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HELICOPTER ENGINE OPTIMIZATION FOR MINIMUM MISSION FUEL BURN

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Abstract

The paper presents an approach for optimizing the design point inlet mass flow rate and overall pressure ratio of a turboshaft engine in order to minimize fuel burn over a specific mission of a medium transport-utility helicopter engine.

The method employs performance models of the helicopter and associated turboshaft engines and is suitable for the preliminary design of a new engine or the re-design of an existing one.

It uses empirical correlations to account for changes in turbomachinery component efficiencies and engine/helicopter weight due to the change of inlet corrected mass flow from a reference value. The turbine cooling flows are adjusted according to the specified upper limit of turbine rotor inlet temperature. The surge margin must be within a specified value while pressure ratio changes must allow the re-introduction of cooling/sealing air flows back into the main flow.

Regarding the mission, the cruise altitude and total distance travelled are fixed while the velocity of best range during cruise and the velocity of best endurance and maximum rate of climb are recalculated based on the new helicopter weight due to changes in engine size and required mission fuel.

The total reduction in mission fuel burn depends on the limits set by the designer.

Nomenclature

E	isentropic efficiency
GBQ	gearbox ratio
H/C	helicopter
HP	high pressure
LP	low pressure
MTOW	maximum take-off weight
NGV	nozzle guide vanes
OPR	overall pressure ratio
P22Q2	LP compressor pressure ratio
P3Q24	HP compressor pressure ratio
PWSD	shaft power delivered
SFC	specific fuel consumption
SL	sea-level
STD	standard
SR	specific range
TOP	take-off power
Tt	total temperature
Vbe	velocity of best endurance
Vbr	velocity of best range
Vx	forward velocity
Vz_{max}	maximum rate of climb
W	mass flow rate
Wc	cooling flow
WF	engine fuel flow rate
WFB	mission fuel burn
W0	helicopter initial weight
XNH	gas generator rotational speed
Δ	difference from reference

Introduction

Aviation currently accounts for around just 2% of man-made CO_2 emissions[1]. However its contribution to total greenhouse gas emissions is higher (~3%) due to other exhaust

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gases emitted during flight as well as contrails.

Future emission levels from aviation will depend on the relative rates of growth and the scale of technological improvements. World-wide traffic is predicted to grow at a rate of 4-5% per year[2]. The CO_2 emissions by worldwide aviation in 2050 would be nearly six times their current level if fuel consumption grows at the same rate.

In awareness of the environmental consequences of continued CO_2 growth, IATA members have agreed in June 2009 to a set of ambitious goals:
• Carbon neutral growth of aviation from 2020
• Improve fuel efficiency by 1.5% the subsequent decade
• Reduce CO_2 emissions by 50% until 2050 compared with 2005 levels.

These targets are planned to be achieved using a four pillar strategy which includes improved technology, effective operations, efficient infrastructure and positive economic measures[3]. Of these four pillars, technology has the best prospect for reducing aviation emissions with advances in engine configurations, aircraft/rotorcraft designs and used materials while significant benefits will be achieved by the implementation of alternative fuels.

Although the helicopter operations sector has currently a relatively small share of the total aviation market, its role is continuously expanding to fulfill the needs of modern society to certain modes of transport (e.g. offshore), medical assistance (air ambulances), law enforcement, search and rescue, fire-fighting, etc. Hence, its future environmental impact would be significant if measures are not taken now to reduce greenhouse gas emissions over the entire mission range.

Previous studies in the public domain on helicopter operation for minimum mission fuel burn have concentrated in trajectory optimization of helicopters[4,5].

In this study, an approach to optimize a turboshaft engine for minimum mission fuel burn of a medium transport/utility helicopter is demonstrated by employing appropriate performance models of the helicopter and its engines. The models have been developed in a commercial simulation environment that allows transparent exchange of information between the models, provides common modelling standards and flexible mathematical model handling. The method is generic and fully configurable so it is well-suited for the preliminary design of a new engine or the re-design of an existing one.

