ERF2015_0106

Conceptual Design and Optimisation of an Advanced Rotorcraft Powerplant Architecture

Fakhre Ali[†], Konstantinos Tzanidakis, Ioannis Goulos, Vassilios Pachidis
Propulsion Engineering Centre, School of Aerospace, Transport and Manufacturing,
Cranfield University, Cranfield, Bedford, MK43 0AL, United Kingdom.
[†]Corresponding author, Email: f.ali@cranfield.ac.uk.

Roberto d'Ippolito
Research & Innovation Manager
NOESIS Solutions
Gaston Geenslaan, 11
B4, 3001 Leuven, BE.

ABSTRACT
This paper demonstrates the application of an integrated rotorcraft multidisciplinary design and optimisation framework, deployed for the purpose of preliminary design and assessment of optimum regenerative powerplant configurations for rotorcraft. The proposed approach comprises a wide-range of individual modelling 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 various gas turbine gaseous emissions. A Single-Objective Particle Swarm Optimizer is coupled with the aforementioned rotorcraft multidisciplinary design framework. The overall methodology is deployed for the design space exploration and optimisation of a reference multipurpose twin-engine light civil rotorcraft, modelled after the Bo105 helicopter, employing two Rolls Royce Allison 250-C20B turboshaft engines. Through the implementation of single-objective optimisation, notionally based optimum regenerative engine design configurations are acquired in terms of engine weight, mission fuel burn and mission gaseous emissions inventory, at constant technology level. The acquired optimum engine configurations are subsequently deployed for the design of conceptual regenerative rotorcraft configurations, targeting improved mission fuel economy, enhanced payload range capability as well as improvements in the rotorcraft overall environmental footprint, while maintaining the required airworthiness requirements. The proposed approach essentially constitutes an enabler in terms of focusing the multidisciplinary design of conceptual rotorcraft powerplants to realistic, three-dimensional operations and towards the realization of their associated engine design trade-offs at mission level.

NOMENCLATURE

T_{out}	Heat exchanger outlet temperature, K
T_{Comp}	Compressor delivery temperature, K
$T_{Exhaust}$	Exhaust air temperature, K
FF_{mean}	Mean fuel flow, kg/sec

Greek Symbols

ΔAUM_i	Delta initial all-up-mass, kg
ΔCO_2	Delta carbon dioxide, kg
$\Delta Fuel\ burn$	Delta mission fuel burn, kg
ΔNO_x	Delta nitrogen oxides, kg
$\dot{W}$	Engine mass flow, kg/sec
$\Delta Weight$	Delta weight, kg

Acronyms

ACARE	Advisory Council for Aeronautics Research in Europe
DOE	Design Of Experiment
FB	Fuel Burn, kg
FF	Fuel Flow, kg/sec
FPT	Free Power Turbine
HEE	Heat Exchanger Effectiveness
HECTOR	HeliCopTer Omni-disciplinary Research-platform
HPC	High Pressure Compressor
LHS	Latin Hypercube Sampling
LPC	Low Pressure Compressor
mPSO	Multi-objective Particle Swarm Optimizer
OPR	Overall Pressure Ratio
PATM	Passenger Air Taxi Mission
PR	Pressure Ratio
RSMs	Response Surface Models
SFC	Specific Fuel Consumption, μg/J
SAR	Specific Air Range, km/kg of fuel
sPSO	Single-Objective Particle Swarm Optimizer
TEL	Twin Engine Light
TEM	Twin Engine Medium
WEBA	Whole Engine Based Approach

1. INTRODUCTION
Since the invention of helicopters, the necessity of required power has been fundamental towards the fulfillment of their operation. Reciprocating engines were the prime choice for power supply in early helicopters. The design requirements for the qualification of these engines were purely based around being able to safely generate the required power. However, as the

helicopter design has evolved over the second half of the 20^{th} century, the requirements to support helicopter operations presented enormous challenges, in order to sustain reliable, safe and robust performance of helicopters. The design incentives over the 20^{th} century were predominantly embraced by the quest to attain higher power to weight ratios, driven by the inherently infeasible design of early reciprocating engines. The introduction of the gas turbines engines in helicopters, provided an unprecedented breakthrough in both weight and size compared to reciprocating engines, yet their transition was accompanied with a trade-off for engine SFC [1], as the advantage of specific fuel consumption still remains with the reciprocating engines.

The state-of-the-art helicopter gas turbine powerplants have exceptionally transformed since the time of their introduction, with a track of remarkable contribution towards the operational portfolio of helicopters [2]. It is however now generally accepted that any further development in the helicopter gas turbine technology, without the consideration to exploit innovative and novel designs may result in diminishing improvement in SFC, as the design philosophies associated with the conventional designs are reaching their saturation. Furthermore, the aviation industry of 21^{st} century has set new standards and challenges that must be complemented by the aero-engines of next generation, in order to satisfy their marketability as well as compliance with the associated legislations. Among other important imperatives of future commercial aviation, the most politically and publically intensified imperative is associated with their contribution towards the environmental degradation, predominantly caused through the gaseous emissions emitted through the combustion of fossil fuel in aero-engines.

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) [3].

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 [3]. 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 [4]. 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 [5] 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 [3]. 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, as elaborated in [6]. 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.

1.2 Rotorcraft mission profile management and trajectory optimisation

To address option (c) several initiatives are underway specifically in Europe under the Seventh Framework Programme of the European Community. Aircraft flight trajectory optimisation 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 [7]. Their work was focused on the simulation and multidisciplinary optimisation 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. Also, Lawson et al deployed a multidisciplinary optimisation framework for minimum rotorcraft fuel burn and air pollutants at mission level [8]. Their work included single and multi-objective optimisations for mission block fuel burn, CO, UHC and NO_x emissions. Their acquired single objective optimisation results based on a generic representative mission profile suggested a reduction in mission block fuel burn of the order of 3.35% in exchange for a 2% trade-off in mission block NO_x emissions. Their multi-objective optimisation studies also suggested a reduction in mission block fuel burn of the order of 2% and up to 4.7% decrease in mission time, and CO and UHC emissions, followed by a negligible increase of 0.1 % in mission block NO_x emissions.

1.3 Rotorcraft-engine design point cycle parameters optimization

With regards to option (b) as elaborated by Goulos et al in [9], the overall approach can effectively be subcategorized within two major sectors of aerospace related research; airframe–rotor design, and engine cycle optimisation. With respect to the latter approach related to rotorcraft applications, Goulos et al [9], proposed a methodology with the potential to reduce fuel consumption associated with the civil rotorcraft operations at mission level, through optimisation 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 optimisation 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 optimisation 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.

1.4 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 engines. Considering unprecedented improvements in engine fuel efficiency, the most promising candidate is the advanced regenerative turboshaft concept. Rosen elaborated in [2], “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 Saravanamouttoo in [10], 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”. Ali et al in [11], conducted a design feasibility study of a sub-optimum conceptual regenerative engine 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.

