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OPTIMIZED POWERPLANT CONFIGURATIONS FOR IMPROVED ROTORCRAFT OPERATIONAL PERFORMANCE

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ABSTRACT

This paper presents an integrated multidisciplinary rotorcraft design and optimization framework, deployed for the design and assessment of a conceptual rotorcraft powerplant configuration at mission level. The proposed approach comprises a wide-range of individual modeling theories applicable to rotorcraft flight dynamics, gas turbine engine performance and weight estimation as well as a novel physics-based, stirred reactor model for the rapid estimation of gas turbine gaseous emissions. A novel Single-Objective and Multi-Objective Particle Swarm Optimizer is coupled with the aforementioned integrated rotorcraft multidisciplinary design framework. The combined approach is applied to the multidisciplinary design and optimization of a reference Twin Engine Light civil rotorcraft modeled after the Eurocopter Bo105 helicopter, operating on representative mission scenario. Through the application of Single-Objective optimization, optimum engine design configurations are acquired in terms of mission fuel consumption, engine weight and gaseous emissions at constant technology level. Multi-Objective studies are carried out in order to quantify the optimum interrelationship between mission fuel consumption and gaseous emissions for the representative Twin Engine Light rotorcraft operation and a variety of engine configurations. The proposed approach essentially constitutes an enabler in terms of focusing the multidisciplinary design of rotorcraft powerplants to realistic, three-dimensional operations and towards the realization of associated engine design tradeoffs at mission level.

NOTATION

Roman symbols

CO_2	Carbon dioxide
T_{out}	Heat exchanger outlet temperature, k
T_{Comp}	Compressor delivery temperature, k
$T_{Exhaust}$	Exhaust air temperature, k

Greek Symbols

Δ_{Weight}	Delta weight, kg

Acronyms

ACARE	Advisory Council for Aeronautics Research in Europe
AGL	Above Ground Level
AUM	All Up Mass, kg
CBA	Component based approach
DOE	Design Of Experiments
DP	Design Point
EHOC	European Helicopter Operators' Committee
EI	Emissions Index
EMS	Emergency Medical Mission
FPT	Free Power Turbine
HEE	Heat Exchanger Effectiveness
HECTOR	HeliCopTer Omni-disciplinary Research-platform
HPC	High Pressure Compressor
LEM	Law Enforcement Mission
LHS	Latin Hypercube Sampling
LPC	Low Pressure Compressor
MTOW	Maximum Takeoff Weight, kg
mPSO	Multi-objective Particle Swarm Optimizer
OPR	Overall Pressure Ratio
OEW	Operational Empty Weight, kg
PATM	Passenger Air Taxi Mission
PR	Pressure Ratio
RBF	Radial Basis Functions
RSMs	Response Surface Models
SFC	Specific Fuel Consumption, μg/J
SAR	Search And Rescue
sPSO	Single-Objective Particle Swarm Optimizer
TEL	Twin Engine Light
TEM	Twin Engine Medium
WEBA	Whole Engine Based Approach
WSG84	World Geodetic System dated in 1984

INTRODUCTION

Background: The aero-industry has had many significant challenges since the beginning of the 21st century. As elaborated by Colin F. McDonald in (Ref. 1), the most salient ones being the reduction in emissions, improvement in Specific Fuel Consumption (SFC), reduction in noise levels and achievement of efficient and most economical life cycle costs. Over the second half of the 20th century of manned flight, rotorcrafts have been in service and have established themselves as key players in a variety of roles. The helicopter operations resulting from civil and military operations, although comprising a significantly smaller portion of the aircraft market in comparison with the fixed-winged aircraft, are experiencing the same concerns with respect to the amount of gaseous emissions produced.

The rotorcraft plays a specific and inimitable role in air transportation and it is often used for purposes where the environmental concerns are secondary, e.g. medical rescue operations, law enforcement, search & rescue, fire suppression, surveillance, military combat and transport purposes. However, the rotorcraft traffic related to passenger transport/air taxi requirements that up to now has been marginal, is expected to grow rapidly. This is mainly driven by the exponential growth in passenger air travel demand that is foreseen for the 2015 – 2020 period (2 to 3 fold increase) (Ref. 2).

Rotorcraft activities presently amount to roughly 1,500,000 flight hours per year only with respect to European airspace. These represent an annual consumption of the equivalent of 400,000 tons of aviation fuel. Maintaining current rotorcraft technologies is expected to quadruplicate this figure within the next 20 years, this being a direct result of the anticipated traffic augmentation (Ref. 2). The Advisory Council for Aeronautics Research in Europe (ACARE), in an attempt to manage the environmental impact of civil aviation, has set a number of goals to be achieved by the year 2020 (Ref. 3). These goals include, among others, reduction of produced carbon dioxide (CO_2) and nitrogen oxides (NO_x) emissions by the order of 50% and 80%, respectively.

Clarke (Ref. 4) described three potential paths towards limiting the environmental impact of civil aviation: (a) significant reduction in the number of operations, (b) changing the type of deployed aircraft, and (c) deployment of alternative operational rules and procedures. Option (a) is not a feasible direction due to the aforementioned forecasted expansion in air traffic (Ref. 2). With regards to option (b), the associated time scale to commercialise new configurations from the conceptual stage along with all the required airworthiness certifications can reach up to 50 years of time frame, as elaborated in (Ref. 5). Thus, in order to address the targets set by ACARE for the year 2020, emphasis needs currently to be placed towards the design of optimum operational procedures. It is noted however that, although presently the investigation of conceptual designs may not effectively address the relatively short-termed ACARE goals, it still is a viable path towards a longer term solution. Therefore, in order to effectively manage the long-term environmental impact of civil aviation while simultaneously accounting for the expected traffic growth, options concerning both (b) advanced conceptual design configurations as well as (c) incorporating optimum operational procedures, need to be thoroughly explored.

Aircraft trajectory optimization: To address option (c) several initiatives are underway specifically in Europe under the Seventh Framework Programme of the European Community. Aircraft flight trajectory optimization studies corresponding to both fixed and rotary wing aircrafts are being explored, aiming towards lower overall mission fuel burn, emissions and noise levels. Goulos et al provides a brief evaluation of the related literature in their study (Ref. 6). Their work was focused on the simulation and multidisciplinary optimization of complete, three-dimensional rotorcraft operations for fuel burn, chemical emissions, and ground noise impact. Their investigated case studies suggested a potential reduction in total mission fuel consumption of the order of 20% and 7% for a police and a passenger transport operation, respectively, relative to their corresponding suboptimal baselines.

Aircraft-engine design parameters optimization: With regards to option (b) as elaborated by Goulos et al in (Ref. 7), the overall approach can effectively be subcategorized within two major sectors of aerospace related research; airframe–rotor design, and engine cycle optimization. With respect to the latter approach related to rotorcraft applications, Goulos et al (Ref. 7), proposed a methodology with the potential to reduce fuel consumption associated with the civil rotorcraft operations at mission level, through optimization of the engine design point cycle parameters. The design space variables essentially comprised the engine combustor outlet temperature, compressor pressure ratio and total engine mass flow. The proposed methodology was enabled through a comprehensive and computationally efficient optimization strategy, utilizing a novel particle-swarm method and was deployed to investigate two different classes of helicopters, a Twin Engine Light (TEL) and a Twin Engine Medium (TEM) helicopter. Their results, through a multi-objective optimization achieved an increase in maximum take-off power as well as a reduction in fuel consumption of the order of 28% and 10% respectively, for a TEL-EMS mission, relative to the baseline case and an increase in DP shaft power and a reduction in mission fuel burn of the order of 11% and 8% respectively, for a TEM-SAR mission, relative to the baseline.

Advanced alternative engine conceptual design and analysis: Another available approach that can effectively lead to the enhancement of current helicopter engine technology is by adopting advanced cycle engines, that are much more efficient than the conventional Brayton cycle enignes. Considering unprecedented improvements in engine fuel efficiency, the most promising candidate is the advanced regenerative turboshaft concept. Rosen elaborated in (Ref. 8), "the UAVs or helicopters that are intended for extremely long duration missions may require powerplants that are much more efficient than Brayton cycle gas turbine engines". Also Saravanamoutto in (Ref. 9), when discussing regenerative technology, suggests "it is not impossible that regenerative units will appear in the future, perhaps in the form of turboshaft engines for long endurance helicopters". Fakhre et al in (Ref. 10), conducted a parametric study to establish the feasibility of regenerative technology for two classes of helicopters under various operations. Their study suggested that an on board heat exchanger offered substantial reduction in total mission fuel burn. However when considering the added weight of the heat exchanger, the regenerative technology was only found to be promising for long range operations e.g. Oil & Gas, Search And Rescue (SAR) or long range Passenger/Air Taxi (PAT) missions etc.