Description of Models

The amount of fuel consumed by a helicopter during a mission may be evaluated by coupling an engine performance model for off-design analysis with a helicopter performance model so that the following sequence of calculations can be realized:
➢ define the required mission profile in terms of ambient and flight conditions
➢ determine shaft power requirements from helicopter performance model and for the current helicopter weight and mission point
➢ calculate the fuel consumption corresponding to the environmental conditions and engine throttle setting of the current mission point from the engine performance model
➢ update the helicopter weight and calculate the next mission point
➢ sum the fuel consumed at each point to obtain the mission block fuel burn

In this study, both the helicopter and engine performance models are developed in the PROOSIS v3.0 simulation environment[6]. This is a

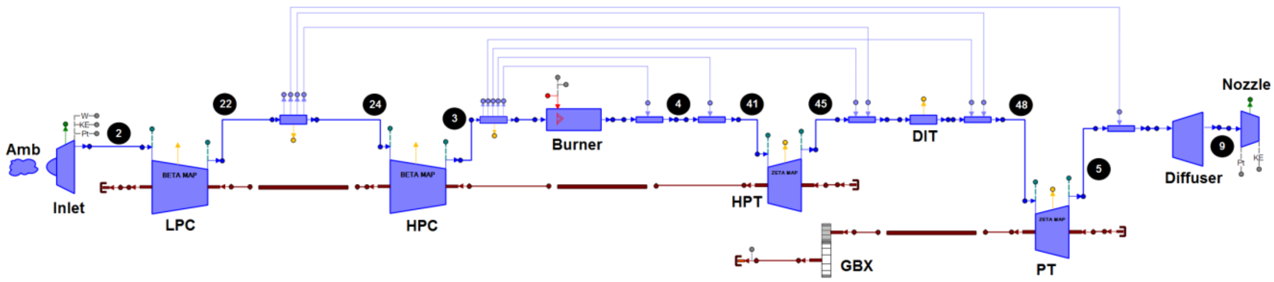

Figure 1: Turboshaft engine PROOSIS schematic diagram & station numbering

stand-alone, flexible and extensible object-oriented tool capable of performing steady-state and transient calculations as well as multi-fidelity, multi-disciplinary and distributed gas turbine engine performance simulations[7]. Different calculation types can be carried out such as mono or multi-point design, off-design, test analysis, sensitivity, optimization, deck generation, etc. It features an advanced graphical user interface allowing for modular model building by 'dragging-and-dropping' the required component icons from one or more library palettes to a schematic window. A component icon, for example, could represent a single engine component (e.g., compressor, turbine, burner, nozzle, etc.), a sub-assembly, a complete engine, a control system, an aircraft model, etc. Components communicate with each other through their ports. Ports define the set of variables to be interchanged between connected components (e.g., mass flow rate, pressure, and temperature in a Fluid port or rotational speed, torque and inertia in a Mechanical port, etc.). The mathematical modelling of components and ports is described with a high-level object-oriented language.

Engine Model

For the work reported here, the TURBO library of engine components available as standard in PROOSIS is used to create the free turbine turboshaft engine model shown in Fig. 1. The library uses industry accepted performance modelling techniques and respects the international standards with regards to nomenclature, interface and object-oriented programming[8].

The engine model has a gas generator consisting of a twin stage centrifugal compressor (LPC and HPC) driven by a single stage axial turbine (HPT). The free power turbine (PT) is a twin axial turbine delivering shaft power through a gearbox (GBX). The model uses appropriate maps to define off-design performance for the turbomachinery components. The burner pressure losses vary with the burner inlet corrected mass flow rate while burner efficiency is a function of burner loading. Inter-turbine duct (DIT) and Diffuser pressure losses vary with inlet swirl angle while the efficiency of the Inlet depends on the inlet mass flow rate. Cooling/sealing flows for the HPT and PT components are extracted from the exit of LPC and HPC as required. Shaft and gearbox transmission losses are also accounted for. Inlet and exhaust (Nozzle) pressure losses, customer power and bleed air extraction (from HPC exit) can be specified.