Also Ali et al [12] conducted a comprehensive preliminary trade-off study through the application of a multidisciplinary design framework. Their study was based on an existing TEL multipurpose helicopter configuration employing conventional engines and sub-optimum conceptual regenerative engines. Their results acquired through the implementation of a representative case study suggested that, the sub-optimum regenerative engine design with a heat Exchanger effectiveness (HEE) of 60%, demonstrated the potential to offer a 34% reduction in mission fuel burnt and consequently a 34% reduction in mission CO_2 inventory, however, resulted in almost two times higher mission NO_x inventory compared to reference simple cycle engine. Their study concluded that, “conceptual regeneration configuration has the potential to significantly improve the CO_2 emissions through the reduction in mission fuel burn, however it may have a detrimental effect on the mission emissions inventory level, specifically for NO_x (Nitrogen Oxides), imposing a trade-off between the fuel economy and environmental performance of the helicopter”. Highlighting the requirement to expand the analytical and design effort of conceptual regenerative engines; to develop integrated optimised solutions that can minimise the trade-off between the rotorcraft operational performance and its environmental impact.

1.5 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 gaseous emissions 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 multidisciplinary design and optimisation framework, targeting the preliminary design of an optimum conceptual regenerative engine design in terms of minimum mission fuel burn, minimum mission NO_x inventory and minimum engine weight. A generic rotorcraft model, representative of a modern civil TEL rotorcraft has been investigated, operating under a representative Passenger Air Taxi Mission (PATM). The design space corresponding to the conceptual regenerative engine thermodynamic cycle parameters as well as engine and mission design outputs in terms of Low Pressure Compressor, High Pressure Compressor, Turbine Entry Temperature, Mass flow, Heat Exchanger Effectiveness (LPC, HPC, TET, W, HEE), engine design point SFC, engine weight, mission fuel burn and mission NO_x emissions inventory has been thoroughly investigated through the application of a Latin Hypercube Sampling (LHS) Design Of Experiment (DOE) approach.

The interdependencies between the various engine designs inputs/outputs are quantified by establishing the corresponding linear correlations between engine inputs/outputs (LPC, HPC, TET, $\dot{W}$, HEE, DP SFC,DP Shaft Power, Engine weight) as well as for the corresponding mission outputs (Fuel burn, and NO_x inventory). A single-objective Particle Swarm Optimizer (sPSO) is employed to derive optimum regenerative engine configurations that correspond to minimum mission fuel burn, minimum mission NO_x inventory and for minimum engine weight.

The acquired optimum regenerative engine configurations through the single-objective optimisation, are subsequently deployed for the design of conceptual rotorcraft engine configurations, 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.

2. SIMULATION METHODOLOGY

2.1 Framework numerical integration and formulation

This study requires the deployment of a multidisciplinary rotorcraft simulation framework, coupled with effective optimisation algorithms in order to allow for efficient design space exploration. The modelling 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 [13, 14], a flight path profile analysis based on the World Geodetic System dated in 1984 (WGS 84) [15], a non-linear trim procedure solving for the aeroelastic behaviour of the main rotor blades as described in [16, 17], an engine performance analysis model and gas turbine emissions model as detailed in [18, 19]. Each of the aforementioned modelling 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 [13, 20], therefore only a brief description of the associated models is provided in this paper. Architectural representation of the integrated rotorcraft multidisciplinary design and optimisation framework deployed in within this study is presented in Fig.1 respectively.

2.2 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 [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 [21, 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.

2.3 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 optimisation studies for example in [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 [12, 24]. Thus, further elaboration shall be omitted.

2.4 Recuperated 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 Heat Exchanger (HE) (shown in Fig.2); 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 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.

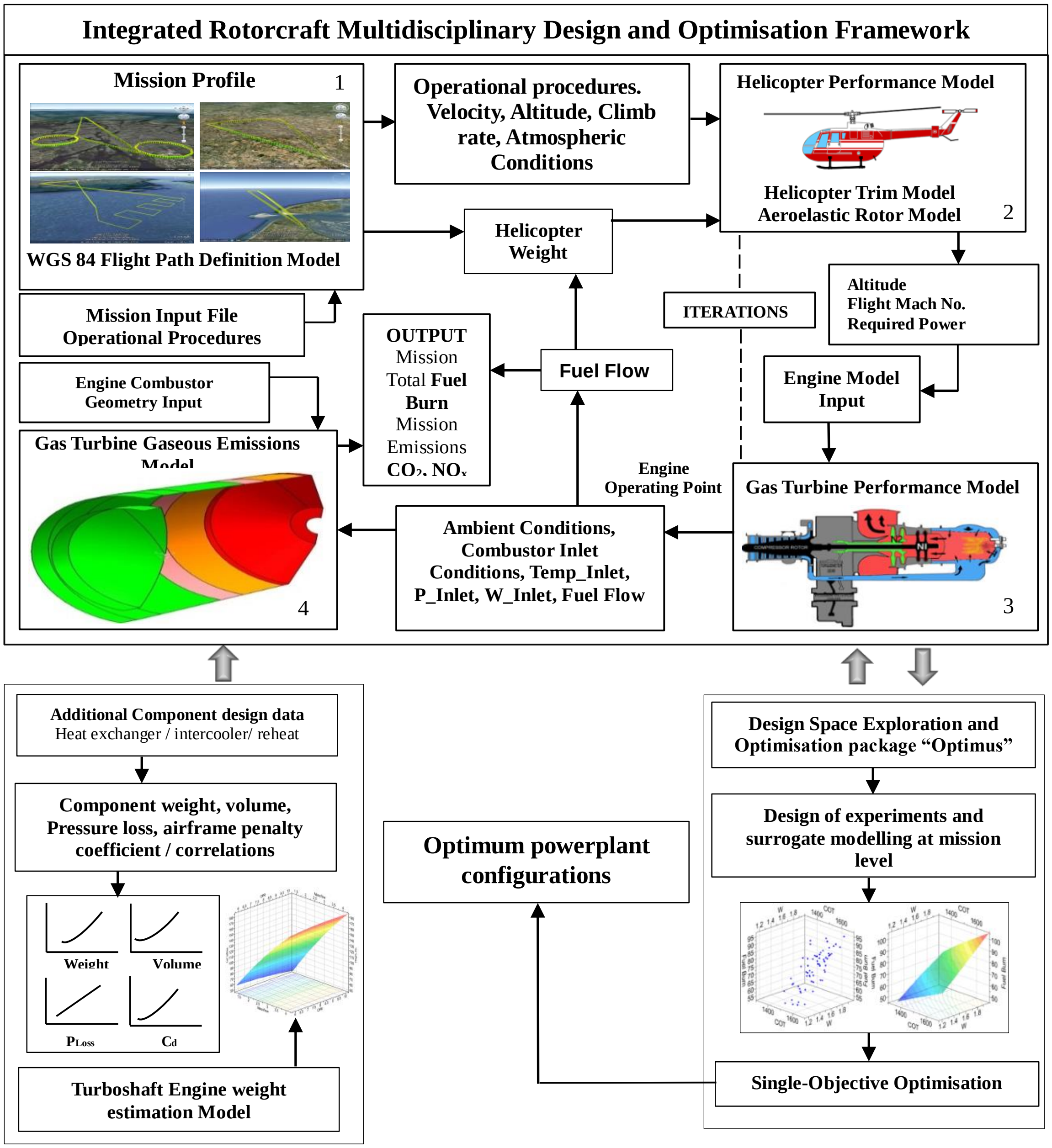

Figure 1: Architecture of integrated rotorcraft multidisciplinary design and optimisation framework; design and analysis of conceptual rotorcraft powerplant configurations.

