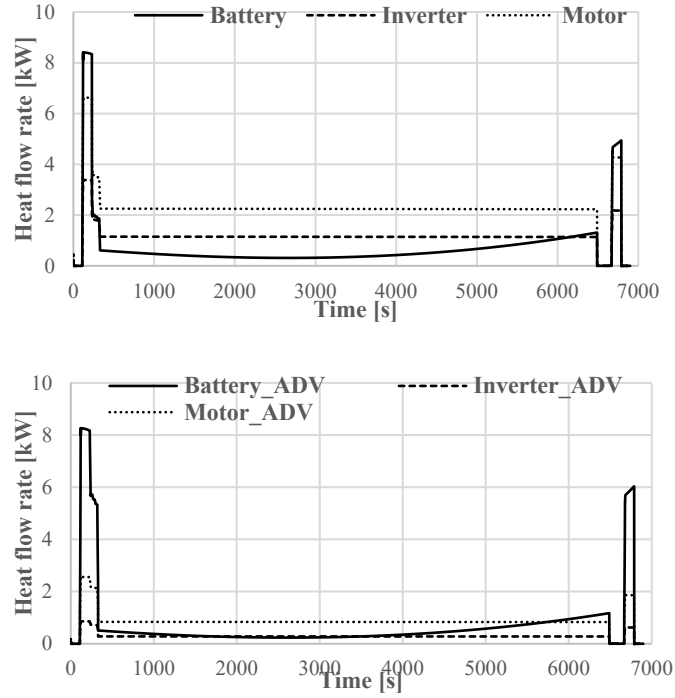


Figure 7. Waste heat profile of the EPS in the design mission – Current (top); Advanced (bottom)

Table 5. TMS design results – Design mission

	Current			Advanced		
	<i>Batt</i>	<i>Inv</i>	<i>Mot</i>	<i>Batt</i>	<i>Inv</i>	<i>Mot</i>
$\dot{Q}_{design}$ [W]	850	1150	2260	800	290	840
W_{air} [kg/s]	0.673	0.289	0.962	0.623	0.069	0.692
U [W/K]	271.8	75.2	67.9	257.5	19.2	36.6
Width [m]	1.00	0.70	0.99	1.00	0.30	0.80
$W_{cold\ plate}$ [kg]	5.51	1.97	2.18	4.58	0.81	1.66
W_{PCM} [kg]	0.00	0.68	2.09	6.14	0.21	0.52
W_{duct} [kg]	8.98			7.69		
W_{TMS} [kg]	21.41			21.42		

The TMS design results for both current and advanced technology levels are presented in Table 5. Unlike the waste heat profile of the inverter or motor, the waste heat flow rate of the battery is not steady during the cruise as the efficiency is dependent on the operating setting and SOC. Hence, the design waste heat flow rate for the battery cold plate is defined at a value close to the maximum, to ensure

the battery operating temperature does not exceed the limit. Also, a fixed area ratio is applied to the parallel coolant line to provide higher coolant mass flow for the battery cold plate as the operating temperature limit is far more stringent than the inverter. Note that the PCM is not required for the battery at the current technology level due to the vast amount of battery weight and hence thermal mass. The operating temperature profile of the component throughout the flight mission is well within the requirement as shown in Figure 8.

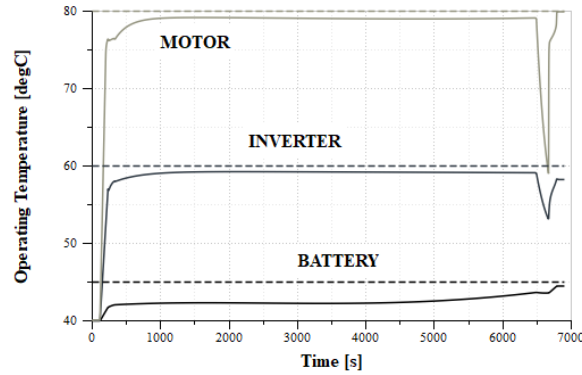


Figure 8. Operating temperature profile in the design mission – Current

FAILURE-MODE REQUIREMENT

During OEI contingency hover for 5 minutes, the ram air inlet is highly limited, hence PCM is used as the TMS for the failure-mode mission. As the battery efficiency is variable with respect to the operating condition, the waste heat load changes accordingly. Moreover, the initial temperature of the PCM would affect the results as its heat capacity is considered as well.

Hence, two scenarios are considered for the failure-mode requirement targeting to capture a wider failure design space:

1. Engine failure occurs at the beginning of the mission (Initial SOC of 95%; $T_{initial} = T_{steady\ state}$)
2. Engine failure occurs at the end of the mission (Initial SOC of 40%; $T_{initial} = T_{melting, PCM}$)

The required PCM weight for each component is calculated and presented in Table 6. As expected, the required PCM weight is higher for the second scenario, hence it is taken as the TMS design result to comply with the failure mode to occur at any point of the mission. It is note that if the core gas turbine is not scaled down, this step will not be necessary.