The deployment of regenerative technology has also been of great interest to enhance the operational capabilities of helicopters for military operations. This dates back to the early 1960's. Various programs have been conducted by both government and private industry to demonstrate the performance and operational capabilities of regenerative technology against conventional technology. The "T63 Regenerative Engine Program" (Ref. 11), conducted by the US Army in 1965 is a remarkable achievement in showcasing regenerative technology potential. The program successfully completed a 50hr flight test in a Light Observation YOH-6A helicopter employing a "Bolted-on-Type" regenerative engine, which resulted in increasing the helicopter's maximum specific range by 25.7% (Ref. 11). Following the "T63 Regenerative Engine Program" various other programs have been reported in the literature. For example a comprehensive evaluation of regenerative power plants has been reported by Colin F. McDonald in "Recuperated Gas Turbine Aero-engines", Part I (Ref. 1),Part II (Ref. 12) and Part III (Ref. 13).

It is evident from the literature that the assessment and evaluation of advanced regenerative cycle was conducted with prime focus on the enhancement of helicopter engine performance and operational benefits, paying little attention towards the assessment of environmental impact resulting from the change in engine technology. One of the reasons for not accounting for helicopter emissions in the past might have been driven by lack of concern for "environmental degradation" by government and associated authorities. However, as the aviation industry has grown over time, the concerns over its impact on the environment have also grown significantly.

Regenerated engines are recognized as one of the most promising alternative (aero-engine) power plant configurations when targeting significant reductions in fuel burn and lower emissions e.g. carbon dioxide (CO_2). The most fundamental advantage offered by the regenerative engine due to its distinct thermodynamic cycle is the reduction in fuel burn, achieved through regeneration of exhaust heat. This advantage is rather significant in the current era, considering the concerns for high fuel prices and the strongly imposed government legislations for maintaining adequate emissions levels. Furthermore, the availability of the technology can now enable the development of light weight and efficient heat exchangers that can fulfill the purpose without penalizing the operational performance of existing rotorcraft. However, it has to be noted that, while regenerative technology offers benefits towards helicopter operational performance and environmental impact through fuel burn reductions, it also calls for an inevitable tradeoff associated with the production of emission species that are of major concern. Fakhre et al in (Ref. 14). conducted an extensive parametric study for an existing TEL multipurpose helicopter configuration employing conventional engines and advanced regenerative engines, within a multidisciplinary framework. The study concluded that "regeneration significantly favors reduction in CO_2 emissions through the reduction in fuel burn. However, it also demands conditions within the combustion chamber that inevitably cause elevated levels of other species e.g. thermal NO_x, by rising equilibrium temperatures at the early stage(s) of the combustion process. This issue can be mitigated with the integration of alternative advanced combustion concepts, e.g. staged combustion, lean burn premixed prevaporised combustion (LPP), rich burn quick quench lean burn combustion (RQL), flameless oxidation or the so-called diluted combustion for abatement of NO_x emission. The above technologies together with an increase in thermal uniformity could make the regeneration concept even more attractive for a future greener aviation".

Scope of the present work: The literature currently available on regeneration technology with regards to its application to rotorcrafts, reveals a gap in knowledge. A complete assessment of the technology in terms of its implications on engine design parameters, engine overall weight and associated effects on fuel burn and emissions inventory has not been addressed in an integrated multi-disciplinary environment, with implicit consideration of the individuality of a complete three-dimensional helicopter mission.

This study proposes an integrated rotorcraft multi-disciplinary design and optimization framework, targeting the preliminary design of an optimum regenerative engine configuration in terms of total mission fuel consumption, emissions inventory, as well as total engine weight. A generic rotorcraft model, based on the Eurocopter Bo105 TEL helicopter is considered under a baseline passenger mission representative of modern helicopter operations. The design space corresponding to the advanced regenerative engine thermodynamic cycle parameters, overall mission fuel burn, engine weight and NO_x emissions is thoroughly investigated through the application of a Latin Hypercube Sampling (LHS) Design Of Experiment (DOE) approach. The design space corresponding to the baseline engine is established by carrying out helicopter flight simulations at mission level, catering for integrated helicopter-engine performance, engine weight, gaseous emissions and operational procedures.

The response surface models corresponding to the respective design space are constructed by utilising the Kriging meta-modeling technique. The interdependencies between the various engine design inputs/outputs are quantified at mission level. A novel single-objective Particle Swarm Optimizer (sPSO) is employed to derive optimum regenerative engine configurations which correspond to minimum mission fuel burn, minimum mission NO_x inventory and for minimum engine weight. It is demonstrated through the acquired single-objective optimization results that, optimising for minimum engine weight, leads to a configuration that offers greater operational improvements compared to benefits realized when optimizing for minimum mission fuel burn and minimum mission NO_x inventory, under the conditions simulated.

Pareto front models are also derived through the application of a novel multi-objective Particle Swarm Optimizer (mPSO) for mission fuel consumption and mission NO_x inventory. The acquired optimum engine models, obtained from the Pareto front, are subsequently deployed for the design of advanced rotorcraft engine cycles, targeting improved mission fuel economy, enhanced payload range capability as well as improvement in the rotorcraft overall environmental impact. The deployed methodology essentially constitutes an enabler in terms of focusing the multidisciplinary design of rotorcraft powerplants to realistic, three-dimensional operations, and towards the realization of associated engine design tradeoffs at mission level.

SIMULATION METHODOLOGY

Framework numerical integration and formulation: This study requires the deployment of a multidisciplinary rotorcraft simulation framework, coupled with effective optimization algorithms in order to allow for efficient design exploration. The modeling methodology deployed for the simulation of complete helicopter operations within this paper comprises a series of dedicated numerical formulations, each addressing a specific aspect of helicopter flight dynamics, engine performance, engine preliminary weight estimation and computation of mission emissions inventory. The proposed simulation methodology herein comprises the Lagrangian rotor blade modal analysis presented in (Ref. 15), a flight path profile analysis based on the World Geodetic System dated in 1984 (WGS 84) (Ref. 16), a non-linear trim procedure solving for the aeroelastic behaviour of the main rotor blades as described in (Ref. 15 and 17). And an engine performance analysis model and gas turbine emissions model as detailed in (Ref. 18 and 19).

Each of the aforementioned modeling methods is integrated together within a standalone framework under the name "HECTOR" (HEliCopTer Omni-disciplinary Research Platform). HECTOR is capable of simulating complete, three-dimensional helicopter missions using a fully unsteady aeroelastic rotor model. HECTOR has been extensively described in (Ref. 20), therefore only a brief description of the associated models is provided in this paper.

Gas turbine performance simulation (Turbomatch): The engine modelling and performance simulation code (Turbomatch) employed for the simulations carried out in this study is a Cranfield University (Ref. 18). in-house code, developed over a number of decades. Turbomatch has previously been utilised in several studies available in the literature for the prediction of Design Point (DP) and Off-Design (OD) performance of gas turbine engines (Ref. 21 and 22). In order to comply with the scope of work presented in this paper, the engine is assumed to be operating at steady-state OD conditions throughout the mission.

Emissions prediction model – Hephaestus: In order to predict the gaseous emissions arising from the fossil fuel combustion in the combustion chamber, the deployment of a robust prediction methodology is necessary. To satisfy this need, a generic emission indices calculation software has been adopted with the integration of Hephaestus, developed by Cranfield University. Hephaestus provides a general prediction methodology based on the stirred reactor concept along with a set of simplified chemical reactions. Hephaestus is capable of accounting for differences in the combustion system. Thus the user can specify a combustor geometry in terms of primary, intermediate and dilution zone volumes as well as the mass flow distribution of a given combustor design. Hephaestus has previously been adopted in several aircraft trajectory optimization studies for example in (Ref. 23). Since the scope of this study is to assess the advancement in the engine technology and its associated trade-offs, details on the emissions modelling methodology have not been included herein, however, the numerical formulation and methodology employed for the purpose of emissions prediction has been separately reported by the authors in the following references (Ref. 14 and 27). Thus, further elaboration shall be omitted.

Regenerated turboshaft engine: For the purpose of this study a TEL helicopter configuration is investigated. The currently installed simple cycle turboshaft engine is notionally modified by adding a HE, demonstrating a regenerated turboshaft engine. The regenerated turboshaft incorporates a HE (shown in Figure. 1); the hot side is placed downstream of the Free Power Turbine (FPT) and the cold side upstream of the combustion chamber. This arrangement enables heat transfer between the exhaust gas and the compressor delivery air prior to combustion chamber.

Depending on the Heat Exchanger Effectiveness (HEE), the ability of the heat exchanger to transfer heat (derived by using equation 1), an increase in (working fluid) compressor delivery air temperature is achieved. This process of preheating upstream of the combustion chamber leads to lower fuel input requirements and essentially results in reduced overall mission fuel burn compared to the baseline simple cycle engine. However, the side-effects resulting from the incorporation of the HE include; i) the additional pressure losses introduced by the heat transfer process and by the installation arrangement of the HE, ii) the increase in inlet temperature of the combustion chamber which increases the tendency to emit higher nitrogen oxides (NO_x) (thermal NO_x) levels and finally, iii) the added weight of the heat exchanger.

The schematic presented in Figure 1 is simply the reflection of how the engine is modeled in Turbomatch (gas turbine performance model) and is purely drawn for demonstration purposes. The schematic may vary depending on the choice and the installation arrangement of the heat exchanger.