Jet-A is used as fuel in this study. The TURBO library in PROOSIS uses three-dimensional linearly interpolated tables for calculating the caloric properties of the working fluid in the engine model. These are generated with the NASA CEA software[9]. Dissociation of combustion products is not taken into account.

For a given set of ambient and flight conditions, the model only needs the power required and rotational speed at the gearbox outlet shaft (assumed fixed) in order to calculate the complete cycle.

The PROOSIS engine model was validated against proprietary data and tools by Turbomeca[10].

Table 1 gives the values of the main engine performance parameters at sea-level standard conditions for the Take-Off (TOP) power rating which is considered as the engine design point in the analysis that follows.

Table 1: Engine parameters at SL/STD take-off power rating (design point)

Parameter	Value	Parameter	Value
PWSD [kW]	1252	E22 [-]	0.809
W2 [kg/s]	4.84	E3 [-]	0.855
P22Q2 [-]	4.76	E45 [-]	0.871
P3Q24 [-]	2.66	E5 [-]	0.899
Tt41 [K]	1360	Wc NGV [%W2]	2.59
XNH [rpm]	40376	Wc Rotor [%W2]	0.25
GBQ [-]	0.286	SFC [kg/kWh]	0.270

The Specific Fuel Consumption (SFC) variation with shaft power PWSD is illustrated in Fig. 2. This is obtained at sea-level standard conditions. At high power conditions SFC remains almost constant but increases sharply at lower powers.

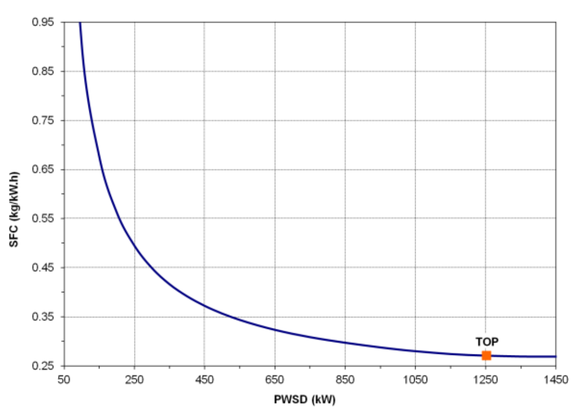

Figure 2: Engine SFC at SL/STD

Helicopter Model

The helicopter performance model is implemented in PROOSIS so as to allow different types of analyses to be carried out. One approach is to define it as a stand-alone PROOSIS component with a Mechanical port that allows it to be connected with the corresponding port of the engine performance component when a fully-integrated model is to be generated –like for example the one presented by Alexiou et al.[11]. Alternatively, it is defined as an internal PROOSIS function that returns the shaft power required for specified ambient/flight conditions and helicopter weight. In this way, the helicopter model can be used from within the engine model. This semi-integrated approach is preferred in this work since it allows both engine as well as coupled engine-helicopter simulations. For example, engine design-point, off-design, parametric and optimization analyses can be carried out at engine level while the helicopter model is only used for the mission analysis.

The helicopter performance model calculates the total helicopter power required according to Leishman[12] and takes into account the main rotor power, the tail rotor power, any customer power extraction needs and the gearbox power losses. The main rotor power comprises the induced, profile, fuselage and potential energy change power terms according to the current helicopter weight, forward and vertical velocity of the helicopter and air density at the current environmental conditions. The type of helicopter is defined through a list of attributes including the number of engines and main rotor blades, the maximum take-off weight, etc. The total power required is then divided by the number of engines to determine the torque needed by each engine for a specified rotor speed.