Table 6. TMS design results – Failure-mode mission

	Current			Advanced		
	<i>Batt</i>	<i>Inv</i>	<i>Mot</i>	<i>Batt</i>	<i>Inv</i>	<i>Mot</i>
Occur at the beginning of the mission						
Heat load [kJ]	7110.4	1617.3	3169.9	6632.8	388.1	1158.5
$SOC_{initial}$ [%]	95%			95%		

$T_{initial}$ [°C]	$T_{steady\ state}$					
$W_{PCM,OEI}$ [kg]	29.93	6.06	10.14	27.92	1.45	4.14
$W_{PCM,total}$ [kg]	48.90			40.39		
Occur at the end of the mission						
Heat load [kJ]	8224.0	1617.3	3169.9	7597.8	388.1	1158.5
$SOC_{initial}$ [%]	40%			40%		
$T_{initial}$ [°C]	$T_{melting,PCM}$					
$W_{PCM,OEI}$ [kg]	36.33	6.95	12.96	33.56	1.67	5.08
$W_{PCM,total}$ [kg]	59.01			47.18		

OFF-DESIGN MISSION

The PHE XV-15 aircraft model with the integrated TMS is assessed at off-design level for a two consecutive 150km flights. Similarly, the PMS is optimized for minimum energy consumption (Table 7). The power ratio is constrained to not exceed the maximum electric motor power available. Note that the PCM is utilized up to the point that it can address an OEI event.

Table 7. Optimized PMS for the off-design mission

Mission phase	Power ratio	
	Current	Advanced
Idle	1.000	1.000
Hover	0.732	0.725
Climb	0.751	0.724
Cruise	0.801	0.781
Descent	0.889	0.815
Hover	0.740	0.731
Idle	1.000	1.000
Hover	0.741	0.840
Climb	0.723	0.729
Cruise	0.782	0.783
Descent	0.867	0.817
Hover	0.764	0.730
Idle	1.000	1.000

The waste heat load profile of the EPS components throughout the mission is presented in Figure 9. As the DOH is increased for the off-design mission, the waste heat load produced by the EPS is significantly increased compared to the design mission.

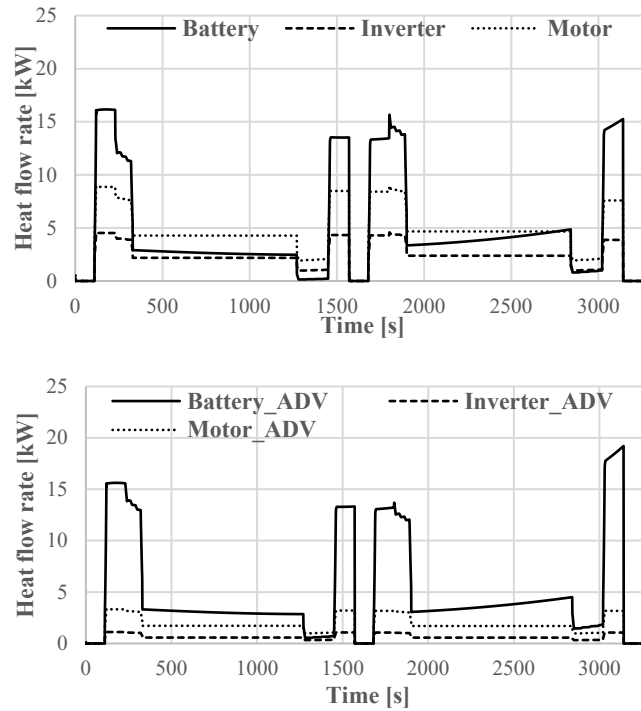


Figure 9. Waste heat profile of the EPS in the OD mission – Current (top); Advanced (bottom)

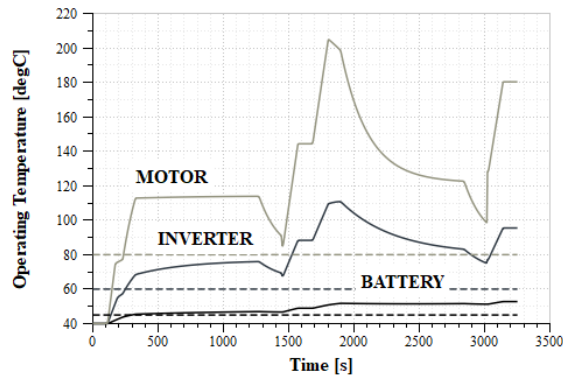


Figure 10. Operating temperature profile in the OD mission – Current

This results inevitably in an increase of the component operating temperatures which exceed the limits (Figure 10). It is thus realized that sizing for the design mission without considering the potential off-design conditions for this configuration results in an infeasible TMS design. In order to tackle this problem, two approaches are considered to bring the operating temperatures of the off-design mission within the assumed limits:

A. Modified PMS

B. Redesigned TMS

Integration of an oversized TMS that will allow to fully utilize the battery (Approach B) or the modification of the energy management at off-design that doesn't require any resizing of the system but penalizes performance as the battery cannot be depleted to the maximum allowable (Approach A).

A. Modified PMS

As a first approach, the EPS-TMS architecture remains unchanged and the PMS is shifted to lower power ratios so that the contribution of the electric system and hence, the heat rejected reduces. This is done by adjusting the power ratio throughout the mission to comply with the operating temperature requirement while ensuring maximum battery energy utilization (Table 8).

Table 8. Modified power management strategy for the off-design mission

Mission phase	Power ratio	
	Current	Advanced
Idle	1.000	1.000
Hover	0.800	0.75
Climb	0.916	0.828
Cruise	0.891	0.870
Descent	0.880	0.900
Hover	0.897	0.850
Idle	1.000	1.000
Hover	1.000	1.000
Climb	0.950	0.950
Cruise	0.892	0.885
Descent	0.870	0.860
Hover	1.000	1.000
Idle	1.000	1.000

With the modified PMS, the operating temperature of the components has complied with the requirements, as shown in Figure 11.