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 Fig. 2 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.

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 HE weight. Variations in engine weight due to changes in engine design parameters during the optimisation 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.

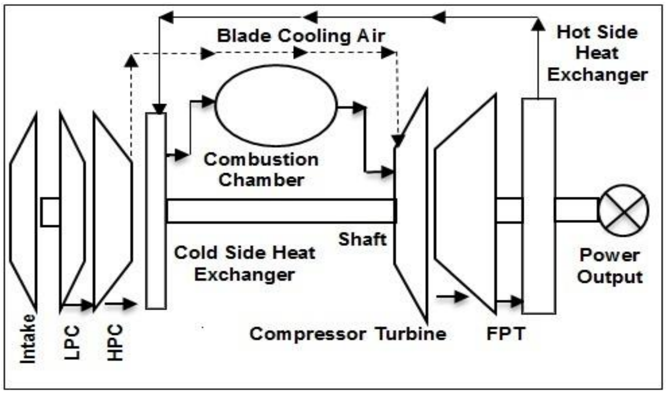

Figure 2: Schematic layout of a two-shaft regenerated turboshaft.

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

2.5 Heat exchanger and engine weight estimation

The HE weight correlation utilized for the purposes of this study is adopted from [25]. The correlation is presented in Fig. 3. 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 optimisation 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 [26, 27], 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 [28]. 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 modelled and simulated in TURBOMATCH according to its component technology level in terms of polytropic efficiency. For this purpose the trends reported in [29] were utilised. To maintain consistency, all engines were modelled at maximum contingency power and their acquired SFC was benchmarked against the readily available data collected from [28].

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 overall methodology and procedure followed for the development of the engine weight estimation model, validation and its integration in HECTOR is presented in Fig. 4 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.

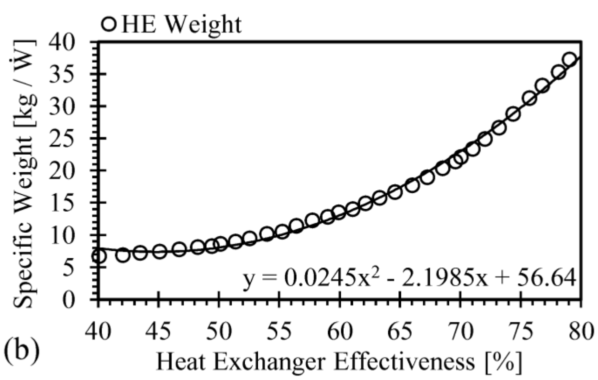

Figure 3: Fixed geometry tubular type heat exchanger specific weight correlation adopted from [25] integrated in HECTOR.

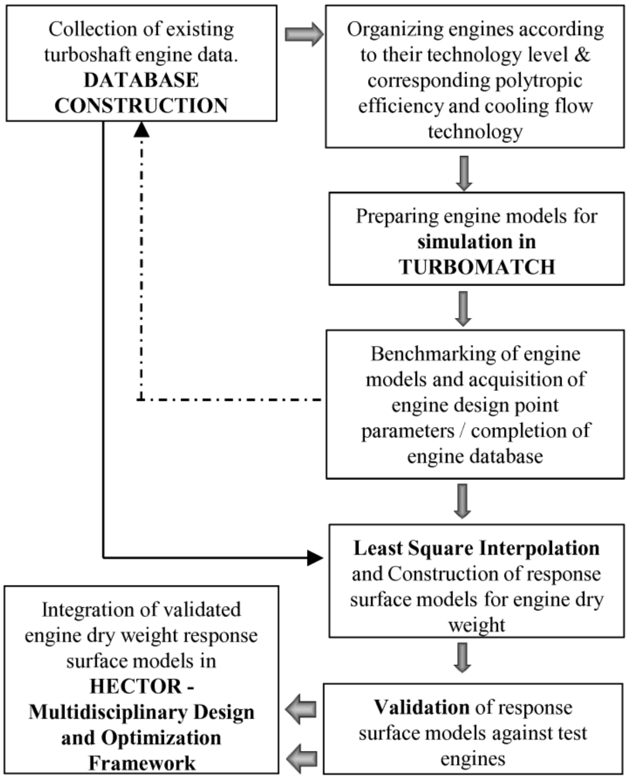

Figure 4: The analysis methodology scheme for; preliminary turboshaft engine dry weight estimation.

Table 1 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 1: Design input/output linear correlation coefficients: Engine cycle design parameters, engine dry weight.

Design input/output	Engine Dry Weight
TET	0.124
$\dot{W}$	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 Fig. 5 (a) and (b). The TET varies from 1000 K – 1470K and the pressure ratio from 6:1 – 10.5:1.

