

CRANFIELD UNIVERSITY

Jesus Ortiz Carretero

**Novel Turboshift Engines Design
and Optimisation for Rotorcraft
Applications**

School of Aerospace, Transport and
Manufacturing

Ph.D. Thesis

Academic Year: Jan 2015 - Feb 2018

Supervisor: Prof V. Pachidis

February 2018

CRANFIELD UNIVERSITY

School of Aerospace, Transport and
Manufacturing

Full time Ph.D.

Academic Year: Jan 2015 - Feb 2018

Jesus Ortiz Carretero

**Novel Turboshift Engines Design and
Optimisation for Rotorcraft Applications**

Supervisor: Prof V. Pachidis

February 2018

This thesis is submitted in partial fulfilment of the requirements for
the degree of Doctor of Philosophy

© Cranfield University, 2018. All rights reserved. No part of this
publication may be reproduced without the written permission of
the copyright owner.

La pelota no se mancha.

Diego A. Maradona

Executive Summary

Since the inception of the first gas turbine engines, scientists and aircraft propulsion designers have attempted to improve engine efficiency, size, weight, fuel consumption and power output. In the current scenario, where the conventional gas turbine technology has almost reached its maturity, companies and researchers are starting to seek new engine architectures and novel cycles to comply with the next future aviation challenges. One of the proposed solutions is the implementation of an auxiliary combustion process into the turbines, known as reheated cycle. In the reheated cycle, the gas from an expansion process (through a turbine stage or a whole turbine) is burned before the next expansion. This concept has the potential to increase the specific work, at the same time the thermal efficiency is improved.

Numerous investigations have been performed on the application of the reheated cycle to turbojet and civil turbofan engines. However, no studies are published about the potential application of this technology in rotorcraft powerplants. Therefore, the aim of this research project is to accomplish an exhaustive analysis and optimisation of a reheated turboshaft engine in terms of thermal efficiency and engine weight. The optimal engines identified at this stage are to be evaluated at mission level in order to assess the final impact on mission fuel abatement.

For this matter, models for the performance simulation of a representative helicopter and for the thermodynamic analysis of the engine architectures have been developed and validated against experimental data. An additional module estimating the cooling flows fluctuation with the engine cycle parameters has been coupled with the engine performance model. Finally, a procedure for the sizing of reheated turboshaft engines have been developed and validated by the author. The different models have been built after a reference Twin-Engine Medium (*TEM*)

helicopter, the Sikorsky UH-60A Black Hawk helicopter powered by two General Electric T700-GE-700 turboshaft engines.

The aforementioned models have been integrated in a common simulation framework for the completion of a preliminary parametric analysis showing the sensitivity of the reheated configurations to changes in the engine design variables. In particular, three distinct reheated architectures have been investigated together with a conventional engine for comparison purposes. The individual response of each engine architecture has been discussed.

The deployed framework has been then coupled with a Genetic Algorithm optimiser to efficiently seek for the best candidate engines in terms of total weight and Specific Fuel Consumption at cruise (SFC_{cr}). At this step, Response Surface Models ($RSMs$) have been developed for the fast estimation of engine weight and coupled with the optimiser routine. It has been proven that the reheated engines have the potential to reduce engine mass and increase thermal efficiency in comparison with the baseline engine, although the optimal conventional engine still shows superior performance under the conditions simulated. This conclusion has been also confirmed by the results obtained at mission level.

A final mission level multi-objective optimisation has been conducted for the conventional engine targeting the minimisation of the overall mission fuel burn and NO_x emissions. A representative model for the prediction of the combustor emissions production has been developed and validated for this purpose. The trade-off existing between fuel efficiency and pollutant depletion has been highlighted along with the potential benefits in block fuel and NO_x inventory.

Acknowledgements

I would like to thank firstly to my supervisor Prof Vassilios Pachidis for his guidance throughout the whole project. Secondly, but not less important, I want to say thanks to the doctors of the team: Ezanne Gires, Julien Enconniere and Alejandro Castillo Pardo. I am specially grateful to Julien and Alex for their help and the good and bad times we shared. This piece of work would not exist without their inputs. I would like to also thank to Dr Ioannis Goulos for his priceless help with the publication matters, and to Salvatore Ippedico for his contribution to the engine weight estimation cause.

I cannot forget here each and every member of the *Txoko*. The regular reunions during these tree years were as awesome as needed. Two members of the group deserve and special mention, my flatmates and friends Dani and Pau. They were always there when I needed them, and I hope they will always be. I could have not survive these three years without them.

Thanks also to Diego Pablo Simeone for teaching us to always believe, to Gerardo Castilla, Jorge San Juan and Carlos Marquez for the countless hours of happiness, and finally, to Bruce Springsteen, Axl Rose and Bono for making the PhD experience more bearable.

Finally, I would like to thank to my parents for the efforts and sacrifices they did to give me the opportunities they never had, and my brother who gave me some very good advice in sketching matters.

Aupa Atleti!

Contents

Executive Summary	v
Acknowledgements	vii
List of Figures	xi
List of Tables	xiii
Nomenclature	xv
1 Introduction	1
1.1 Rotorcraft activity volume	2
1.2 Future aviation environmental regulations	3
1.3 Sustainable rotary-wing aviation initiatives	3
1.3.1 FriendCopter programme	4
1.3.2 <i>NICETRIP</i> project	4
1.3.3 Clean Sky initiative	4
1.3.4 Clean Sky 2 initiative	6
1.3.5 Other initiatives	8
1.4 Novel engine cycles for increased efficiency	10
1.5 Aim & Objectives	12
1.6 Contribution to knowledge	14
1.6.1 Publications	15
1.7 Thesis structure	16
2 Literature Review	19
2.1 Rotorcraft performance simulation	19
2.1.1 Rotorcraft simulation frameworks	20
2.1.1.1 <i>Phoenix</i> platform	20
2.1.1.2 <i>HECTOR</i> platform	22
2.2 Gas turbine engine simulation	24
2.2.1 Gas turbine engine performance modelling	25
2.2.2 Gas turbine engine gaseous emissions modelling	27
2.2.3 Gas turbine engine weight modelling	31

2.3	Performance optimisation of turboshaft engines for rotorcraft applications	34
2.4	Reheated engine concept	39
3	Methodology	47
3.1	Helicopter performance modelling	49
3.2	Powerplant performance modelling	51
3.2.1	Baseline engine architecture	51
3.2.2	Reheated engine architectures	52
3.2.3	Engine model description	53
3.2.4	Cooling flow sizing module	57
3.3	Powerplant weight modelling	59
3.3.1	Turboshaft intake sizing	60
3.3.2	Turbomachinery disk temperature distribution	62
3.3.3	Engine materials library expansion	65
3.3.4	Centrifugal compressor weight prediction capability	65
3.3.4.1	Centrifugal compressor gas path designation	66
3.3.4.2	Annulus design	67
3.3.4.3	Blade weight	75
3.3.4.4	Impeller disk weight	76
3.3.4.5	Minimum volume impeller	79
3.3.4.6	Centrifugal compressor shroud and diffuser weight	81
3.3.5	Free turbine shaft weight prediction capability	81
3.3.6	Auxiliary reheaters sizing	82
3.3.6.1	Reheater design	84
3.3.7	Reheated engines weight modelling	88
3.4	Combustor emissions modelling	91
4	Models Validation	95
4.1	Helicopter performance model validation	95
4.2	Powerplant performance model validation	98
4.3	Weight prediction validation	101
4.3.1	ATLAS updates validation	101
4.3.1.1	General Electric CT7-8A weight model verification	102
4.3.1.2	Turbomeca Makila 1A1 weight model verification	104
4.3.2	Powerplant weight model validation	105
4.4	Combustor emissions model validation	109
5	Results & Discussion	111
5.1	Parametric analysis of powerplant performance	112
5.1.1	<i>CONV</i> engine	114
5.1.1.1	Effect on SFC_{cr}	114
5.1.1.2	Effect on $EI NO_{xcr}$	115
5.1.1.3	Effect on SP_{MP}	116
5.1.1.4	Effect on W_{eng}	117

5.1.2	<i>ITR</i> engine	118
5.1.2.1	Effect on SFC_{cr}	118
5.1.2.2	Effect on SP_{MP}	120
5.1.2.3	Effect on W_{eng}	121
5.1.3	<i>SEQ</i> engine	122
5.1.3.1	Effect on SFC_{cr}	123
5.1.3.2	Effect on SP_{MP}	124
5.1.3.3	Effect on W_{eng}	124
5.1.4	<i>ITR</i> + <i>SEQ</i> engine	126
5.1.4.1	Effect on SFC_{cr}	126
5.1.4.2	Effect on SP_{MP}	127
5.1.4.3	Effect on W_{eng}	128
5.2	Powerplant optimisation	129
5.2.1	Surrogate engine weight modelling	132
5.2.1.1	Engine weight surrogate models validation	135
5.2.2	Optimised conceptual engines	136
5.2.2.1	Optimised <i>CONV</i> engine	136
5.2.2.2	Optimised <i>ITR</i> engine	137
5.2.2.3	Optimised <i>SEQ</i> engine	139
5.2.2.4	Optimised <i>ITR</i> + <i>SEQ</i> engine	142
5.2.2.5	Optimal configurations comparison	144
5.3	Engine performance assessment at mission level	147
5.3.1	Flight mission description	148
5.3.2	Calculation of flight segment optimal TRs	149
5.3.3	Performance assessment at mission level results	151
5.3.4	Mission level optimisation for <i>CONV</i> engine. W_{fuel} vs. W_{NO_x}	154
6	Conclusions & Future Work	159
6.1	Summary & conclusions	159
6.1.1	Helicopter aerodynamic performance	159
6.1.2	Turboshaft engine performance	159
6.1.3	Turboshaft engine weight	160
6.1.4	Turboshaft engine NO_x emissions	161
6.1.5	Integration of a global engine optimisation tool	162
6.1.6	Key findings	162
6.2	Future work	163
6.2.1	ATLAS code updating	163
6.2.2	HEPHAESTUS code updating	166
6.2.3	<i>HECTOR</i> code updating	167
6.2.4	Turboshaft performance analysis and optimisation	168

Bibliography**175**

List of Figures

1.1	Global CO_2 emissions share by sector	1
1.2	<i>NICETRIP</i> tilt-rotor design concept	5
1.3	Clean Sky <i>ITDs</i> organisation	5
1.4	<i>GRC</i> contributions with respect to the <i>Vision 2020</i> goals	6
1.5	<i>GRC</i> technologies integration scheme	6
1.6	Clean Sky 2 programme structure	7
1.7	Clean Sky 2 rotorcraft concepts	8
1.8	Large civil <i>VTOL</i> transport candidates	9
1.9	Large Civil Tiltrotor turboshaft engine layout	9
1.10	Robinson R22 Beta II helicopter	10
2.1	Decomposition of helicopter forces and moments	20
2.2	<i>Phoenix</i> architecture scheme	22
2.3	<i>HECTOR</i> architecture scheme	23
2.4	Turbofan engine modelling in <i>GSP</i>	25
2.5	Examples of predefined engine configurations in GasTurb	26
2.6	Turbojet engine brick modelling in TURBOMATCH	27
2.7	Conventional combustor scheme in HEPHAESTUS	30
2.8	ATLAS turbofan 2D cutaway comparison	33
2.9	<i>ADT</i> architecture scheme	36
2.10	Single-spool recuperated turboshaft scheme	38
2.11	Correlation for heat exchanger weight estimation	38
2.12	Thermal efficiency comparison	40
2.13	Ericsson cycle diagram	41
2.14	Comparison of thermodynamic cycles with and without turbine- burner	42
2.15	T - S diagrams for different <i>ITB</i> cycles	43
2.16	<i>UHEGT</i> concept configuration 2.17	44
2.17	Baseline engine vs. Reheated architectures	45
3.1	Engine performance analysis tool scheme	48
3.2	Cooling flowrate calculation scheme	49
3.3	Sikorsky UH-60 Black Hawk helicopter 3-view drawing	50
3.4	General Electric T700-GE-700 engine	51
3.5	General Electric T700/CT7 engine family architecture	52
3.6	Turboshaft engine conceptual configurations	53