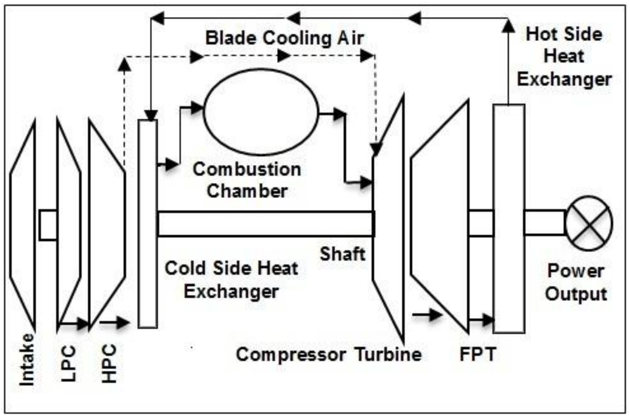


Figure 1: Schematic Layout of a two-spool Regenerated Turboshaft

$$HE\ effectiveness = \frac{T_{out} - T_{Comp}}{T_{Exhaust} - T_{Comp}}\ \% \qquad [1]$$

The associated design effects of the regenerative engine on the helicopter performance are mainly captured in terms of weight deltas compared to a baseline engine. An empirical correlation derived from previously published studies is incorporated to account for the onboard Heat Exchanger (HE) weight. Variations in engine weight due to changes in engine design parameters during the optimization process are catered for by adopting a Whole Engine Based Approach (WEBA). A preliminary weight estimation method for turboshaft engines, exclusively developed for the execution of this study has been adopted. Both weight estimation approaches are briefly discussed in the following section of this paper.

Heat exchanger and engine weight estimation: The HE weight correlation utilized for the purposes of this study is adopted from [Ref. 24]. The correlation is presented in Figure 2. The helicopter configuration investigated in this study represents a TEL configuration, therefore the gross heat exchanger weight for the helicopter was extended to "two engines" during the simulations to establish weight deltas between the baseline and the regenerated engines.

The calculation of engine weight due to varying engine design parameters during the optimization process demands a comprehensive weight estimation methodology. The weight estimation of aero-engines is challenging and can turn into a laborious exercise. Generally speaking, two types of approaches can be adopted, a Component Based Approach (CBA) and a WEBA.

The employment of CBA is expected to have more credibility and accuracy, as correlations for individual components are acquired based on their design parameters. This approach is mostly favoured for fixed wing aircrafts e.g. turbofans, where the quantity of components comprised is large. With regards to WEBA, the approach is fast and is based on simple correlations between the engine cycle design parameters and the overall engine weight. This particular approach enables the investigator to establish a rapid engine weight estimation that can be utilised for design assessments at the preliminary conceptual design level. WEBAs are discussed in the following references [Ref. 25 and 26], mainly applied to fixed wing aero-engines. In the context of this study a WEBA was followed to derive a fast, qualitative estimate of engine weight rather than detailed component-by-component calculations.

A database was constructed for turboshaft engines of up to 1000shp. The upper limit of 1000Shp was set to restrict the database to be specifically representative of Single Engine Light and Twin Engine Light helicopter variants and to maintain engine design consistency from the preliminary design point of view. This was also done to limit the scope and application of the desired engine weight model to be only compatible for baseline engine and similar type of engine variants. In total 48 turboshaft engines were collected from the available public domain source (Ref. 33). Only the information specific to engine design parameters e.g. mass flow, TET and OPR along with SFC at maximum contingency power as well as engine dry weight were recorded for each engine. Once sufficient data was collected, each engine was then modeled and simulated in Turbomatch according to its component technology level in terms of polytropic efficiency. For this purpose the trends reported in (Ref. 34) were utilised. To maintain consistency, all engines were modeled at maximum contingency power and their acquired SFC was benchmarked against the readily available data collected from (Ref. 33).

Upon the completion of the database, the next step was to organise and assign each engine design point cycle parameters as inputs, and its respective dry weight as an output. A standard method of Least-Squares interpolation technique was then applied to develop response surface models (RSMs) for engine dry-weight, as a function of engine design point cycle parameters e.g. mass flow, TET and OPR. The acquired RSMs for engine dry-weight and its validation is further elaborated under the results and discussion section of this paper. The overall methodology and procedure followed for the development of the engine weight estimation model, validation and its integration in HECTOR is presented in Figure 3 respectively.

It should be emphasized that the specific task of engine weight estimation included within this work was not oriented towards establishing a verification of any engine weight estimation analysis tool. Rather, the analysis performed was focused on establishing a mathematical function that can provide a rapid estimation of engine dry weight, based on the basic engine design point cycle parameters. This was needed to be integrated into the HECTOR framework to develop (1) a more credible and consistent design space and (2) corresponding engine weight deltas between the baseline and conceptual cycle engine.

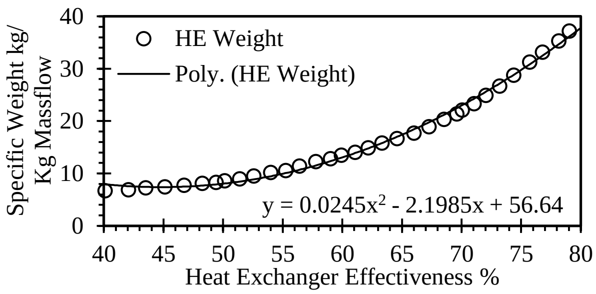


Figure 2: Fixed geometry tubular type heat exchanger specific weight correlation adopted from (Ref. 24) integrated in HECTOR

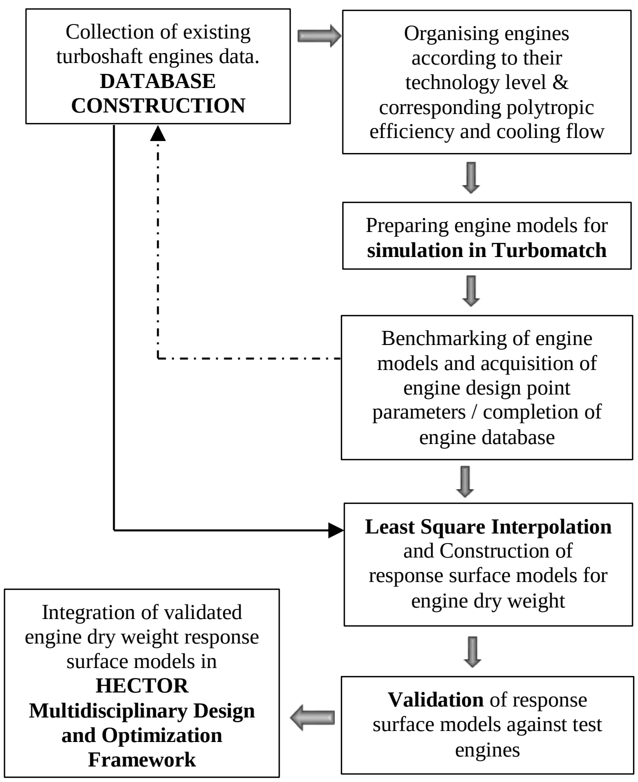


Figure 3: The analysis methodology scheme for; preliminary turboshaft engine dry weight estimation

OPTIMIZATION STRATEGY

It is evident that the nature of the problem addressed within this study requires formulation of the various disciplines, solved subsequently in an integrated multidisciplinary manner. This is a major step forward in rotorcraft mission analysis and generally in engineering design as it builds the foundations for accounting for synergies between the multiple disciplines. However, this comes with a considerable increase in computational cost. On top of that, following the usual practice of trial-and-error with such multidisciplinary problems is deemed as prohibiting as it is carried out in a multi-variable and multi-output context; it is considerably challenging to make decisions on the grounds of multiple competing outputs without the use of a robust optimisation strategy. In order to tackle the aforementioned complexities a consistent optimisation strategy is required. Taking into account the computational expenses that might be incurred by running HECTOR numerous times as well as realising the highly non-linear relations between the multitude of inputs and outputs, two major tasks were regarded as appropriate; firstly, the exploration and approximation of the design space and secondly, the actual optimisation of the system.

Design space exploration: Although experts are normally involved in the process of engineering design it is often essential to explore the design space of the problem at hand. In this way, a first mapping is achieved on how the discipline-specific models behave within a multidisciplinary system. It is, therefore, crucial to choose an appropriate DOE technique that will effectively capture in a systematic way the maximum possible information of the system's response. For the purposes of this task the LHS is employed as it has proven to be an effective design space-filling technique (Ref. 28), particularly for complex systems like the current rotorcraft mission analysis problem. The LHS method segregates the design space into a hypercube grid and fills it by avoiding any confounding effect of the experiments. Nonetheless, this can be regarded only as the first step in the two-stage process of design space exploration and approximation.

The second stage is focused on the approximation of the system's response. In this step, a Response Surface Model (RSM) is built. This comprises of developing a meta-model aiming to describe the complex relationship between the multiple inputs and outputs of the system technique used. Hence, these two steps should be considered as strongly related and complementary to each other.