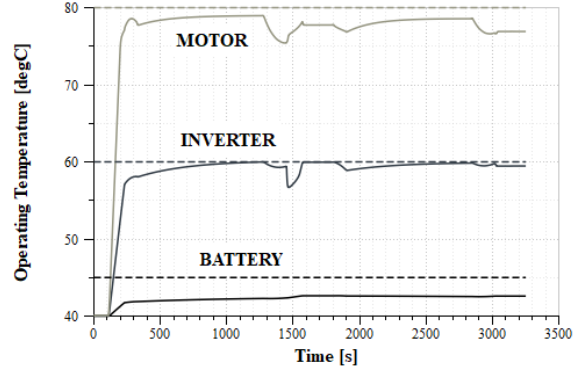


Figure 11. Operating temperature profile in the OD mission – Approach A; Current

B. Redesigned TMS

The second approach involves the TMS redesign with respect to the heat loads generated at the off-design mission. The redesigned TMS results for both current and advanced technology levels are presented in Table 9. As seen in the obtained results, the weight penalty of the redesigned TMS is significant highlighting that oversizing the TMS for off-design may not be the best approach.

Table 9. TMS redesign results – Off-design mission

	Current			Advanced		
	<i>Batt</i>	<i>Inv</i>	<i>Mot</i>	<i>Batt</i>	<i>Inv</i>	<i>Mot</i>
Q_{design} [W]	4790	2450	4800	4020	580	1730
W_{air} [kg/s]	2.777	0.309	3.085	2.451	0.272	2.724
U [W/K]	1807.9	218.7	141.8	1469.0	33.7	73.9
Width [m]	1.50	0.65	1.02	1.50	0.67	1.01
$W_{cold\ plate}$ [kg]	100.7	16.0	18.1	66.1	1.7	7.9
W_{PCM} [kg]	28.8	11.4	14.5	29.8	2.3	3.7
W_{duct} [kg]	13.4			12.6		
W_{TMS} [kg]	202.8			124.0		
ΔW_{TMS} [%]	847%			479%		

In both cases, the components operating temperature profile are within the assumed limits, as shown in Figure 12. However, the degree of battery energy utilization in Approach A is significantly lower as shown in Table 10. Based on existing literature [12], hybrid electric configurations perform better at lower ranges as the battery energy utilization increases. It is however demonstrated here that utilizing

the battery at its maximum energy capability would result in excessive heat loads that cannot be handled by the TMS designed for the design mission. The effect of the TMS requirements on the overall mission performance is analyzed hereafter.

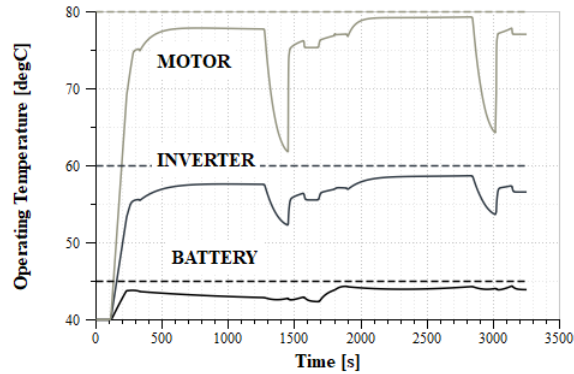


Figure 12. Operating temperature profile in the OD mission –Approach B; Current

Table 10. Battery state of charge at the end of the mission

Technology level	SOC_{end} [%]	
	Approach A	Approach B
<i>Current</i>	65.9%	20.3%
<i>Advanced</i>	61.4%	20.9%

IMPACT OF TMS INTEGRATION

Impact on weight

As discussed, the TMS is designed based on three different design and off-design sizing conditions. The TMS weight increases progressively when moving from the design mission to the off-design sizing conditions (Figure 13). From this figure, the significant sensitivity of the sizing conditions selected on the overall system weight is obvious. For current technology levels, designing the TMS based on the conventional sizing mission results in approximately 43 kg of added system weight. The OEI condition results in more than three times the weight of the system sized at the design mission. This is due to the fact that a considerable amount of PCM weight is required to comply with the failure mode mission capability. If the system is oversized to comply with the off-design heat loads (Approach B), then its weight result to almost zeroing any payload capability.

Based on the TMS design and mission level assessment, the weight breakdown of the XV-15 aircraft with integrated TMS is shown in Table 11. As the GTOW for the mission has been fixed, the amount of available payload is varied with respect to the designed aircraft

system and fuel burnt during the mission. The weight breakdown results indicate the impact of the technology level is significant on the designed system weight of the PHE aircraft.