3.7	Engine model bricks structure in TURBOMATCH	54
3.8	Original disk temperature distribution	63
3.9	Updated disk temperature distribution	64
3.10	Axial & centrifugal compressor weight parts	66
3.11	Gas path stations numbering	67
3.12	Centrifugal disk and blade geometry	70
3.13	Velocity triangle at centrifugal impeller outlet	74
3.14	Impeller blade and disk discretisation for stress analysis	77
3.15	Analysis of the centrifugal compressor discretisation sensitivity	78
3.16	Centrifugal disk temperature distribution	79
3.17	Types of power shaft connection	82
3.18	<i>FT</i> shaft length calculation options	83
3.19	Reheater layout	83
3.20	Reheater weight parts	84
3.21	Reheated engines cutaways generated by ATLAS	89
3.22	Turbomachinery configuration options in ATLAS	91
3.23	General Electric T700/CT7 combustor cross section	92
3.24	Engine combustor model layout	92
4.1	Resonance chart of the Sikorsky UH-60A main rotor	96
4.2	Shape of the main rotor vibration modes	97
4.3	Rotorcraft model flight dynamics trim results	97
4.4	General Electric T700-GE-700 experimental results comparison	100
4.5	General Electric T700-GE-700 simulation results comparison	101
4.6	Cutaways of the turboshaft engines for validation	102
4.7	General Electric CT7-8A engine cutaway comparison	103
4.8	Turbomeca Makila 1A1 engine cutaway comparison	105
4.9	General Electric T700-GE-700 engine cutaway	106
4.10	General Electric T00-GE-700 engine cutaway comparison	107
4.11	Combustor model NO_x results comparison	109
4.12	<i>ET</i> variation at <i>FF</i> and <i>PZ</i>	110
5.1	Effect of <i>CONV</i> engine design inputs on SFC_{cr}	115
5.2	Effect of <i>CONV</i> engine design inputs on $EI NO_{x_{cr}}$	116
5.3	Effect of design inputs on combustor parameters at cruise	116
5.4	Effect of <i>CONV</i> engine design inputs on SP_{MP}	117
5.5	Effect of <i>CONV</i> engine design inputs on W_{eng}	118
5.6	Effect of <i>ITR</i> engine design inputs on SFC_{cr}	120
5.7	Effect of <i>ITR</i> engine design inputs on SP_{MP}	121
5.8	Effect of <i>ITR</i> engine design inputs on W_{eng}	122
5.9	Effect of <i>SEQ</i> engine design inputs on SFC_{cr}	124
5.10	Effect of <i>SEQ</i> engine design inputs on SP_{MP}	125
5.11	Effect of <i>SEQ</i> engine design inputs on W_{eng}	125
5.12	Effect of <i>ITR</i> + <i>SEQ</i> engine design inputs on SFC_{cr}	127
5.13	Effect of <i>ITR</i> + <i>SEQ</i> engine design inputs on SP_{MP}	128

5.14	Effect of $ITR + SEQ$ engine design inputs on W_{eng}	129
5.15	Powerplant multi-objective optimisation scheme	131
5.16	Engine weight RSM generation	133
5.17	$LOOCV$ results for the engine weight $RSMs$	135
5.18	Pareto front for $CONV$ engine optimisation. SFC_{cr} vs. W_{eng} . . .	137
5.19	Pareto front for ITR engine optimisation. SFC_{cr} vs. W_{eng}	138
5.20	Cooling flow fractions comparison for ITR optimal engines	140
5.21	Pareto front for SEQ engine optimisation. SFC_{cr} vs. W_{eng}	140
5.22	Cooling flow fractions comparison for SEQ optimal engines	142
5.23	Pareto front for $ITR + SEQ$ engine optimisation. SFC_{cr} vs. W_{eng} .	142
5.24	Cooling flow fractions comparison for $ITR + SEQ$ optimal engines	144
5.25	Comparison of optimal engine configurations. SFC_{cr} vs. W_{eng} . . .	145
5.26	Optimal engines percentage savings in W_{eng} and SFC_{cr}	145
5.27	Engine shaft power at forward flight	146
5.28	Assessment of conceptual powerplants at mission level flowchart . .	148
5.29	Flight mission definition	149
5.30	Engine power vs. time. Comparison between baseline and optimal engines	150
5.31	Effect of TR_2 on SFC at each mission flight segment	151
5.32	Optimal engines performance at mission level	152
5.33	Optimal engines cycle parameters at mission level	153
5.34	Engine power vs. time (Baseline engine)	155
5.35	$CONV$ engine optimisation at mission level scheme	155
5.36	Pareto front for $CONV$ engine optimisation at mission level. W_{fuel} vs. W_{NO_x}	156
5.37	Optimal conventional engines performance at mission level	158
6.1	Lycoming T53 turboshaft layout	164
6.2	Detail of the disk-shaft coupling in the General Electric T700 engine	164
A.1	Turbine stage cooling schematic diagram	171

List of Tables

2.1	Review of empirical methods for NO_x estimation	29
2.2	Review of <i>WEB</i> approaches	32
2.3	ATLAS turbofan engine weight prediction comparison	33
2.4	Real Brayton cycle reference data	40
3.1	Rotorcraft model design parameters	51
3.2	Cooling flows sizing parameters	59
3.3	General Electric T700-GE-700 <i>GGT</i> cooling flows distribution . . .	59
3.4	Ellipse semi-axis values for centrifugal blade sizing	69
3.5	Combustor model design parameters	93
4.1	General Electric T700-GE-700 <i>DP</i> at <i>MCP SL-ISA</i> + 0 conditions	99
4.2	General Electric T700-GE-700 experimental test conditions	99
4.3	General Electric CT7-8A performance parameters at 30-Second <i>OEI</i> <i>SL - ISA</i> + 0 conditions	103
4.4	General Electric CT7-8A engine mass comparison	104
4.5	General Electric CT7-8A engine mass breakdown	104
4.6	Turbomeca Makila 1A1 performance parameters at <i>TO SL-ISA</i> + 0 conditions	105
4.7	Turbomeca Makila 1A1 engine mass comparison	105
4.8	Turbomeca Makila 1A1 engine mass breakdown	106
4.9	General Electric T700-GE-700 performance parameters at <i>MP SL-</i> <i>ISA</i> + 0 conditions	106
4.10	General Electric T700-GE-700 engine mass comparison	107
4.11	General Electric T700-GE-700 engine mass breakdown	107
5.1	Engine design space exploration operating points	112
5.2	Design parameter boundaries for space exploration	113
5.3	<i>CONV</i> engine reference data	114
5.4	Reheated engines reference data	119
5.5	Cycle variables used in the optimisation of each engine architecture	130
5.6	Optimal <i>CONV</i> engines design variables	137
5.7	Optimal <i>CONV</i> engine results	137
5.8	Optimal <i>ITR</i> engines design variables	138
5.9	Optimal <i>ITR</i> engine results	139
5.10	Optimal <i>SEQ</i> engines design variables	140

5.11	Optimal <i>SEQ</i> engine results	141
5.12	Optimal <i>ITR</i> + <i>SEQ</i> engines design variables	143
5.13	Optimal <i>ITR</i> + <i>SEQ</i> engine results	143
5.14	Operating conditions of the mission flight segments	149
5.15	Optimal engines results at mission level	153
5.16	Design parameters boundaries for <i>CONV</i> engine optimisation . . .	154
5.17	Optimal <i>CONV</i> engines data at mission level	157
5.18	Optimal <i>CONV</i> engines results at mission level	157

Nomenclature

Acronyms & abbreviations

ACARE Advisory Council for Aeronautics Research in Europe

ADT Aeroelasticity Decoupling Tool

AFR Atomising Flow Ratio —

BL Blade Length *m*

BPR By-Pass Ratio —

CB Component Based

CF Cooling Flows

CH Connecting Hardware

CIP Combustor Inlet Pressure *atm*

CIT Combustor Inlet Temperature *K*

CO Carbon monoxide

CO₂ Carbon dioxide

CONV Conventional cycle

COP Combustor Outlet Pressure *Pa*

DCFR Dome Cooling Flow Ratio —

DH Centrifugal blade radial dimension *m*

DL Centrifugal blade axial dimension *m*

<i>DOE</i>	Design of Experiments	
<i>DP</i>	Design Point	
<i>DZ</i>	Dilution Zone	
<i>EI</i>	Emission Index	$g/kg\ fuel$
<i>EPNdB</i>	Effective Perceived Noise in dB	
<i>ET</i>	Equivalent Temperature	K
<i>F</i>	Flap vibration mode	
<i>FAR</i>	Fuel to Air mass Ratio	—
<i>FF</i>	Flame Front	
<i>FN</i>	Net thrust	kN
<i>FOCA</i>	Federal Office of Civil Aviation	
<i>FT</i>	Free Turbine	
<i>GAs</i>	Genetic Algorithms	
<i>GGT</i>	Gas Generator Turbine	
<i>GQR</i>	Gaussian Quadratic Regression	
<i>GRC</i>	Green RotorCraft	
<i>GSP</i>	Gas Turbine Simulation Program	
<i>H₂O</i>	Water vapour	
<i>HE</i>	Heat Exchanger	
<i>HECTOR</i>	HEliCopTer Omni-disciplinary Research platform	
<i>HPC</i>	High Pressure Compressor	
<i>HPT</i>	High Pressure Turbine	
<i>IADP</i>	Innovative Aircraft Demonstrator Platform	
<i>IGV</i>	Inlet Guide Vane	

<i>IPT</i>	Intermediate Pressure Turbine	
<i>ISA</i>	International Standard Atmosphere	
<i>ITB</i>	Inter-Turbine-Burner	
<i>ITD</i>	Integrated Technology Demonstrator	
<i>ITR</i>	Inter-Turbine Reheated	
<i>IZ</i>	Intermediate Zone	
<i>L</i>	Lag vibration mode	
<i>LHS</i>	Latin Hypercube Sampling	
<i>LOOCV</i>	Leave-One-Out Cross Validation	
<i>LPC</i>	Low Pressure Compressor	
<i>LPT</i>	Low Pressure Turbine	
<i>MCP</i>	Maximum Continuous Power	
<i>MP</i>	Maximum Power	
<i>MR</i>	Main Rotor	
<i>NICETRIP</i>	Novel Initiative Competitive Effective Tilt Rotor Integrated Project	
<i>NM</i>	Nautical Miles ($1\text{ NM} = 1,852\text{ m}$)	
NO_x (<i>NO</i> and <i>NO</i> ₂)	Nitrogen oxides	
<i>OD</i>	Off-Design	
<i>OEI</i>	One Engine Inoperative	
<i>OPR</i>	Overall Pressure Ratio	—
<i>P</i>	Rotational speed multiple	
<i>PCN</i>	Non-dimensional rotational speed	%
<i>PhoeniX</i>	Platform Hosting Operational & ENvironmental Investigations for Rotorcraft	

<i>PR</i>	Pressure Ratio	—
<i>PZ</i>	Primary Zone	
<i>PZER</i>	Primary Zone Equivalence Ratio	—
<i>R</i>	Rotor	
<i>REACH</i>	Registration, Evaluation, Authorisation and Restriction of Chemicals	
<i>RMEM</i>	Rotorcraft Mission Energy Management	
<i>RSM</i>	Response Surface Model	
<i>RTO</i>	Research and Technology Organisation	
<i>S</i>	Stator	
<i>SAGE</i>	Sustainable and Green Engines	
<i>SEL</i>	Single-Engine Light	
<i>SEQ</i>	Sequential combustor	
<i>SF</i>	Safety Factor	—
<i>SFC</i>	Specific Fuel consumption	$\mu g/J, g/kN \cdot s$
<i>SGO</i>	Systems for Green Operations	
<i>SHE</i>	Safran Helicopter Engine	
<i>SHP</i>	Shaft horsepower	kW
<i>SL</i>	Sea Level	
<i>SO_x</i>	Sulphur oxides	
<i>SP</i>	Specific Power (Work)	kJ/kg
<i>ST</i>	Specific Thrust	$N \cdot s/kg$
<i>T</i>	Torsional vibration mode	
<i>TE</i>	Technology Evaluator	
<i>TEL</i>	Twin-Engine Light	

<i>TEM</i>	Twin-Engine Medium	
<i>TET</i>	Turbine Entry Temperature	<i>K</i>
<i>TO</i>	Take-Off	
<i>TOP</i>	Trade-Off Parameter	—
<i>TOW</i>	Take-Off Weight	<i>kg</i>
<i>TR</i>	Tail Rotor	
<i>TR₁</i>	Temperature Ratio in 1 st reheater	—
<i>TR₂</i>	Temperature Ratio in 2 nd reheater	—
<i>TR_c</i>	Compressor Temperature Ratio	—
<i>UHC</i>	Unburned hydrocarbons	
<i>UHEGT</i>	Ultra-High Efficiency Gas Turbine	
<i>vol</i>	Volume	<i>m</i> ³
<i>VTOL</i>	Vertical Take-Off and Landing	
<i>WATE</i>	Weight Analysis of Turbine Engines	
<i>WEB</i>	Whole Engine Based	
<i>WS</i>	Working Split	—
Roman symbols		
<i>A</i>	Area	<i>m</i> ²
<i>a</i>	Ellipse semi-axis on the <i>x</i> direction	<i>m</i>
<i>B</i>	Cooling flow fraction	%
<i>b</i>	Ellipse semi-axis on the <i>y</i> direction	<i>m</i>
<i>Bi_{metal}</i>	Metal Biot number	—
<i>C_P</i>	Power coefficient [$P_{MR}/\rho A_{MR}(\Omega R_{MR})^3$]	—
<i>C_T</i>	Thrust coefficient [$T_{MR}/\rho A_{MR}(\Omega R_{MR})^2$]	—