Choosing the most appropriate technique to build the meta-model often requires some insight on both the engineering problem at hand as well as the potency of the approximating method. The former is relatively appreciated through the application of the systematic LHS design. In respect to the latter, Kriging meta-modeling has been proven to be among the most promising approaches for the approximation of highly non-linear problems (Ref. 29 and 31) and is chosen in this work to construct the required RSMs. The process integration as well as the necessary tools required for the aforementioned steps were realised through the multi-purpose simulation platform "NOESIS Optimus" (Ref. 30).

Optimization approach: Once the design space exploration and approximation stages are successfully completed what remains is to determine the optimum designs of the complex rotorcraft mission analysis problem. Usually such problems involve characteristics that impose barriers in approximating the true optimum solutions. In (Ref. 32) several complexities were indicated in this context; multi-modality, deceptive peaks, noisy landscapes and isolated optima can significantly affect convergence to the optimal solutions. Due to the several complexities and non-linear relations between the multiple inputs and outputs of the problem at hand, a comprehensive and effective global optimiser is required.

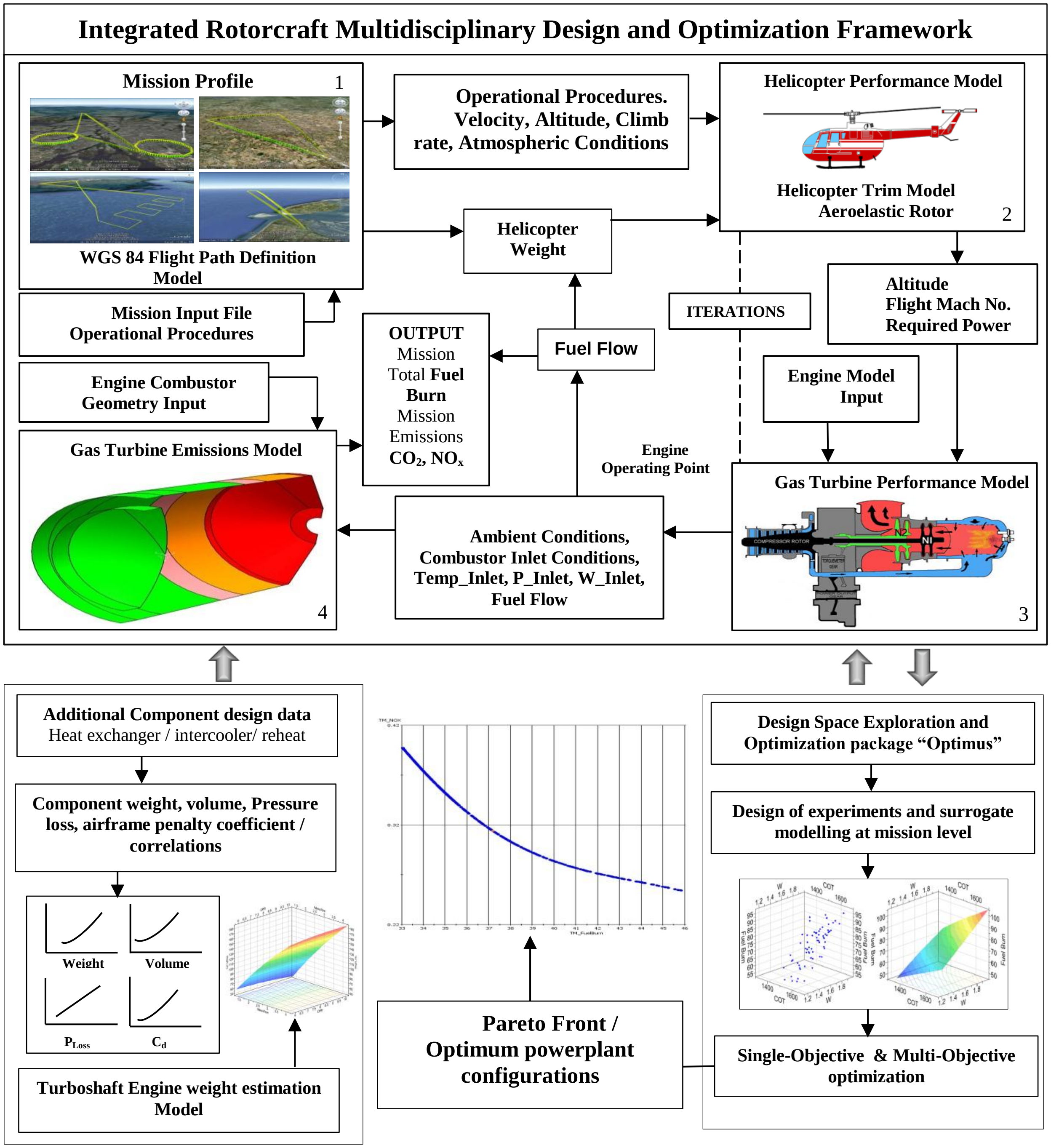


Figure 4: Architecture of integrated rotorcraft multidisciplinary design and optimization framework; design and analysis of conceptual rotorcraft powerplant configurations

In this work two novel single-objective and multi-objective (sPSO and mPSO), global optimizers are employed in order to deal with the complexities mentioned above. In (Ref. 33) the authors have elaborated on the advantages and disadvantages of multiple state-of-the-art optimization algorithms when utilised in multidisciplinary environments and have exemplified the effectiveness of the novel PSO optimisers both in terms of convergence and computational demands.

Overall, the PSO algorithm is based on individuals that imitate the flocking of birds or schooling of fish populations. Its behaviour is driven by the "self-awareness" of an individual which promotes the exploration of the design space and by the "social-awareness" which encourages exploitation of promising areas in the design-space. Further information on the development of the novel PSO used in this study can be found in (Ref. 33). In this work, the novel PSO was deployed both for single-objective optimisation studies, as well as building the corresponding Pareto fronts of two design outputs as shown in the following sections.

Compilation of helicopter and engine configuration: The aircraft deployed for the purpose of this study is modeled after the Eurocopter Bo105 helicopter. The Eurocopter Bo105 is a TEL utility multipurpose helicopter equipped with two Rolls Royce Allison 250C20B turboshaft engines rated at 313 kW maximum contingency power. Table 1 presents the helicopter model characteristics.

The Allison 250C20B engine is equipped with a single-spool gas generator including a six-stage axial compressor followed by a centrifugal compressor. The engine configuration is outlined in Table 2. The maximum contingency power setting is selected as the design point for the respective Turbomatch model. The model has been matched at design point conditions with public domain data (Ref. 34) in terms of SFC. The configuration of the Bo105 as well as its performance characteristics have been extensively documented and analysed in [35] thus, further elaboration shall be omitted. A detailed description of the Allison 250C20B engine family can be found in (Ref. 33).

Case study definition: A generic three-dimensional reference mission representative of a modern TEL helicopter was designed in the context of a PATM. The incorporated operational procedures in terms of geographical location selection, deployed airspeed, altitude, climb/descent rates and idle times have been defined with input from the European Helicopter Operator's Committee (EHOC).

The geographical representation, in terms of global coordinates, along with the deployed operational procedures, are illustrated in Figures 5(a) and 5(b) respectively. The PATM designed for the purpose of this study assumes that the helicopter takes off from a heliport in Germany to pick up the designated passenger(s) from a secondary location. It subsequently transfers them to a nearby hotel and transits back to the heliport where it originated from.

Design space definition: Having established the corresponding engine design and mission parameter values for the integrated Bo105 helicopter–engine systems (presented in Table 3), the design space corresponding to the engine size, as well as various thermodynamic cycle parameters can be defined for the regenerated engine. The overall aim is to acquire thermodynamic cycle parameters related to a regenerated engine, which can lead to lower overall mission fuel burn and emissions with minimum engine weight, while maintaining the DP shaft power and payload-range capability of the baseline engine.

Table 1: Baseline design parameters: Reference Bo105 twin-engine light helicopter configuration

Design Parameter	Value	Units
Max Gross Weight	2500	kg
OW	2200	kg
Number of blades	4	-
Blade chord	4.91	M
Blade twist	8	Degree
Rotorspeed	44.4	Rad/sec

Table 2: Baseline design parameters: Reference Bo105 twin-engine light helicopter engine

Design Parameter	Value	Units
TET	1470	K
W	1.56	Kg/sec
LPC PR	2.73	-
HPC PR	2.6	-
DP shaft power	313	kW
DP SFC	109.98	μg/J

Table 4 presents the design variable bounds set for the reference Allison 25C20B engine, respectively. The design space variables correspond to the engine's LPC, HPC pressure ratio, TET, Air Mass Flow (W) and the notional Heat Exchanger Effectiveness (HEE). The variable bounds have been defined as such, so that they reflect medium-term engine redesign. Throughout the course of the optimization process, a constant technology level is assumed in terms of maximum allowable TET as well as engine component polytropic efficiencies. A constant polytropic efficiency of the order of 87% is assumed for the axial compressor and 83% for the centrifugal compressor, at DP operation (Ref. 9).