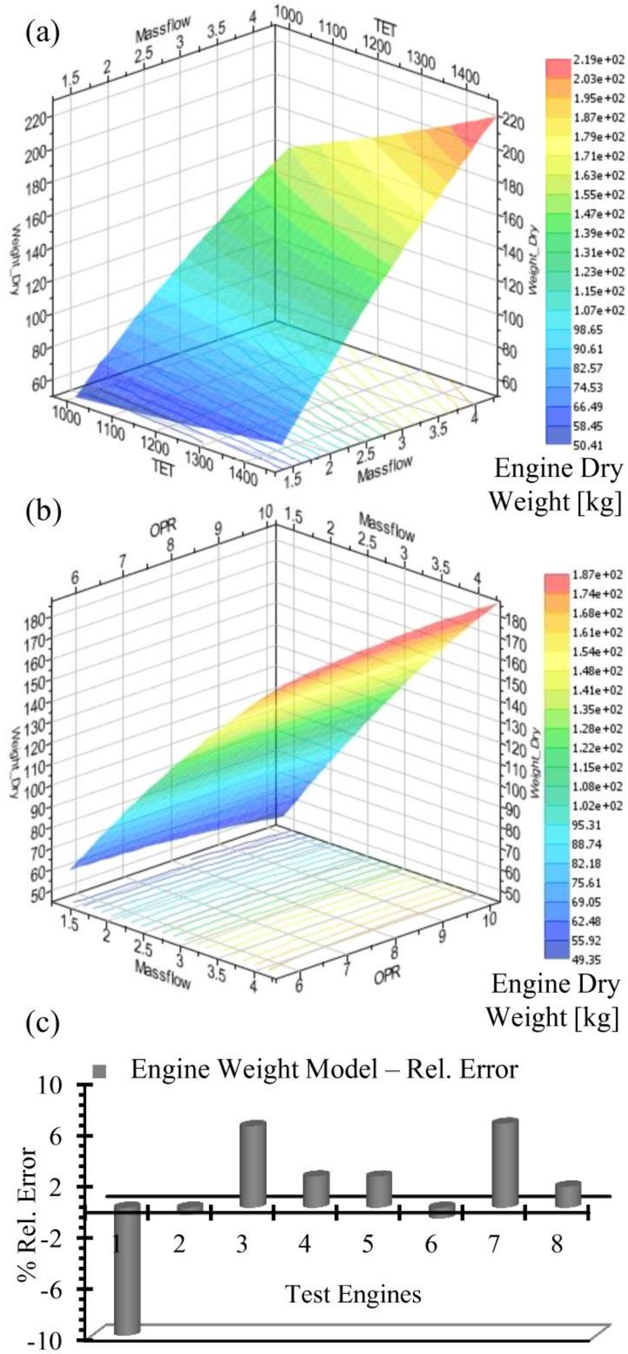

Figure 5: (a) Engine dry weight response surface model; engine mass flow and turbine entry temperature; (b) Engine dry weight response surface model; engine mass flow and overall pressure ratio, (c) Engine dry weight response surface model prediction; relative error for the test engines

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 5(c) 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 acceptable 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 optimisation process.

3. OPTIMISATION 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.

3.1 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 [30], 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-modelling has been proven to be among the most promising approaches for the approximation of highly non-linear problems [31, 32] 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" [33].

3.2 Optimisation 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 [34] 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.

In this work a novel single-objective (sPSO), global optimizer is employed in order to deal with the complexities mentioned above. In [35] the authors have elaborated on the advantages and disadvantages of multiple state-of-the-art optimisation 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 PSO used in this study can be found in [35]. In this work, the PSO was deployed for single-objective optimisation studies respectively.

3.3 Compilation of helicopter and engine configuration

The aircraft deployed for the purpose of this study is modelled after the Airbus-Helicopters Bo105 helicopter. The Airbus-Helicopters Bo105 is a TEL utility multipurpose helicopter equipped with two Rolls Royce Allison 250-C20B turboshaft engines rated at 313 kW maximum contingency power. Table 2 presents the helicopter model characteristics.

The Allison 250-C20B 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 [28] in terms of SFC. The configuration of the Bo105 as well as its performance characteristics have been extensively documented and analyzed in [36] thus, further elaboration shall be omitted. A detailed description of the Allison 250C20B engine family can be found in [28].

Table 2. a) Baseline Bo105 helicopter characteristics, b) baseline engine design point parameters, c) mission outputs parameters for PATM, simulated in HECTOR.

a) Bo105 helicopter characteristics		
Parameter	Value	Units
Max gross weight	2500.00	kg
OW	2200.00	kg
Number of blades	4.00	-
Blade radius	4.91	M
Blade twist	8.00	Degree
Rotor speed	44.40	rad/sec
b) Baseline engine design point parameters		
TET	1470.00	K
$\dot{W}$	1.56	kg/sec
LPC PR	2.73	-
HPC PR	2.60	-
DP shaft power	313.00	kW
DP SFC	109.98	μg/J
c) PATM parameters for Bo105 helicopter		
Time	1725.00	seconds
Range	36.22	km
Fuel Burn	59.99	kg
CO_2	191.92	kg
H_2O	74.60	kg
NO_x	0.287	kg

3.4 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 representations in terms of global coordinates along with the deployed operational procedures are illustrated in Figure 6 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.

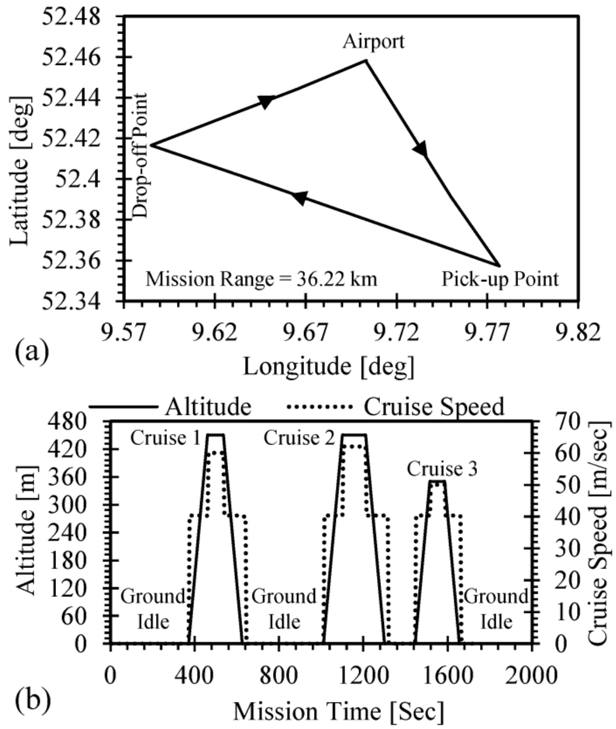

Figure 6: a) Reference PATM geographical definition; (b) time variations of deployed operational airspeed and altitude.

3.5 Design space definition

Having established the corresponding engine design and mission parameter values for the integrated Bo105 helicopter–engine systems (presented in Table 1(c)), 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 3 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 optimisation 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.

The maximum allowable DP TET is limited to the baseline value as presented in Tables 2. Thus, although the impact of a higher technology level in terms of maximum allowable TET has been accounted for throughout the DOE method (lower and upper bounds for TET are shown in Table 3), the aforementioned effect is excluded from the optimisation 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 2. This constraint is applied so that the payload carrying capability of the reference helicopter is not changed during the optimisation process.

Table 3: 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	-
$\dot{W}$	1	2	kg
TET	1300	1600	K
HEE	40	80	%
Pressure loss	5	9	%

4. RESULTS AND DISCUSSION

4.1 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 hundred-twenty simulations have been performed for the reference Bo105 model within the HECTOR framework, presented in Fig.1. 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. 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.

The linear correlation coefficients in terms of the systems response are presented in Table 4 corresponding to the regenerated Bo105 helicopter. The correlations suggest that HEE favours the reduction in overall mission fuel burn, however it has a detrimental effect on NO_x. 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.Theses interdependencies between the engine design parameters against engine and mission output parameters are also shown in Fig.7, in the form of a simple sensitivity analysis, established based on the acquired linear correlations presented in Table 4. It is to be highlighted that, although the sensitivity analysis suggest a linear relationship between the various engine design input parameters against the engine and mission output parameters. These correlations may well not be linear when analyzed explicitly for a specified application. However, for the problem at hand the interdependencies presented in Fig.7, serve as a fair basis for making a sound judgment about the level of influence some of the key engine parameters have on its design outputs as well as on the mission output parameters.