C_p	Specific heat of air at constant pressure	$J/kg \cdot K$
D	Diameter	m
f	Area of the cylindrical section of the blade	m^2
h	Specific enthalpy	J/kg
h/t	Hub to tip ratio	—
h/t_{exit}	Inlet hub to outlet tip radius ratio	—
K	Combustion pattern factor	—
K_{cool}	Cooling flow factor	—
L/D	Axial blade length and outlet blade tip diameter ratio	—
l/h	Length to height ratio	—
M	Mach number	—
n_b	Number of blades	
P	Pressure	atm
P_{MR}	Main rotor power	W
R	Radius	m
r	Radial variable	m
R_{air}	Universal gas constant of air	$J/kg \cdot K$
S	Entropy	$J/kg \cdot K$
S	Mixing parameter	—
s_f	Slip factor	—
T	Temperature	K
t	Thickness	m
T_{metal}	Maximum metal temperature	K
T_{MR}	Main rotor thrust	N

U	Speed	m/s
V	Velocity	m/s
W	Weight	kg
w	Mass flow	kg/s

Greek symbols

α	Absolute velocity angle	deg
Δ	Increment	
ϵ_0	Blade cooling effectiveness	—
ϵ_f	Film cooling effectiveness	—
η	Isentropic efficiency	—
$\eta_{c,int}$	Internal cooling efficiency	—
η_{poly}	Polytropic efficiency	—
η_{th}	Thermal efficiency	—
γ	Specific heat ratio	—
Ω	Main rotor rotational speed	rad/s
ω	Blade natural frequency or rotational speed	rad/s
ϕ	Equivalence ratio	—
Ψ	'Space-chord' ratio	—
ψ	Stage loading coefficient	—
ρ	Density	kg/m^3
σ	Solidity	—
σ_y	Yield stress	MPa
σ_θ	Tangential stress	MPa
σ_{UTS}	Ultimate Tensile stress	MPa

σ_{VM} Von Mises stress MPa

Subscripts

1 1^{st} Turbine, 1^{st} Compressor

2 2^{nd} Turbine, 2^{nd} Compressor

3 3^{rd} Turbine

abs Absolute

amb Ambient conditions

ato Atomisation

av Average

ax Axial

b Burner/combustor

be Blade element

bl Blade

bore Disk bore

c Compressor

cool Coolant

cr Cruise

d Dome

dc Dome cooling

diff Diffuser

dsk Disk

e Disk element

eng Engine

eq Equivalent

<i>f</i>	Fuel
<i>fan</i>	Fan
<i>g</i>	Gas
<i>h</i>	Hub
<i>I</i>	Inlet
<i>int</i>	Intake
<i>m</i>	Mean
<i>max</i>	Maximum
<i>min</i>	Minimum
<i>nom</i>	Nominal
<i>O</i>	Outlet
<i>pass</i>	Cooling passage
<i>rad</i>	Radial
<i>ref</i>	Reference
<i>rel</i>	Relative
<i>rh</i>	Reheater
<i>rim</i>	Disk rim
<i>rot</i>	Rotor
<i>s</i>	Shroud
<i>sh</i>	Shaft
<i>st</i>	Static
<i>sta</i>	Stator
<i>stoi</i>	Stoichiometric
<i>swirl</i>	Swirl flow

t	Tip
tot	Total
tur	Turbine
u	Upper
w	Whirl (tangential)

Chapter 1

Introduction

Carbon dioxide (CO_2) is well-known to be one of the principal contributors to the greenhouse effect and so, to the global climate change. Currently, aviation sector has a minor role in the global share with only a 2% of the total human related CO_2 emissions (Fig. 1.1). If the transport sector is uniquely considered, the pie diagram shows that aviation is responsible for 11% of the sector emissions, compared to the 74% coming from road transportation [1]. However, air transport is expected to grow at 4-5% each year over the next two decades [2]. This increased activity implies an escalating contribution to the CO_2 total production.

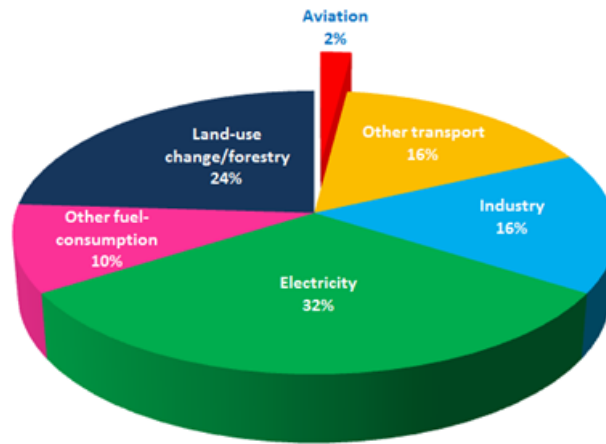


FIGURE 1.1: Global CO_2 emissions share by sector [3]

Apart from the CO_2 emissions, aviation is also responsible for the production of nitrogen oxides, commonly known as NO_x . This pollutant, not only contribute to the formation of photochemical smog at ground level, but also add to the problem of acid rain. At global level, NO_x produced by aircraft engines is less than 3%

of the total share [4]. However, the main concern is that the NO_x generated as a result of the operation of stationary gas turbines, and subsonic aircraft and rotorcraft leads to the formation of ozone in the troposphere. A prolonged exposure to high concentrations of ozone has been proven to be harmful for the human health.

A third side effect of aviation activity is its noise pollution. In fact, the impact of aviation on residential communities is an important factor that may limit the aviation growth. For this matter, the application of effective noise abatement procedures is vital to increase the degree of acceptance to airport activities [5].

In sum, the environmental issues that has been listed constitute a challenge for the future of the aviation sector and its sustainability. In this context, this project is focused on the reduction of the environmental footprint of air transport activities, in particular, those concerning with the operation of rotorcraft.

1.1 Rotorcraft activity volume

Due to their exclusive lifting method, rotorcraft are able to undertake missions that are impossible for other platforms. Thus, they have found a wide range of applications where the ability to fly stationary at a low speed constitutes a decisive advantage, such as medical evacuation, search and rescue, off-shore oil platforms service, law enforcement and passenger transportation in densely or isolated areas.

On average, rotorcraft are accountable for a total of 1,500,000 flight hours per year, which are still far from the 10,000,000 hours flown by the European commercial airlines. However, rotorcraft activities are expected to rise sharply in the short term, especially those related with medical services since the helicopter is regarded as the most efficient vehicle for the safe and fast transport of patients and organs for transplantation. Besides, the passenger transport by rotorcraft is expected to rise due to the ongoing increase in air travel demand [6].

Therefore, as a result of the rising relevance of rotorcraft operation within the aviation sector, the environmental impact of this activity cannot be longer ignored, and new technologies have to be developed to support a sustainable rotorcraft growth.

1.2 Future aviation environmental regulations

To cope with the aviation activity forecast and to ensure that proper initiatives are undertaken to tackle its environmental impact, different government authorities and regulatory bodies are elaborating new policies oriented towards the depletion of the aviation environmental footprint. In this context, the Advisory Council for Aeronautics Research in Europe (*ACARE*) has established a list of targets as part of the *Vision 2020* strategy which includes [7]:

- 50% reduction of CO_2 emissions
- 80% reduction of NO_x emissions
- 50% reduction of perceived noise levels
- Green product life including design, manufacturing, maintenance and disposal/recycling

The mentioned targets are defined with respect to year 2000 technology. More recently, *Flighthpath 2050* defined a list of more ambitious goals to be met by year 2050 technology [8]:

- 75% reduction of CO_2 emissions
- 90% reduction of NO_x emissions
- 65% reduction of perceived noise levels
- Design of air vehicles and components to be fully recyclable

In order to fit the expected growth in operations within this environmental legislation framework, novel powerplant and vehicle solutions have to be explored.

1.3 Sustainable rotary-wing aviation initiatives

In this context, several research programmes have been and are launched to deliver suitable solutions for improved rotorcraft performance and environmental footprint. A brief summary of some of them is included herein to enhance the description of the context of this project.

1.3.1 FriendCopter programme

This project was a development programme completed in September 2009 under the European Community 6th Framework. The main aim of its research activity was to tackle the rotorcraft noise and vibrations issues, and attenuate the emission of NO_x pollutants in order to enhance the public acceptance and the environmental compliance of helicopters. For that purpose, the FriendCopter project set four main goals: (i) reduction of acoustic footprint areas, (ii) fuel consumption saving for high speed flight, (iii) cabin noise similar to conventional airliner flight levels, and (iv) cabin vibration comparable to the smooth operation of jet aircraft.

In order to achieve the mentioned goals, the investigation focused on the development of optimum noise abatement flight procedures, the design of an innovative engine intake and outlet nozzle, the modification of conventional power gearboxes, and the full scale testing of active rotor blade control technology. A more detailed description of the FriendCopter programme is available in [9].

1.3.2 *NICETRIP* project

The Novel Initiative Competitive Effective Tilt Rotor Integrated Project (*NICETRIP*) was an initiative undertaken in parallel with the FriendCopter programme under the European Community 6th Framework and completed in December 2013. The project aimed at the acquisition of new knowledge and the development of novel technologies towards the design and sustainable operation of a civil tilt-rotor aircraft with the architecture shown in Fig. 1.2. The main objectives of *NICETRIP* covered the validation of new components on full-scale rigs, the testing of a large-scale full-span powered model in a wind tunnel, and the assessment of different aspects of the tilt-rotor integration in the european air traffic system. More information about this programme can be found in [10].

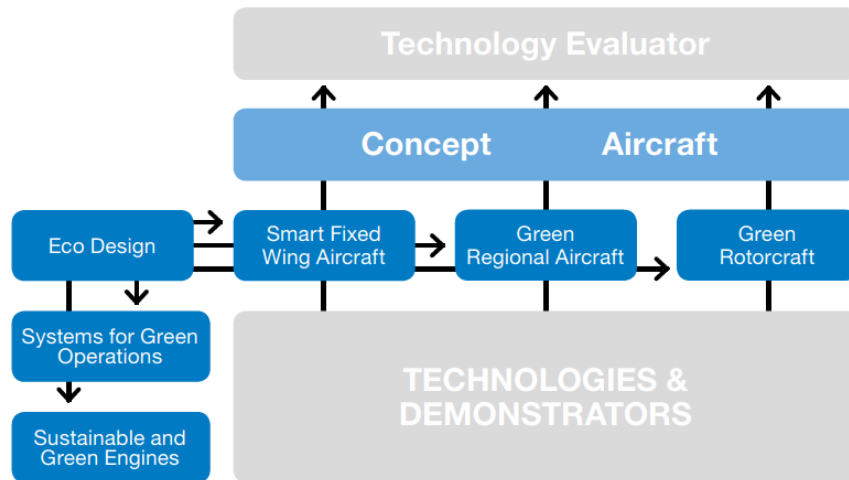
1.3.3 Clean Sky initiative

This programme was started in 2008 as part of the European Union Horizon 2020 research and innovation programme. It became the largest European research project with a budget of €1.6 billion and more than 600 partners from 24 European

FIGURE 1.2: *NICETRIP* tilt-rotor design concept [11]

countries. It targeted the reduction of gas emissions and acoustic levels produced by aircraft through the development of advanced technologies [12].

The research activity of Clean Sky was distributed in six Integrated Technology Demonstrators (*ITDs*) and a Technology Evaluator (*TE*), as shown in Fig. 1.3. The technologies developed by the different *ITDs* are intended to be applied to new aircraft configurations, including large commercial aircraft, regional jet and turboprop airplanes, small business jet concepts, and multiple-size helicopters.

FIGURE 1.3: Clean Sky *ITDs* organisation [12]

From the different *ITDs* within the Clean Sky schedule, the Green RotorCraft (*GRC*) *ITD* was the one responsible of deliver innovative solutions for the next rotorcraft generation. Its final aim was to meet the environmental targets described in Fig. 1.4, which are in line with the *ACARE* environmental objectives set for 2020.

The research activity of the *GRC ITD* was divided into different sub-projects representative of areas where rotorcraft technologies can be improved. As observed

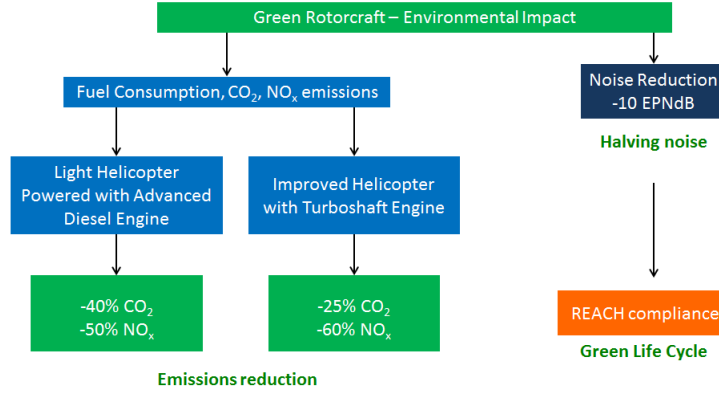


FIGURE 1.4: *GRC* contributions with respect to the *Vision 2020* goals [13]

in Fig. 1.5, all the research areas were connected with the *TE* which assessed the environmental impact of the newly developed technologies in an impartial and consistent manner.