The maximum allowable DP TET is limited to the baseline value as presented in Tables 1 and 2. Thus, although the impact of a higher technology level in terms of maximum allowable TET has been accounted for throughout the DOE method (Table 4), the aforementioned effect is excluded from the optimization process in order to comply with the limitations of a constant technology level approach. It is noted that any potentially optimum regenerated engine designs need to comply with specific airworthiness specification requirements, such as acceptable One Engine Inoperative (OEI) performance for Category A helicopter operations. The maximum engine take-off power at DP is therefore constrained to the designated baseline values for both engine configurations as described in Tables 1 and 2. This constraint is applied so that the payload–range capability of the reference helicopter is not changed during the optimization process.

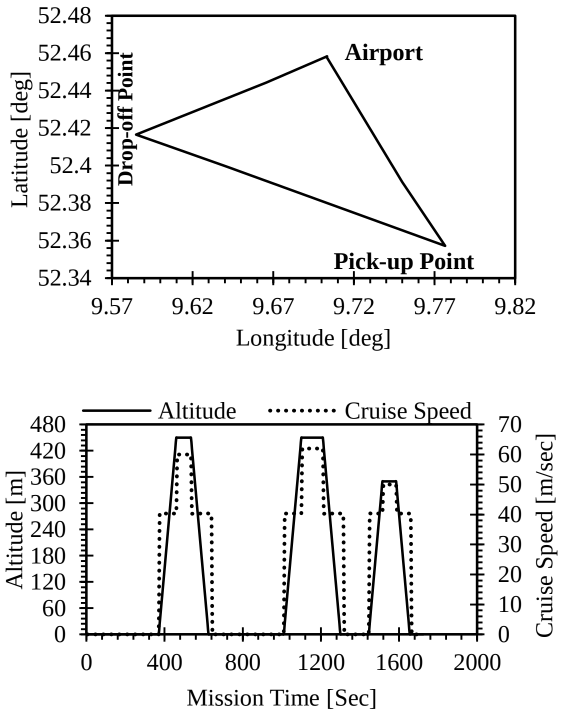


Figure 5: Reference Passenger Air Taxi Mission: (a) geographical definition; (b) time variations of deployed operational airspeed and AGL altitude

Table 3: Total mission parameters fort baseline engine design

Mission Parameter	Value	Units
Time	1725	Seconds
Range	36.22	km
Fuel Burn	59.99	kg
EI CO_2	191.92	Kg
EI H_2O	74.6	kg
EI NO_x	0.287	kg

Table 4: Bounds for DP engine size and thermodynamic cycle variables

Design Parameter	Low Bound	High Bound	Units
LPCPR	1.3	3.1	-
HPCPR	1.3	3.3	-
W	1	2	Kg
TET	1300	1600	k
HEE	0.4	0.8	%

RESULTS AND DISCUSSION

Engine Weight estimation RSM and validation: As discussed in the earlier section of this paper. A WEBA was employed within this study with the objective to derive a mathematical function for engine dry weight in terms of engine cycle design parameters. This was required to enable a sound and consistent prediction for engine dry weight from a preliminary engine design point of view. The computation for variations in the engine weight during the design space exploration and optimization process is of great importance for the development of a multi-fidelity system design approach. Since the weight deltas between the baseline engine and the conceptual engine design can have a significant impact on the rotorcraft AUM, and therefore the required power, overall mission fuel burn as well as on the mission emissions inventory e.g. CO_2, NO_x emissions.

Table 5 presents the interdependencies between each engine design parameter inputs (W,PR,TET) and output (Engine Dry Weight) in terms of the linear correlation coefficients, derived from the developed engine dry weight RSMs. The acquired correlation coefficients enable to establish the amount and type of average dependency amongst each design input and output. The value of such coefficients range between -1 to 1, the sign indicates the nature of relation while the absolute value defines the magnitude of relation.

Table 5: Design input/output linear correlation coefficients: Engine cycle design parameter, engine dry weight

Design input/output	Engine Dry Weight
TET	0.124
W	0.936
OPR	0.247

As expected, all the engine design input parameters result in a positive correlation towards the engine dry weight. However, it is worthy to note that among all the engine design inputs, the mass flow has the dominant effect on the overall engine dry weight. Increasing or decreasing the engine design mass flow has an overall influence on both the turbomachinery size and design, as well as on the physical size of the overall engine. With regards to the OPR and the TET, a rather moderate effect is observed. This can be attributed to the fact that, the engine database was only limited to engines with shaft power of up to 1000shp. Therefore, the pressure ratio and TET variation are not too large to have a major impact on engine weight, as shown in Figure 6 and 7. The TET varies from 1000 K – 1470K and the pressure ratio from 6:1 – 10.5:1.

Figure 6: Engine dry weight response surface model; engine mass flow and turbine entry temperature

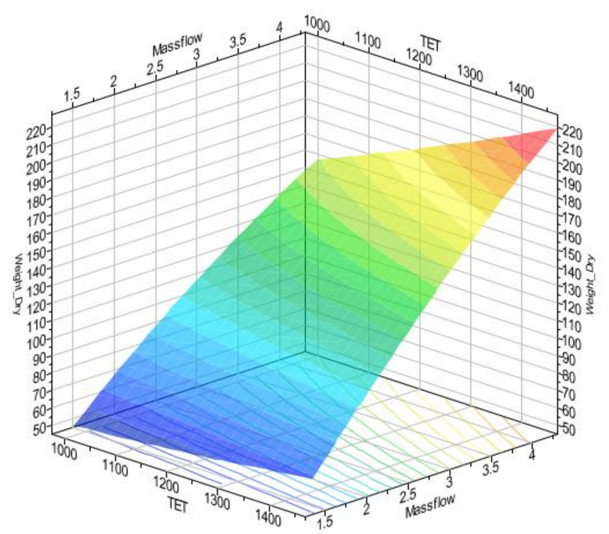
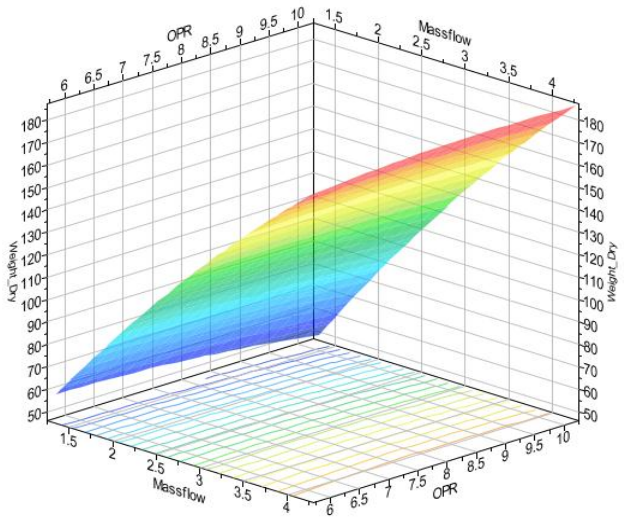


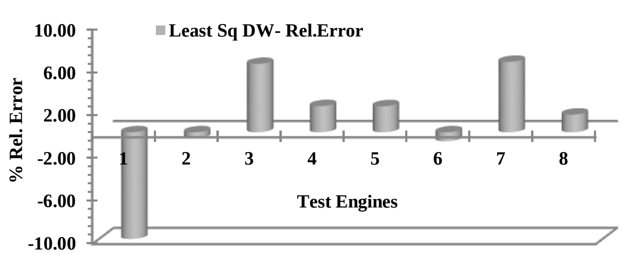
Figure 7: Engine dry weight response surface model; engine mass flow and overall pressure ratio



A separate set of carefully selected test engines was used to verify the validity of the developed engine dry weight RSMs hypothesis against the real engine data. The test engines selection was made as such to include engines that represent wide range of technology level, engine weight, as well as engine design point cycle parameters. This approach was followed to ensure the robustness and predictive variability of the developed engine dry weight RSMs across a diverse range of representative engines.

Figure 8 shows the variation in the prediction capability of the developed RSMs. The model demonstrated a strong predictive capability across the wide range of selected test engines. The observed relative error for all test cases was within $\pm 10\%$, and are deemed accepatable for the purpose and scope of this study. The developed engine dry weight RSMs were incorporated accordingly in HECTOR to cater for the engine weight estimation during the DOE and optimization process.

Figure 8: Engine dry weight response surface model prediction; relative error for the test engines



LHS and RSM approach results: The design space corresponding to the established baseline engine design parameter boundaries can now be implemented, as sufficient mission data is now readily available for the execution of HECTOR coupled with the DOE process. For the purpose of this study, a total of one hundered simulations have been performed for the reference Bo105 model within the HECTOR framework, presented in Figure 7. With the successful completion of the DOE process, the interdependencies between each engine design parameter inputs and outputs are acquired in terms of the linear correlation coefficients as previously established and discussed for engine dry weight RSMs.

The linear correlation coefficients in terms of the systems response are presented in Table 6. The correlations suggest that HEE favours the reduction in overall mission fuel burn, however HEE has a detrimental effect on NO_x. Also, as shown in Figure 2, the weight of the heat exchanger is a function of its design effectiveness and mass flow, which is strongly captured by the DOE process. The acquired linear correlation for mass flow, suggests that it has a strong influence on engine weight and design point power. Furthermore, the effect of LPC and HPC on engine thermal efficiency and design point shaft power is also evident to be well captured by the DOE process. As both the LPC and HPC have a strong correlation for DP shaft power as well as a significant impact on the reduction of mission fuel burn.

