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Rotorcraft Architectures

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Preliminary design of hybrid-electric propulsion systems for emerging UAM rotorcraft architectures

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ABSTRACT

The increasing demands for air-taxi operations together with the ambitious targets for reduced environmental impact have driven significant interest in alternative rotorcraft architectures and propulsion systems. The design of Hybrid-Electric Propulsion Systems (HEPSs) for rotorcraft is seen as being able to contribute to those goals. This work aims to conduct a comprehensive design and trade-off analysis of hybrid

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powerplants for rotorcraft, targeting enhanced payload-range capability and fuel economy. An integrated methodology for the design, performance assessment and optimal implementation of HEPSs for conceptual rotorcraft has been developed. A multi-disciplinary approach is devised comprising models for rotor aerodynamics, flight dynamics, HEPS performance and weight estimation. All models are validated using experimental or flight test data. The methodology is deployed for the assessment of a hybrid-electric tilt-rotor, modelled after the NASA XV-15. This work targets to provide new insight in the preliminary design and sizing of optimally designed HEPSs for novel tilt-rotor aircraft. The paper demonstrates that at present, current battery energy densities (250Wh/kg) severely limit the degree of hybridization if a fixed useful payload and range are to be achieved. However, it is also shown that if advancements in battery energy density to 500Wh/kg are realized, a significant increase in the level of hybridization and hence reduction of fuel burned and carbon output relative to the conventional configuration can be attained. The methodology presented is flexible enough to be applied to alternative rotorcraft configurations and propulsion systems.

Keywords: Urban Air Mobility (UAM), tilt-rotor, Hybrid-Electric Propulsion System (HEPS), optimization, energy-management, preliminary design.

1. INTRODUCTION

1.1 Background

Urban Air Mobility (UAM), often referred to as the ‘mobility in the third dimension’, is an emerging market in the aviation industry that is expected to grow sharply in the next 30 years. It is estimated to reach a market worth of \$500 billion by 2030 [1]. Researchers anticipate that the increasing rates in population and urbanization are essentially leading to a continuous rise of the congestion in urban areas [2]. Air-taxis target to ease this congestion, increase mobility reach and reduce the environmental impact stemming from urban transportation. This application is envisioned to make use of up to 15-passenger, hybrid or all-electric Vertical Take-Off and Landing (VTOL) configurations, operating in an intra-urban environment for short and frequent flights of 50 nautical miles or less [3].

At the same time, the ambitious targets established by the Advisory Council for Aviation Research and Innovation in Europe (ACARE) [4] and National Aeronautics and Space Administration (NASA) [5] to reduce fuel consumption, chemical emissions, as well as noise, have led to a growing interest in exploring alternative energy sources than fossil fuels. Electric propulsion is being considered for VTOL aircraft in particular due to improved flexibility of power distribution, redundancy in case of engine failure, perceived potential for lower maintenance costs and capability to reduce the need for engine oversizing [6].

However, these benefits come at a cost of increased Empty Weight (EW) and complexity of the propulsion system [6,7], and are therefore evaluated in this paper.

Up to this time, approximately 200 unique architectures have been proposed for UAM applications from industry and academia [8]. These configurations that involve fundamentally different aircraft designs, as well as propulsion architectures, are considered complex and unconventional [9].

1.2 Hybrid-electric propulsion

The integration of hybrid-energy systems disrupts the conventional design and sizing process and is associated with new design challenges. This is explained by the fact that, HEPs introduce new design variables and constraints. These relate mainly to technology parameters, degrees of hybridization associated with the various electrical components and application. Additionally, the existence of two energy sources introduces the need to adopt alternative power management strategies, which must be optimised during preliminary design.

HEPs for fixed-wing aircraft have been investigated widely in the literature. Several authors have presented methodologies for sizing of different hybrid configurations [10-13]. Additionally, authors have examined the electrification of small [14] and larger aircraft [15] and stated that improvements in overall performance can be achieved as battery technology advances. On the contrary, Epstein et al. [16] concluded that unless radical technology developments are made, electric propulsion cannot contribute significantly to the reduction of CO₂ emissions for large airliners. However, smaller aircraft, such as regional, general aviation and UAM may be more suitable for electrification.

Isikveren et al. [10] formulated simplified expressions that can describe various HEPs using only two parameters – the degrees of hybridization of energy and power. Riboldi et al. [17] proposed an optimal approach for the preliminary design of a hybrid-electric aircraft, targeting to minimize the aircraft Take-Off Weight (TOW) subjected to multiple operational, certification and safety constraints.

Preliminary design studies of HE architectures for civil rotorcraft configurations are limited [6-7,18]. Roumeliotis et al. [6] presented an assessment of a battery-boosted parallel architecture for a twin-engine medium helicopter using a realistic mission scenario. Based on his analysis, the HE configuration resulted in improved overall mission Specific Fuel Consumption (SFC) relative to the conventional helicopter, especially at part load operations. This improvement was mainly associated with the gas turbine down-sizing, which allowed to operate closer to the maximum power settings.

Danis et al. [7] presented a conceptual design methodology for emerging rotorcraft configurations. In this study, various hybrid tilt-rotor variants were investigated, by re-configuring the NASA XV-15 with a HEPS. A parallel-hybrid configuration provides improved overall performance due to the battery-boosting capability. This resulted in a maximum reduction of 10% in cruise fuel relative to the conventional configuration. This is achieved with a weight penalty of 25% increase in EW compared to the baseline tilt-rotor. Hendricks et al. [9] presented a multidisciplinary approach for the optimization of UAM vehicle propulsion systems applied to a tilt-wing configuration.

The limitations of these studies [6-7,9,18] are mainly associated with the exclusion of power management strategies from the preliminary design process. Additionally, HE rotorcraft are investigated considering fixed electric-system architectures and not exploring the design space [6,18]. It is therefore realized that, although the aforementioned studies provide insight to vehicle and propulsion designers, the evaluation of new architectures necessitates the adoption of a holistic approach. This will enable the potential to investigate the trade-offs between mobility, aircraft performance, as well as environmental impact and economics in a multidisciplinary analysis environment.

1.3 Scope of present work

Hybrid-electric VTOL configurations for UAM applications have drawn significant attention by industry and academia. The literature currently available on comprehensive design and sizing assessments of HEPS for novel rotorcraft reveals a gap in knowledge. A complete assessment of a HE VTOL in terms of its implications on battery weight, powerplant

weight, and associated effects on payload-range capacity has not been addressed in a multidisciplinary environment, with implicit consideration of power management strategies.

The objective of this work is to develop an integrated methodology for the design and optimal implementation of alternative propulsion systems for novel civil rotorcraft. Additionally, this work aims to provide a comprehensive preliminary design and trade-off study for a HE novel UAM configuration through the utilization of a multi-disciplinary framework. The investigated rotorcraft is modelled after a generic tilt-rotor.

An integrated methodology for the preliminary design and analysis of HEPS for rotorcraft configurations is developed. Models for rotor aerodynamics, flight dynamics, mission analysis, gas turbine performance [20] and battery performance are developed and validated for each respective discipline.

The analysis is based on exploring the design space of HE architectures for rotorcraft with an integrated process targeting to establish an optimum synergy between electric and thermal components, while also accounting for aspects such as weight, payload-range capacity and technology level sensitivity. This work aims to provide insights and guidance for preliminary design and sizing of HEPS for novel tilt-rotor aircraft.

2. METHODOLOGY

The modelling approach comprises a series of comprehensive modules, each applicable to the different aspects of rotorcraft and powerplant performance. These contain a rotorcraft performance model coupled with models for engine off-design performance analysis [20], gaseous emissions estimation [21] and weight estimation. The engine performance software is coupled with a battery simulation model [22] and an energy management module to enable analysis of HE architectures. The different simulation modules are integrated within a mission analysis model, incorporating a flight path model based on the World Geodetic System dated in 1984 (WGS 84) [23], as reported by Goulos et. al [24].

2.1 Rotorcraft performance

The integrated approach intends to incorporate the features of the 1st level of Padfield's hierarchical paradigm, appropriate for performance studies [25]. The rotorcraft performance model comprises a set of aerodynamic models for the different aircraft components; rotor, fuselage, empennage and wings as well as a nonlinear trim model.

2.1.1 Rotor aerodynamics

The rotor model implemented combines the basic principles from blade element theory and momentum theory for the evaluation of steady blade airloads within the desired level of fidelity. The combined Blade Element Momentum Theory (BEMT) is well documented in textbooks [26-28].

The rotor model utilizes steady-state blade non-linear BEMT, coupled with different inflow models for axial propeller flight and flight with an edgewise flow component. The rotor is treated as an actuator disk, discretized radially and azimuthally into several non-interacting elements so that each blade section acts as a 2D airfoil to produce aerodynamic forces. The blades are assumed to be rigid bodies, with no flap deflection, which is considered a valid approximation for small-diameter rotors [26].

In hovering or high-speed cruise flight, the flow field is azimuthally axisymmetric. For this operating condition, the rotor aerodynamics model is coupled with Bhagwat's induced inflow model formulated for propellers [30]. The method caters for the inclusion of the swirl induced velocity component, which is ignored by traditional momentum theory [26, 28]. Swirl induced velocity is significant for highly-loaded rotors, such as those of tilt-rotor aircraft [30,31]. The employed method solves simultaneously for the axial and tangential induced velocity components, from the implementation of conservation of axial and angular momentum. The equations presented by Bhagwat [30], originally developed for hovering flight conditions, are modified accordingly to account for flight conditions with $V_\infty > 0$.

In conversion mode, the applicability of BEMT to non-axial flow conditions is limited, as the rotors experience non-axisymmetric flow. Therefore, the rotor aerodynamics model is coupled with the Mangler and Squire inflow model. The Mangler and Squire inflow model is

a simple potential flow based analytical model that has found extensive application [32, 33].

The definition of the inflow is described by Bramwell [27].

Simple BEMT is further modified by replacing the small angles approximation with a large angle approach, as prop-rotors are expected to have large inflow angles [34]. Corrections for tip loss effect are introduced with Prandtl's tip-loss function [35]. Look-up tables for the two-dimensional airfoil sections as a function of local angle of attack and Mach number are used. A globally convergent Powell hybrid method [36] is used to solve the non-linear system of equations expressed by the conservation of tangential and axial momentum.