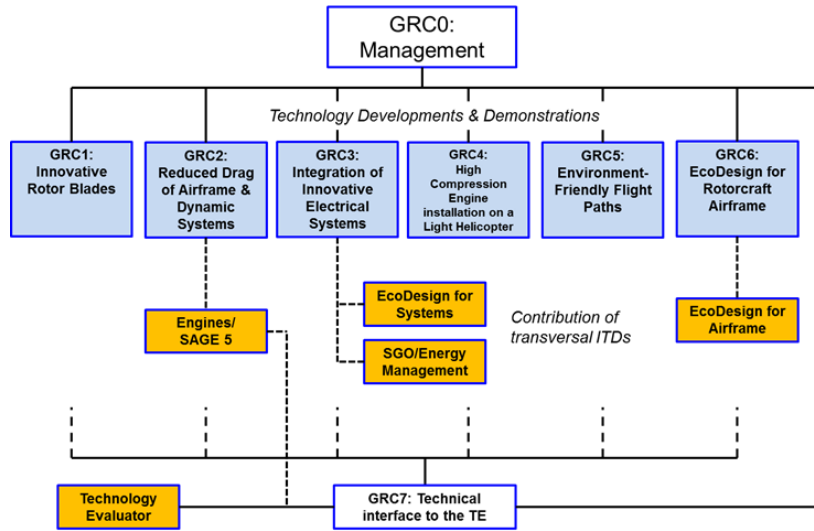


FIGURE 1.5: *GRC* technologies integration scheme [13]

The Clean Sky programme finished on 2016 exceeding the expectations established at the beginning of the programme. Thus, it was decided to continue its research activity under the Clean Sky 2 project.

1.3.4 Clean Sky 2 initiative

With a budget exceeding €4 billion, the Clean Sky 2 project took-off on 2014 as a 10-year project to continue with the development and demonstration of novel technologies for the civil market to cut aircraft and rotorcraft emissions and noise.

In particular, the specific objectives of the project include the reduction of CO_2 and NO_x emissions by 20-30% in comparison with the 'state of the art' aircraft entering into service in 2014, and up to 75% reduction in noise affected communities [14]. To achieve those goals Clean Sky 2 includes new approaches in the areas of large passenger aircraft, regional aircraft, fast rotorcraft, airframes, engines, embedded systems, and small air transport activity.

The structure of the project is summarised in Fig. 1.6. Clean Sky 2 comprises four different elements. The first element is formed by the three Innovative Aircraft Demonstrator Platforms (*IADPs*) responsible for the demonstration and simulation at full vehicle level of Fast Rotorcraft, Large Passenger Aircraft and Regional Aircraft solutions. The second element includes three *ITDs* devoted to the development of airframe designs, engine solutions and on-board systems. As in the Clean Sky programme, the *TE* is also present to assess the environmental impact and benefits of the technologies developed in the *IADPs* and *ITDs*. Finally, the structure of the project is completed by two transverse activities: Eco-Design and Small Air Transport, that integrate the knowledge acquired by the *ITDs* and the *IADPs* for specific applications [15].

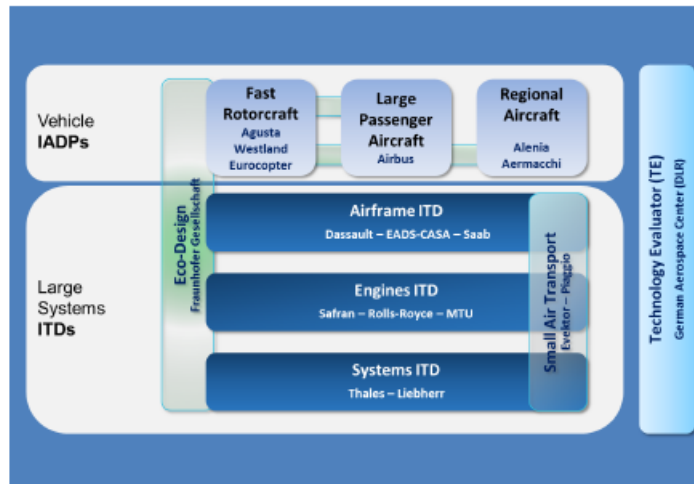


FIGURE 1.6: Clean Sky 2 programme structure [15]

Focusing on the rotorcraft aspect of the project, two revolutionary architectures are envisaged: a tiltrotor (NextGenCTR) and a compound rotor (LifeRCraft) as shown in Fig. 1.7. These two demonstrator aircraft are conceived to combine the speed and range capabilities of fixed-wing aircraft with the vertical take-off and landing, and the hover capacity of rotary-winged aircraft. The development

program of these architectures includes the design, construction and flight test, making use of the technologies developed in Clean Sky 1 and by the Clean Sky 2 *ITDs*. Therefore, both architectures will be equipped with updated rotor blades, improved engine intakes, and innovative electric systems.

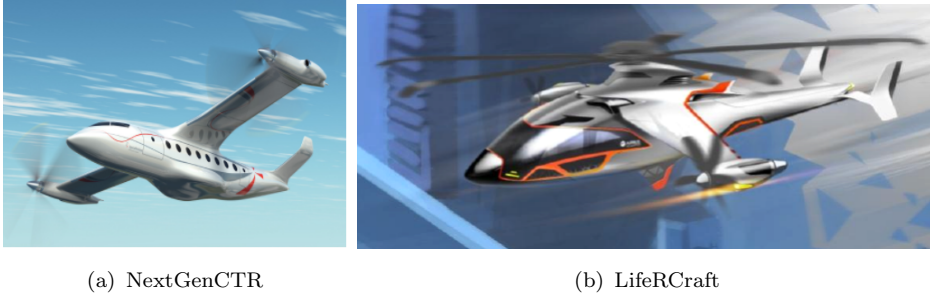
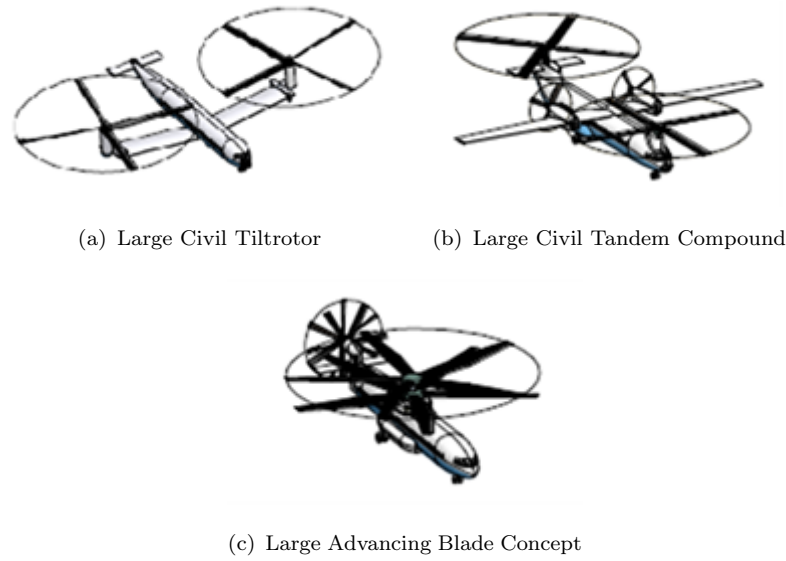


FIGURE 1.7: Clean Sky 2 rotorcraft concepts [16, 17]

1.3.5 Other initiatives

Apart from the research activities launched in Europe, at the other side of the Atlantic, members of NASA and the Army research community are leading rotorcraft innovation programmes, as well, to envision potential rotorcraft applications in 2025 and to identify potential technology barriers. For example, as part of the NASA Heavy Lift Rotorcraft System Investigation, three different configurations are assessed as candidates for a civil Vertical Take-Off and Landing (*VTOL*) transport capable of carrying 120 passengers at a cruise speed of 350 *knots* at 30,000 *ft* altitude (Mach number 0.6) with a range of 1,200 *NM* [18, 19]. These vehicle configurations are shown in Fig. 1.8.

These architectures were identified by the industry as the most promising candidates for civil applications. The Large Civil Tiltrotor has two tilting rotors at the wing tips, a low wing and a horizontal tail. The Large Civil Tandem Compound has two main rotors in a tandem configuration, two pusher propellers for cruise propulsion, a high wing and a horizontal tail. Finally, the Large Advancing Blade Concept is equipped with two main rotors in a coaxial configuration, a rear pusher propeller, and horizontal and vertical tails for cruise trim. The three candidates were compared in terms of mission gross weight, installed engine power, mission fuel, purchase price, and direct operating cost, where the Large Civil Tiltrotor (Fig. 1.8(a)) showed as the most promising concept.

FIGURE 1.8: Large civil *VTOL* transport candidates [19]

The powerplant design of this latter configuration was investigated in [20]. It comprises four turboshaft engines, two in each propulsion nacelle, with a 2-spool configuration plus a free power turbine (Fig. 1.9). A parametric analysis was conducted to find the Specific Fuel Consumption (*SFC*) trend of the engine. Compressor (*OPR*) was varied from 20 to 150 whilst combustor exit temperature ranged from 1,588 to 2,033 *K*. Turbine cooling was considered throughout the engine cycle investigation.

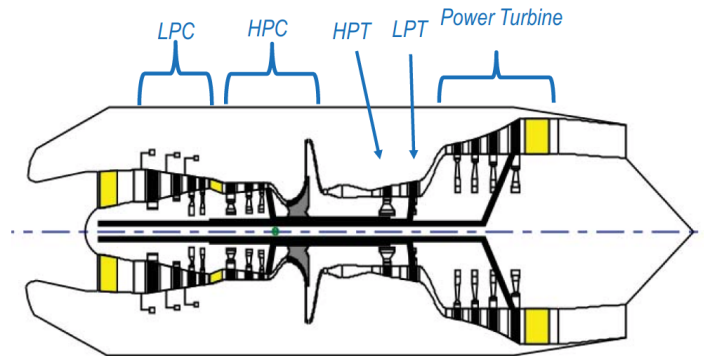


FIGURE 1.9: Large Civil Tiltrotor turboshaft engine layout [20]

The engine cycle selected from the previous analysis resulted in a 37% and 15% reduction in mission fuel and NO_x , respectively. A 10% reduction in vehicle empty weight was also achieved through the implementation of this engine cycle.

Another interesting initiative conducted in the NASA Ames Research Center deals with the design of an all-electric powerplant as replacement for the piston engine used in ultra-light utility class helicopters. The project aimed to reduce fuel consumption, diminish the dependency on hydrocarbon fuels, and contribute to the CO_2 emissions abatement. In particular, the authors used the Robinson R22 Beta II helicopter as baseline (Fig. 1.10), and attempted to replace the conventional piston engine by an all-electric powerplant. The design of the major components of this electric powerplant, namely: the fuel cell, the compressor-expander, the hydrogen storage, the lithium-ion batteries and the electric motor is addressed, and the principal technical barriers identified. More information about this initiative can be found in [21].

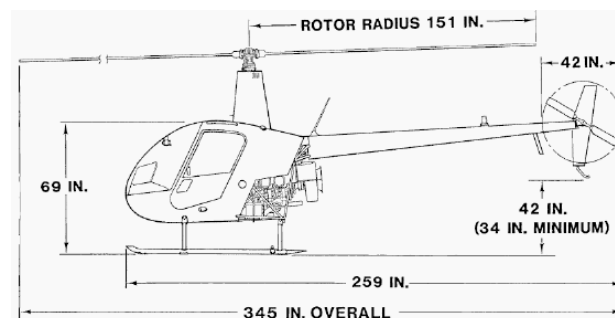


FIGURE 1.10: Robinson R22 Beta II helicopter [21]

1.4 Novel engine cycles for increased efficiency

The research activity described before focused on the integration of new technologies, radical rotorcraft configurations and the installation of alternative powerplants. However, aviation is expected to rely on gas turbines and liquid hydrocarbons well into the 21st century [22–24]. Bearing in mind the maturity level of current aircraft powerplants, significant efficiency improvements need novel engine architectures, such as recuperated, intercooled, intercooled-recuperated and reheated cycles [25]. According to thermodynamics, all the previous concepts have the potential to increase thermal efficiency. A brief description of the working principles of these thermodynamic cycles is included herein:

- **Recuperated cycle**

The rationale behind this concept is the addition of a recuperator device at the exit of the engine, so heat is transferred from the turbine outlet to the

compressor outlet. This way, the remaining heat of the hot exhaust stream is employed to rise the temperature of the air at the inlet of the combustor which reduces the heat addition needed to attain the required Turbine Entry Temperature (TET). As a result, the fuel consumption is reduced, whilst the useful work remains unchanged. However, the installation of the recuperator comes with a penalty in engine weight and increased pressure losses that must be thoroughly accounted.

- Intercooled cycle

In this concept, the compression process is split into two or more sections and heat is extracted at the exit of each of the compressors, so the work required by the subsequent compressor is reduced. As a result, the total compression work is lessened, and the useful work increases. However, the drawback of intercooling is that, for a fixed compression ratio, the air delivered into the combustor will be at a lower temperature, and thus, a higher heat addition is required to achieve the desired TET . This problem may be solved by combination of intercooling with recuperation, so the extra heat required is obtained without a fuel consumption penalty. However, the reduced temperature of the air delivered by the compressor has also a positive side effect on the sizing of the turbine cooling flows. Due to the lower temperature of the cooling gas, the amount of coolant required in the hot section of the engine may be potentially reduced, which ultimately translates in a higher thermal efficiency.