Following the successful execution of LHS, RSMs have been structured based on the DOE results, using interpolation based on the Kriging technique. The acquired RSMs describe the mathematical relationship between the engine design inputs (HPC PR, LPC PR, W, TET, PR, HEE) and outputs, mission fuel consumption, DP shaft power, engine weight and mission emission index of NO_x. The developed RSMs are subsequently used as drivers throughout the optimization process, presented in this paper.

Table 6: Regenerated engine cycle design input/output linear correlation coefficients: reference Bo105 helicopter/ passenger mission

Design input/output	DP shaft Power	Fuel burn	Engine Weight	Nitrogen Oxide (NO_x)
LPC PR	0.372	-0.476	0.021	0.388
HPC PR	0.476	-0.447	0.103	0.432
TET	0.430	-0.041	0.041	-0.114
W	0.630	0.303	0.690	-0.436
HEE	-0.047	-0.620	0.574	0.495

Single-Objective Optimizations: Having established the design space and the associated numerical formulation of RSMs, three respective single-objective optimizations were decided to be performed. This was done to acquire the optimum engine configuration for minimum mission fuel consumption and the optimum engine configuration for minimum engine dry-weight as well as to acquire the optimum engine configuration corresponding to minimum mission NO_x inventory. For the optimization purpose the sPSO algorithm was deployed, since it was already established from a revious study that PSO is a strong candidate amongst other techniques e.g. SAE (Ref. 7 and 32).

The maximum attainable DP TET was limited to the baseline value in order to represent a constant technology level. The DP shaft power of each engine was also constrained to its baseline value in order to account for sufficient OEI performance, and therefore comply with airworthiness certification requirements. Table 7 presents the bounds for engine thermodynamic design parameter inputs and design constraint applied for both aforementioned single-objective optimizations.

In order to check the reliability of the constructed RSMs, separate HECTOR simulations were performed for all three acquired optimum configurations. Tables 8,9 and 10 presents the percentage reduction in fuel burn and NO_x obtained through RSM and HECTOR simulations, with respect to the baseline engine configuration. An average relative error of up to 2% was achieved between the RSMs and HECTOR simulations in terms of mission fuel consumption and NO_x inventory, corresponding to all three optimum configurations.

Table 11 presents the optimum engine design parameters acquired for minimum mission fuel burn against the baseline engine. The optimum configuration has around 10.21% lower engine OPR, 16.53% lower mass flow and 69.44% higher engine weight, with the potential to reduce the mission fuel consumption by approximately 51.5%, and results in two times more NO_x emissions compared to baseline engine.

Table 7: Bounds for DP engine size and thermodynamic cycle- Single-objective optimization

Design Parameter	Low Bound	High Bound	Units
LPCPR	1.3	3.1	-
HPCPR	1.3	3.3	-
W	1	2	Kg
TET	1470		K
HEE	40	80	%
Constraints for single-objective optimisation			
DP Power	313000		W
TET	1470		K

Table 8: RSM relative error and minimum fuel burn optimization results

	RSM	HECTOR	RSM rel.error %
Baseline	58.8	59.99	-1.98
Optimized	29.10	29.25	-0.53
Reduction	-50.52	-51.24	Avg rel. error 1.25%

Table 12 presents the optimum engine design parameters acquired for minimum mission NO_x inventory against the baseline engine. The optimum configuration has around 48.24% lower engine OPR, 5.1% increased mass flow and 16.67% higher engine weight, with the potential to reduce the mission NO_x inventory by almost 59.55%, while simultaneously reducing the mission fuel burn by around 4.62%.

Table 13 presents the optimum engine design parameters acquired for minimum engine weight against the baseline engine. The optimum configuration has around 23.19% lower engine OPR, 15.21% reduced mass flow, equal overall engine weight, with the potential to reduce the mission fuel burn by approximately 26.99%, and results in only 11.5% increase in mission NO_x inventory compared to the baseline engine.

Table 9: RSM relative error and minimum NO_x optimization results

	RSM	HECTOR	RSM rel. error %
Baseline	0.282	0.287	-1.74
Optimized	0.116	0.119	-2.44
Reduction	-58.83	-58.54	Avg rel.error 2.09%

Table 10: RSM relative error and minimum weight optimization results

	RSM	HECTOR	RSM rel. error
Baseline	58.8	59.99	-1.98
Optimized	43.8	44.16	-0.82
Reduction	-25.89	-26.39	Avg rel. error -1.39

Table 11: Comparison between baseline and optimum engine cycle parameters for minimum mission fuelburn: Bo105 helicopter/passenger mission

Design parameter	Baseline	Optimum	Units	Rel.baseline %
LPC_PR	2.73	2.31	-	-15.54
HPC_PR	2.6	2.76	-	6.30
OPR	7.098	6.37	-	-10.21
TET	1470	1470	k	0.00
W	1.56	1.30	g	-16.53
HEE	0	0.80	%	80.00
Weight	144	244	kg	69.44
Mission output parameters				
Fuel Burn	59.99	29.10	kg	-51.50
NO_x	0.287	0.87	kg	202.44

Table 12: Comparison between baseline and optimum engine cycle parameters for minimum NO_x inventory: Bo105 helicopter/passenger mission

Design parameter	Baseline	Optimum	Units	Rel.baseline %
LPC_PR	2.73	2.20	-	-19.43
HPC_PR	2.6	1.67	-	-35.76
OPR	7.098	3.67	-	-48.24
TET	1470	1470	k	0.00
W	1.56	1.64	g	5.17
HEE	0	0.40	%	40.00
Weight	144	168	kg	16.67
Mission output parameters				
Fuel Burn	59.99	62.76	kg	4.62
NO_x	0.287	0.116	kg	-59.55

Table 13: Comparison between baseline and optimum engine cycle parameters for minimum engine weight: Bo105 helicopter/passenger mission

Design parameter	Baseline	Optimum	Units	Rel.baseline %
LPC_PR	2.73	2.34	-	-14.11
HPC_PR	2.6	2.33	-	-10.57
OPR	7.098	5.45	-	-23.19
TET	1470	1470	k	0.00
W	1.56	1.32	g	-15.21
HEE	0	0.40	%	0.40
Weight	144	144	kg	0.00
Mission output parameters				
Fuel Burn	59.99	43.8	kg	-26.99
NO_x	0.287	0.32	kg	11.50

An interesting observation can be made by comparing the optimized configurations acquired for minimum mission fuel burn with the minimum mission NO_x inventory. It is well understood that the objective functions corresponding to both aforementioned configurations lead to conflicting design requirements. Designing an engine to attain minimum mission fuel burn requires an engine design that corresponds to maximum attainable thermal efficiency, under the imposed design criterion. However, on the other hand minimization of mission NO_x inventory requires an engine design with the lowest possible thermal efficiency, specifically for the problem under consideration.

For the problem at hand, the overall thermal efficiency is mainly dependent on the OPR, TET and HEE. Since, the optimization was perfomed at fixed TET to comply with constant technology engine redesign rule. Therefore, only the engine OPR and HEE have the influence on the overall engine thermal efficiency.

It is evident from the results presented in Tables 11 and 12 that the optimized solution for minimum fuel burn has significantly higher OPR compared to the solution acquired for minimum mission NO_x inventory. Furthermore, the HEE for minimum fuel burn solution corresponds to the upper limit of the design space bounds set for HEE, presented in Table 4 i.e. 80%, while for the minimum NO_x solution, lowest possible value of the HEE is achieved i.e. 40%.

Optimized configurations operational benefits: To identify which configuration offers the greatest value at operational level, further quantification of the realized benefits is required, e.g. operational benefits in terms of payload, range and environmental impact.

It is noted that the acquired single-objective optimum configurations are optimized towards specific design objective. Therefore, they should not be cross-compared amongst each other. However, a general comparison of the implications of each configuration on the rotorcraft overall operational capability can be made based on a generic design criterion.

In general the choice and selection of the powerplant is mainly based around the imposed design criterion of the desired rotorcraft. The design merits and qualification of the rotorcraft is mainly driven by the required mission/ operation that the desired rotorcraft is destined to serve e.g. civil, military etc. Along with many other design parameters, the design parameters such as All-Up-Mass (AUM), payload, range and environmental impact are of prime importance, specifically when designing an engine for civil rotorcraft e.g. executive travel.

As the focus of this study is dedicated towards the multidisciplinary design and assessment of civil rotorcraft conceptual powerplant. The most promising configuration is considered as one that offers maximum fuel savings, while simultaneously resulting in minimum engine weight and minimum mission NO_x inventory, under the simulated design space, constraints and operational conditions. Fuel savings can only be used as either an increase in payload capacity of the rotorcraft and/or towards increasing the range of the rotorcraft. In order to establish a consistent comparison between the acquired optimum configurations, it is assumed that the acquired fuel savings are used towards increasing the overall range capability of the rotorcraft.