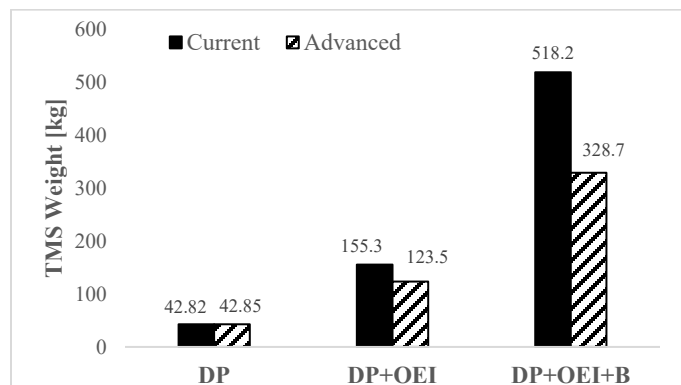


Figure 13. TMS sizing results progression

Table 11. Overall weight breakdowns

Weight breakdown [kg]		BSE	HEP	HEP B	BSE ADV	HEP ADV	HEP B ADV
Aircraft structure [10]		1511.4	1511.4	1511.4	1410.6	1410.6	1410.6
Fixed equipment group [10]		1017.4	1017.4	1017.4	1017.4	1017.4	1017.4
Propulsion group		1564.9	2681.4	3044.2	1460.6	1959.7	2164.9
Powerplant total		492.1	349.7	349.7	459.3	326.4	326.4
EPS		-	1103.6	1103.6	-	508.7	508.7
TMS		-	155.3	518.2	-	123.5	328.7
Auxiliary system [10]		1072.7	1072.7	1072.7	1001.2	1001.2	1001.2
Empty weight		4116.3	5232.8	5595.7	3911.3	4410.4	4615.6
DP	GTOW	6500.0	6500.0	6500.0	6500.0	6500.0	6500.0
	Fuel burnt	685.6	598.0	612.6	624.5	540.9	552.6
	Payload available	1698.1	669.2	291.7	1964.2	1548.7	1331.8
OD	GTOW	6000.0	6000.0	6000.0	6000.0	6000.0	6000.0
	Fuel burnt	329.8	288.5	267.5	300.3	258.7	241.0
	Payload available	1553.9	478.7	136.8	1788.4	1330.9	1143.4

The impact on the TMS integration is illustrated in Figure 14, where the value represents the payload available deviation against the identical aircraft system without considering TMS. The top figure represents the impact on the payload available for the design point mission, whereas the bottom figure refers to the off-design mission. The redesigned TMS (Approach B) at off-design mission is assessed

for the design point mission, which is indicated as “DP+OEI+B” case in the figure. The penalty in payload for current technology is greater than the advanced technology and becomes maximum when the TMS is oversized at off-design mission (Approach B).

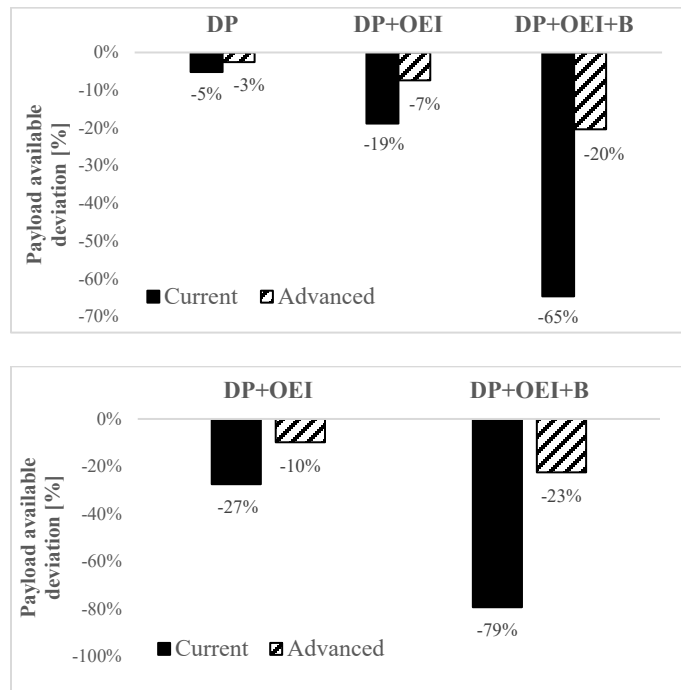


Figure 14. TMS integration impact on the payload available – DP mission (top); OD mission (bottom)

Impact on induced drag

The impact of the TMS integration on fuel consumption is presented in Table 12. Again, the value in the table represents the fuel burnt deviation against the identical aircraft system with the same PMS, but without considering TMS integration. As expected, the fuel burn penalty caused by the redesigned TMS (Approach B) is much greater, as the ram air mass flow rate inlet to the redesigned TMS is significantly higher, and the induced drag power is proportional to the air mass flow rate.

Table 12. TMS integration impact on the fuel consumption

<i>Sizing mission</i>	Fuel burn deviation [%]		
	<i>TMS Sizing point</i>	<i>Current</i>	<i>Advanced</i>
DP	DP+OEI	0.8%	0.5%
	DP+OEI+B	3.3%	2.7%
OD	DP+OEI	0.6%	0.4%
	DP+OEI+B	2.3%	2.4%

Impact on normalized fuel consumption

The normalized fuel consumption with respect to the payload available in the system is shown in Figure 15. The effect of TMS added weight and hence, reduction of useful load for each sizing condition has a significant effect on the fuel per passenger. It is noted that the 'No TMS' corresponds to the PHE tiltrotor without a cooling system for the electrical components. It is an unfeasible design used for comparison purposes. Specifically, when the TMS is sized for the baseline design mission, a 5% penalty in fuel per payload is reported relative to the hybrid configuration without TMS consideration. The necessity to carry enough PCM to comply with the OEI requirement and withstand the heat spikes during this condition translates into a 14% penalty relative to the configuration without TMS consideration. For advanced technology, these penalties reduce to 3% and 9% for the design mission and OEI respectively.

Next, the off-design condition was assessed. It has been already reported that the designed TMS is not able to comply with the operating temperature requirement at off design mission, which is highlighted in Figure 15 (bottom). It is observed that despite the higher fuel burn reduction by the maximum battery energy utilization in Approach B, the normalized fuel consumption with respect to payload is considerably increased due to the significant increment in the redesigned TMS weight. The penalty in fuel per passenger of Approach A is 15% relative to the identical system which is assumed to be capable of depleting battery energy to the desired level. Approach B leads to almost four times higher normalized fuel highlighting that the effect of added weight is significant and hence,

oversizing the TMS for OD conditions is not appropriate for this case. This effect in fuel becomes less pronounce as technology level advances with calculated penalties of 0.5% (Approach A) and 16% (Approach B).