- Reheated or sequential combustion cycle

Reheating has been traditionally used for thrust augmentation in military gas turbine engines. In such applications, the sequential combustion takes place at a lower pressure compared with that at the main burner, and so, the increase in engine thrust is attained with a reduction in the overall thermal efficiency. However, if the additional heating process is moved within the turbines, where the pressure is higher, the cycle efficiency can be potentially increased. In fact, in the Carnot cycle, which sets the upper limit on the achievable efficiency, the expansion occurs at constant temperature. Therefore, the ideal option is to organize the heating process so that the total temperature in the turbine section remains constant. A practical method to

implement this concept in real engines is the sequential combustion. Similarly to intercooling, reheating may be combined with recuperation for further thermal efficiency enhancement. In this case, as a result of the heat addition in a latter expansion stage, the exhaust stream may be hotter than in the conventional engine, and so, this increased temperature can be used to preheat the gas at the inlet of the combustor to reduce the fuel input at the main burner. The implementation of the reheating comes with a penalty in engine mass and complexity, an additional pressure loss in the auxiliary burners and a redistribution of the engine cooling flows, aspects that have to be considered in the final thermal efficiency assessment.

Finally, it is worth to mention herein another research path for helicopter mission fuel reduction, which is based on the variable-speed main rotor concept. Garavello *et al* [26] showed that the power consumed by the helicopter can be reduced by almost 30% if the rotational speed of the main rotor is adapted to the forward flight speed. However, they also found that the variable rotational speed of the free turbine has a detrimental effect on engine *SFC* which completely cancels the helicopter power savings. Therefore, they point out that to exploit the potential benefits of an optimum-speed rotor system, it is necessary to design a wide-speed-range turboshaft engine.

1.5 Aim & Objectives

The main goal of this project is the exploration of new gas turbine engine designs that can bring performance benefits in comparison with current rotorcraft powerplant technology. To that end, the performance of the conventional turboshaft engine is studied and optimised in order to reduce the environmental footprint of the helicopter in terms of fuel consumption and NO_x emissions. Apart from the conventional turboshaft architecture, the merits of novel cycle engines are to be assessed as alternative candidates for rotorcraft propulsion. In particular, the reheated cycle engine is investigated herein. Although reheated cycles have been recently implemented in stationary gas turbines for power generation, and some investigations about their possible applications in turbofan engines are available in the public literature, there is a general lack of knowledge about the potential of the reheated cycle in rotorcraft turboshaft engines. Therefore, one of the main aims

of the present project is to explore different reheated cycle architectures, evaluating not only the potential benefits in thermal efficiency, but also addressing the integration of the powerplant in the helicopter in terms of the weight fluctuation introduced by the conceptual engines.

In order to fulfil the aforementioned targets, the following objectives were set:

- Create and validate a turboshaft engine model to be used for the performance analysis of the conventional engine and the reheated cycle architectures.
- Implement a blade cooling sizing model able to adapt the engine cooling flow needs to the cycle parameters i.e. total temperature and pressure, and mass flow at compressors and turbines. This model must capture the effects that the additional burners have in the engine cooling flow distribution so the merits of the reheated cycles can be thoroughly assessed.
- Develop a model to simulate the NO_x emissions of the conventional engine combustor. The model must be able to capture the impact of the engine design variables on NO_x formation.
- Develop a validated tool to estimate the mass of the different powerplant architectures. The tool must be sensitive to the changes in engine cycle parameters, has to be able to account for the weight added by the extra burners, and must keep the computational cost low so it can be used for design space exploration and optimisation purposes.
- Integrate all the tools and models employed as a unique multidisciplinary framework with an input-output structure. This framework is to be coupled with a multi-objective optimiser in order to identify the optimal engine configurations in terms of efficiency, and weight, and also in terms of NO_x emissions for the conventional engine.
- Create a rotorcraft model representative of an existing helicopter design in order to analyse the performance of the optimal candidate engines over a realistic flight mission.

1.6 Contribution to knowledge

The main contribution of this piece of work is the assessment and optimisation of the reheated cycle turboshaft engine performance, which constitutes the first attempt to analyse the potential benefits of reheated cycle engines for helicopter propulsion applications. Not only the thermal efficiency and its impact on mission fuel and CO_2 emissions are investigated, but also the effect of the auxiliary burners on powerplant weight is examined. For the thorough assessment of the reheated cycle engines performance, an exhaustive cooling flow sizing routine was also integrated. This constitutes another novelty since previous work in reheated engines was conducted with little or no concern regarding cooling flows fluctuation and its subsequent impact on engine performance.

The estimation of the turboshaft engine weight demanded an upgrade of the software ATLAS in order to include the distinct features characteristic of turboshaft engines in the weight computation. This task was undertaken by the author, which resulted in a more versatile tool that can be now utilised for the weight estimation of turbofan, turboshaft and power generation gas turbine engines. Once the ATLAS update was completed, different surrogate models were created in order to alleviate the high computational cost of the engine weight computation through ATLAS. This 'hybrid' method constitutes also a novel contribution to knowledge, since it offers a completely new approach for a fast and accurate weight prediction for gas turbine engines.

In parallel with his main research activity, the author also contributed to the Clean Sky project as an active member of the *GRC TE*. During this collaboration, the author assessed the impact of innovative rotorcraft technologies developed by the Clean Sky partners on rotorcraft fuel burn, and gaseous and noise emissions. Besides, the author assisted in the elaboration of several reports containing the main results and findings produced by the *TE*. All this activity was developed in close collaboration with the Clean Sky partners, namely, Safran Helicopter Engines (former Turbomeca), Leonardo Helicopters, Airbus Helicopters and the Netherlands Aerospace Centre.

1.6.1 Publications

During his research period, the author has contributed to the following publications:

- **Journal articles**

- Ortiz-Carretero, J., Castillo Pardo, A., Pachidis V., and Goulos, I., 2018, *Impact of Adverse Environmental Conditions on Rotorcraft Operational Performance and Pollutant Emissions*, Journal of Engineering for Gas Turbines and Power, Vol. 140, No. 2.
- Enconniere, J., Ortiz-Carretero, J., and Pachidis, V., 2017, *Mission Performance Analysis of a Conceptual Coaxial Rotorcraft for Air Taxi Applications*, Aerospace Science and Technology 69, p. 1-14.
- Enconniere, J., Ortiz-Carretero, J., and Pachidis, V., 2017, *Mission Optimisation of a Conceptual Coaxial Rotorcraft for Taxi Applications*, Aerospace Science and Technology 72, p. 14-24.

- **Conference articles**

- Ortiz-Carretero, J., Castillo Pardo, A., Pachidis V., and Goulos, I., *Assessment of the Effect of Environmental Conditions on Rotorcraft Pollutant Emissions at Mission Level*, GT2017-63440, Proceedings of ASME Turbo Expo 2017, Charlotte (North Carolina, USA).
- Ortiz-Carretero, J., Enconniere, J., Pachidis, V., Stevens, J., Smith, C., Thevenot, L., and d'Ippolito, R., *Clean Sky Technology Evaluator Green Rotorcraft Mission Environmental Performance Analysis*, AEGATS2016_12, AEGATS 2016, Paris (France).
- Enconniere, J., Ortiz-Carretero, J., Goulos, I., Pachidis, V., Smith, C., Stevens, J., d'Ippolito, R., and Thevenot, L., *The Impact of Clean Sky Technology on Future 3500 lb Single Engine Light Rotorcraft*, ISABE 2017, Manchester (UK).

1.7 Thesis structure

The Thesis has a conventional structure with introduction, literature review, methodology, models validation, results and discussion, and conclusions and future work sections.

The first chapter introduces the current context of the project, and briefly describes several initiatives and alternatives deployed to tackle the targets imposed by the aviation industry with special emphasis on rotorcraft technology. The aim and the list of project objectives are also included in this section, together with the contribution of the author to knowledge.

The second chapter, devoted to the literature survey accomplished by the author, includes a review on the following topics:

- Helicopter flight dynamics and performance simulation
- Gas turbine engine performance simulation
- Gas turbine engine gaseous emissions modelling
- Gas turbine engine weight modelling
- Optimisation of turboshaft engines for rotorcraft applications
- Reheated cycle engine concept

The *Methodology* section covers the detailed description of the simulation models developed, and their integration within the simulation framework. This chapter has four parts, each one devoted to a specific aspect of the helicopter-powerplant system simulation:

- Helicopter flight dynamic performance simulation in *HECTOR*
- Powerplant performance simulation in TURBOMATCH
- Powerplant weight modelling in ATLAS
- Combustor NO_x emissions modelling in HEPHAESTUS

The different updates introduced in ATLAS for the weight estimation of turboshaft reheated engines are fully explained within this section, as well.

The exhaustive validation and verification of the tools and models developed is included in the fourth chapter.

Once the methods developed by the author are explained and validated, the results achieved during this project are showcased in the *Results & Discussion* section. For a better understanding, the chapter is divided in the following parts:

- The parametric analysis of the powerplant performance. In this initial section, the sensitivity of each engine architecture to changes in the design variables is first studied.
- The optimisation of the rotorcraft powerplant. The trade-off between engine size and thermal efficiency is assessed in this section for each conceptual engine. The generation of surrogate models for fast engine weight estimation is also included herein.
- The assessment of the optimal engine configurations at mission level. From the previous optimisation analysis, the most optimal solutions are evaluated over a representative flight mission to calculate their impact on total mission fuel.
- The optimisation of the conventional engine at mission level in terms of fuel consumption and NO_x emissions. The trade-off between these two parameters is highlighted, and an optimal engine design is suggested as a suitable solution for simultaneous mission fuel and NO_x reduction.

To conclude the report, a final chapter is included to summarise the key findings of the project, and enumerate the limitations of the work with suggestions and recommendations for future work.

Chapter 2

Literature Review

This section summarises the literature exploration completed by the author as the starting point of the study presented in this Thesis. First, a brief review of simulation of rotorcraft performance and the tools employed for that purpose is presented. Secondly, the different aspects of gas turbine engine simulation studied by the author are covered, namely, performance simulation, combustor emissions production and powerplant weight modelling. After that, a general review of the previous work completed in turboshaft engines optimisation for rotorcraft applications is presented. Finally, a section discussing the principles of the reheated cycle and the most notable studies carried out so far on this topic is included.

2.1 Rotorcraft performance simulation

The nature of the studies completed in this project requires the exhaustive calculation of the power consumed by the helicopter flying at a certain flight condition, and the fuel flow and NO_x emissions produced by the engine at that operating point.

The helicopter is an intricate combination of several subsystems that interact with each other. The computation of the different forces and moments coming from fuselage, empennage, tail rotor, powerplant and main rotor (Fig. 2.1) have to be thoroughly accounted in order to derive the final trim state of the helicopter, and thus, the power demanded to the powerplant to flight at that equilibrium condition.

Once the shaft power required per engine is derived, the fuel flow and the gas pressure and temperature at each engine station at that particular operating condition can be computed. This information is then employed to compute the emissions produced by the engine combustor at the given flying state.

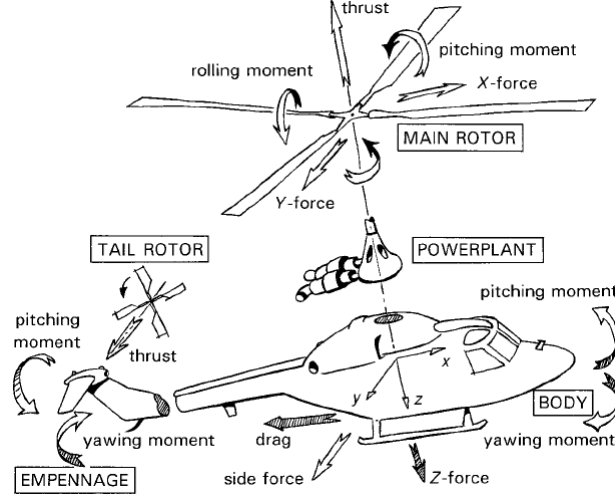


FIGURE 2.1: Decomposition of helicopter forces and moments [27]

Therefore, the complete simulation of the helicopter performance is a multidisciplinary problem that requires the implementation of different routines that solves the rotorcraft flight mechanics, the thermodynamic engine performance and the combustor emissions production in an integrated manner. In the next subsection, two representative rotorcraft simulation frameworks employed for the thorough assessment of conventional helicopter performances are reviewed.

2.1.1 Rotorcraft simulation frameworks

2.1.1.1 *Phoenix* platform

Phoenix (Platform Hosting Operational & ENvironmental Investigations for Rotorcraft) was designed by the Clean Sky *GRC ITD* for the assessment of the performance and environmental benefits of the conceptual rotorcraft technologies developed by the *GRC* partners. *Phoenix* is capable of calculate the fuel burn, pollutant emissions and ground noise levels of a given rotorcraft platform. An exhaustive description of the tool can be found in [28], herein a brief description of the framework architecture and the modules integrated is provided. The tool comprises the following modules:

- A routine to solve the helicopter flight mechanics (EUROPA).
- A deck called *SHE* (Safran Helicopter Engine) for the simulation of the engine performance and the combustor emissions.
- A module to simulate the auxiliary power and bleed off-takes required by the secondary systems called *RMEM* (Rotorcraft Mission Energy Management).
- A routine to estimate the on-ground levels of perceived noise (HELENA).