2.1.2 Non-linear trim procedure

The tilt-rotor is a laterally symmetric vehicle which allows to de-couple the roll-yaw degrees-of-freedom [37]. Experimental look-up tables expressed as functions of incidence and side-slip angles are utilized for the prediction of fuselage forces and moments. The forces and moments of the wing-pylon and empennage are also represented using experimental look-up tables. The aerodynamic coefficients for those are expressed as functions of angle-of-attack, Mach number as well as nacelle-angle (i_N) and flap setting (δ_F). The aerodynamic data of the aforementioned components are extracted for the XV-15 tilt-rotor from Refs. [38-40]. Figure 1 illustrates the two-dimensional longitudinal rigid-body tilt-rotor model.

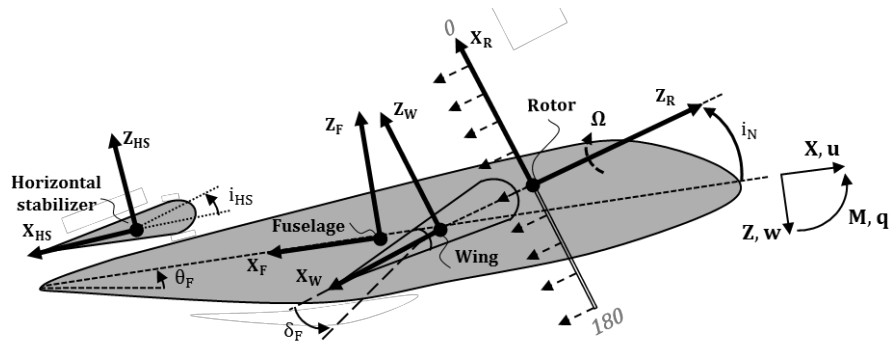


Figure 1 Tilt-rotor rotorcraft free-body diagram

The downwash of the rotor is accounted for on the wing and empennage. The wing is modelled considering the portion of it in the freestream and slipstream separately. The area of the wing affected by the rotor wake depends on the wake contraction ratio and is evaluated based on the nacelle angle and freestream velocity, and rotor thrust coefficient

[41,42]. A Newton-Raphson approach is applied to obtain the control and fuselage attitude angles required to achieve an equilibrium trim state for a specified operating condition.

2.2 Gas turbine performance (TURBOMATCH)

The engine performance simulation model TURBOMATCH (Cranfield University) [20], has been incorporated within the simulation framework. TURBOMATCH utilizes zero-dimensional aerothermal analysis, employing discrete component maps. It is a well-validated software that has found extensive application in the literature for gas turbine design and off-design analysis [43]. Additionally, TURBOMATCH has been successfully deployed in several studies for turboshaft engines for rotorcraft applications [44,45].

2.3 HEP performance & energy management

In order to model innovative HEPs, the development of a steady-state performance model for any user-defined architecture and degree-of-hybridization is essential. TURBOMATCH is coupled with a battery simulation model for Lithium-Ion type (Li-Ion) batteries developed based on [11]. Fixed efficiency models are assumed for the rest of the electrical components using an electric power chain approach as proposed by Isikveren [10]. A flexible energy management module is incorporated within the HEPs module to allow the control of the power distribution between the two sources at aircraft and mission level.

2.4 Propulsion system weight estimation

In order to assess HE configurations, it is essential to include weight models for the engines and the different electric components. For a retrofit rotorcraft, the turboshaft engines can be scaled down with the introduction of the electric support. The engine scaling effect on the overall engine weight is evaluated using ATLAS, a software tool developed at Cranfield University [49]. Moreover, the integration of weight models for electric components is essential. The weights for the electric components are highly dependent on the technology advancements. The battery weight is estimated based on the maximum energy and power. The motors and generators are sized based on the method reported in

[50]. In the absence of validated real system data, an additional mass factor of 10% of the propulsion system is added to account for weight of cables and mounting as proposed in [6].

2.5 Integrated mission analysis

Figure 2 presents the incorporation of the models within a mission analysis procedure. The overall process, originally reported by Goulos et al. [44], has been adapted for dual-energy rotorcraft architectures. A three-dimensional treatment of the trajectory is employed with a flight path profile model that enables the conversion between a global geographical system to local Cartesian coordinates [23].

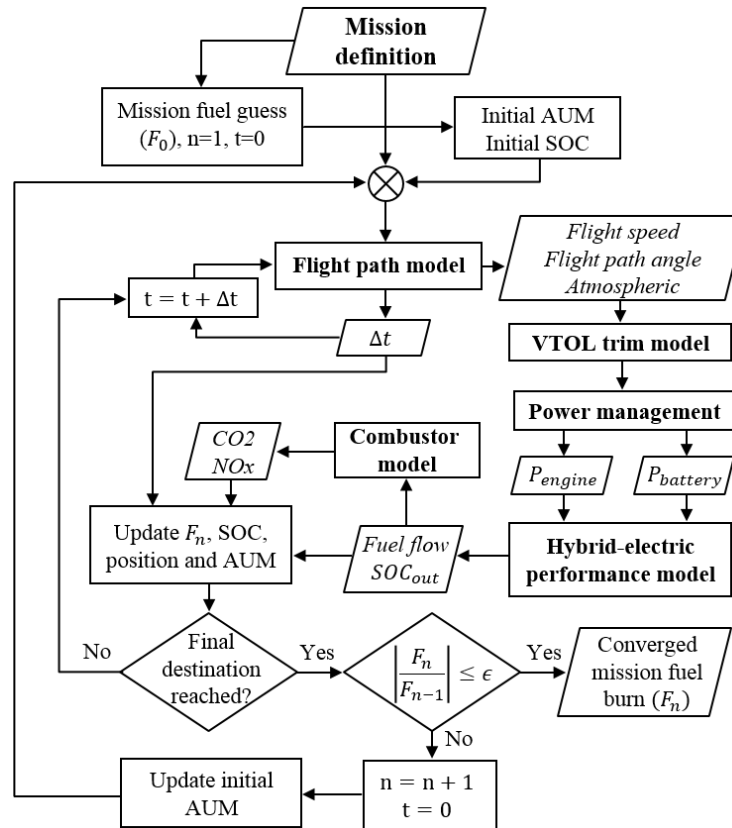


Figure 2 Integrated mission analysis model flowchart reported in [24] adapted for hybrid-electric VTOL configurations

The integrated mission analysis model is based on the adoption of modular blocks corresponding to the different mission task elements; idle, payload, hover, cruise, and loiter. The mission profile is truncated into discrete segments based on the user-defined time step (Δt). The analysis process is initiated with a user-defined mission fuel burn and battery State-of-Charge (SOC) for hybrid architectures, which is used to set the initial All-Up-Mass

(AUM). The rotorcraft and hybrid-powerplant are assumed to operate at steady-state trim conditions and steady-state off-design conditions for the whole mission, respectively.

The flight dynamics code is used to trim the rotorcraft in each segment, providing the power requirement to the hybrid-electric performance. The power management module is used to distribute the demanded power on the two sources based on specified power-split schedules. The engine fuel flow and SOC of the battery are calculated subsequently. The engine fuel flow and SOC of the battery are calculated subsequently. The weight and position of the rotorcraft are updated based on the fuel burned and flight conditions in each segment respectively. This allows to capture the gradual weight reduction during a mission. The battery weight is assumed to remain constant throughout the whole mission and hence, only the fuel flow is contributing to the rotorcraft operating weight reduction. A fixed-point iteration scheme is employed to solve for mission fuel burn, as per Goulos et al. [44]. The integration of the models resulting in the complete platform is illustrated in Figure 3.

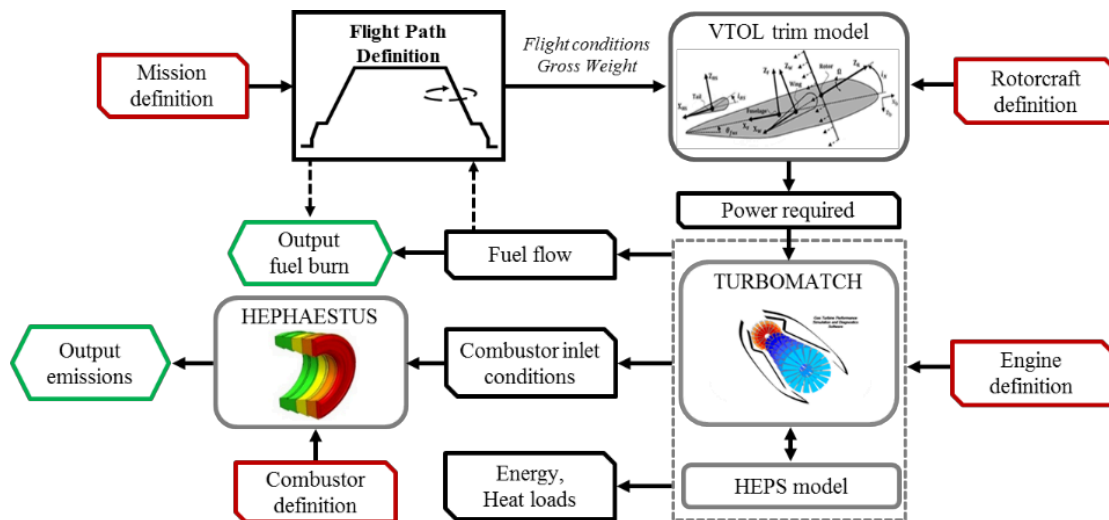


Figure 3 Multi-disciplinary hybrid-electric rotorcraft framework

3. PRELIMINARY HEPS DESIGN AND OPTIMIZATION

The devised framework is used to explore the design space of the investigated conceptual tilt-rotor. The overall design and sizing process consists of two stages; an initial DSE and an optimization approach. The first stage is a systematic procedure targeting to explore the design space and obtain the maximum amount of information on it and its regions of

272 infeasibility. The second stage is to be subsequently deployed to search for potentially
 273 optimum solutions utilizing knowledge from stage one.