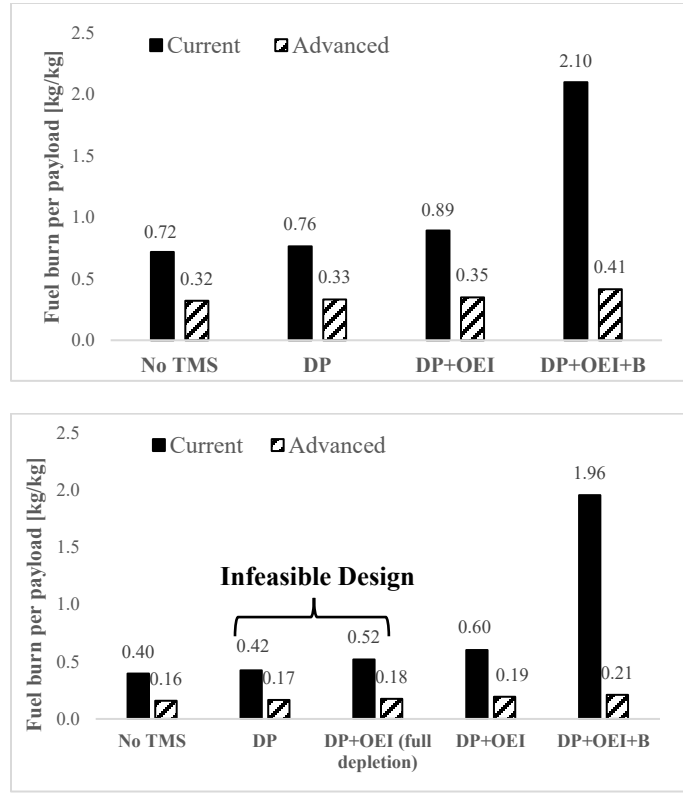


Figure 15. Fuel consumption per payload – Design mission (top); OD mission (bottom)

OVERALL PERFORMANCE ASSESSMENT

In this section, the performance impact of the designed PHE VTOL aircraft with an integrated TMS model at the mission level is presented.

It has been discussed that the payload penalty due to aircraft electrification is significant due to the energy density level of the electric storage system [10], [22]. The payload penalty due to the weight of the designed TMS is considerable likewise (Figure 14).

Thus, normalized parameters with respect to the payload available are used to assess the overall performance of the PHE VTOL aircraft. The energy to revenue work (ETRW) ratio defined in Equation (8) is used to assess the energy efficiency of the hybrid electric system [22], [37]. The pollutant gaseous emission reduction is also calculated. The CO₂ emission is directly calculated based on the fuel consumption, while NO_x emission is calculated based on P3T3 method which is described in [22]. Similarly, CO₂ per payload and NO_x per payload are used to assess the greenhouse gas emission of the PHE tilt rotor.

$$ETRW = \frac{m_f \cdot LHV + E_{electric}}{m_{payload} \cdot g \cdot Range} \quad (8)$$

Besides, it is important to note that the actual payload that can be carried by aircraft would not be identical to the amount of useful load available. This could be either due to the passenger seat configuration or due to the volumetric constraint in the aircraft fuselage. Thus, the maximum payload of the tiltrotor is capped at 1,000kg in the study, which is approximately equivalent to 10 PAX for a commercial aircraft mission [38]. Note that the remaining weight allowance is assumed to be filled with extra fuel to have longer range capability.

In order to assess the benefits via the electrification, the analyzed performance parameters against the baseline configuration without electrification are evaluated and depicted in Figure 16 and Figure 17.

Figure 16 represents the obtained results for the design mission – 700km ranges with a 6,500kg GTOW mission. Likewise with the fuel block analysis, the redesigned TMS approach is analyzed for comparison. The obtained results indicate that energy efficiency and greenhouse gas emission per payload are penalized by 30% through aircraft hybridization at the current technology level, whereas approximately 10% improvement is expected at the advanced technology level for the considered mission. Moreover, the weight penalty from the redesigned TMS has significant impact on the performance for current technology, but the impact is significantly reduced at advanced technology level.

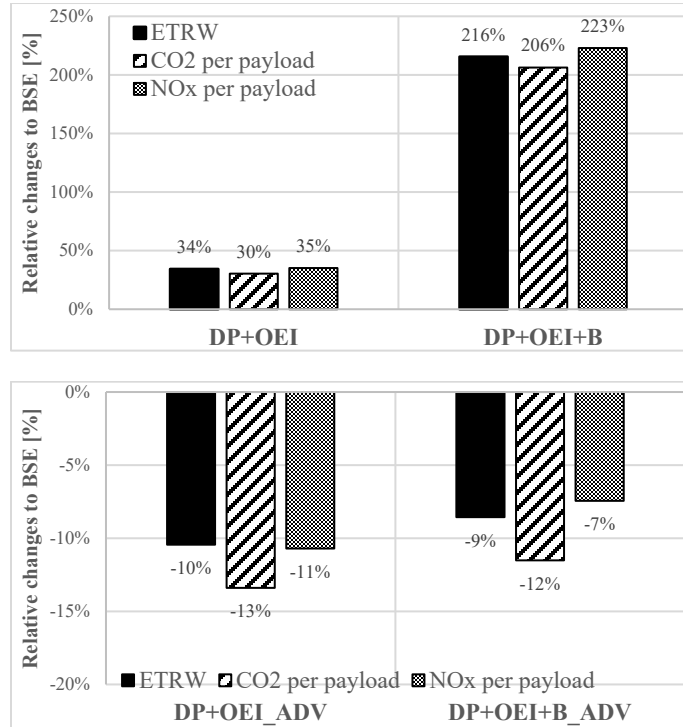


Figure 16. Performance parameter changes to Baseline configuration– Design mission; Current (top) and Advanced (bottom)

As discussed, two approaches are used at off-design to comply with the electric component operating temperature limits. The performance parameters against the baseline configuration for both approaches are presented in Figure 17.