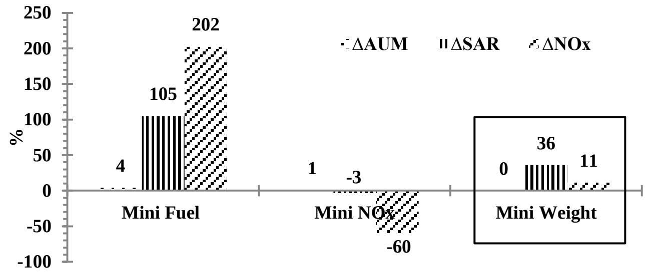
Table 14 presents the key parameters associated with the aforementioned imposed design criterion:- Specific Air Range (SAR), All-Up-Mass (AUM) and the NO_x inventory deltas, establised for all three acquired optimized configurations, with respect to the baseline. It is evident from Figure 9 that, the operational benefits offered by optimized miminum engine weight configuration can be placed close to the aforementioned imposed design criterion. This configuration offers an increase in rotorcraft range capability by 36.02% (at mission cruise conditions), no weight penalty due to chage in engine design, and increases the mission NO_x inventory by only 11%, with respect to the baseline configuration.

Table 14: Comparison between baseline and optimum engine configurations; single-objective results; mission level parameters and deltas; Bo105 helicopter/passenger mission

Parameter	Baseline	Mini Fuel burn	Mini NOx	Mini Weight	Units
Specific Air Range	2.299	4.714	2.224	3.128	km/kg of fuel
$Mean_{ff}$	0.018	0.009	0.018	0.013	kg/sec
AUM	2500	2400	2476	2500	kg
ΔAUM	-	4.0	1.0	0.0	%
ΔSAR	-	105.03	-3.29	36.02	%
ΔFuel burn	-	-50.5	4.62	-26.99	%
ΔNO_x	-	202.44	-59.55	11.50	%

It is therefore demonstrated through the acquired single-objective optimization results. Optimizing for minimum engine weight, essentially leads to a configuration that offers greater operational improvements compared to the benefit realized, when optimizing for minimum mission fuel burn and minimum mission NO_x inventory, under the simulated design space constraints and operational conditions.

Figure 9: Comparison between optimized engine configurations single-objective results; mission level parameters and deltas ;Bo105 helicopter/passenger mission



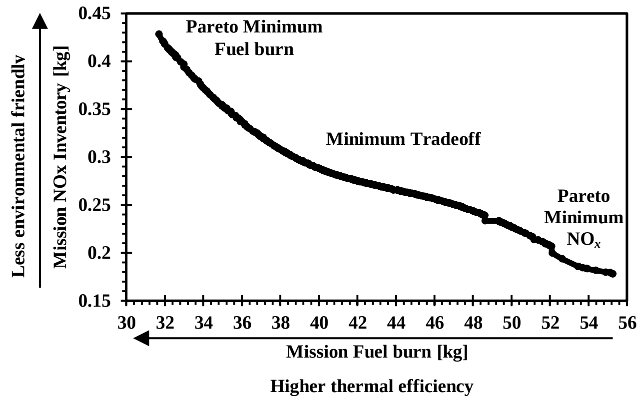
Multi-objective Optimization and results: The optimum configurations acquired through the application of sPSO correspond to the solutions that are optimized for specific design objectives e.g. mission fuel burn, engine weight and NO_x inventory. In order to effectively implement the additional criterion of fuel economy, the associated trade-off between fuel burn and mission NO_x inventory needs to be thoroughly addressed and quantified. For that purpose a multi-objective approach is utilised through the deployment of mPSO, using the developed RSMs.

A Pareto front has been structured for minimum mission fuel burn and minimum mission NO_x inventory. The objective functions of the Pareto front formation process dictate simultaneous minimization of mission fuel burn with minimization of mission NO_x inventory level. The Pareto front acquired is presented in Figure 10 for the conceptual regenerative Bo105 helicopter, under the design space limits presented in Table 7. It is apparent from the Pareto front that the engine mission NO_x inventory increases almost exponentially with minimization of mission fuel burn. This behaviour is mainly attributed to the conflicting design requirements attached with both objective functions. As mentioned earlier designing for minimum fuel burn requires an engine design that corresponds to maximum attainable thermal efficieny, under the imposed design criterion. However, on the other hand minimization of mission NO_x inventory requires an engine design with lowest possible thermal efficiency, specifically for the problem under consideration.

The part of the Pareto front that corresponds to minimum fuel burn employ heat exchangers with higher effectiveness and higher OPR. On the other hand the corresponding configurations acquired for minimum mission NO_x inventory have minimum attainble OPR and HEE, under the imposed design constriants.

The acquired Pareto front can be regarded as preliminary guide with respect to the engine design process. The span of the front allows for engine sizing as well as for selection of thermodynamic cycle parameters in an optimum manner, using a single design criterion. The associated trade-off between mission fuel economy, payload–range capacity and the environmental impact are the compromises that the designer has to accept.

Figure 10: Comparison between optimised engine configurations; mission level parameters and deltas ;Bo105 helicopter/passenger mission



As implemented during the single-objective study, the focus of multi-objective optimization is also directed towards acquiring an optimum engine configuration for civil rotorcraft application. Therefore the most promising configuration is considered as one that offers maximum fuel savings, while simultaneously resulting in minimum engine weight and minimum mission NO_x inventory, under the simulated design space, constraints and operational conditions. Considerting the acquired Pareto front presented in Figure 10, three configurations are of prime importance, highlighted in Figure 10. Insights into the aforementioned configurations engine cycle design parameters will help to established in-depth understanding of the associated tradeoffs and interrelationship between the mission fuel burn and mission NO_x inventory, as well as their influence on the respective engine cycle design parameters.

Similar to the single objective optimization, the Pareto models were also based on the structured RSMs, and therefore it was expected that the same RSM error would propagate to the respective Pareto models. Before proceeding with further analysis it is imperative to validate the quality of the Pareto models acquired. For that purpose the aforementioned three representative points were selected. Separate HECTOR simulations were performed for all three optimum configurations of choice. An average Pareto model relative error calculated is up to -1.94%, presented in Tables 15, 16 and 17, for all three selected configurations respectively.

Table 15: RSM relative error and minimum fuel burn optimization results: Bo105 helicopter/passenger mission

	RSM	HECTOR	RSM rel.error %
Baseline	58.8	59.99	-1.98
Optimized	31.67	32.10	-1.34
Reduction	-46.14	-46.49	Avg rel. error -1.66

Table 16: RSM relative error minimum NO$_x$ inventory optimization results: Bo105 helicopter/passenger mission

	RSM	HECTOR	RSM rel.error %
Baseline	0.282	0.287	-1.74
Optimized	0.178	0.179	-0.56
Reduction	-36.88	-37.63	Avg rel. error -1.15

Table 17: RSM relative error minimum tradeoff configuration for fuel burn and NO$_x$ inventory; Bo105 helicopter/passenger mission

	RSM	HECTOR	RSM rel.error %
Baseline	58.8	59.99	-1.98
Optimized	43.78	44.63	-1.90
Reduction	-25.54	-25.60	Avg rel. error -1.94

Table 18: Comparison between baseline and optimum engine cycle parameters; Pareto model for minimum mission fuel burn: Bo105 helicopter/passenger mission

Design parameter	Baseline	Optimum	Units	Rel.baseline %
LPC_PR	2.73	3.1	-	13.55
HPC_PR	2.6	1.44	-	-44.48
OPR	7.098	4.48	-	-36.95
TET	1470	1470	k	0.00
W	1.56	1.46	g	-6.63
HEE	0	0.80	%	80.00
Weight	144	259.95	kg	80.52
Mission output parameters				
Fuel Burn	59.99	31.67	kg	-47.21
NOx	0.287	0.428	kg	49.13

Table 19: Comparison between baseline and optimum engine cycle parameters; Pareto model for minimum mission NO$_x$ inventory : Bo105 helicopter/passenger mission

Design parameter	Baseline	Optimum	Units	Rel.baseline %
LPC_PR	2.73	3.10	-	13.55
HPC_PR	2.6	1.30	-	-50.00
OPR	7.098	4.03	-	-43.22
TET	1470	1470	k	0.00
W	1.56	1.52	g	-2.36
HEE	0	0.40	%	40.00
Weight	144	157.24	kg	9.89
Mission output parameters				
Fuel Burn	59.99	55.24	kg	-7.92
NOx	0.287	0.178	kg	-37.98

Table 20: Comparison between baseline and optimum engine cycle parameters; Pareto model for minimum tradeoff between mission fuel burn and NO$_x$ inventory: Bo105 helicopter/passenger mission

Design parameter	Baseline	Optimum	Units	Rel.baseline %
LPC_PR	2.73	3.10	-	13.54
HPC_PR	2.6	1.30	-	-50.00
OPR	7.098	4.03	-	-43.23
TET	1470	1470	k	0.00
W	1.56	1.54	g	-1.08
HEE	0	0.60	%	60.00
Weight	144	186.24	kg	39.33
Mission output parameters				
Fuel Burn	59.99	43.78	kg	-27.02
NOx	0.287	0.266	kg	-7.32

Table 18 presents the Pareto optimum engine design parameters acquired for minimum mission fuel burn against the baseline engine. The optimum configuration has 36.5% lower engine OPR, 6.63% lower mass flow and 80.52% higher engine weight, with the potential to reduce the mission fuel consumption by approximately 47.21%, and results in increasing the mission NO$_x$ inventory by 49.13%, compared to the baseline engine.