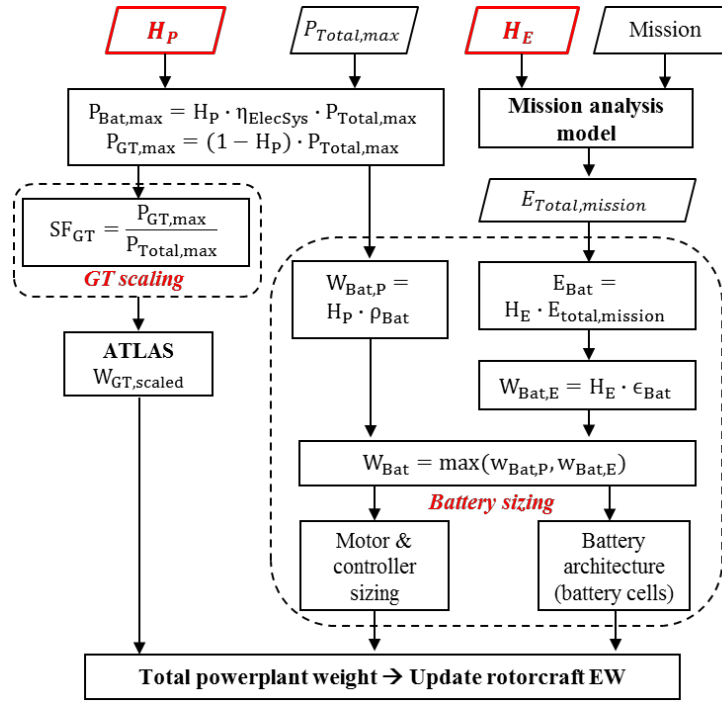


Figure 4 Stage 1 - HE powerplant initial design space exploration

276 Figure 4 displays an illustrative flowchart that describes the sizing approach considered in
 277 the first stage of the present methodology. The initial DSE makes use of two design variables,
 278 which are considered fundamental in the design of HE powerplants. These consist of the
 279 Degree-of-Hybridization (DOH) of Power (H_P) and DOH of Energy (H_E) introduced by
 280 Schmitz and Hornung [47].

$$H_P = \frac{P_{Bat,max} [kW]}{P_{total,max} [kW]} \quad (1)$$

$$H_E = \frac{E_{Bat} [kWh]}{E_{total,mission} [kWh]} \quad (2)$$

281 In order to retrofit any rotorcraft model with a hybrid powerplant, several design and
 282 performance requirements must be fulfilled. The certification and safety constraints
 283 considered are:

- 284 • One Engine Inoperative (OEI) capability
- 285 • Fixed payload-range capacity
- 286 • Constant engine power ratings

• Constant Maximum Take-Off Weight (MTOW)

The DOH of power (H_P) is used to evaluate the maximum available electric power. Introducing a second energy system enables the capability to avoid gas turbine over-sizing. For this reason, the baseline engine can be scaled down based on the maximum available power of the electric system using a scaling factor $SF_{GT} = \frac{P_{GT,max}}{P_{total,max}}$ which depends on the H_P . The scaling is done by changing the design inlet mass flow while keeping the same cycle in terms of specific power [6].

The DOH of energy (H_E) is used to evaluate the maximum energy available in the battery. H_E is expressed as a ratio of battery installed energy (E_{Bat}) to the total required energy during the mission ($E_{total,mission} = \frac{\text{Propulsive Energy}}{\eta_{ElecSys}}$). The propulsive energy is an integral calculated numerically by running the mission analysis framework. The $E_{total,mission}$ is calculated based on the assumption that the electric system has a constant efficiency of $\eta_{ElecSys} = 90\%$. This is a simplified assumption that is essential for this calculation, as the battery efficiency varies throughout the mission based on its discharge characteristics.

The design space is constrained by the SOC of the battery at the beginning and the end of the mission, corresponding to 95% and 15% SOC, respectively [9]. The minimum SOC (SOC_{min}) is selected to avoid severe reduction in battery life. Table 1 presents the propulsion components assumptions.

Table 1 Propulsion component technology level assumptions

Parameter	Unit	Value
Battery specific energy (e_{Bat})	Wh/kg	250
Battery specific power (ρ_{Bat})	W/kg	900
Motor specific power	kW/kg	8
Motor efficiency	-	0.97

A reference mission is selected in order to construct the payload-range diagram for the conventional rotorcraft which is used as a baseline model. This mission consists of a 5-min idle period where all the power is produced by the gas turbine, a take-off and climb segment up to a reference altitude and a subsequent descent and land segment. The mission finishes with 30-minutes fuel reserves at maximum endurance speed and a battery with 30% SOC.

312 The total energy of the battery (E_{Bat}) is therefore calculated. The battery is then sized based
 313 on the two aforementioned variables based on Equation (3).

$$W_{Bat} = \max(W_{Bat,P}, W_{Bat,E}) \quad (3)$$

314 Having calculated the battery weight, the battery pack architecture (number of cells) is
 315 evaluated based on the technology assumptions regarding battery specific power and energy
 316 [22]. The gas turbine is scaled based on the battery selected while ensuring that maximum
 317 intermittent and OEI capabilities are maintained. The weight of the new powerplant is
 318 calculated and the tilt-rotor EW is updated accordingly.

319 Stage 1 is based on the assumption that the ratio of power provided by the gas turbine to
 320 the total power requirement (Eq. (4)) is fixed for the whole mission. It is noted however that,
 321 this might not be the optimum power management approach. For this reason, in stage 2 a
 322 new time-dependent variable is introduced; the power split (PS) which is defined as the
 323 ratio of power supplied by the battery to the total power at any time in the mission (Eq. (4)).
 324 The H_P and H_E are design parameters used to define gas turbine scaling, battery and
 325 powerplant architecture. On the other hand, PS is used during off-design performance to
 326 handle power between the two energy sources. During the mission, the gas turbine works
 327 at maximum continuous power rating and the electric motor is used to assist when the
 328 power demand is higher. When the power request is lower than the gas turbine DP power,
 329 PS is used to optimally split power between two sources.

$$PS(t) = \frac{P_{GT}(t) [kW]}{P_{total}(t) [kW]} \quad (4)$$

330 Next, stage 2 makes use of the information obtained from the DSE in stage 1. The overall
 331 sizing process is illustrated in Figure 5. In this stage, the payload capacity is calculated based
 332 on the updated EW and the maximum fuel capacity (W_{Fuel}) which remains constant and
 333 equal to that of the baseline model. The variations in powerplant weight are accommodated
 334 by adjusting payload weight to never exceed the MTOW. During this stage, a payload-range
 335 diagram is constructed, which is based on the assumption that all energy sources are
 336 consumed with respect to certification and safety margins.

337

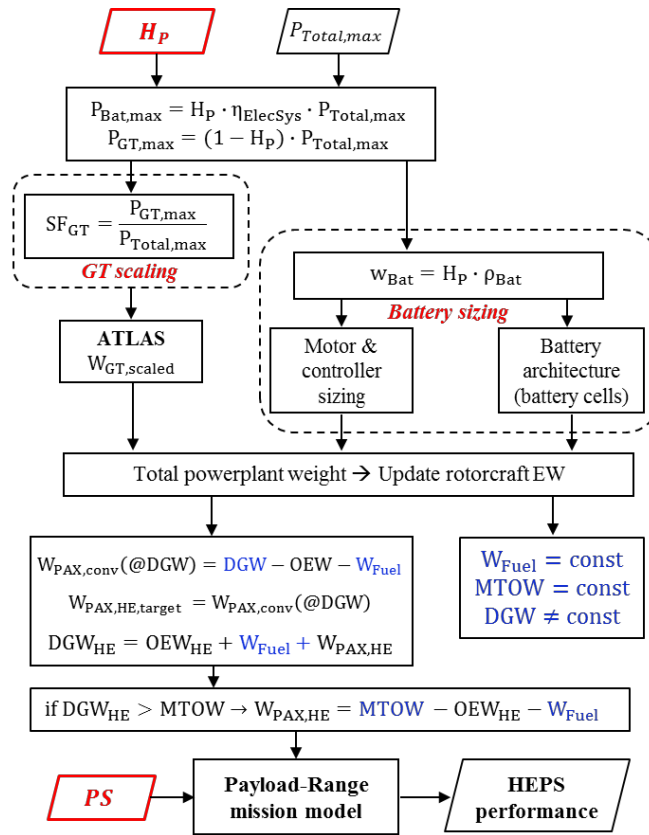


Figure 5 Stage 2 - HE powerplant optimization

At this stage, a global optimization algorithm is used to determine optimum battery and engine size as well as power management strategy. The developed performance framework is coupled with a global optimization algorithm for the derivation of a payload-range chart. The optimization algorithm selected for the present work is the non-dominated sorting genetic algorithm II (NSGA-II) [48]. The employed optimization approach is selected to avoid getting trapped in locally optimum solutions across the design space, as described in [46].

For a conventional rotorcraft, the payload-range capacity is a trade-off between specific power and thermal efficiency. For a hybrid-electric configuration, new DOFs are introduced in the design process and hence, it is essential to establish an appropriate metric to capture this trade-off. For this reason, a holistic metric that encompasses energy efficiency throughout the entire payload-range mission is introduced.

The ratio of energy liberated to revenue work done (ETRW) is a key metric reported by Poll [51]. It depends on the amount of fuel burned to carry a given payload for a given distance ($ETRW = \frac{m_f \cdot LHV}{\text{Payload} \cdot \text{Range}}$). For a hybrid-electric configuration, a 'modified' version of

354 this metric is introduced (METRW) to account for the two energy sources (Eq. (5)). This
 355 metric allows expressing the required energy efficiency of a HE system throughout the entire
 356 mission to maintain payload-range capacity. The optimization of a HE powerplant targeting
 357 enhanced payload-range capacity can be attained by minimizing the METRW.

$$\text{METRW} = \frac{m_f \cdot \text{LHV} + e_{\text{Bat}} \cdot W_{\text{Bat}} \cdot (\text{SOC}_{\text{max}} - \text{SOC}_{\text{min}})}{\text{Payload} \cdot \text{Range}} \quad (5)$$

358 4. RESULTS AND DISCUSSION

359 4.1 Test case: XV-15 tilt-rotor

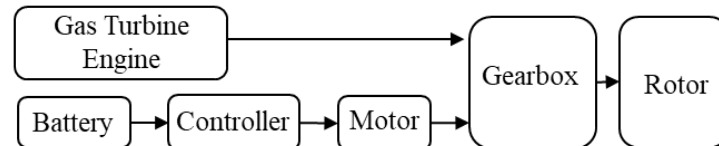
360 The aircraft configuration selected for the purpose of this study is a representative
 361 emerging civil rotorcraft architecture, modelled after the NASA XV-15 tilt-rotor. This
 362 configuration is equipped with two nacelle-mounted turboshaft engines [52] and has been
 363 extensively documented and analyzed in Refs. [53-55]. The baseline aircraft model in this
 364 case study is equipped with two General Electric T700-GE-700 turboshaft engines. The
 365 corresponding engine has been modelled and validated in TURBOMATCH by Ortiz-Carretero
 366 et al. [45]. The maximum contingency power setting is set as the Design Point (DP). Table 2
 367 presents the rotorcraft main design parameters.