At the current technology level (top figure), neither modifying PMS (Approach A) nor redesigning TMS (Approach B) can outperform the baseline configuration. Moreover, the performance penalties of electrification are considerably higher than that of the design mission. Besides, it is interesting to note that the performance penalties caused in Approach B are far more significant than that in Approach A, despite much higher battery energy utilization. This is due to the significant weight penalty induced by the TMS redesign in Approach B.

In contrast, at the advanced technology level, the performance improvement generated in Approach B is higher than that in Case A, as shown at the bottom of Figure 17. Moreover, both cases have outperformed the baseline configuration at the advanced technology level.

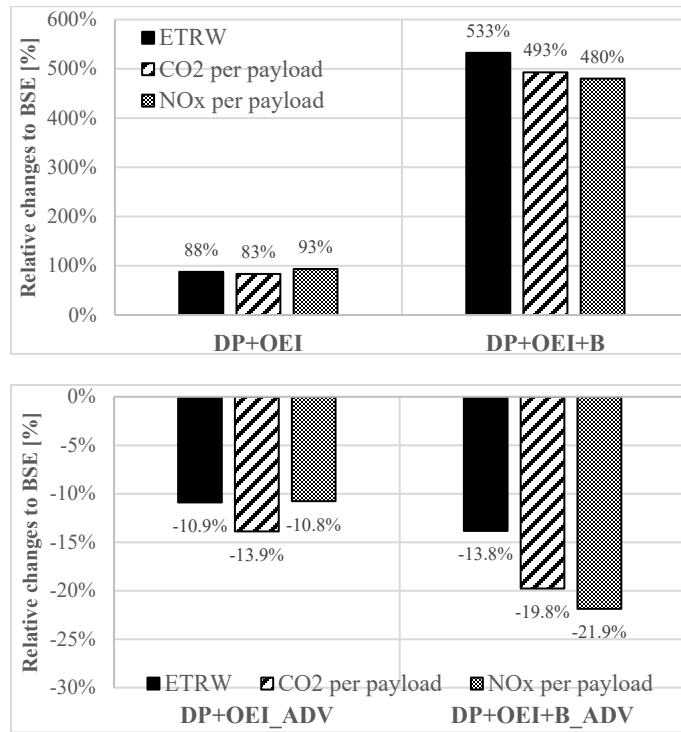


Figure 17. Performance parameters changes to Baseline configuration – OD mission; Current (top) and Advanced (bottom)

CONCLUSION

This study discusses the integration of a TMS into a PHE tiltrotor aircraft modeled after the NASA XV-15 and evaluates the resulting performance. The expected benefits of electrification are evaluated in the short term and long term by considering both current and advanced technology levels. The TMS is evaluated under three sizing conditions: 1) a design mission with a range of 700km and a

GTOW of 6,500kg; 2) a failure-mode requirement of OEI contingency hover for 5 minutes at the beginning and end of the mission; 3) an off-design mission with two consecutive 150km range and a GTOW of 6,000kg.

The designed TMS consists of a direct ram air-cooled cold plate as a cooling device and PCMs as a heat storage system. The EPS and TMS are sized for the design mission, and then the PCMs added to comply with the failure mode requirements. Followingly, the designed PHE VTOL aircraft system with an integrated TMS model is assessed at mission level for both design and off-design missions.

The results of the design analysis showed that the proposed TMS is able to effectively handle the amount of waste heat produced by the electric components for the design mission. The fuel penalty by TMS integration at aircraft level is 5% fuel compared to the PHE configuration without considering TMS. This penalty increases to 14% when OEI capability is considered. These penalties are reduced to 3% and 9% for the design mission and OEI respectively for advanced technology.

However, the designed TMS was not capable of maintaining the component operating temperature within the limit in the off-design mission, as the DOH is increased. Two approaches are conducted to comply with the operating temperature requirement: Approach A – modify the PMS to reduce the waste heat load of the components; Approach B – redesign the TMS based on the off-design mission requirement. Both approaches are managed to comply with the operating temperature requirement at a cost, insufficient battery energy utilization in Approach A and increased weight penalty via redesigning TMS in Approach B. The significant weight penalty from redesigning TMS is due to the small temperature difference between the ram air inlet and the battery operating temperature limit. Approach A results in a 15% fuel penalty per payload relative to the system sized at the design mission with OEI capability. On the other hand, Approach B leads to almost four times higher fuel per payload due to the significant added weight of the redesigned TMS. Therefore, it is not recommended to oversize the TMS for off-design. These results highlight the importance of selecting appropriate TMS design points and the need to assess its performance against different mission requirements.

Furthermore, normalized performance parameters associated with energy efficiency and greenhouse gas emission per payload are quantified to assess the electrification benefits of the PHE-TMS tiltrotor aircraft model. The results indicate that benefits are expected for future technology, providing 10% improvement in performance and emissions.