The basic parameters of the helicopter considered in this study are presented in Table 2. The statistical method presented Rand and Khromon[13] is employed in order to determine typical values. The basis of these calculations is the helicopter maximum take-off weight, MTOW.

Table 2: Helicopter parameters

Parameter	Symbol	Value	Units
Maximum Take-off Weight	MTOW	7400	kg
Weight Empty	WE	4105	kg
Fixed Useful Load[‡]	FUL	200	kg
Fuel Capacity	VFu	1.45	m^3
Number of Engines	Neng	2	-
Number of Rotor Blades	Nb	4	-
Main Rotor Diameter	D	15.2	m
Main Rotor Blade Chord	c	0.49	m
Main Rotor Solidity	σ	0.08	-
Rotor Blade Tip Speed	U	223	m/sec
Rotor Speed	NR	280	rpm
Equivalent Flat Plate Area	SCx	3.0	m^2
Power Extraction	Pex	10	kW

There are two important values of helicopter forward velocity; velocity for best range Vbr and velocity for best endurance Vbe.

Vbr results to maximum specific range SR which is defined as forward speed divided by total fuel flow rate[14]. The value of Vbr increases with increasing helicopter weight and altitude.

Vbe is the velocity corresponding to minimum fuel consumption. At this speed, power required is minimum hence excess power available is maximum and hence the maximum rate of climb Vz_{max} can be accomplished.

As altitude increases less excess power is available and Vz_{max} occurs at higher Vbe. Although at higher altitudes less power is required at higher forward speeds at the same time less power is available from the engines compared to that at sea-level. For the examined helicopter, the variation of specific range (SR) and fuel flow (WF) with forward speed (Vx) is presented in Fig. 3 for MTOW and sea-level standard conditions. Vbr and Vbe are marked on the graph.

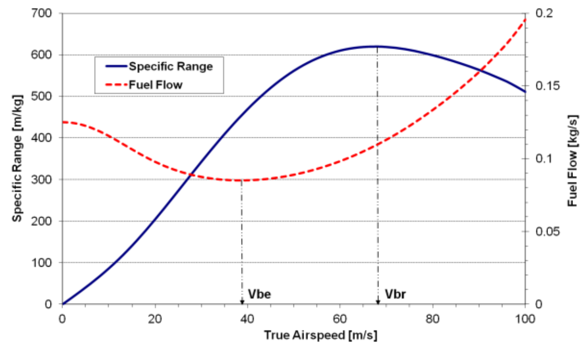

Figure 3: SR and WF versus Vx at MTOW and SL/STD

The Mission

The mission considered for demonstrating the optimization approach consists of three segments representing the climb, cruise and descent phases. The helicopter is assumed to climb from sea-level to 1000 m with Vbe and Vz_{max}, then cruise at this altitude for 400 km with Vbr and descent vertically to 0 m at a constant rate of 12.5 m/s. Vbr corresponds to the mid-cruise helicopter weight (weight at start of mission – fuel burn during climb – half of cruise fuel burn) while Vbe and Vz_{max} are based on the initial helicopter weight at half of the cruise altitude. The initial helicopter weight W0 comprises the operating empty weight (OEW), the payload (fixed), the mission fuel WFB and reserve fuel equal to 10% of WFB.

The values of the main mission parameters for the reference engine design are summarized in Table 3.

‡ Crew + trapped oil and fuel.