368 *Table 2 Tilt-rotor main design parameters.*

Design parameter	Value	Unit
Empty Weight	4116	kg
MTOW	6803	kg
Disk loading	13.2	lb/ft ²
Take-off power	1156	kW
Number of blades	3	-
Blade radius	3.81	m
Root cut-out	0.0875	%
Blade chord	0.3557	m
Blade twist	41	deg
Rotor solidity	0.089	-
Nominal rotor speed	61.7	rad/s
Turboshaft engine	T700-GE-700	-
Number of engines	2	-

369 The conceptual tilt-rotor is designed to carry up to 9 passengers for eight sequential UAM
 370 short flights of 93 km (8x93). A parallel hybrid propulsion system is selected for the hybrid

371 tilt-rotor, as illustrated in , on account of high flexibility and reduced complexity with respect
 372 to potential other hybrid-electric configurations. A DC electrical bus allows to eliminate the
 373 mechanical cross-shafting of the baseline tilt-rotor [7]. The present powerplant comprises
 374 two energy sources; fossil fuel and battery, providing energy to the gas turbine and motor
 375 respectively. The gas turbine and battery-motor system are connected with a gearbox that
 376 acts as a power splitter.



377
 378 *Figure 6 Conceptual hybrid-electric powerplant*

379

380 4.2 Comparison with experimental and flight test data

381 The integrated simulation framework is based on a multidisciplinary approach comprising
 382 models for the different aspects of rotorcraft and hybrid-electric powerplant performance.
 383 In order to gain confidence in the developed framework, the individual models have been
 384 validated using experimental and flight test data.

385

386 4.2.1 Rotor performance analysis

387 Validation of the propeller aerodynamics model has been conducted for both inflow
 388 models. First, the developed method is compared with QPROP, a validated MIT-developed
 389 code, which adopts a BET-vortex theory formulation [58]. This model-to-model comparison
 390 aims to verify the prediction of the radial distributions of induced velocities by the current
 391 model. A limitation of the QPROP model is the capability to use only linear aerodynamic
 392 characteristics including the Prandtl-Glauert compressibility correction for the calculation of
 393 the local coefficients of the airfoils [58]. For this reason, the same characteristics are used in
 394 the developed model for this comparison.

395 Figure 7 presents the radial velocity variations for the NASA SR-2 propeller for an indicative
 396 case. Good agreement is observed with a Root-Mean-Square (RMS) error of 1.74% and 2.96%
 397 relative to QPROP for the axial and tangential induced velocity, respectively. In terms of

propeller performance parameters, such as thrust, power, torque and efficiency, a maximum prediction error of $\pm 3\%$ is noted.

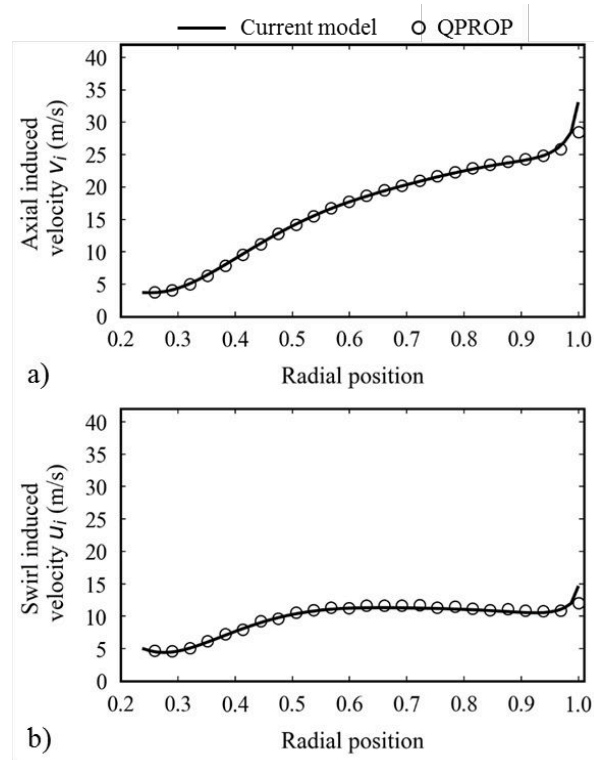


Figure 7 Radial distributions of induced velocities for NASA SR-2 propeller calculated from the simulation model and QPROP program a case $M_\infty = 0.2$, $\theta_0 = 10^\circ$, $\Omega = 7503.2$ rpm at Sea Level ISA a) axial induced velocity, b) swirl induced velocity

The extended BEMT propeller model is then validated with experimental data for the NASA SR-2 propeller [56] extracted from Ref. [57]. It is a well-documented low-speed propeller considered appropriate for this study. NASA SR-2 is an eight-bladed propeller of 0.622 m diameter with a linear twist and variable chord distribution as reported in [57]. NACA 16-series airfoil characteristics are used for the blades of the propeller.

Figure 8 presents the performance predictions based on simulations performed for the NASA SR-2 at Mach 0.20 and 0.34 respectively. Results are presented as functions of pitch angle and advance ratio. Comparison with experimental data extracted from Ref. [57] has been included for validation purposes. Good correlation between the model predictions and the experimental data can be observed for both flight conditions with an overall $C_p = P/(\rho A(\Omega R)^3)$ RMS error of 6.47% and 8.52% at Mach 0.2 and 0.34, respectively. Good agreement is illustrated with the correlation being best for low pitch angles and high advance

ratios. The larger discrepancies observed at low advance ratios and high loading cases are attributed to insufficiently accurate airfoil data. This is due to the fact that at this operating region, a larger portion of the blade is in stall. This is a common limitation of the employed modelling fidelity [29], as the experimental airfoil characteristics are typically extrapolated in the non-linear region [64].

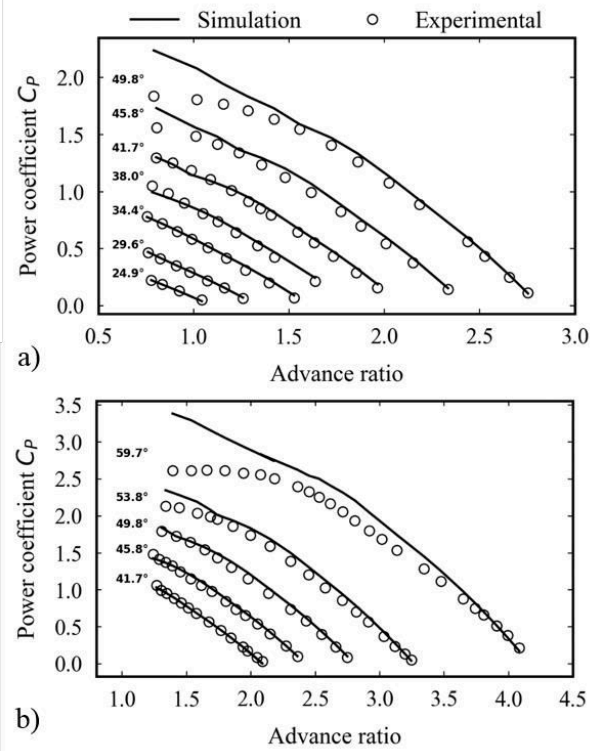


Figure 8 Performance map of the NASA SR-2 propeller – comparison with experimental data extracted from [57] a) Mach 0.2, b) Mach 0.34

The performance of the rotors for non-axisymmetric flight conditions is quantified using the Mangler and Squire inflow model [33]. Figure 9 presents comparisons with existing methods and flight test data for the NASA 2-meter Rotor Test System (2MRTS) [59]. The experimental data are extracted from inflow measurements over the rotor disk for the 2MRTS rotor as reported in [59-61]. The superimposed data are simulated data extracted from [26].

The main discrepancies are observed near the leading and trailing edge, as well as close to the hub (Figure 9). In those regions, the variation of the inflow can only be accurately predicted by vortex theory [26]. Good correlation is observed for the rest of the rotor disk with a RMS error of 9.23% and 5.45% relative to the experimental data for the longitudinal

and lateral inflow, respectively. Moreover, the discrepancies between predictions and experimental data are because in the latter case there is an interaction of the rotor with the ROBIN fuselage [65] which is not included in the current model.

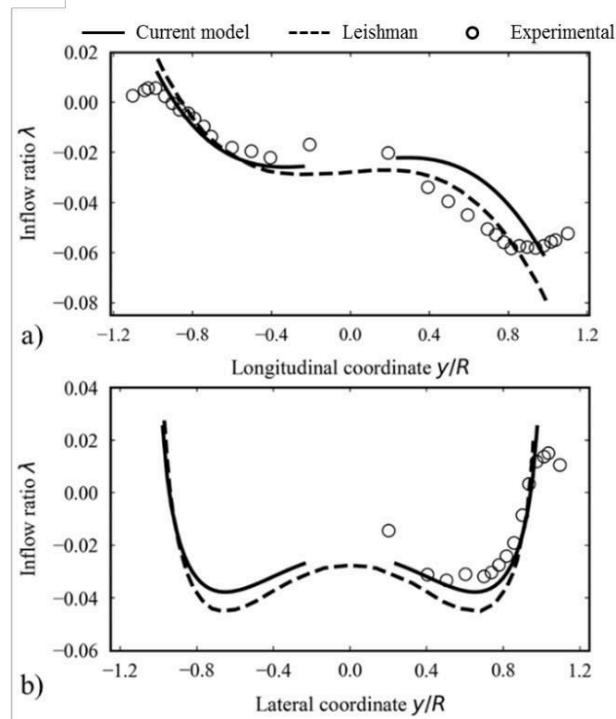


Figure 9 Variation of the a) longitudinal and b) lateral inflow for the 2MRTS rotor at $\mu=0.15$, $C_T = 0.008$, α_s (shaft tilt) = -3° – comparison with simulated data extracted from [26] and experimental data extracted from [59-61]

4.2.2 XV-15 isolated rotor performance

Figure 10 presents predictions based on simulations performed for the XV-15 isolated rotor. The blades use NACA 64 series airfoils (Table 3), as reported in Ref. [62]. Comparisons with experimental data extracted from Refs. [19,55] have been included for validation. Good agreement can be observed between predictions and experimental data with a RMS error of 6.16% relative to the experiments.