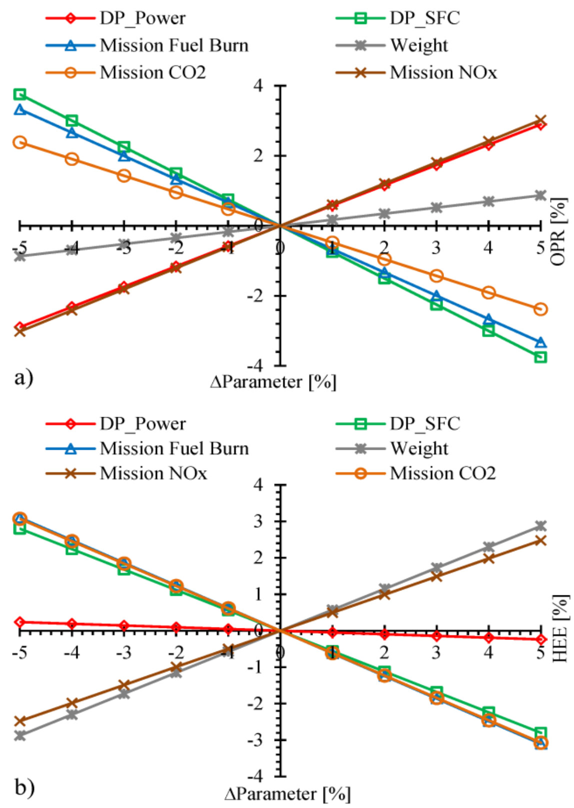

Figure 7: (a) Sensitivity analysis for various engine and mission output parameters against engine OPR; (b) Sensitivity analysis for various engine and mission output parameters against heat exchanger effectiveness.

Table 4: Regenerated engine cycle design input/output linear correlation coefficients: reference Bo105 helicopter/ passenger mission.

Input/output	LPC PR	HPC PR	OPR	TET_{DP}	$\dot{W}$	HEE	SFC_{DP}	P_{DP}	MFB	CO_2	NO_x	EW
LPC_PR	**1.000**	0.027	0.669	0.037	-0.023	-0.002	-0.547	0.669	-0.476	-0.477	0.388	0.045
HPC_PR	0.027	**1.000**	0.743	0.037	0.070	-0.003	-0.510	0.743	-0.448	-0.448	0.432	0.092
OPR	0.669	0.743	**1.000**	0.045	0.015	0.008	-0.751	0.579	-0.665	-0.666	0.604	0.174
TET_{DP}	0.037	0.057	0.045	**1.000**	-0.210	-0.050	-0.231	0.431	-0.212	-0.197	-0.197	0.021
$\dot{W}$	-0.023	0.070	0.015	-0.045	**1.000**	0.010	0.003	0.630	0.303	0.312	-0.437	0.691
HEE	-0.002	-0.003	0.008	-0.050	0.010	**1.000**	-0.560	-0.048	-0.621	-0.614	0.496	0.575
SFC_{DP}	-0.547	-0.510	-0.751	-0.231	0.003	-0.560	**1.000**	-0.514	0.914	0.913	-0.675	-0.397
P_{DP}	0.372	0.447	0.579	0.431	0.630	-0.048	-0.514	**1.000**	-0.175	-0.712	-0.054	0.496
MFB	-0.476	-0.448	-0.665	-0.042	0.303	-0.621	0.914	-0.175	**1.000**	1.000	-0.820	-0.187
CO_2	-0.477	-0.448	-0.666	-0.046	0.312	-0.614	0.913	-0.172	1.000	**1.000**	-0.821	-0.177
NO_x	0.388	0.432	0.604	-0.114	-0.437	0.496	-0.675	-0.540	-0.820	-0.821	**1.000**	-0.005
EW	-0.013	0.104	0.066	0.021	0.691	0.575	-0.397	0.496	-0.187	-0.177	-0.005	**1.000**

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, $\dot{W}$, TET, PR, HEE) and outputs, DP shaft power, DP SFC, mission fuel burn, engine weight and mission emission inventory of NO_x. The developed RSMs are subsequently used as drivers throughout the optimisation process, presented in this paper. The corresponding response surface models for some of the engine design and mission output parameters are presented in Figs. 8 -10, respectively. The RSMs presented in Fig.8, shows that the increase in engine LPC and HPC results in reducing the engine SFC, however, has a detrimental effect on mission NO_x inventory. This can also be seen in the sensitivity analysis presented in Fig. 7, for the engine OPR against various engine and mission output parameters. The acquired trends show that an increase in engine OPR results in increased engine DP power and lower SFC; the reduction in the latter parameter results in reducing the mission fuel burn and CO_2 emissions. However, the benefit in mission fuel burn through reduction in engine DP SFC (through increase in engine OPR) is attained with a penalty in mission NO_x emissions and a moderate increase in engine weight.

In Figure 7 similar parametric sensitivities are presented for the on-board heat exchanger effectiveness, again it can be established that, as the on-board heat exchanger effectiveness is increased, the engine thermal efficiency improves, therefore the engine SFC, mission fuel burn and mission CO_2 emissions are reduced, however, improvements in aforementioned parameters are attained with a penalty in mission NO_x emissions as well as in engine weight, as shown in Fig. 7 the engine weight and NO_x emissions increases with increase in HE effectiveness.

It can also be established from the results presented in Figs. 7-10 that, for a constant technology regenerative engine employing a conventional technology combustion system, the engine fuel efficiency is attained with a trade-off for engine NO_x emissions. In terms of engine design; this means that designing an engine to achieve minimum mission fuel burn will require an engine that corresponds to maximum attainable thermal efficiency, under the imposed design criterion. However, on the other hand, where minimization of mission NO_x inventory is an engine design objective; this will require an engine design that corresponds to lowest possible thermal efficiency. These conflicting design challenges require the implementation of a robust multi-objective design strategy in order to systematically reach a balanced compromise between the design objective functions. As in terms of the civil rotorcraft application both of the aforementioned engine design objectives present strong influence on the rotorcraft operational and environmental performance.

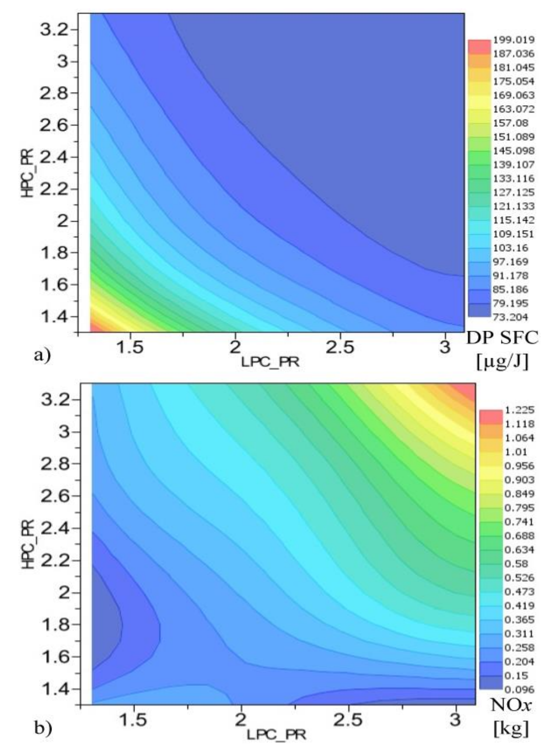

Figure 8: (a) RSM for engine SFC versus engine LPC and HPC; (b) RSM for mission NO_x versus engine LPC and HPC.