Table 3: Mission parameters for reference engine design

Parameter	Value	Unit
W0	6627.7	kg
Vbe	37.8	m/s
Vz_{max}	20.6	m/s
Vbr	67.3	m/s
SR	713.6	m/kg
WFB	570.7	kg
Time	6043.4	s

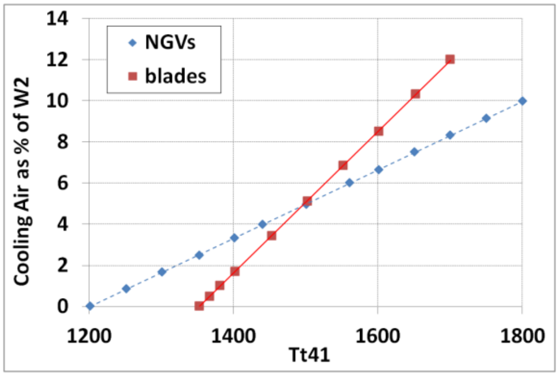

Figure 4: Variation of turbine cooling flows with Tt41

Analysis and Results

The engine design point parameters selected to optimize for minimum mission fuel burn are the inlet air mass flow rate W2 (related to engine size) and the overall pressure ratio OPR which is established from the LPC pressure ratio P22Q2 and the HPC pressure ratio P3Q24. Bounds must be imposed on these parameters in order to obtain feasible designs. For this study the bounds selected are: -30%<W2<30%, -30%<P22Q2<50% and -20%<P3Q24<100% of their corresponding baseline values.

During the optimization, any change in either compressor pressure ratio must allow turbine cooling/sealing flows to be re-introduced back into the main flow (total pressure of secondary flows must be greater than static pressure of main flow at the return location). In addition, there is a lower limit for the surge margin. An upper limit to the turbine rotor inlet temperature Tt41 is also imposed including the option to fix it at its reference value. Based on Tt41 value the turbine cooling flows (for NGVs and rotor blades) are re-calculated[15] as shown in Fig. 4. Finally, the effect of changing W2 on turbomachinery component efficiencies and engine weight is taken into account through empirical correlations, as shown in Fig. 5.

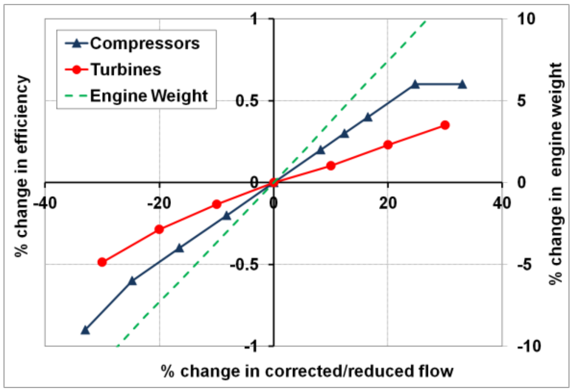

Figure 5: Assumed variation of compressor & turbine efficiency and engine weight with change in W2

The optimization calculation sequence is depicted in Fig. 6 and comprises the following steps:

➢ The engine performance at the reference design point (Table 1) is obtained first followed by a mission analysis (Table 3) in order to establish baseline values at engine and helicopter mission level. The mission calculation is iterative since the initial helicopter weight depends on the required fuel for the mission and the velocities of best range and best endurance depend on helicopter weight.

➢ The optimizer (Simplex[16]) adjusts the values of the engine design parameters (W2, P22Q2 and P3Q25) and for every new set of values a thermodynamic analysis is performed to establish the new design values of turbomachinery component efficiencies according to the relevant correlations of Fig.5.