Table 3 XV-15 rotor blade airfoil sections

Blade Airfoil section (r/R)	Airfoil
0.0875	NACA 64-X25
0.25	NACA 64-X25
0.5268	NACA 64-X18
0.8093	NACA 64-X12
1	NACA 64-X08

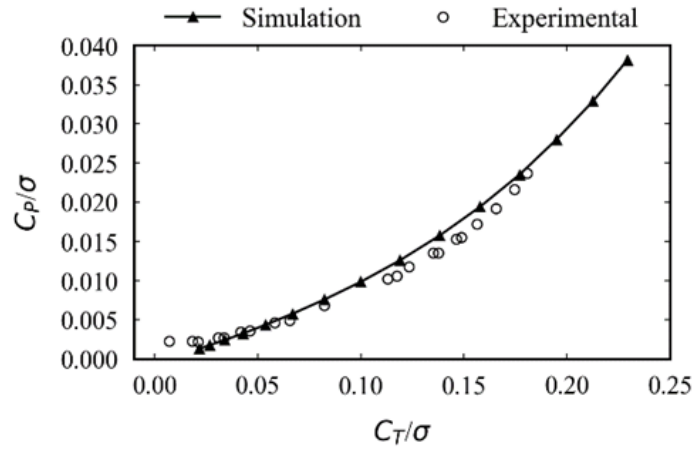


Figure 10 Hover results for XV-15 isolated rotor at $M_{tip}=0.69$ – comparison with experimental data found in [55]

Figure 11 presents a comparison between simulations and measurements of the XV-15 rotor in conversion mode for four different pylon angles. The superimposed data are evaluated using CAMRAD software [63] and are extracted from [55]. The nacelle angles vary from 15 to 75 degrees. For each experimental point, the rotational speed remains constant and the blade pitch is trimmed to match the thrust coefficient. The incorporated model shows good correlation with the experimental and CAMRAD simulation data.

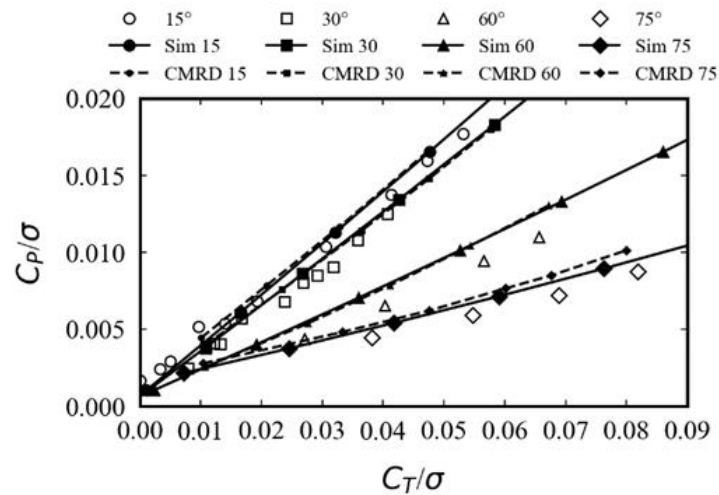


Figure 11 Conversion mode results for XV-15 rotor at $M_{tip}=0.65$, $\mu=0.32$ – comparison with experimental and CAMRAD data [55]

Better predictive accuracy is observed for low tilt angles and low loading cases with a maximum RMS of 5%. Larger discrepancies between predictions and experimental data are noted for $\varphi_{tilt}=60^\circ$ and $\varphi_{tilt}=75^\circ$ with RMS 13% and 7.8% respectively. The potential source

of disagreement in this region can be associated with the Mangler and Squire inflow model. It is noted however that, CAMRAD simulations exhibit the same behavior and hence, the discrepancies are generated from compressible and viscous effects and hence, are associated with the airfoil maps.

4.2.3 XV-15 flight dynamics trim analysis

Figure 12 presents predictions for the fuselage pitch angle and prop-rotor power requirement for the XV-15 tilt-rotor operating in straight and level flight. The results are presented as functions of i_N and δ_F . Comparisons with flight test data reported in Ref. [19, 39] are included for validation purposes. Figure 13 illustrates a schematic representation of the wing and pylon, highlighting the positive directions of these angles.

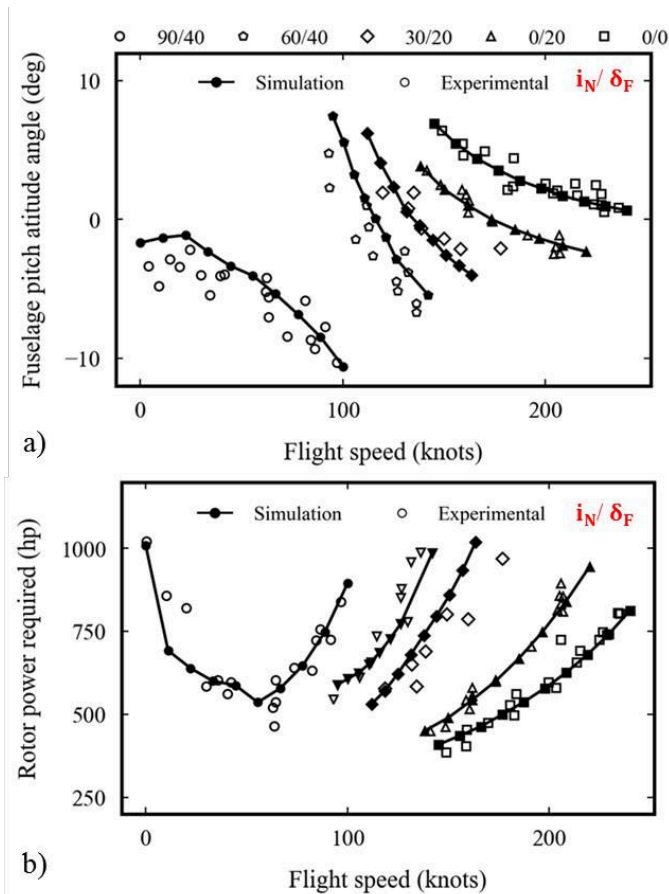


Figure 12 Flight dynamics trim results for the XV-15 tilt-rotor – comparison with flight test data extracted from [19,39] (a) Fuselage pitch angle, (b) Single rotor power required at different nacelle angles/flap angles at SLS conditions

Overall a very good agreement between simulation and flight test data is shown for the prop-rotors at near-vertical orientation (90/40) as well as near-horizontal orientation (0/20, 0/0), with a 5.1% and 3.8% RMS error relative to the flight test data respectively. The larger discrepancies observed for the intermediate nacelle angles (30 and 60°) are associated with component interactions which are significant for these nacelle angles. The employed method uses a simplified interaction approach that may not be able to capture the magnitude of these effects.

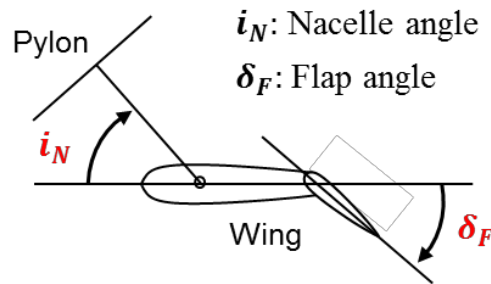


Figure 13 Schematic representation of wing-pylon configuration

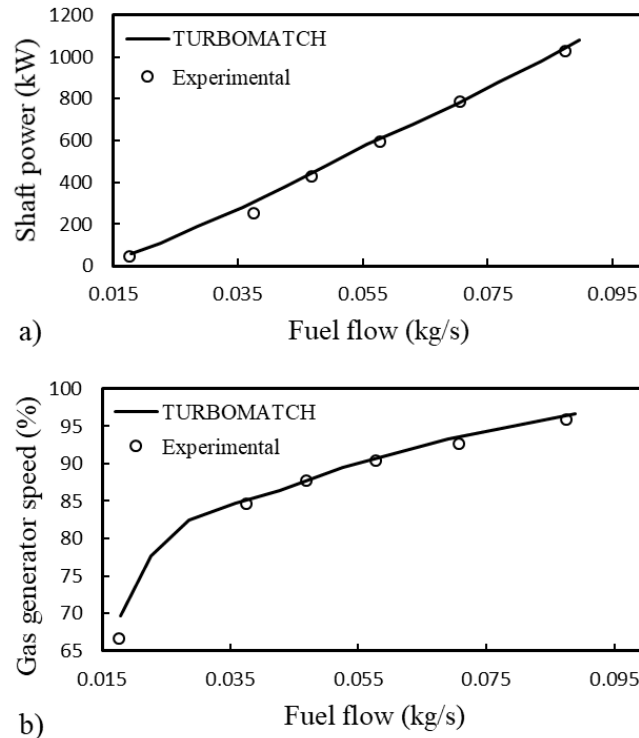


Figure 14 TURBOMATCH engine model validation: (a) gas generator speed versus fuel flow and (b) shaft power versus fuel flow extracted from [65]

4.2.4 Gas turbine performance

Figure 14 presents the respective engine performance validation based on simulations performed using the TURBMATCH T700-GE-700 model as reported in [45]. It is observed that the TURBOMATCH model matches very well with the experimental data [65] for the whole operating envelope.

4.3 Design space exploration

Having developed a simulation model for the preliminary design and analysis of HE novel rotorcraft, an initial DSE is undertaken to investigate the impact of the HE design variables on powerplant architecture.

The impact of the hybridization degrees of H_P and H_E on gas turbine sizing and hybrid-powerplant design is illustrated in Figs. 15-17. Each of the presented contour plots comprises approximately 400 simulation points. Each point on the contour plots represents a unique powerplant, sized based on the methodology described in Section 3. For each simulation point, the non-linear mission analysis framework described in Section 2.5 is deployed to size the powerplant and calculate the take-off AUM (AUM_{TO}). A payload-range mission analysis is then carried out for each point. The sizing mission selected is based on the payload-range capability of the conventional tilt-rotor, representing the baseline architecture (Sec. 4.1).

The hashed regions indicate designs for which the rotorcraft EW exceeds the MTOW ($EW > MTOW$). It is highlighted that the highly-constrained design space is an outcome of the limitations associated with primarily battery energy density. Additionally, the design space is further constrained with a red dashed line, which represents designs that the AUM_{TO} exceeds the MTOW ($AUM_{TO} > MTOW$) for zero payload (0 PAX). With an increasing number of passengers, the feasibility region progressively shrinks. As a result, the design space is further constrained with a black dashed line representing the designs that the $AUM_{TO} > MTOW$ for a fixed payload of nine passengers (9 PAX).