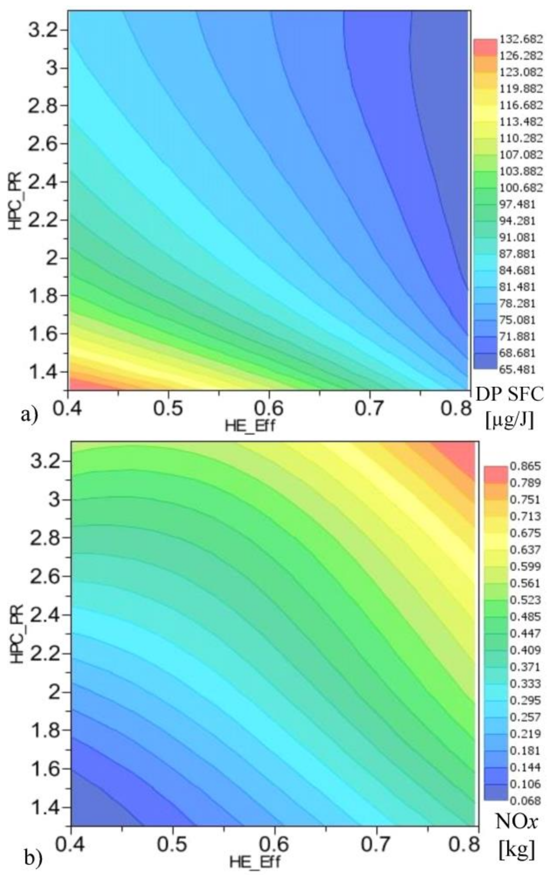

Figure 9: (a) RSM for engine SFC versus engine HPC and heat exchanger effectiveness; (b) RSM for mission NO_x versus engine HPC and heat exchanger effectiveness.

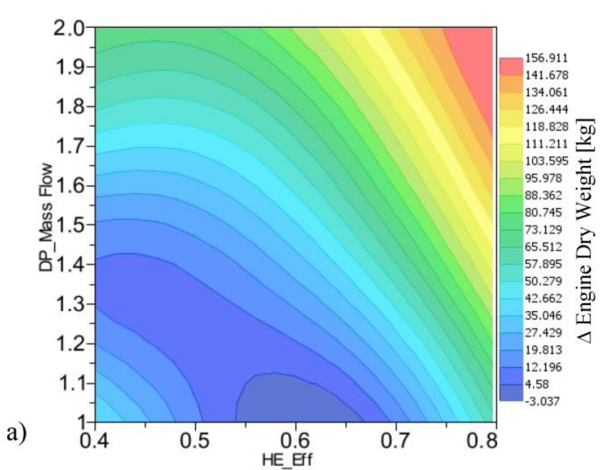

Figure 10: RSM for Δengine dry weight versus engine DP mass flow and heat exchanger effectiveness.

4.2 Single-Objective Optimisations

Having established the design space and the associated numerical formulation of RSMs, three respective single-objective optimisations were 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 optimisation purpose the sPSO algorithm was deployed, since it was already established from previous study that PSO is a strong candidate amongst other techniques e.g. SAE [7, 9, 35].

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 5 presents the bounds for engine thermodynamic design parameter inputs and design constraints applied for both aforementioned single-objective optimisations.

In order to check the reliability of the constructed RSMs, separate HECTOR simulations were performed for all three acquired optimum configurations. Table 6 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 7 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.

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 increasing the mission fuel burn by around 4.62%.

Finally, 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.

An interesting observation can be made by comparing the optimised 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 optimisation was performed 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. The results presented in Fig.11 shows a comparison between the engine design cycle parameters corresponding to respective acquired optimum configurations.

Table 5: Bounds for DP engine size and thermodynamic cycle- Single-objective optimisation.

Design Parameter	Low Bound	High Bound	Units
LPCPR	1.3	3.1	-
HPCPR	1.3	3.3	-
$\dot{W}$	1	2	kg/sec
TET	1470		K
HEE	40	80	%
Constraints for single-objective optimisation			
DP Power	313000		W

Table 6. RSM relative error for all three single-objective based optimized configurations: Bo105 helicopter/passenger mission.

RSM relative error for minimum mission fuel burn				
Configuration	RSM	HECTOR	Units	RSM rel.error %
Baseline	58.80	59.99	kg	-1.98
Optimized	29.10	29.25	kg	-0.53
Reduction	-50.52	-51.24	%	Avg rel. error 1.25
RSM relative error for minimum mission NO_x inventory				
Baseline	0.282	0.287	kg	-1.74
Optimized	0.116	0.119	kg	-2.44
Reduction	-58.83	-58.54	%	Avg rel. error 2.09
RSM relative error for minimum engine weight				
Baseline	58.8	59.99	kg	-1.98
Optimized	43.8	44.16	kg	-0.82
Reduction	-25.83	-26.39	%	Avg rel. error -1.39

Table 7. Comparison between baseline engine cycle design parameters against the acquired optimized minimum fuel burn, NO_x and engine weight configurations; Bo105 helicopter/PAT mission.

Design parameter	Baseline	Min Fuel	Min NO_x	Min Weight	Units	Min Fuel Rel. %	Min NO_x Rel. %	Min Weight Rel. %
LPC_PR	2.73	2.31	2.20	2.34	-	-15.54	-19.43	-14.11
HPC_PR	2.60	2.76	1.67	2.33	-	6.30	-35.76	-10.57
OPR	7.09	6.37	3.67	5.45	-	-10.21	-48.24	-23.19
TET	1470.00	1470.00	1470.00	1470.00	K	0.00	0.00	0.00
$\dot{W}$	1.56	1.30	1.64	1.32	kg/sec	-16.53	5.17	-15.21
HEE	0.00	80.00	40.00	40.00	%	80.00	40.00	40.00
Engine and mission output parameters								
Engine Weight	144.00	244.00	168.00	144.00	kg	69.44	16.67	0.00
Fuel Burn	59.99	29.10	62.76	43.80	kg	-51.50	4.62	-26.99
NO_x	0.287	0.870	0.116	0.32	kg	202.44	-59.55	11.50

It can be established from the results shown that, the optimised 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%.