The mentioned tools are linked together with a flexible design environment tool called OPTIMUS. A general overview of the integration of the different modules is presented in Fig. 2.2. For the simulation of the complete helicopter mission, the flight profile is divided in a series of sampling points based on time steps declared by the user. The calculation is started with an initial guess for the fuel supply in order to establish the helicopter Take-Off Weight (*TOW*). At each sampling point, EUROPA computes the power required by the rotors and updates the vehicle position following the mission profile. The output from EUROPA is fed into the *SHE* module to calculate the performance of the engine including the secondary system power and bleed off-takes estimated by the *RMEM* routine. The fuel burnt during this time step is subtracted from the helicopter mass to simulate the progressive weight reduction over the mission. Once this procedure is applied to all the sampling points, the obtained mission fuel is compared with the initial guess. If the discrepancy is above a certain margin, the initial fuel supply guess is updated and the whole process is repeated. Once convergence is reached, the mission emissions levels of CO_2 and NO_x are evaluated and the aerodynamic results from EUROPA are transferred to HELENA for the noise footprint calculation.

PhoeniX was successfully utilised in the environmental impact assessment of single and twin-engine helicopter configurations as part of the Clean Sky programme [30].

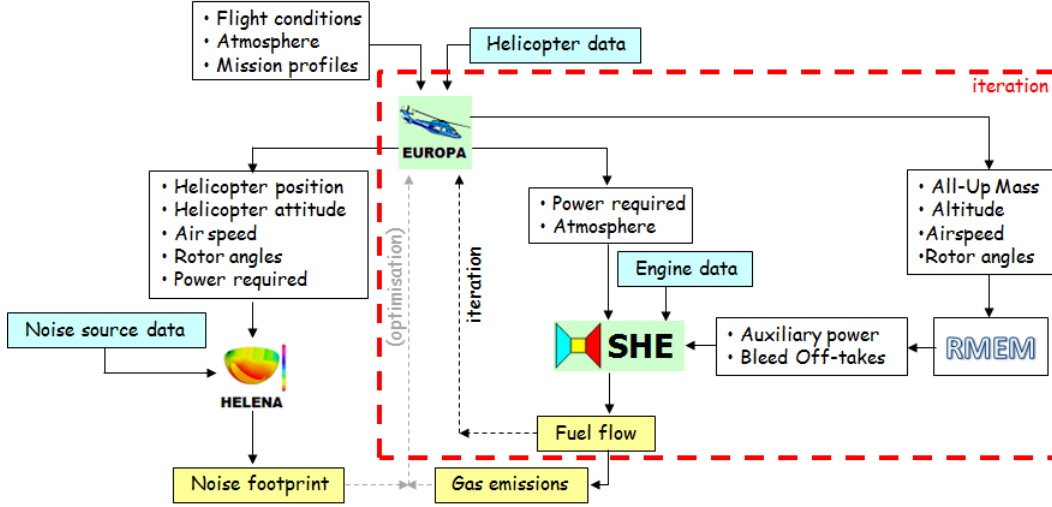


FIGURE 2.2: *Phoenix* architecture scheme, adapted from [29]

2.1.1.2 *HECTOR* platform

HECTOR (HEliCopTer Omni-disciplinary Research platform) was developed by Goulos [31] at Cranfield University. A detailed description of the modelling principles and the routines integrated in the code can be found in the cited reference. *HECTOR* is a standalone simulation framework with an integrated range of capabilities applicable to helicopter performance evaluation and mission analysis. This code can be applied to helicopter trim (steady state) performance as well as dynamic conditions simulation to assess take-off and landing performance. Figure 2.3 shows *HECTOR* structure with the inputs required by the code highlighted in orange, and the outputs in green. A module is implemented to process the mission profile which basically truncates the flight path into discrete segments and defines the flight parameters and ambient conditions for the rest of the framework tools. The flight mechanics of the helicopter are solved by the helicopter performance model in a similar fashion *EUROPA* does in *Phoenix*. The outcomes of this module are transferred to *TURBOMATCH*, which is the software employed for the engine performance simulation. *TURBOMATCH* calculates the engine fuel flow and the thermodynamic parameters of the engine which are fed into *HEPHAESTUS* for the computation of CO_2 and NO_x emissions. A thorough description of *TURBOMATCH* and *HEPHAESTUS* is included in Sections 2.2.1 and 2.2.2, respectively. Note that, similarly to *Phoenix*, the fuel consumed by the helicopter powerplant is internally employed by the code to update the weight of the rotorcraft during the mission. The final fuel result is compared with the initial value guessed and, if they do not match, the process is repeated with an updated

helicopter initial mass. This iterative calculation is executed until convergence is reached.

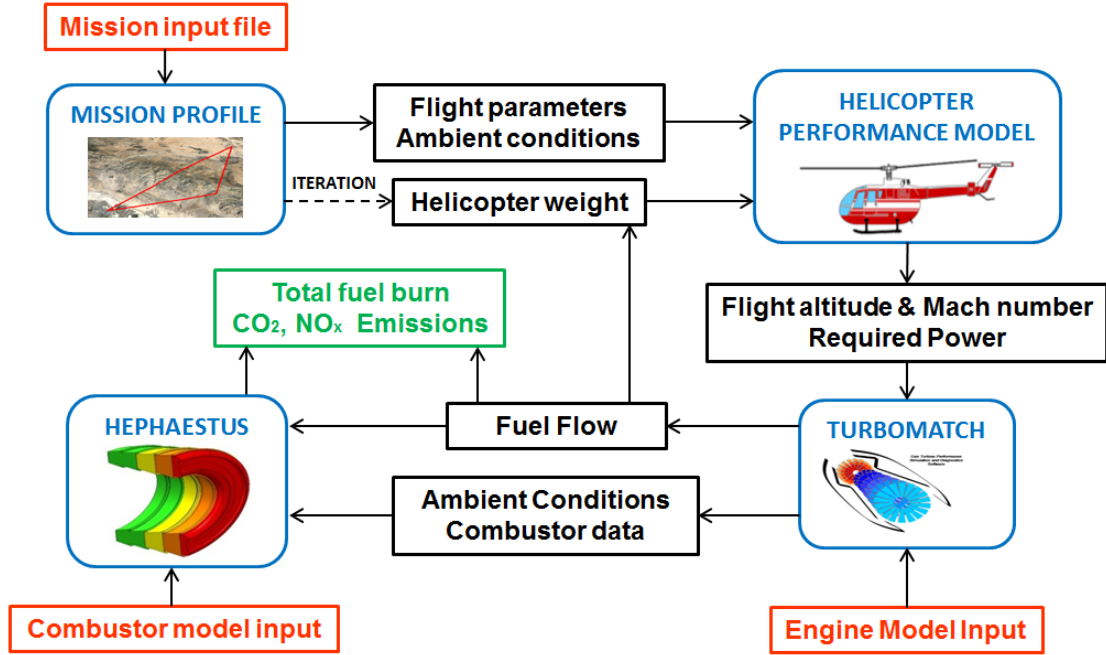


FIGURE 2.3: *HECTOR* architecture scheme, adapted from [32]

HECTOR has been widely used in the past for the assessment of rotorcraft performance at mission level [33], the optimisation of rotorcraft flight paths [28], and the optimisation of conventional and conceptual rotorcraft powerplants [34, 35]. This proves the reliability of *HECTOR* to be used in rotorcraft powerplant optimisation studies, and so it is the tool to be used by the author in the present study.

Although the scope of the present work is the design and optimisation of turboshaft engines, a brief description of the helicopter performance code *HECTOR* used to calculate the power requirement is included herein. A more detailed description can be found in refs. [36, 37].

HECTOR models the helicopter as a series of components: fuselage, main rotor, tail rotor, and vertical and horizontal stabilisers, which mutually interact with each other. The fuselage, in turn, is modelled as a 6 degrees of freedom rigid body. The first step of the helicopter power computation is the trimming process that defines a state of equilibrium between the overall forces and moments acting at a given set of flight conditions (airspeed, altitude, helicopter mass, climb rate, sideslip and turning angle). A linear version of Euler's equations is used to

describe the kinematics of the rigid body. These equations are solved iteratively, in order to derive the combination of control inputs (collective pitch, longitudinal cyclic, lateral cyclic and tail rotor collective pitch) corresponding to the equilibrium state. To simulate the aerodynamic performance of the main rotor, its blades are modelled as elastic bodies connected to the hub with three degrees of freedom (flapping, lagging and feathering); the aerodynamic performance of the main rotor is determined then using standard blade element theory. The rotor disc is divided in the azimuthal and radial directions into a series of blade elements, and the application of the corresponding aerofoil information at each radial station provides the aerodynamic loading on the blade. In the calculation, the rotor induced flow, and the unsteady aerodynamics and elastic kinematics of the blade are properly modelled. Besides, rotor-fuselage interaction and ground effect are also catered by the code. A similar approach but more simplified is used to simulate the aerodynamic performance of the tail rotor. Finally, the horizontal and vertical stabilisers are modelled using a set of maps built with experimental data. Note that the flow field at the stabilisers is corrected to account for distortions induced by the main rotor. Once the trim state of the helicopter is derived and the aerodynamic coefficients (thrust, torque and power coefficients) of each rotor are known, the power consumed by the rotors is computed. The mechanical losses incurred in the transmission system are modelled through two gearbox efficiency values (one for each rotor) declared by the user. The corrected power value is then, introduced in TURBOMATCH. Note that the current version of the code neglects the power consumed by the secondary systems of the helicopter.

2.2 Gas turbine engine simulation

The performance assessment described in this Thesis requires the evaluation of the thermal efficiency, NO_x emissions and weight of the different conceptual engine architectures analysed. In this section, different approaches and methods to simulate the mentioned aspects of the gas turbine performance are reviewed with special attention to the tools to be used by the author.

2.2.1 Gas turbine engine performance modelling

In order to estimate the thermodynamic parameters and the thermal efficiency of the different engine configurations assessed, a zero-dimensional¹ computational tool is required that can accurately predict the performance of the baseline engine at the same time it is flexible enough to accommodate the novel cycle features into the gas path. An exhaustive list of zero-dimensional tools for the numerical simulation of gas turbine performance was elaborated by the Research and Technology Organisation (*RTO*) of NATO [38]. Herein, a brief review of some of these tools will serve to offer a general idea about the numerical simulation of gas turbine engines performance.

One of these simulation environments is the Gas Turbine Simulation Program (*GSP*) developed by the National Aerospace Laboratory of the Netherlands [39]. Steady-state and transient simulation of different types of gas turbine architectures can be accomplished by the particular arrangement of component bricks in a visual interface as the one shown in Fig. 2.4.

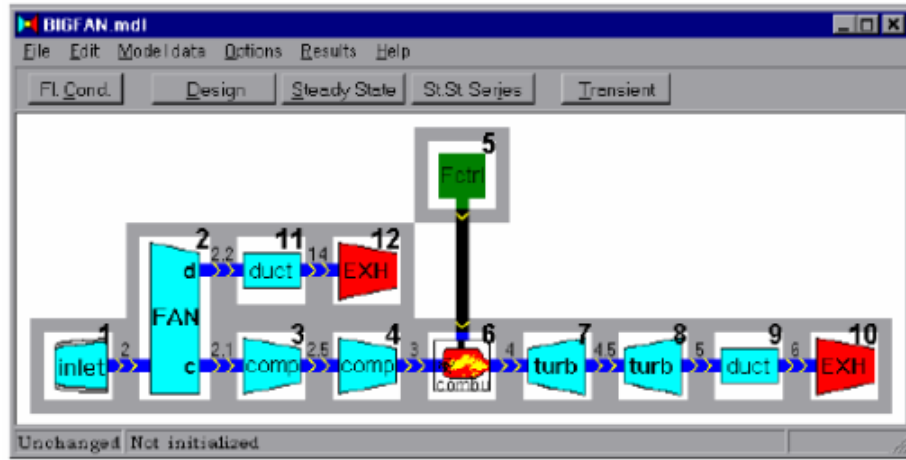


FIGURE 2.4: Turboprop engine modelling in *GSP* [38]

Effects such as installation losses, engine components deterioration and /or malfunctions of control and other subsystems can be simulated by the tool. Apart from that, *GSP* is a flexible simulation platform with a large library of component models that also allows the user to introduce his own custom components models. This software has been used in the past in the assessment of rotorcraft powerplant performance [26].

¹The flow properties are averaged over the flow cross section areas at the inlet and outlet surfaces of the component.

Another widely extended performance simulation code is GasTurb, first developed by Kurzke [40]. The main feature of this software is its user-friendly graphical interface. By using predefined engine configurations the mathematics of the problem are hidden from the user, and so, it can be accessed by a wider audience. A vast library of predefined configurations is implemented in the code for both jet propulsion and power generation. Some examples are reproduced in Fig. 2.5.