As a future work, establishment of a multidisciplinary optimization methodology to consider the TMS design impact in the early design phase is suggested.

ABBREVIATIONS

ρ	Density
A	Area

ACM	Air Cycle Machine
ADV	Advanced
BSE	Baseline
c_p	Specific heat capacity
C_q	Flow coefficient
DOH	Degree of Hybridization
E	Energy
EPS	Electric Propulsion System
ETRW	Energy to Revenue Work
g	Gravitational constant
GTOW	Gross Take-off Weight
HEPS	Hybrid Electric Propulsion system
ISA	International Standard Atmosphere
K_{spill}	Empirical spillage coefficient
LHF	Latent Heat of Fusion
LHV	Lower heating value
LHV	Lower heating value
LP	Low-pressure
m	Mass
NLPQL	Nonlinear Programming Quadratic Line search
OD	Off Design
OEI	One Engine Inoperative
P	Power
PAX	Passengers

PCM	Phase change material
PMS	Power Management Strategy
SFC	Specific Fuel Consumption
SOC	State of Charge
T	Temperature
TEC	Thermoelectric Cooling
TMS	Thermal Management System
U	Overall heat transfer coefficient
UAM	Urban Air Mobility
v	Velocity
VTOL	Vertical Take-off and Landing
W	Mass flow rate

SUBSCRIPTS

f	Fuel
GT	Gas Turbine

REFERENCES

- [1] Holden J. and Goel N., “Fast-Forwarding to a Future of On-Demand Urban Air Transportation,” pp. 1–98, 2016, [Online]. Available: <https://www.uber.com/elevate.pdf>
- [2] Volocopter, “THE ROADMAP to scalable urban air mobility White paper 2.0,” 2021.
- [3] Johnson W., Silva C., and Solis E., “Concept vehicles for VTOL air taxi operations,” *Proceedings of the AHS International Technical Meeting on Aeromechanics Design for Transformative Vertical Flight 2018*, 2018.
- [4] Chapman J. W., Hasseeb H., and Schnulo S., “Thermal management system design for electrified aircraft propulsion concepts,” *AIAA Propulsion and Energy 2020 Forum*, pp. 1–23, 2020, doi: 10.2514/6.2020-3571.
- [5] Annappagada S. R., MacDonald M., A. Sur, Mahmoudi R, and Lents C., “Hybrid Electric Aircraft Battery Heat Acquisition System,” *2018 AIAA/IEEE Electric Aircraft Technologies Symposium, EATS 2018*, pp. 1–13, 2018, doi: 10.2514/6.2018-4992.
- [6] Kellermann H., Lüdemann M., Pohl M., and Hornung M., “Design and optimization of ram air-based thermal management systems for hybrid-electric aircraft,” *Aerospace*, vol. 8, no. 1, pp. 1–22, 2021, doi: 10.3390/aerospace8010003.

- [7] Rheume J. M. and Lents C. E., "Design and Simulation of a Commercial Hybrid Electric Aircraft Thermal Management System," *2018 AIAA/IEEE Electric Aircraft Technologies Symposium, EATS 2018*, pp. 1–9, 2018, doi: 10.2514/6.2018-4994.
- [8] Rheume J. M. and Lents C. E., "Commercial hybrid electric aircraft thermal management system design, simulation, and operation improvements," *AIAA Propulsion and Energy Forum and Exposition, 2019*, no. August, pp. 1–23, 2019, doi: 10.2514/6.2019-4492.
- [9] Rheume J. M. and Lents C. E., "Commercial hybrid electric aircraft thermal management sensitivity studies," *AIAA Propulsion and Energy 2020 Forum*, pp. 1–6, 2020, doi: 10.2514/6.2020-3558.
- [10] Danis R. A., Green M. W., Freeman J. L., and Hall D. W., "Examining the Conceptual Design Process for Future Hybrid-Electric Rotorcraft," *Nasa/Cr—2018–219897*, no. May, 2018.
- [11] Roumeliotis I. *et al.*, "Integrated Systems Simulation For Assessing Fuel Thermal Management Capabilities for Hybrid-Electric Rotorcraft," *ASME Turbo Expo 2020*, pp. 1–13, 2020.
- [12] Saia C. A., Goulos I., Roumeliotis I., Pachidis V., and Bacic M., "Preliminary Design of Hybrid-Electric Propulsion Systems for Emerging Urban Air Mobility Rotorcraft Architectures," *J Eng Gas Turbine Power*, vol. 143, no. 11, Nov. 2021, doi: 10.1115/1.4052057.
- [13] Saia C. A., Roumeliotis I., Goulos I., Pachidis V., and Bacic M., "Design Exploration and Performance Assessment of Advanced Recuperated Hybrid-Electric Urban Air Mobility Rotorcraft," *J Eng Gas Turbine Power*, vol. 144, no. 3, Mar. 2022, doi: 10.1115/1.4052955.
- [14] Padfield G. D., *Helicopter Flight Dynamics*. Oxford, UK: Blackwell Publishing Ltd, 2007. doi: 10.1002/9780470691847.
- [15] Leishman J. G., *Principles of Helicopter Aerodynamics*, 2nd ed. 2016.
- [16] Bhagwat M. J., "Optimum Loading and Induced Swirl Effects in Hover," *Journal of the American Helicopter Society*, vol. 60, no. 1, pp. 1–14, Jan. 2015, doi: 10.4050/JAHS.60.012004.
- [17] Mangler K. W., "Calculation of the induced velocity field of a rotor," Royal Aircraft Establishment, Report Aero 2247.
- [18] Harendra P. B., Joglekar M. J., Gaffey T. M., and Marr R. L., "V/STOL tilt rotor study. Volume 5: A mathematical model for real time flight simulation of the Bell model 301 tilt rotor research aircraft," 1973.
- [19] Hangiu R. P., Filip A. T., Martis C. S., and Biro K. A., "System-level Modeling and Simulation of a Permanent Magnet Synchronous Motor for an Integrated Starter Alternator," *Journal of Electrical and Electronics Engineering*, vol. 5, no. 2, pp. 67–70, 2012.
- [20] Petit M., Marc N., Badin F., Mingant R., and Sauvart-Moynot V., "A tool for vehicle electrical storage system sizing and modelling for system simulation," in *2014 IEEE Vehicle Power and Propulsion Conference, VPPC 2014*, 2014. doi: 10.1109/VPPC.2014.7007010.
- [21] Roumeliotis I., Mourouzidis C., Zafferetti M., Unlu D., Broca O., and Pachidis V., "Assessment of Thermo-electric power plants for rotorcraft application," *J Eng Gas Turbine Power*, vol. 142, no. 5, pp. 1–12, 2020, doi: 10.1115/1.4045103.
- [22] Kang S., Roumeliotis I., Zhang J., Broca O., and Pachidis V., "Assessment of Engine Operability and Overall Performance for Parallel Hybrid Electric Propulsion Systems for a Single-Aisle Aircraft," *J Eng Gas Turbine Power*, vol. 144, no. 4, Apr. 2022, doi: 10.1115/1.4052880.
- [23] Duffy K. P., "Electric Motors for Non-Cryogenic Hybrid Electric Propulsion," Jul. 2015. doi: 10.2514/6.2015-3891.
- [24] National Academies of Science, Engineering and Medicine, "Electric Propulsion," in *Commercial Aircraft Propulsion and Energy Systems Research: Reducing Global Carbon Emissions*, Washington, D.C.: National Academies Press, 2016. doi: 10.17226/23490.
- [25] Pagonis M., "Electrical power aspects of distributed propulsion system in Turbo-electric powered aircraft," PhD, Cranfield University, 2015.