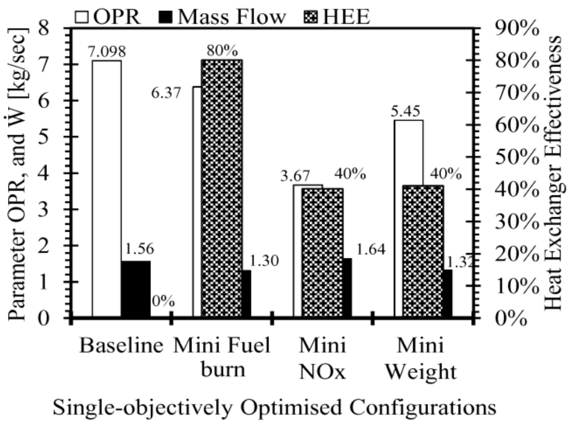

Figure 11: Comparison between the optimised engine design cycle parameters; single-objective optimisation results TEL Bo105 helicopter, passenger Air Taxi Mission.

4.3 Optimised 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 optimised 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 Initial All-Up-Mass (AUM_i), payload, range and environmental impact are of prime importance, specifically when designing an engine for civil rotorcraft e.g. executive travel. Since the focus of this study is dedicated towards the multidisciplinary design and assessment of a 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 8 presents the key parameters associated with the aforementioned imposed design criterion; Specific Air Range (SAR), AUM_i and the NO_x inventory deltas, established for all three acquired optimised configurations, with respect to the baseline. It is evident from Fig. 12 that, the operational benefits offered by optimised minimum 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 change in engine design, and increases the mission NO_x inventory by only 11%, with respect to the baseline configuration.

Figures 13 show the fuel flow and NO_x rate against mission time for all three acquired optimised 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.

Table 8: Comparison between baseline and optimum engine configurations, mission level parameters and deltas; Bo105 helicopter/passenger mission**.**

Parameter	Baseline	Mini FB	Mini NO_x	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_i	-	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	%

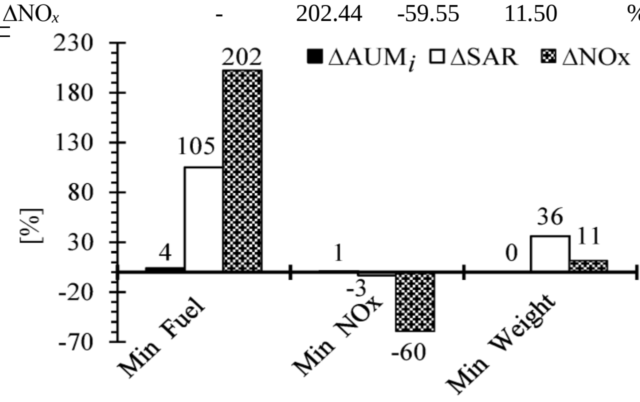

Figure 12: Single-objective results; mission level parameters and deltas; Bo105 helicopter/passenger mission.

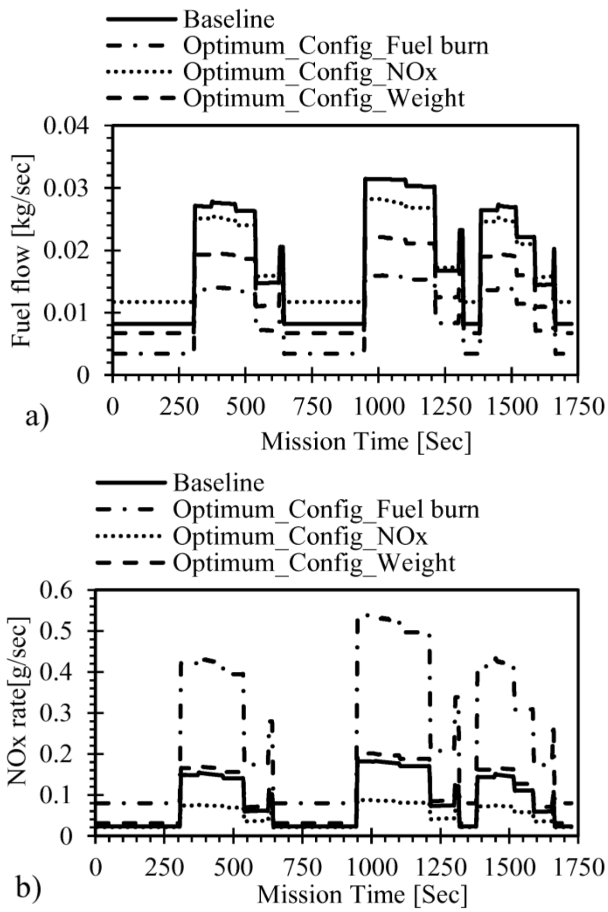

Figure 13: a) Fuel flow and b) NO_x emissions production rate; comparison between baseline and optimally designed engine configurations, Reference Bo105/passenger mission.

It can be therefore established from the acquired single-objective optimisation results that, 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.

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.

5. CONCLUSIONS

An innovative multidisciplinary design and optimisation 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 rotorcraft–engine system within any designated operation has been deployed.

A comprehensive and computationally efficient optimisation strategy, utilizing a Particle-Swarm Optimizer, has been implemented. The overall methodology has been applied to the multidisciplinary design and optimisation of a reference twin-engine light civil rotorcraft modelled after the Airbus-Helicopters Bo105 helicopter, operated on a representative mission scenario. Through the application of a single-objective Particle-Swarm Optimizer, notionally based engine design specifications, optimised in terms of mission fuel burn, engine weight and NO_x emissions have been acquired at constant technology level. It has been established, through a comprehensive comparison between the acquired optimisation 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. It has been demonstrated that the aforementioned optimised regenerative configuration in terms of engine weight has the potential to increase the reference rotorcraft specific range capability by 36.02% (at mission cruise conditions), has no associated weight penalty due to change in engine design, however, has a mission NO_x inventory penalty of the order of 11%, with respect to the baseline configuration.

Finally, it has been 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.

ACKNOWLEDGEMENTS

The authors would like to acknowledge Professor Pericles Pilidis, Dr Vishal Sethi, Dr Hugo Pervier, Dr Tashfeen Mahmood and Mr Atma Prakash from the Propulsion Centre of Cranfield University, for their insightful advice and continuing support.

REFERENCES

[1]. Sample, R. D., "Research requirements for development of regenerative engines for helicopters" NASA technical report, NASA CR-145112.

[2]. Rosen, K. M., "A Prospective, The Importance of Propulsion Technology to the Development of Helicopter Systems with a Vision for the Future" The 27th Alexander A. Nikolsky Lecture, Journal of the American Helicopter Society.

[3]. Stevens, J., Roberto d'Ippolito., Pachidis, V., Berta, A., Goulos, I., and Rizzi, C., 2012, "A Multidisciplinary Simulation Framework for Optimization of Rotorcraft Operations and Environmental Impact," Second International Conference on Engineering Optimization.