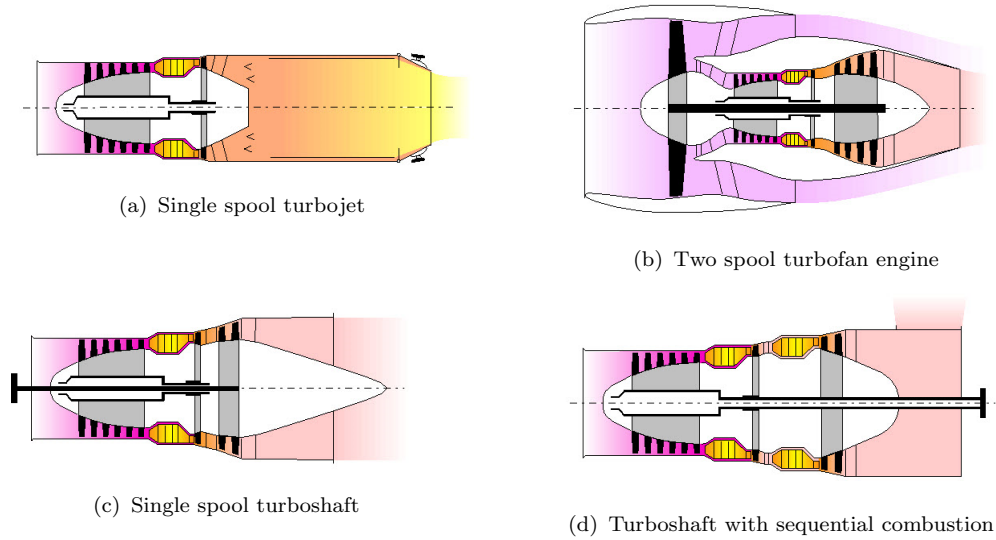


FIGURE 2.5: Examples of predefined engine configurations in GasTurb [40]

Apart from the many engine configurations that can be simulated in GasTurb, this software allows to calculate the performance of the engine at Design-Point (*DP*) and Off-Design (*OD*) conditions, as well as at transient. Besides, its graphical interface allows the user to easily complete parametric analysis and comparative performance studies.

Finally, the tool to be employed in the current study, also included in the *RTO* list, is the Cranfield University in-house gas turbine performance simulation code TURBOMATCH, first developed by MacMillan [41]. This software is a component-based gas turbine simulation environment, where the different engine components are modelled as bricks. These bricks are then combined by the user to recreate the engine structure desired. An example of the brick modelling of a one-spool turbojet engine is reproduced in Fig. 2.6. This component-based approach is similar to the one implemented in *GSP*.

Some of the key points of TURBOMATCH are listed below:

- Long standing and validated tool.

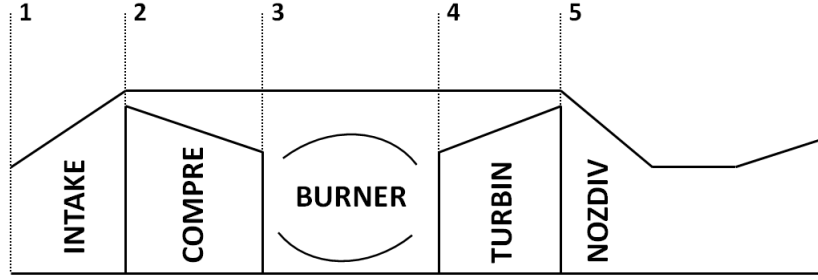


FIGURE 2.6: Turbojet engine brick modelling in TURBOMATCH [42]

- Capability to simulate a wide range of civil/military aero and land based engines.
- Steady state/transient and design/off-design engine performance analysis.
- Ability to include air bleedings for turbine cooling and/or cabin use.
- Capability to simulate the effects of turbomachinery components deterioration.
- Capacity to implement and simulate inter-stage turbine reheating. This upgrade in the code was implemented by Pellegrini *et al* [43].

TURBOMATCH has been employed in several studies for the prediction of *DP* and *OD* gas turbine engine performance [44, 45], and it is the engine performance simulation tool implemented in *HECTOR* (Fig. 2.3).

2.2.2 Gas turbine engine gaseous emissions modelling

As mentioned by Lefebvre *et al* [4], the exhaust of an aircraft gas turbine is composed of carbon monoxide (CO), CO_2 , water vapour (H_2O), unburned hydrocarbons (UHC), smoke (mainly carbon particles), NO_x , and excess atmospheric oxygen and nitrogen. In stationary gas turbine engines, other pollutants of concern are sulphur oxides (SO_x), formed when the sulphur-containing compounds in the residual fuels these engines tend to burn react with the air oxygen. From this list, only CO_2 , water vapour and NO_x are emitted in appreciable amounts, whilst CO , UHC and smoke emissions have been significantly reduced at all power conditions in modern engines. Therefore, the latter components are not further considered in the emissions analysis.

Although CO_2 and water vapour cannot be considered as pollutants, since they are the products of complete combustion of hydrocarbon fuels, they still contribute to the greenhouse effect. The emission of these products into the atmosphere can only be reduced by burning less fuel. Therefore, the engine fuel flow provided by TURBOMATCH will be used as an indicator of CO_2 and water vapour emissions.

On the other hand, NO_x (NO and NO_2) constitutes a combustion pollutant *per se*. The production of NO_x is not only related with the fuel consumption, but the conditions in the engine combustor, such as air pressure and temperature, combustor residence time and fuel atomization [4]. The NO_x formation is a complex process that involves four different production mechanisms: thermal NO , nitrous oxide mechanism, prompt NO and fuel NO . The elaboration of each of these methods is avoided herein for conciseness, but more information can be found in [4].

Different approaches have been followed in the past to model NO_x emissions, ranging from simplistic empirical and semi-empirical models and simplified physics-based stirred reactor models to more complex CFD based simulations.

The empirical and semi-empirical models are based solely upon changes in combustor inlet parameters and strongly depend on the design features and geometry of the combustor employed to derive the correlations. Hence, they are not able to account for changes in combustor geometry and fuel properties (temperature and fuel low heating value). Since empirical methods do not require a great number of inputs, they may be a good choice when little information about the combustor being analysed is available, or when a low computational cost solution is required. Odgers and Kretschmer [46] compiled some of the empirical approaches employed to measure NO_x emissions in gas turbines. These methods are summarised in Table 2.1. The correlation parameters employed in these methods include Combustor Outlet Pressure (COP), Combustor Inlet Temperature (CIT), combustor air mass flow (w_b), and Fuel to Air mass Ratio (FAR).

Also within this category, the Swiss Federal Office of Civil Aviation ($FOCA$), in order to fill the gap in rotorcraft emissions assessment, derived a set of empirical functions to calculate the fuel flow and emissions (NO_x , UHC , CO and carbon particles) of representative rotorcraft engines [47]. The empirical functions were developed from experiments conducted on a wide range of helicopters. However, this method has a lack on accuracy of $\pm 15\%$ for fuel prediction and an estimated

TABLE 2.1: Review of empirical methods for NO_x estimation [46]

Method	Year	Correlation factors
Marchiona <i>et al</i>	1973	COP, CIT, w_b, FAR
Davis <i>et al</i>	1973	COP, CIT, w_b, FAR
Vermes	1974	COP, CIT, w_b, FAR
Marshall	1975	COP, CIT, FAR
Cohen	1974	COP, CIT, w_b, FAR
Diehl-Biaglow	1974	COP, CIT, w_b, FAR
Mularz <i>et al</i>	1975	COP, CIT, w_b, FAR
Sullivan	1976	COP, CIT, w_b, FAR

error of a factor of two or more for emissions calculation. The correlation derived for the estimation of NO_x emissions is as follows:

$$EI NO_x \approx 0.2113 \cdot SHP^{0.5677} \quad (2.1)$$

Where EI is the Emission Index in $g/kg \text{ fuel}$, and SHP is the shaft horsepower in kW . It is observed that the expression derived by *FOCA* is uniquely a function of engine shaft power, which implies that the fluctuation of environmental conditions from the experiment conditions are not accounted and neither the variation of engine cycle parameters, such as combustor pressure and temperature. Therefore, this approach is too simplistic to be used in an exhaustive engine design space exploration.

On the other hand, CFD simulations represent the most complex approach for NO_x emissions estimation. The numerical simulation of the turbulent reacting flow inside the combustor comes with a high computational cost. This high computational time prevents the use of the numerical method in a simulation environment oriented to the optimisation of rotorcraft engine cycles. Besides, the CFD approach requires a detailed 3-D model of the combustor geometry, which is not available at the preliminary stage of the engine design.

Finally, physics-based stirred reactor models can be seen as a trade-off between the two previous approaches. These models use a set of assumptions to deal with the turbulent flow reacting within the combustor, and enable to accurately compute the time-dependent chemistry of pollutants formation. In fact, physics-based stirred reactor models are able to account for changes in engine cycle parameters

and combustor geometry, and offer an estimation of NO_x emissions in a computational effective cost manner. Therefore, this approach is the most suitable to be used in the multidisciplinary optimisation study proposed by the author.

The physics-based stirred reactor model to be employed is the Cranfield University in-house software for gas turbine emissions prediction developed by Celis and called HEPHAESTUS [48, 49]. A full description of the code theory and validation is provided in the cited references. In HEPHAESTUS the combustor is divided in four regions, *i.e.* Flame Front (FF), Primary Zone (PZ), Intermediate Zone (IZ) and Dilution Zone (DZ), as observed in Fig. 2.7. The dimensions of each region are defined by the user to set the basic combustor geometry, together with the operating conditions, namely, ambient air conditions, air pressure and temperature at combustor inlet, fuel properties, and air flow distribution within the combustor liner.

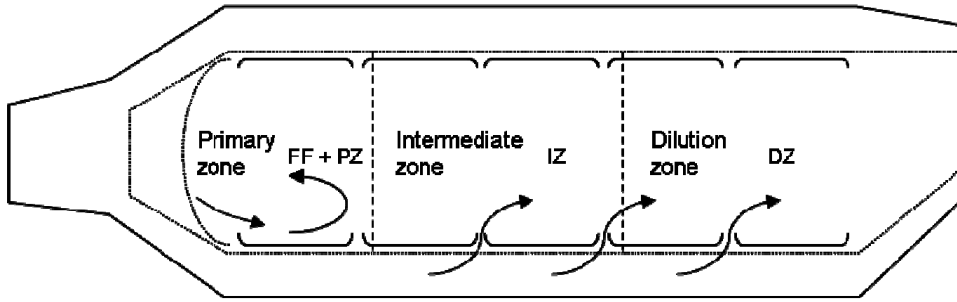


FIGURE 2.7: Conventional combustor scheme in HEPHAESTUS [49]

HEPHAESTUS has been satisfactorily used in past conceptual design studies and environmental assessments for fixed-winged [49] and rotating-winged aircraft [33].

The current version of the software assumes a standard composition for the reacting air, which is a valid assumption for conventional gas turbine combustors. However, this premise may limit the application for the inter-turbine burners NO_x estimation, as the reacting air in these components includes the products of previous combustions in its composition. In fact, the combustion using this vitiated air as reactant has been proven beneficial for the NO_x emissions reduction. The higher heat capacity of the CO_2 and H_2O gases produced during combustion helps to lower the flame temperature, which in turns reduces the production of thermal NO_x [50].

Therefore, HEPHAESTUS as is cannot be applied to the emission analysis of the reheated cycle architectures. Instead it will be only applied for the calculation of

NO_x emissions produced by the conventional cycle engine configurations, whilst the required HEPHAESTUS upgrade is suggested as future work.

2.2.3 Gas turbine engine weight modelling

The weight estimation of gas turbine engines at a preliminary design stage is not a straightforward task as little information, if any, about the engine components geometry is available at that design step. However, in the aerospace industry, a first evaluation of the engine weight and size is paramount in order to deal with the integration of the powerplant with the rest of the aircraft components, and to measure the effect of engine weight in the whole aircraft/rotorcraft performance. Traditionally, two types of strategies have been adopted to tackle this problem, *i.e.* Whole Engine Based (*WEB*) methods that estimate the weight of the whole engine as a unique system, and Component Based (*CB*) methods that predict the total weight by calculating individually the contribution of each engine component.

The *WEB* approaches were developed to provide a fast engine weight prediction during the conceptual design stage, when there is a general lack of data regarding the engine components design. These methods are based on empirical models built with an existing engine database that correlates a set of engine design parameters with its total weight. Therefore, the *WEB* approaches can only be applied on engines that are similar to the database cases and so, they cannot be used for the weight estimation of novel cycle engines. Several existing *WEB* approaches have been reviewed by Lolis *et al* [51] and listed in Table 2.2. It is observed that all of them rely on a short set of engine parameters (no more than four) available at the conceptual design stage, such as engine mass flow (w_{eng}), net thrust (FN), By-Pass Ratio (BPR), OPR or fan diameter (D_{fan}). It is noted, as well, that the methods reviewed are oriented to either turbojet or turbofan engines weight estimation, which evinces the lack of *WEB* estimation methods for rotorcraft turboshaft engines.