- [26] Jansen R. H., Bowman C., Jankovsky A., Dyson R., and Felder J., "Overview of NASA electrified aircraft propulsion research for large subsonic transports," in *53rd AIAA/SAE/ASEE Joint Propulsion Conference, 2017*, 2017. doi: 10.2514/6.2017-4701.
- [27] Siemens, "Simcenter Amesim Software." 2022.
- [28] Wikner E. and Thiringer T., "Extending battery lifetime by avoiding high SOC," *Applied Sciences (Switzerland)*, vol. 8, no. 10, Oct. 2018, doi: 10.3390/app8101825.
- [29] Pointon J. M., "Thermal management of electromechanical actuation on an all-electric aircraft," MSc Thesis, Cranfield University, 2007.
- [30] Scipy, "SciPy documentation." Oct. 19, 2022.
- [31] Sharma A., Tyagi V. v., Chen C. R., and Buddhi D., "Review on thermal energy storage with phase change materials and applications," *Renewable and Sustainable Energy Reviews*, vol. 13, no. 2. pp. 318–345, Feb. 2009. doi: 10.1016/j.rser.2007.10.005.
- [32] NIST, "NIST Chemistry WebBook," 2022. <https://doi.org/10.18434/T4D303> (accessed Dec. 16, 2022).
- [33] ChemSrc, "Hexacosane," Jan. 23, 2022. https://www.chemsrc.com/en/cas/630-01-3_594658.html (accessed Dec. 16, 2022).
- [34] Maisel M., "Tilt Rotor Research Aircraft Familiarization Document," 1975.
- [35] Louwerse M. C., Jansen H. v, Groenendijk M. N. W., and Elwenspoek M. C., "Nozzle fabrication for micropropulsion of a microsatellite," *Journal of Micromechanics and Microengineering*, vol. 19, no. 4, p. 045008, Apr. 2009, doi: 10.1088/0960-1317/19/4/045008.
- [36] Maisel M., Giulianetti D. J., and Dugan D. C., "The History of the XV-15 Tilt Rotor Research Aircraft from Concept to Flight," 2000. [Online]. Available: <https://www.researchgate.net/publication/24295050>
- [37] Agarwal R. K. and Zhang Z., "Optimization of ETRW (Energy liberated during a flight/revenue work done) of an airplane for minimizing its environmental impact," in *SAE Technical Papers*, 2011. doi: 10.4271/2011-01-2524.
- [38] EASA, "EASA review of standard passenger weights in 2022 shows no significant change," *EASA*, Nov. 09, 2022. Accessed: Dec. 22, 2022. [Online]. Available: <https://www.easa.europa.eu/en/newsroom-and-events/news/easa-review-standard-passenger-weights-2022-shows-no-significant-change>



DSPACE

<https://dspace.org/>

On the design of integrated thermal management systems for eVTOL aircraft

Kang, Sangkeun; Saias, Chana Anna; Roumeliotis, Ioannis; Broca, Olivier

2023-09-28

American Society of Mechanical Engineers (ASME)

<https://doi.org/10.1115/GT2023-103097>

Kang S, Saias C, Roumeliotis I, Broca O. (2023) On the design of integrated thermal management systems for eVTOL aircraft. In: ASME Turbo Expo 2023: Turbomachinery Technical Conference and Exposition, Volume 5, 26-30 June 2023, Boston, Massachusetts, Paper number GT2023-103097