The present parametric analysis is based on the assumption that the available battery energy is consumed based on a constant DOH of power (H_P) strategy throughout the whole mission. It is noted however that, this is a sub-optimum strategy to manage the power

available from the two sources. This is explained by the fact that a constant power ratio between the two sources may force the engine to operate at part load and hence, high SFC. For the baseline tilt-rotor, the cruise segment represents approximately 65% of the block fuel consumed in a typical mission profile with 740 km range. Thus, it is expected that the main fuel burn reduction due to the introduction of a hybrid powerplant will occur in cruise segment.

Figures 15-16 illustrate the impact of the H_E and H_P on the total battery installed mass (W_{Bat}) and rotorcraft EW, respectively. As expected, increasing the DOH results in higher W_{Bat} and concurrently rotorcraft EW. It is observed that for a hybridization of energy above 0.3, the rotorcraft EW exceeds the MTOW ($EW > MTOW$). This can be explained by the fact that current state-of-the-art batteries are limited in terms of energy density. Similarly, the feasible region is limited to a maximum H_P of 0.9 which is a direct result of the battery capacity and electric motor specific power. The design space boundaries become even stricter when payload increases, resulting in a feasible region corresponding to maximum values of $H_E=0.13$ and $H_P=0.24$. It is therefore realized that developments in battery technology are considered key-enablers in the design of hybrid rotorcraft vehicles.

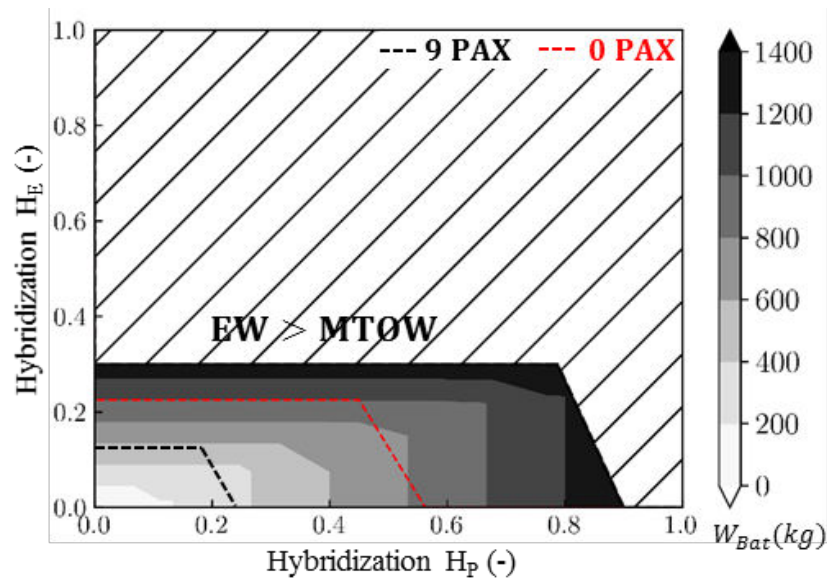


Figure 15 Impact of H_E and H_P on battery weight (W_{Bat})

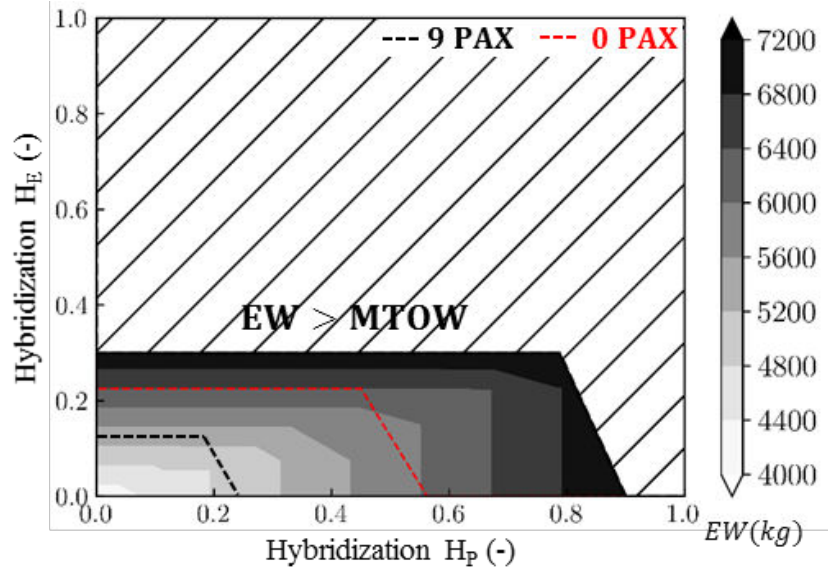


Figure 16 Impact of H_E and H_P on rotorcraft EW

Figure 17 shows the impact of the hybridization factors on gas turbine scaling (SF_{GT}). The scaling is done assuming the same cycle in terms of specific power and scaling the design point inlet mass flow. It can be observed that in order to cope with the specifications of the tilt-rotor for maximum available power, the gas turbine has to be scaled from 100% to a minimum value of 60%. As the gas turbine is scaled down and hence SF_{GT} decreases, W_{Bat} increases to ensure the maximum required power ratings. As a result, the engine 'average' operating point for the sizing mission moves towards higher relative power and hence, lower SFC.

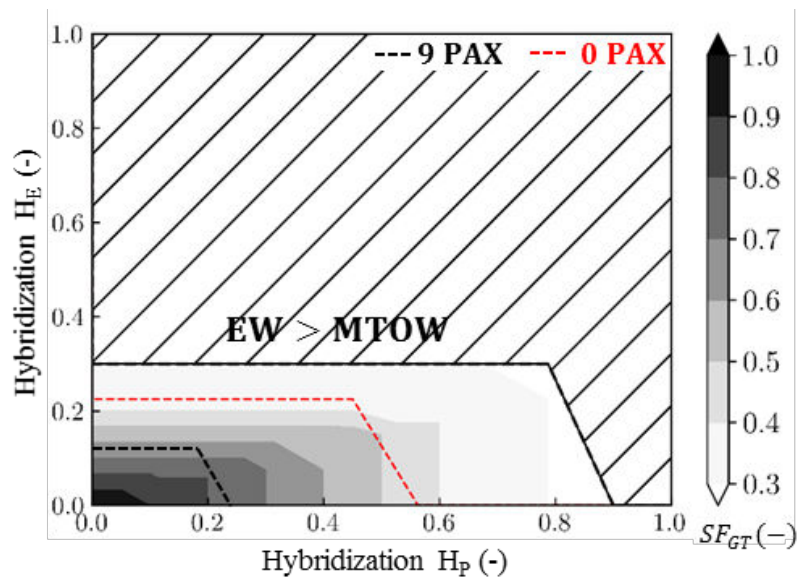
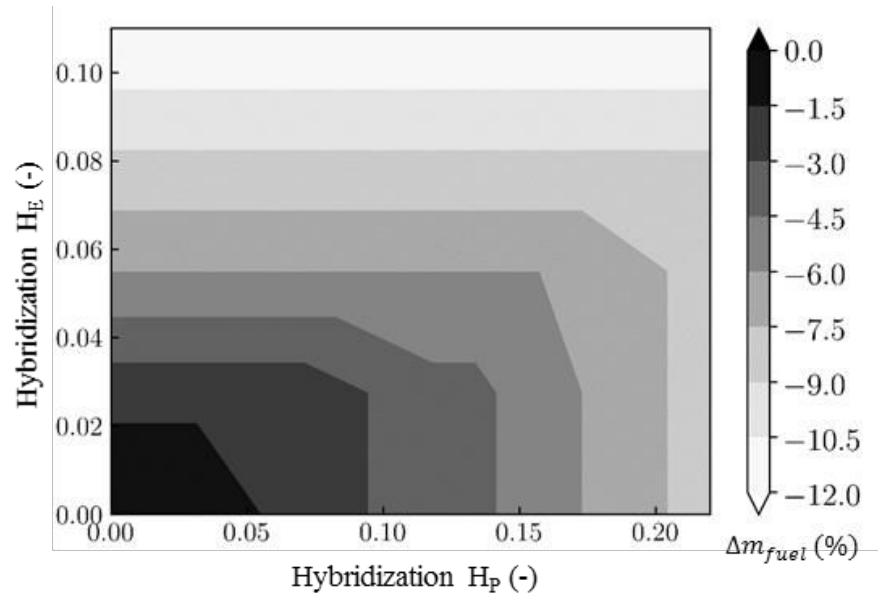


Figure 17 Impact of H_E and H_P on engine scaling index (SF_{GT})

553 The impact of the design variables (H_E, H_P) on battery installed mass (W_{Bat}) and
 554 rotorcraft EW is similar. As the size of the gas turbine decreases (Fig. 17) to a maximum
 555 scaling value of $SF_{GT} = 0.38$, its weight ($W_{GT,scaled}$) reduces accordingly. At the same time,
 556 the W_{Bat} increases (Fig.15) and the rotorcraft EW increases (Fig.16). It is therefore realized
 557 that the weight of the powerplant and consequently the rotorcraft EW is dominated by the
 558 installed battery and not the gas turbine weight.

559 The H_E of a mission strongly depends on the mission itself. For the technology level
 560 assumed in the present work, the contribution of energy from the battery to the total energy
 561 consumed is relatively low for a 740-km range mission ($H_E=0.1-0.24$). The H_E can be
 562 potentially increased if a battery recharging system is added. However, the trade-off is not
 563 straight forward as an on-board charging system would increase the weight and complexity
 564 of the system. For this reason, in the present analysis, the batteries are only charged when
 565 the aircraft is getting refueled.

566 Figure 18 illustrates the impact of the hybrid design variables on the mission fuel burn
 567 relative to the conventional tilt-rotor (Δm_{fuel}) for the design mission of 650-km and 9
 568 passengers. It is highlighted that only the feasible designs are plotted here. As H_P and H_E
 569 increase, the reduction in fuel burn also increases, reaching a maximum value of -12%
 570 relative to the conventional (Figure 18).