[4]. Advisory Council for Aeronautics Research in Europe Report, "Strategic Research Agenda" Volume 1, October

2002. Available at: http://ec.europa.eu/research/transport /pdf/acare_strategic _research _en.pdf.

[5]. Clarke, J. P., "The Role of Advanced Air Traffic Management in Reducing the Impact of Aircraft Noise and Enabling Aviation Growth," J. Air Transp. Manag., 9, pp. 161–165.

[6]. Brooker, P., "Civil Aircraft Design Priorities: Air Quality? Climate Change? Noise?," Aeronaut. J., 110(1110), pp. 517–532.

[7]. Goulos, I., Pachidis, V., D'Ippolito, R., Stevens, J., and Smith, C., "An Integrated Approach for the Multidisciplinary Design of Optimum Rotorcraft Operations," ASME J. Eng. Gas Turbines Power, 134(9), p. 091701.

[8]. Linares, C., Lawson, C. P. and Smith, H., "Multidisciplinary optimisation framework for minimum rotorcraft fuel and air pollutants at mission level" Aeronautical Journal, July 2013, Volume 117 No 1193.

[9]. Goulos, I., Hempert, F., Sethi, V., Pachidis, V., D'Ippolito, R and Massimo D'Auria., "Rotorcraft Engine Cycle Optimization at Mission Level," ASME J. Eng. Gas Turbines Power, 135(9), p. 091202-1.

[10]. Cohen, H., Rogers, G.F.C., Saravanamoutto, H.I.H., "Gas turbine theory" 4th edition. Padstow, Cornwall.

[11]. Ali, F., Pachidis, V., Goulos, I., Mahmood, T., and Pericles, P., "Helicopter mission analysis for a regenerated turboshaft engine". Proceedings of ASME Turbo Expo 2013: Power for Land, Sea and Air, June 03 - 07, 2013, San Antonio, Texas, USA.

[12]. Ali, F., Goulos, I., and Pachidis, V., "An Integrated Methodology to Assess the Operational and Environmental Performance of a Conceptual Regenerative Helicopter" The Aeronautical Journal, Manuscript Number: AeroJ-D-13-04065. Vol. 119, (1211), January 2015.

[13]. Goulos, I., 2012, "Simulation Framework Development for the Multidisciplinary Optimization of Rotorcraft," Ph.D. thesis, Cranfield University, Cranfield, Bedfordshire, UK.

[14]. Goulos, I., Pachidis, V., and Pilidis, P., "Lagrangian Formulation for the Rapid Estimation of Helicopter Rotor Blade Vibration Characteristics," Aeronautical Journal, Vol. 118, (1206), August 2014.

[15]. European Organization for the Safety of Air Navigation and Institute of Geodesy and Navigation, 1998, WGS 84 Implementation Manual, Eurocontrol, Brussels, Belgium.

[16]. Goulos, I., Pachidis, V., and Pilidis, P., "Flexible Rotor Blade Dynamics for Helicopter Aeromechanics Including Comparisons with Experimental Data," Aeronautical Journal, Vol. 118, (1210), December 2014.

[17]. Goulos, I., Pachidis, V., and Pilidis, P., "Helicopter Rotor Blade Flexibility Simulation for Aeroelasticity and Flight Dynamics Applications," J. Am. Helicopter Soc., Journal of the American Helicopter Society, Vol. 59, (4), October 2014.

[18]. Macmillan, W. L., 1974, "Development of a Module Type Computer Program for the Calculation of Gas Turbine Off Design Performance," Ph.D. thesis, Cranfield University, Cranfield, Bedfordshire, UK.

[19]. Celis, C., 2010, "Evaluation and Optimisation of Environmentally Friendly Aircraft Propulsion Systems," Ph.D. thesis, Cranfield University, Bedfordshire, UK.

[20]. Goulos, I., Giannakakis, P., Pachidis, V., And Pilidis, P., 2013. "Mission Performance Simulation of Integrated Helicopter–Engine Systems Using an Aeroelastic Rotor Model," ASME J.Eng. Gas Turbines Power,135(9), 091201-1.

[21]. Li, Y. G., Marinai, L., Gatto, E. L., Pachidis, V., and Pilidis, P., "Multiple-Point Adaptive Performance Simulation Tuned to Aero-engine Test-Bed Data," Journal of Propulsion and Power, Vol. 25(3), 2009, pp. 635–641.

[22]. Pachidis, V., Pilidis, P., Marinai, L., and Templalexis, I., 2007, "Towards a Full Two Dimensional Gas Turbine Performance Simulator," Aeronaut. J.,111(1121), pp. 433–442.

[23]. Celis, C., Long, R., Sethi, V., and Zammit Mangion, D., 2009, "On Trajectory Optimisation for Reducing the Impact of Commercial Aircraft Operations on the Environment", ISABE-2009 1118, 19th Conference of the International Society for Air Breathing Engines, Montréal, Canada.

[24]. Goulos, I., Ali. F., Tzanidakis, K., Pachidis, V., and d'Ippolito, R., "A Multidisciplinary Approach for the Comprehensive Assessment of Integrated Rotorcraft-Powerplant Systems," ASME J. Eng. Gas Turbines and Power, Vol. 137, (1), January 2015.

[25]. Nicolas C. Kailos (1967): Increased helicopter capability through advanced power plant technology Journal of the American Helicopter Society, Volume 12, Number 3, 1 July, pp. 1-15(15).

[26]. Clavier, J. (2008), Aero gas turbine engine design Project (AVIC). Ultra high by-pass ratio (12-14) study (MSc Thesis), School of Engineering, Cranfield University, Cranfield.

[27]. Svoboda, C. (2000), "Turbofan engine database as a preliminary design tool", Aircraft design 2000, 3, p.17-31.

[28]. JANE'S International Aero-engines, "Aircraft Engines of the World" Volume 20. Page 260-261.

[29]. Mattingly, J. D., Heiser, W.H., and PRATT, D.T., "Aircraft Engine Design" 2nd ed. Virginia: AIAA, 2002.

[30]. Iman, R. L. (2008). Latin hypercube sampling. John Wiley & Sons, Ltd.

[31]. Kleijnen, J. P. (2009). Kriging meta-modeling in simulation: a review. European Journal of Operational Research, 192(3), 707-716.

[32]. Wang, G. G., & SHAN, S. (2007). Review of meta-modeling techniques in support of engineering design optimization. Journal of Mechanical Design, 129, 370.

[33]. NOESIS Solutions, (2012), OPTIMUS REV 10.6 Manual, Leuven, Belgium.

[34]. Deb, K. (1999), Multi-objective genetic algorithms: Problem difficulties and construction of test problems, Evolutionary computation, vol. 7, no. 3, pp. 205-230.

[35]. Tzanidakis, K., Pachidis, V., D'Ippolito, R. and D'Auria, M. (2013). Optimisation in a multidisciplinary environment – a turbomachinery application, NAFEMS World Congress, 9-12 June, Salzburg, Austria.

[36]. STALEY, J. A., "Validation of Rotorcraft Flight Simulation Program through Correlation with Flight Data for Soft-in-plane Hingeless Rotor," USAAMRDL-TR-75-50, 1976.