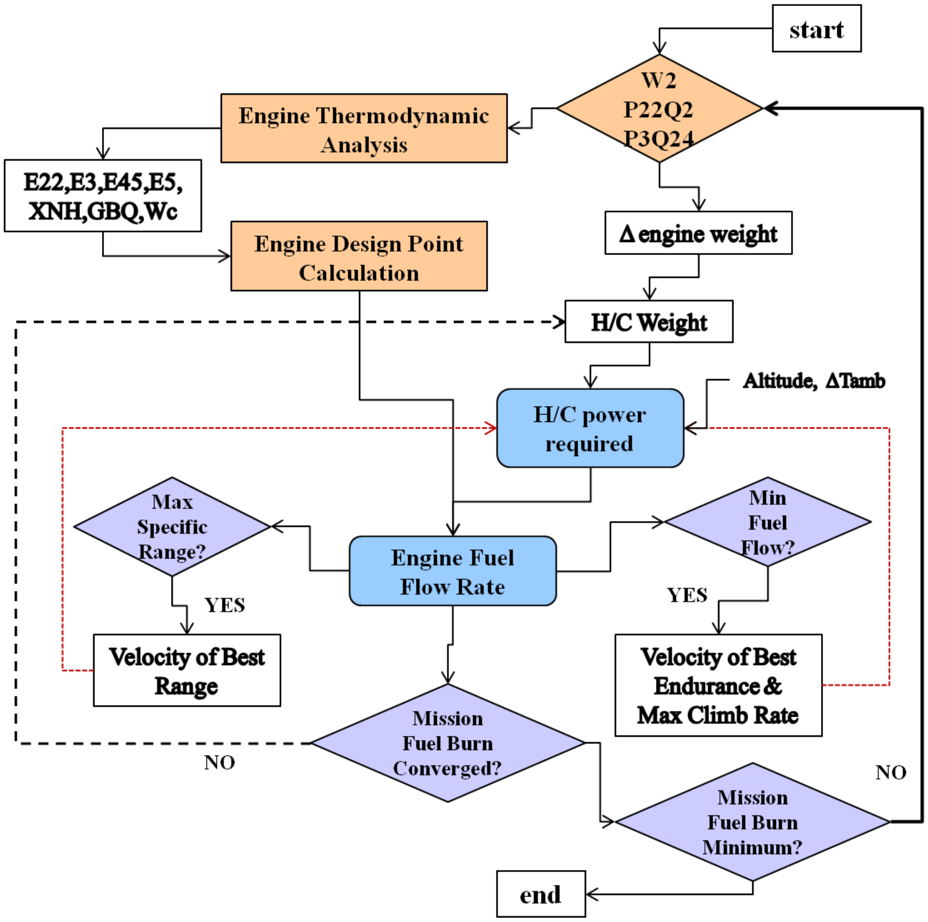

Figure 6: Optimization calculation flow chart

New gas generator and power turbine design rotational speeds are also determined so that the product of corrected flow with the square of corrected rotational speed is constant. This means that a new gearbox ratio is also established. Within this calculation an iterative scheme is included to determine the turbine cooling flows from Tt41 according to Fig. 4.

- For the reference value of shaft power PWSD, the engine design point calculation is then carried out that scales the turbomachinery component maps according to the new design values of mass flow rate, pressure ratio, isentropic efficiency and rotational speed. The position of the design point on the maps remains fixed. Burner efficiency and component (inlet, burner, diffuser, inter-turbine duct) pressure losses are kept constant at their reference values. Turbine cooling flows are established as for the thermodynamic analysis.
- For this new engine design the mission is carried out in the same iterative way as for the reference case but now also including the change in helicopter initial weight due to the change in engine weight which is calculated from the relevant correlation of Fig. 5 for the new value of W2.
- The optimum combination of engine design values is the one that produces the minimum mission fuel burn without violating the imposed constraints.

Following this procedure, the engine design parameters and the benefit in mission fuel burn are presented in Table 4 as percentage differences from the corresponding reference case and for three different Tt41 values.

Table 4: Mission fuel burn optimization results

	Tt41 [K]		
Parameter	**1360**	**1450**	**1600**
ΔW2 [%]	3.90	-7.39	-19.35
ΔP22Q2 [%]	-30.00	-27.16	-17.76
ΔP3Q24 [%]	100.00	100.00	100.00
ΔOPR [%]	40.02	45.68	64.49
ΔWFB[%]	-5.82	-7.74	-9.35

As expected, mission fuel burn reduces as the engine thermal efficiency improves and this occurs when OPR and Tt41 both increase. The change in engine mass flow rate W2 (and hence engine size) depends on Tt41. When Tt41 is fixed at its reference value (1360 K), the optimum value of cruise specific fuel consumption occurs at a higher value of W2 compared to the reference one while the two compressor pressure ratios are driven to their specified lower and upper limits respectively. This is presented graphically in Fig. 7, that shows for Tt41=1360K the variation of cruise SFC with ΔP3Q24 (from 0 to 100%) and for different values of ΔP22Q2 (-30, 0 and +30%). The variation of ΔW2 with ΔP3Q24 is also included for ΔP22Q2=-30%.