571 Figure 18 Impact of H_E and H_P on mission fuel burn relative to the nominal rotorcraft based on the reference mission and 9
 572 passenger capacity
 573

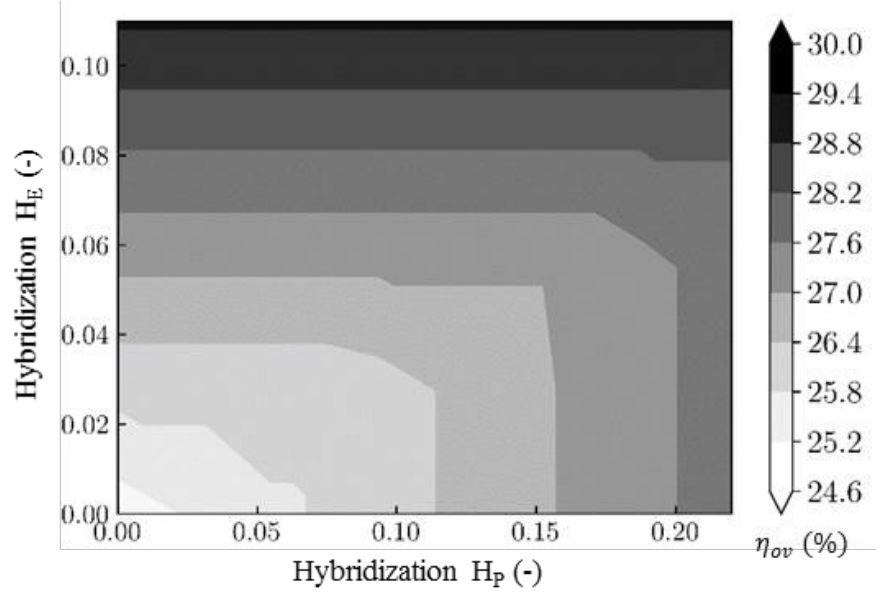


Figure 19 Impact of H_E and H_P on overall system efficiency relative to the nominal rotorcraft based on the reference mission and 9 passenger capacity

Figure 19 presents the effect of the hybrid design variables on the overall system efficiency (η_{ov}) averaged for the whole mission (Eq. (6)). This is expressed as an average ratio of propulsive energy to the energy consumed for the mission. The propulsive energy is an integral calculated numerically based on the overall mission. Similarly, the heat input and electric energy are calculated based on the fuel and battery energy consumed throughout a mission. From Fig. 18, the η_{ov} of the conventional tilt-rotor based on the payload-range mission is approximately 25%. The introduction of a 400kg battery leads to an increase of overall system efficiency to 30% based on the same mission. Roumeliotis et. al introduced a corrected overall efficiency ($\eta_{corr,ov}$) by utilizing a battery grid efficiency of 50% to account for the electric energy production and transfer to charging stations as reported in [6] (Eq. (7)).

$$\eta_{ov} = \frac{\text{Propulsive energy (kWh)}}{\text{Heat input(kWh)} + \text{Electric energy(kWh)}} \quad (6)$$

$$\eta_{corr,ov} = \frac{\text{Propulsive energy (kWh)}}{\text{Heat input(kWh)} + \frac{\text{Electric energy(kWh)}}{\text{Grid efficiency (-)}}} \quad (7)$$

With the introduction of this correction, the overall system efficiency gain due to hybridization is reduced. Specifically, the $\eta_{corr,ov}$ of each design decreases by approximately

1-1.5%. Thus, the maximum $\eta_{corr,ov}$ that can be attained is 28.7%. It is noted that the introduction of a 50% battery grid efficiency has a small effect on η_{ov} . The variation of η_{ov} in the design space depends on the installed battery and the gas turbine scaling that allows the engine to work closer to its maximum power setting and hence, lower SFC. It is therefore realized that the dominant effect in the improvement of η_{ov} is the fact that the battery allows avoiding gas turbine oversizing. Thus, the gas turbine is scaled down and operates at more favorable conditions throughout the mission.

Figure 20 illustrates the impact of DOH on engine scaling index (SF_{GT}), battery installed energy (E_{Bat}), system overall efficiency (η_{ov}), take-off weight (AUM_{TO}) and mission fuel burn (m_{fuel}). The results are obtained based on the methodology described in Section 3. For each unique powerplant on this diagram, the non-linear mission analysis framework is used to obtain the AUM_{TO} , m_{fuel} and η_{ov} based on the payload-range mission of the baseline conventional tilt-rotor. It is observed that the maximum E_{Bat} is limited to 110kWh, a limitation that is directly related to the energy density of the battery. As the E_{Bat} increases, the gas turbine reduces and the take-off weight increases. The benefit associated with increased hybridization is the improvement in overall efficiency ‘averaged’ over the whole mission. For a powerplant architecture (68%,112kWh), the AUM_{TO} exceeds MTOW. The weight distribution of three indicative designs (from Fig. 20) are illustrated in Figure 21.

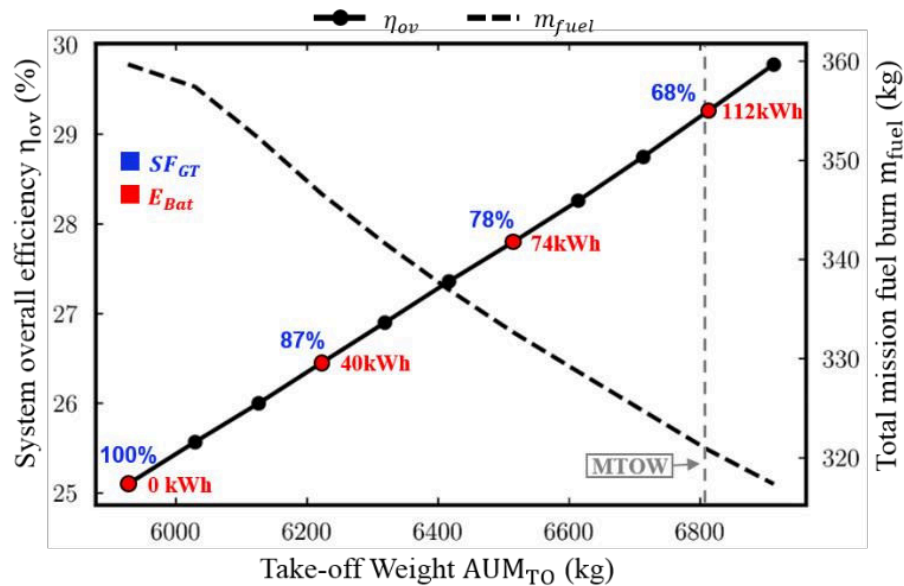


Figure 20 Impact of the HEPS powerplant on system overall efficiency, mission fuel burn and take-off weight based on the reference sizing mission

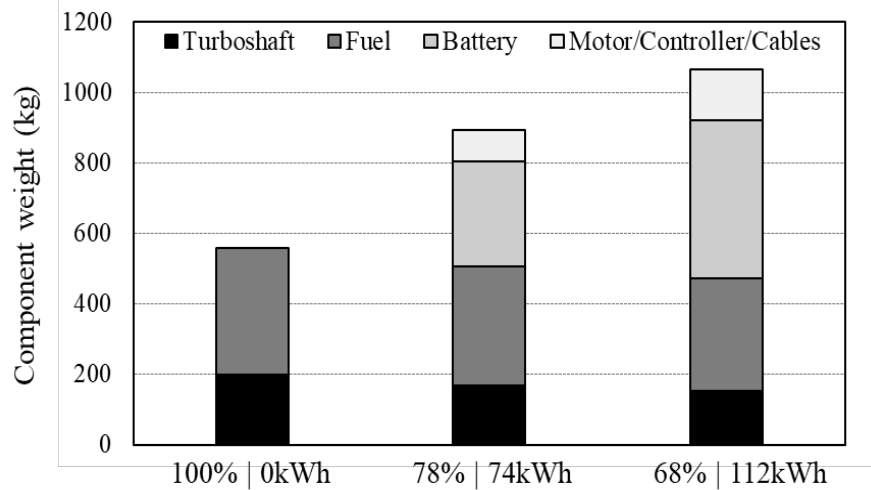


Figure 21 Weight distribution chart for different powerplant configurations

4.4 Effect of HE design on payload-range capacity

In the second stage, the power split (PS) ratio (Eq. (4)) – a time-dependent variable, is introduced, targeting an optimal approach for power management during the mission. Additionally, the maximum payload that the aircraft can carry is integrated in the sizing process.

Figure 22 illustrates the impact of PS and DOH (battery mass) on the maximum range that can be attained. For each point in the contour plot, the non-linear mission analysis framework described in Section 2.5 is deployed to size the powerplant, calculate the take-off AUM (AUM_{TO}) and the maximum allowed payload capacity. A payload-range mission is then carried out, assuming a fixed fuel capacity (W_{Fuel}) to obtain the maximum range, as described in Section 3.

It is observed that the maximum mission range is increased with the introduction of a HE architecture that acts as a range-extender. The increase in maximum range is achieved with a trade-off in the payload capacity. Specifically, for batteries larger than 380kg, the rotorcraft loses the capability to carry 9 passengers. When introducing larger batteries, the maximum available payload calculated at DGW is reduced to 7 and 5 passengers for a battery above 440kg and 490kg respectively.

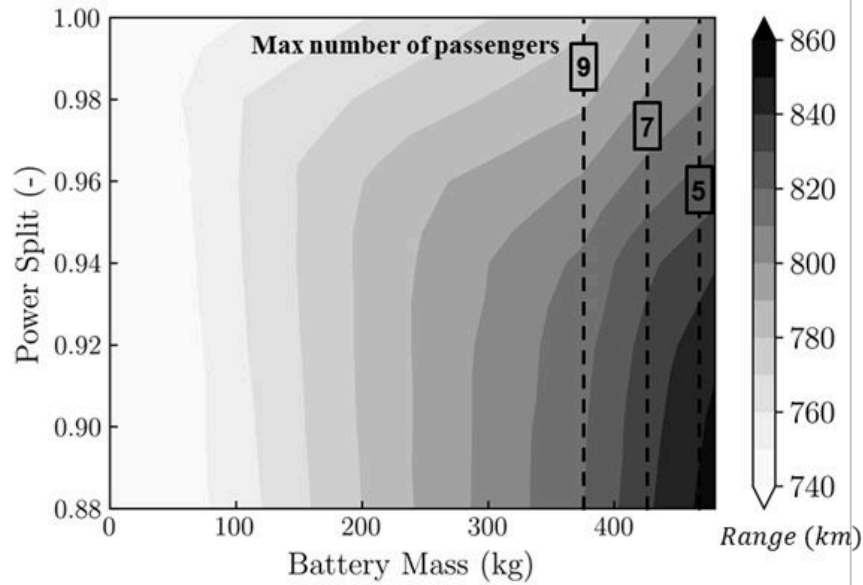


Figure 22 Impact of power split and battery weight on mission payload and maximum range for the reference sizing mission

Figure 23 illustrates the maximum payload and overall efficiency as a function of maximum range for the conventional (GT) and hybrid configuration (HE), respectively. Two technology levels are considered for this comparison, batteries with $\epsilon_{Bat}=250\text{Wh/kg}$ and $\epsilon_{Bat}=500\text{Wh/kg}$. In order to calculate the maximum available payload, it is assumed that the rotorcraft starts the mission with $AUM_{TO} = MTOW$ and the payload is then evaluated based on the weight of the HEPS. For the conventional tilt-rotor, the AUM_{TO} remains constant for all the designs considered by increasing the available fuel with respect to the reduced payload. Each hybrid-electric design on this figure is sized based on the methodology presented in Section 3 and the power split (PS) is optimized for minimum METRW (Eq. (5)). As presented in Figure 23, the conventional rotorcraft can carry a maximum useful load of 1880 kg for a flight of 680 km with an average overall efficiency of 25.5%. As payload progressively decreases, the fuel capacity increases and hence, the maximum range of the conventional rotorcraft increases. For a fixed useful load of 940 kg and increased fuel by 400 kg per engine, the maximum mission range that can be attained is 1498km for the conventional tilt-rotor.