Table 19 presents the Pareto optimum engine design parameters acquired for minimum mission NO$_x$ inventory against the baseline engine. The optimum configuration has 43.22% lower engine OPR, 2.36% lower mass flow and 9.89% higher engine weight, with the potential to reduce the mission fuel consumption by approximately 7.92%, while simultaneously reducing the mission NO$_x$ inventory by 37.98%, compared to the vbaseline engine

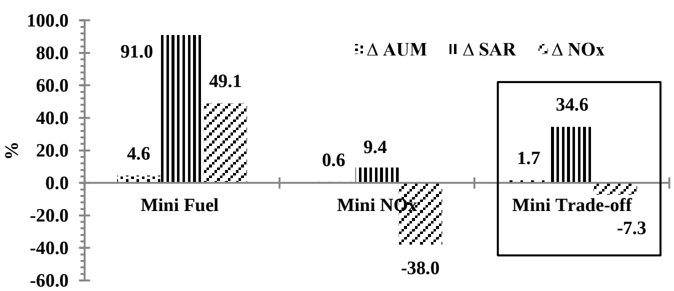
Table 20 presents the Pareto optimum engine design parameters acquired for the configuration corresponding to the boptimum engine configuration that represents the minimum tradeoff between mission fuel burn and mission NO$_x$ inventory, against the baseline engine. The optimum configuration 43.23% lower engine OPR, 1.08% lower mass flow and 39.33% higher engine weight, with the potential to reduce the mission fuel consumption by approximately 27.02%, while simultaneously reducing the mission NO$_x$ inventory by 7.02%, compared to the baseline engine.

Table 21 presents the key parameters derived from the benefits realized from the aforementioned acquired optimum configurations. Specific Air Range (SAR), All-Up-Mass (AUM) and the NO$_x$ inventory deltas are establised for all three acquired optimized configurations, with respect to the baseline configuration. It is evident from Figure 11 that the operational benefits offered by the configuration corresponding to minimum tradeoff between the mission fuel burn and mission NO$_x$ inventory can be placed close to the imposed design criterion. This particular configuration offers an increase in rotorcraft range capability by 36.02% (at mission cruise conditions), increases the AUM of the rotorcraft by only 1.7% and reduces the mission NO$_x$ inventory by 7.3%, with respect to the baseline configuration.

Table 21: Comparison between baseline and optimum engine configurations multi-objective results mission level parameters and deltas; Bo105 helicopter/passenger mission

Parameter	Baseline	Mini Fuel burn	Mini NOx	Mini tradeoff	Units
Specific Air Range	2.299	4.39	2.52	3.09	km/kg of fuel
Mean$_{ff}$	0.018	0.009	0.016	0.013	kg/sec
AUM	2500	2384	2485	2457	kg
ΔAUM	-	4.6	0.6	1.7	%
ΔSAR	-	91	9	35	%
ΔFuel burn	-	-47.21	-7.92	-27.02	%
ΔNO_x	-	49.13	-7.32	-37.98	%

Figure 11: Comparison between optimized engine configurations multi-objective results; mission level parameters and deltas; Bo105 helicopter/passenger mission



Figures 12 and 13 show the fuel flow and NO_x rate against mission time for all three acquired optimized solutions against the baseline configuration, corresponding to the respective passenger mission. It is evident from both the Figures presented that, the configurations corresponding to minimum mission fuel consumption and minimum mission NO_x inventory behave in an opposite manner to each other. It can be established that, the solution with minimum mission fuel burn results in minimum fuel flow requirements, yet has the highest production rate for NO_x emissions. Similarly, the solution for minimum mission NO_x inventory has the lowest production rate for NO_x emissions but has the highest fuel flow requirements.

Figure 12: Fuel flow production rate comparison between baseline and optimally designed engine configurations: multi-objective results; Reference Bo105/passenger mission

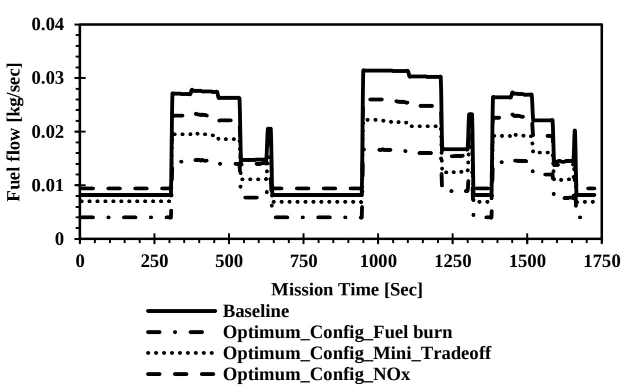
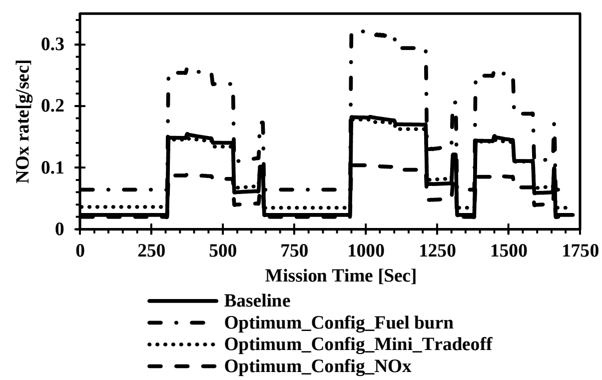


Figure 13: NO_x production rate comparison between baseline and optimally designed engine configurations: multi-objective results; Reference Bo105/passenger mission



Considering the operational benefits presented in Figure 11 for the acquired optimum configurations, through the multi-objective optimization for mission fuel burn and NO_x inventory. In the case where the employment of the regenerated engine is favourable, it merely depends upon the evaluation criteria and the level of discretion the engine designer holds towards the various tradeoffs, imposed by the design requirements. Most certainly, the initial indication from the results is to focus on developing technologies towards minimizing weight and the production of NO_x to mitigate their associated penalties within tolerable limits. These technologies may well be witnessed by introducing lighter materials to construct the on-board heat exchangers, and by introducing advanced or novel combustion designs to offset and stabilize the elevated level of temperatures caused within the combustion chamber, encouraged by the heat exchange process.

For the current technology level to mitigate the associated weight and NO_x tradeoffs are an immediate challenge to enable the regenerated turboshaft engines to be an attractive candidate for future aviation. Nevertheless given the current technology level, there may be some scope for their utilization under certain operations e.g. Aearch And Rescue (SAR), police law enforcement or military transport operations, where the ultimate value for the onboard heat exchanger can be realized due to their long range, and the associated environmental penalties can be negotiated.

It is therefore demonstrated that the deployed methodology can be applied to identify advanced regenerative optimum design specifications for rotorcrafts in terms of sizing and thermodynamic cycle parameters, using a single design criterion. The respective trade-offs that the designer must accept are between mission fuel economy, payload–range capacity as well as the environmental impact.

CONCLUSIONS

An innovative multidisciplinary design and optimization methodology has been proposed for the conceptual design and analysis of alternative rotorcraft powerplant configurations. A multidisciplinary integrated simulation framework capable of computing the flight dynamics, engine performance, engine weight as well as gaseous emissions of any defined helicopter–engine system within any designated operation has been deployed.

A comprehensive and computationally efficient optimization strategy, utilizing a novel particle-swarm optimizer, has been implemented. The overall methodology has been applied to the multidisciplinary design and optimization of a reference Twin-Engine Light civil rotorcraft modeled after the Eurocopter Bo105 helicopter, operated on a representative mission scenario. Engine design specifications, optimized in terms of mission fuel consumption, engine weight and gaseous emissions, have been acquired at constant technology level, through the application of a novel single-objective Particle Swarm Optimizer. It is demonstrated, through the acquired single-objective optimization results that, optimizing for minimum engine weight, leads to a configuration that offers greater operational improvements compared to the benefits realized when optimizing for minimum mission fuel consumption and minimum mission NO_x inventory, under the simulated design space, constraints and operational conditions.

The optimum interrelationship and the respective tradeoff between mission fuel burn and mission NO_x emissions inventory has been quantified through the employment of a novel multi-objective Particle Swarm Optimizer. The acquired optimum Pareto front models corresponding to minimum mission fuel burn and minimum mission NO_x inventory suggest that mission NO_x inventory increases almost exponentially with minimization of mission fuel burn.

It is emphasized that, in the case where the employment of the regenerated engine is favourable, it merely depends upon the evaluation criteria and the level of discretion the engine designer holds towards the various tradeoffs, imposed by the design requirements. Finally, it was demonstrated that the proposed methodology can be applied to identify advanced regenerative optimum design specifications for rotorcrafts in terms of sizing and thermodynamic cycle parameters using a single design criterion. The respective trade-offs that the designer must accept are between mission fuel economy, payload–range capacity as well as the environmental impact.

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