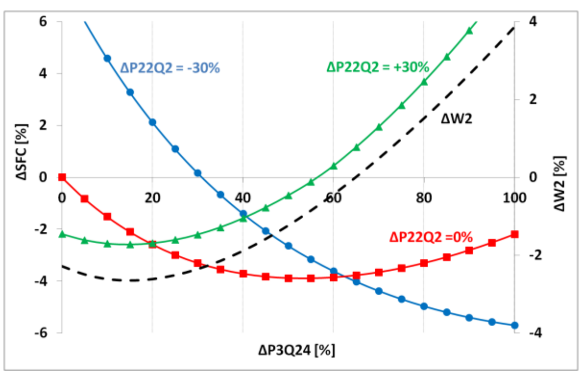

Figure 7: Effect of varying compressor pressure ratio on cruise SFC and W2

If Tt41 is allowed to increase from its reference value then a lower compared to reference W2 value is obtained at even higher OPR values leading to further mission fuel burn benefits due to improved engine performance as well as due to engine weight reduction.

Tables 5 and 6 summarize the engine design point and mission parameters respectively for all three values of Tt41.

Table 5: Engine parameters for optimized cycle

	Tt41 [K]		
Parameter	**1360**	**1450**	**1600**
W2 [kg/s]	5.03	4.48	3.90
OPR [-]	17.74	18.46	20.84
P22Q2 [-]	3.33	3.47	3.92
P3Q24 [-]	5.32	5.32	5.32
E22 [-]	0.810	0.808	0.806
E3 [-]	0.861	0.859	0.854
E45 [-]	0.867	0.865	0.863
E5 [-]	0.899	0.897	0.895
XNH [rpm]	39611	41956	44959
GBQ [-]	0.291	0.261	0.230
Wc NGV [%]W2	2.59	4.07	6.51
Wc Rotor [%]W2	0.25	3.30	8.36
SFC [kg/kWh]	0.256	0.253	0.249

Table 6: Mission parameters for optimized engine design

	Tt41 [K]		
Parameter	**1360**	**1450**	**1600**
W0 [kg]	6597.4	6567.2	6538.0
Vbe [m/s]	37.8	37.7	37.6
Vz_{max} [m/s]	20.8	20.9	21.1
Vbr [m/s]	66.6	66.2	65.8
SR [m/kg]	757.7	773.4	787.1
WFB [kg]	537.4	526.5	517.3
Time [s]	6105.1	6144.9	6178.2

From table 6, it can be seen that the optimized engine results in lower helicopter weight and hence lower velocity of best range which in turn causes an increase in mission time. This may not be an acceptable solution if for example the mission deals with either a search and rescue or a medical emergency operation. In such case, the mission time can be an additional constraint that requires also a change in the mission parameters (e.g. flight altitude).

The optimization calculation was also repeated for Tt41=1600K and assuming that either (a) there is no effect on component efficiencies from changing W2 or (b) that the effect is twice that shown in Fig 5. In both cases the effect of W2 on engine weight is the same as before. In the former case, the mission fuel burn benefit amounts to 11.55% while in the latter case it is 7.38%.

Summary and Conclusions

A procedure has been proposed that allows the designer to optimize the engine cycle for minimum fuel burn of a helicopter mission. The approach takes into account changes in the turbomachinery component efficiencies and engine weight due to engine inlet flow rate changes. Limits are imposed for turbine rotor inlet temperature, surge margin and pressure ratio. Turbine cooling/sealing flows are established according to the turbine rotor inlet temperature.