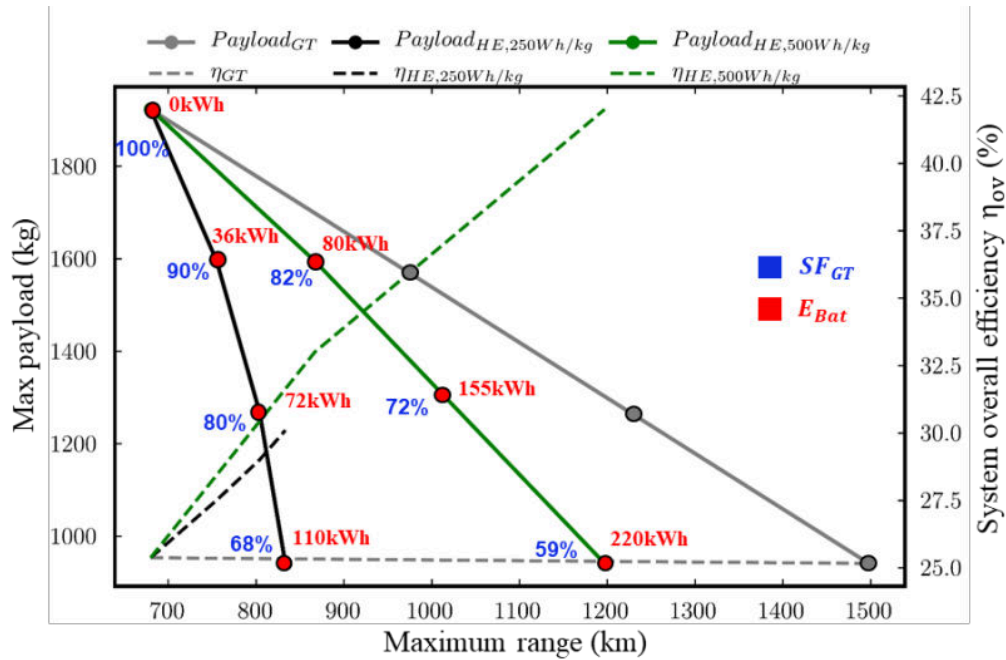


Figure 23 Impact of the HEPS powerplant on maximum payload, range and system overall efficiency based on the reference sizing mission

Similarly, for the HE configuration, as the DOH increases, the maximum useful load gradually reduces with the introduction of larger batteries. At the same time, the maximum range that can be attained increases. For a battery of 250Wh/kg, at a fixed 940 kg useful load, the maximum range that can be attained is 833 km. For a battery of 500Wh/kg, a significant improvement is observed compared to current battery technology levels. At a fixed payload of 940kg, the maximum range that can be attained is 1198 km. This improvement is mainly associated with the engine downsizing to a 59% of the baseline engine due to the introduction of the electric system. This allows the gas turbine to operate at more favorable conditions, especially during cruise.

It is observed that based on the conventional tilt-rotor, when the useful payload is replaced with fuel, the maximum range that can be attained is superior to the hybrid designs (Fig. 23). Specifically, for a fixed payload of 940 kg, the conventional tilt-rotor operates for 25% longer relative to the HE design with $\epsilon_{Bat}=500Wh/kg$. For this payload, the HE configurations operate with 75 kg fuel/passenger. On the contrary, the conventional tilt-rotor operates with 155 kg fuel/passenger and hence, its gaseous emissions will be approximately double compared to the hybrid designs.

671 This chart can be used as a guide in preliminary design and sizing of optimally designed
 672 HEPS for novel tilt-rotor aircraft in the early design phase. For example, in order to design a
 673 HE tilt-rotor with a useful load of 1230 kg and a range of 780 km, it is essential to install
 674 72kWh batteries and scale the gas turbine to 80% if current level of technology is considered.
 675 The present trade-off strongly depends on battery energy density. For this reason, the
 676 battery system is expressed using installed battery energy (E_{Bat}) and not battery mass
 677 (W_{Bat}). It has been demonstrated that, the installation of batteries with higher energy
 678 density (500Wh/kg) allows to further improve payload capacity. This chart can be extracted
 679 using the present methodology for different configurations and sizing missions.

680 5. CONCLUSION

681 This work presented a comprehensive preliminary design exploration and assessment of
 682 a parallel HEPS for a civil rotorcraft designed for UAM applications. The design space
 683 exploration conducted using two fundamental hybridization parameters; the DOH of power
 684 and energy is used to quantify corresponding trade-offs between fuel economy, energy
 685 efficiency, engine weight and rotorcraft weight, as well as, payload-range capacity. A
 686 multidisciplinary approach for rotorcraft performance, HEPS performance, mission analysis
 687 and power scheduling has been developed and validated. A conventional tilt-rotor
 688 configuration and its HE variant equipped with a novel parallel HEPS were developed to
 689 enable this assessment.

690 It has been demonstrated that improvements up to 12% in mission fuel burn can be
 691 attained for a fixed sizing mission under the simulated design space, constraints and
 692 operational requirements. At the same time, an increase of 20% in overall system efficiency
 693 relative to the conventional rotorcraft can be achieved. This improvement is more obvious
 694 for low mission ranges. The increase in energy efficiency is mainly associated with the gas
 695 turbine scaling and hence, operating at points of low SFC. The scaling is done while
 696 maintaining the required payload at DGW, the corresponding initial AUM and airworthiness
 697 requirements (OEI) of the baseline rotorcraft.

698 It has been shown that the introduction of a HEPS modifies the payload-range capacity of
 699 the conventional rotorcraft with battery energy density being the dominant parameter

affecting this trend. The maximum range can increase to 4.65% and 7.92% relative to the conventional rotorcraft for batteries of 68kWh and 95kWh, respectively with current battery energy densities. Battery energy densities of 500Wh/kg result in improvements in maximum range up to 25% and 50% relative to the conventional rotorcraft for batteries of 80kWh and 155kWh, respectively. It has been demonstrated that, the replacement of the batteries with kerosene fuel can further increase the maximum range of the aircraft. However, the fuel consumed and hence, the carbon footprint of the vehicle is almost double compared to the HE variants. At the same time, the HE variants provide improved flexibility and reduced maintenance cost.

This work can be used as a guide in preliminary design and sizing of optimally designed HEPSs for novel rotorcraft configurations. It can assist vehicle and propulsion designers at the early design phase followed by detailed design and integration studies. The methodology is flexible and hence, this analysis can be reproduced for any rotorcraft configuration and HEPS. The results presented in this paper provide new insight in powerplant design for UAM novel tilt-rotor aircraft, highlighting trade-offs between hybridization, gas turbine size, payload-range capacity and weight.

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NOMENCLATURE

ACRONYMS

AUM	All-Up-Mass
BEMT	Blade Element Momentum Theory
CO ₂	Carbon dioxide
DOH	Degree-of-Hybridization
DSE	Design Space Exploration
ETRW	Energy to Revenue Work Ratio
EW	Empty Weight
HEPS	Hybrid-Electric Propulsion System
METRW	Modified ETRW
MTOW	Maximum Take-Off Weight
SFC	Specific Fuel Consumption

SOC	State-of-charge
VTOL	Vertical Take-Off and Landing
UAM	Urban Air Mobility

Roman symbols

i_N	Nacelle angle	deg
δ_F	Flap angle	deg
H_P	Degree-of-Hybridization of Power	-
H_E	Degree-of-Hybridization of Energy	-
ρ_{Bat}	Battery specific power	kW/kg
ϵ_{Bat}	Battery specific energy	kWh/kg
$P_{total,max}$	Hybrid maximum available power	kW
$P_{GT,max}$	Gas turbine maximum contingency power	kW
$P_{Bat,max}$	Battery maximum power	kW
SF_{GT}	Gas turbine scaling factor	%
P_{total}	Total power requirement	kW
$P_{ElecSys}$	Power supplied by the electric system	kW
$E_{total,mission}$	Total mission electric energy demand	kW
E_{Bat}	Battery installed energy	kWh
$\eta_{ElecSys}$	Total efficiency of the electric system	%
Ω	Rotor speed	rad/s
σ	Rotor solidity	-
V_∞	Free stream velocity	m/s
C_T	Rotor thrust coefficient	-
C_P	Rotor power coefficient	-
Δm_{fuel}	Fuel burn delta, $100 (m_{fuelHE} - m_{fuelconv})/m_{fuelconv}$	%
η_{ov}	Overall system efficiency	%
$\eta_{corr,ov}$	Overall corrected system efficiency	%

Subscripts

$()_{GT/Bat}$	Referring to the gas turbine or battery, respectively
$()_{ElecSys}$	Referring to the electric system (battery, motor, controllers)
$()_{conv/HE}$	Referring to the conventional tilt-rotor or hybrid-electric tilt-rotor, respectively

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Figure Captions List

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724 **Table Caption List**

Table 1	Propulsion component technology level assumptions
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Table 3	XV-15 rotor blade airfoil sections

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Preliminary design of hybrid-electric propulsion systems for emerging urban air Mobility rotorcraft architectures

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