For the specific engine-helicopter-mission combination, the total fuel burn benefit ranged from 5.8% to 9.4%, depending on the maximum value of turbine rotor inlet temperature that can be tolerated.

Although the optimization study is implemented in a specific simulation environment, it is also possible to export it as a deck in the form of an executable (with or without a graphical user interface), a DLL or C/C++ source code. All aspects of the analysis can be defined externally by the user including the engine design parameters (inclusive of turbomachinery component maps and fluid model properties), the helicopter attributes, the mission description, the variation of engine weight and turbomachinery component efficiencies with inlet corrected flow, the variation of turbine cooling flows with turbine rotor inlet temperature, the optimization algorithm, the constraints and the objective function.

In addition, given that the helicopter performance model is implemented as a function it can be replaced with a different one (e.g. of higher fidelity) provided the final interface is the same.

Hence, the proposed approach is generic allowing the optimization of the engine as well as the helicopter for different combinations of engines and helicopters and different missions or combinations of missions and according to the objectives and limitations set by the designer.

Acknowledgements

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References

1. ACARE, 2010, “Aeronautics and Air Transport: Beyond Vision 2020 (Towards 2050)”.
2. Leahy, J., 2010, Airbus Global Market Forecast 2010-2029.
3. IATA, 2009, “The IATA Technology Roadmap Report”, 3rd Edition.
4. d’Ippolito, R., Stevens, J., Pachidis, V., Berta, A., Goulos, I., and Rizzi, C., 2010, “A Multidisciplinary Simulation Framework for Optimization of Rotorcraft Operations and Environmental Impact,” 2nd International Conference on Engineering Optimization, Lisbon, Portugal.
5. Goulos, I., Pachidis, V., D’Ippolito, R., Stevens, J., and Smith, C., 2010, “An Integrated Approach for the Multidisciplinary Design of Optimum Rotorcraft Operations”, Journal of Engineering

for Gas Turbines and Power, Vol. 134(9).
6. http://www.proosis.com/
7. Alexiou, A., Baalbergen, E. H., Kogenhop, O., Mathioudakis K. and Arendsen, P., 2007, "Advanced Capabilities for Gas Turbine Engine Performance Simulations", ASME Paper No. GT-2007-27086.
8. Alexiou, A. and Tsalavoutas, A., 2011, "Introduction to Gas Turbine Modelling with PROOSIS", 1st Edition, Empresarios Agrupados Internacional (EAI) S.A., Madrid Spain.
9. Gordon, S. and McBride, B.J., 1994, "Computer Program for Calculation of Complex Chemical Equilibrium Composition and Applications", NASA RP1311, National Aeronautics and Space Administration, Washington DC.
10. Grapin, J. C., 2010, Pre-Design Team Engineer, Turbomeca, private communication.
11. Alexiou, A., Tsalavoutas, A., Pons, B., Aretakis, N., Roumeliotis, I. and Mathioudakis K., 2012, "Assessing Alternative Fuels for Helicopter Operation", Journal of Engineering for Gas Turbine and Power, 134 (11).
12. Leishman, J. G., 2006, "Principles of Helicopter Aerodynamics", 2nd Edition, Cambridge University Press.
13. Rand, O. and Khromon, V., 2002, "Helicopter Sizing by Statistics", 58th AHS Forum, Montreal, Canada.
14. Stepniewski, W. Z. and Keys, C. N., 1984, "Rotary-Wing Aerodynamics", Dover Publications, New York.
15. Walsh, P.P. and Fletcher, P., 2004, "Gas Turbine Performance", 2nd Edition, Blackwell Science, Oxford.
16. Nelder, J.A. and Mead R, 1965, "A simplex method for function minimization", Computer Journal 7, 308–313.