On the other hand, *CB* methods estimate the engine total weight as the sum of the different components weight. The lack of publicly available information about the weight of engine components inhibits the development of empirical *CB* methods similar to the *WEB* approaches in Table 2.2. Instead, *CB* methods rely on the preliminary design of the engine gas path that is then used to estimate the volume of the different components. This approach offers a more accurate weight estimate

TABLE 2.2: Review of *WEB* approaches [51]

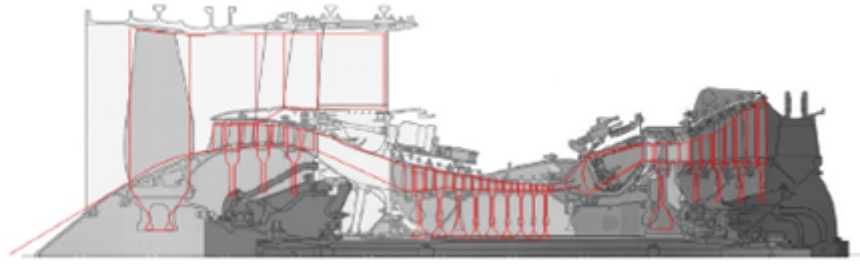
Method	Year	Correlation factors	Gas Turbine Class
Whitehead & Brown	1953	w_{eng}	Turbojet
Pennington	1959	FN	Turbojet
Torenbeek	1975	FN, w_{eng}, BPR, OPR	Turbofan
Raymer	1989	FN, BPR	Turbofan
Jenkinson <i>et al</i>	1999	FN, BPR	Turbofan
Svoboda	2000	FN	Turbofan
Guha <i>et al</i>	2012	D_{fan}	Turbofan

at the price of an increased complexity, as more inputs are needed, and an extended computational time. The modular approach of *CB* methods is appropriate for the weight estimation of novel engine concepts, since the weight variations due to changes in engine cycle parameters are properly captured. Besides, the weight of novel engine components, such as intercoolers or heat exchangers, can be added into the final engine weight estimate if appropriate design methods are integrated in the *CB* method routine.

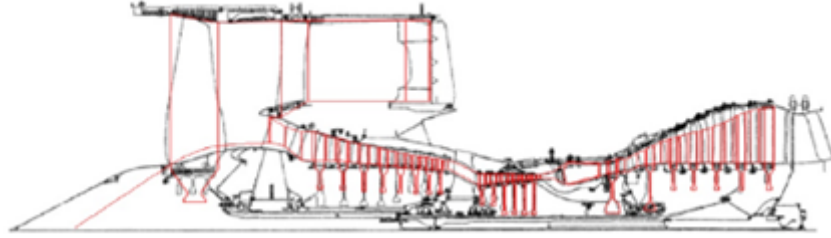
Similarly to the *WEB* approaches, Lolis *et al* [51] examined the main *CB* methods developed until the date. This list only includes two approaches, *i.e.* the Sagerser method (1971) [52] and the NASA *WATE* (Weight Analysis of Turbine Engines) method (1977) [53]. These approaches are mentioned herein for completion, more information can be found in the cited references.

The nature of the turboshaft engine design and optimisation to be completed by the author requires the use of a *CB* method to account for the modifications in engine cycle parameters, and especially for the implementation of the auxiliary inter-turbine reheaters. The *CB* approach to be used is *ATLAS*, first developed by Lolis [54] at Cranfield University. This code, built upon the NASA *WATE* principles, was aimed to provide a robust, accurate, flexible and expandable tool for the study of novel turbofan engine configurations.

ATLAS was validated by Lolis against two different turbofan architectures [54], namely, a 2-spool configuration, the CFM56-7B27 turbofan, and a 3-spool one, the Rolls-Royce Trent 892. The 2D engine layouts derived by *ATLAS* are plotted together with the engine original cutaways in Fig. 2.8. The weight estimates for those engines are listed in Table 2.3.



(a) CFM56-7B27



(b) Rolls-Royce Trent 892

FIGURE 2.8: ATLAS turbofan 2D cutaway comparison [54]

TABLE 2.3: ATLAS turbofan engine weight prediction comparison [54]

Engine model	Original [kg]	Estimated [kg]	Error [%]
CFM56-7B27	2,405	2,286	-4.97
Trent 892	5,942	5,609	-5.6

An additional model for the Rolls-Royce Trent 1000 was created and validated by Mourouzidis [55]. The design of this engine is similar to the Trent 892, so the validation results for the Trent 1000 are omitted herein.

ATLAS has been successfully employed in the study of modern turbofan engine configurations [55]. No turbojet engine weight estimation has been completed, although it could be undertaken thanks to the modular nature of the code. Neither the sizing of turboshaft engines nor industrial gas turbines have been attempted. In fact, for these latter engines, ATLAS fails to size some of its components *i.e.* the Free Turbine (*FT*) shaft, the engine intake and the compact centrifugal turbomachinery. The code also lacks the capacity to model the geometry of the additional burners. Therefore, in order to convert ATLAS into a thorough gas turbine engine sizing tool, some additional features are to be added and validated by the author first.

One of the drawbacks of *CB* methods is their associated high computational cost

that may become too large for a vast design space exploration and/or an engine design optimisation. To alleviate the computational time while keeping the high accuracy, Lolis introduced the idea of the 'hybrid method' [54]. This approach consists on the application of a *CB* method to generate a weight database that is then used to create a fast regression model that captures the effect of the design variables in the engine weight. This regression model can be implemented within a design space exploration or an optimiser routine for the engine weight estimation, instead of the time consuming *CB* method. Note that a different regression surface must be generated for each engine architecture.

The described hybrid approach, that has never been used in the weight analysis of aircraft gas turbine engines, is to be applied in this project for the design of conceptual turboshaft engines. Therefore, ATLAS will be deployed to generate a 'cloud' of engine weight data under the design assumptions defined for each engine architecture. These databases will be then used to interpolate the weight of the candidate engines utilising the cycle design variables as inputs.

2.3 Performance optimisation of turboshaft engines for rotorcraft applications

In this section, a brief review of existing optimisation studies for rotorcraft turboshaft engines is presented. The main merits of these investigations are highlighted and the limitations and gaps identified.

As highlighted by Clarke [5] there are three potential ways to reduce the environmental footprint of civil aviation, namely:

- a) Significant reduction in number of operations.
- b) Design modification of the aircraft/rotorcraft fleet.
- c) Enhancement of the flight operational rules and procedures.

Due to the expected rise in aviation activity, approach (a) is not viable. Thus, the rotorcraft performance research paths reviewed by the author are in line with solutions (b) and (c).

Within category (b), the work completed by Linares *et al* [56] can be mentioned. These authors developed a multidisciplinary framework for the analysis of rotorcraft performance encompassing a rotorcraft mission energy management model, a helicopter mission performance model, the engine performance simulation code TURBOMATCH, and a module for the engine emissions prediction based on the *FOCA* correlations mentioned in Section 2.2.2. The rotorcraft mission energy management model implemented in the routine simulates the power requirements of secondary power systems of the helicopter. This routine was used for the optimisation of a Single-Engine Light (*SEL*) Bell 206L-4 helicopter (equipped with an Allison 250-C30R turboshaft engine) over a passenger transport mission. The optimisation used mission flight parameters, such as altitude and forward speed at different mission segments, to seek the best mission profile management for reduced mission fuel and NO_x . The results of this analysis showed that reductions up to 3.35% are attainable at the expense of a 2% increase in NO_x emissions, which demonstrate the trade-off between these two mission parameters. This compromise between fuel efficiency and NO_x emissions will be also proven by the results derived by the author in Section 5.3.4, where simultaneous optimisation of mission fuel and NO_x for the conventional turboshaft engine is attempted. Note, also, that one of the limitations of this work is that the NO_x emissions are computed with *FOCA* methods which may have an uncertainty of a factor of two or more for pollutant levels prediction.

Within this category, the contribution of Goulos *et al* [28] can be also mentioned. The work accomplished in that reference targets the design of optimum rotorcraft operations in terms of fuel burn, gaseous emissions, and ground noise impact. The variables employed in the optimisation include the airspeed and altitude of each mission task, and also a limited fluctuation of the original flight path coordinates. To model that, each cruise task element is divided into five individual segments with four corresponding intermediate waypoints. This way, the variables used in the optimisation are the latitude and longitude coordinates of the respective waypoints. The authors used the tool *Phoenix* for the rotorcraft performance simulation, whilst for the optimisation deployed they first used a Design of Experiments (*DOE*) technique based on the Latin Hypercube Sampling (*LHS*) method [57]. The results obtained from the *DOE* were then used to build a Response Surface Model (*RSM*) able to accurately approximate the response of the simulation methodology with a reduced computational time. The optimisation case study is based on a Twin-Engine Light (*TEL*) rotorcraft model based on the configuration

of the Airbus Helicopters Bo105 equipped with two Rolls-Royce Allison 250-C20B turboshaft engines. Two baseline missions were employed *i.e.* a police surveillance and a passenger transport mission. These authors proved that substantial reductions in fuel burn, gaseous emissions (CO_2 and NO_x) and ground noise impact can be achieved simultaneously through the optimisation of the flight procedures and the path route.

On the other hand, several research projects which lie within approach (c) can be mentioned herein. The first one is the conventional turboshaft engine optimisation at mission level performed by Goulos *et al* for mission fuel reduction [34]. For the helicopter performance simulation the software *HECTOR* was used together with an Aeroelasticity Decoupling Tool (*ADT*). The formulation of the *ADT* is based on the assumption that the weight of the overall fuel burn change due to the incorporation of a thermodynamic engine cycle different to the one used for the baseline configuration is small enough so as not interfere with the helicopter trim procedure. Hence, the time variations of engine inlet Mach number, operational altitude, and shaft power requirement throughout the course of a designated operation are independent of the engine configuration. The architecture of the *ADT* is shown in Fig. 2.9.

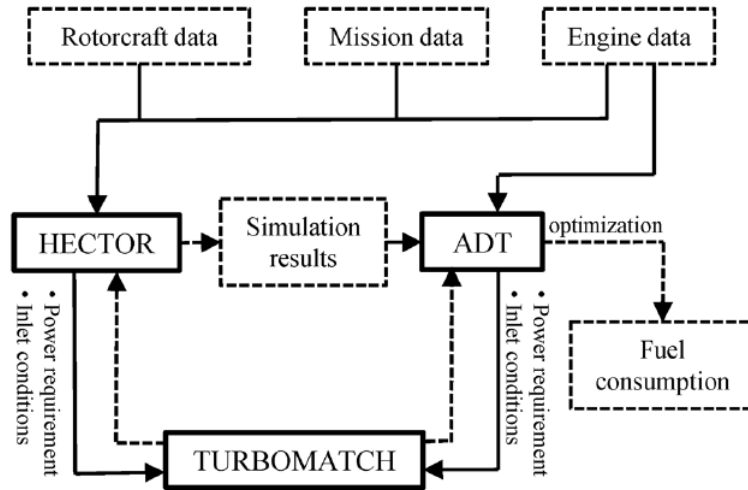


FIGURE 2.9: *ADT* architecture scheme [34]

As observed, the engine performance of any defined thermodynamic cycle is assessed throughout the mission using the stored time variations of engine inlet conditions and shaft power requirement, with no further *HECTOR* simulations. The mission fuel consumption is calculated with the time-integration of the fuel flows calculated over the mission. The implementation of the *ADT* highly reduces

the computational time associated with the assessment of any given engine configuration within a complete mission. This idea will be applied by the author to accelerate the optimisation of the conventional turboshaft engine at mission level (see Section 5.3.4).

Similar approach to the one deployed in [28] was used by the authors to explore the design space and to build the *RSMs* required. For the optimisation itself the variables used include the *LPC* (Low Pressure Compressor) and *HPC* (High Pressure Compressor) Pressure Ratios (*PRs*), and the engine mass flow. A constant technology level was assumed, so the *TET* was kept fixed. The *DP* power was also maintained so that the rotorcraft payload-range capability was not penalised. Models of the *TEL* helicopter used in [28] and the Twin-Engine Medium (*TEM*) Aérospatiale SA330, powered by two Turbomeca Turmo IV-C turboshaft engines, were used for the optimisation analysis. The multi-objective optimisation completed between fuel consumption and *DP* shaft power showed that it is possible to increase shaft power and reduce mission fuel in comparison with the baseline engine, and that, for the optimal engines, the mission fuel consumption increases almost linearly with maximum engine shaft power. A limitation of this work is that even though *TET* is fixed constant, the variation in compressor *PRs* affects the turbine cooling flows requirements, as the condition of the coolant gas is altered. This factor was not considered by the authors.

It is also worth to mention the efforts of Fakhre *et al* towards the optimisation of a single-spool recuperated turboshaft engine for rotorcraft applications [32, 35, 58–61]. *HECTOR* was used throughout the whole analysis. The authors studied this particular cycle due to its potential to reduce the fuel consumption at part-power operation. The concept analysed is shown in Fig. 2.10.

A first trade-off analysis was completed for a *TEL* and a *TEM* helicopter (the Airbus Helicopters Bo105 and the Aérospatiale SA330, respectively). The results from this trade-off study showed that the conceptual regenerative helicopter can offer significant improvement in the payload-range capability, although it may have a detrimental effect on the mission NO_x emissions inventory. In fact, the recuperator/heat exchanger (*HE*) triggers the formation of NO_x due to its contribution to increase the equilibrium temperature at early stages of the combustion process. As another outcome of the analysis, it was proven that the effectiveness of the *HE* has a large impact on the level of acquired benefits.

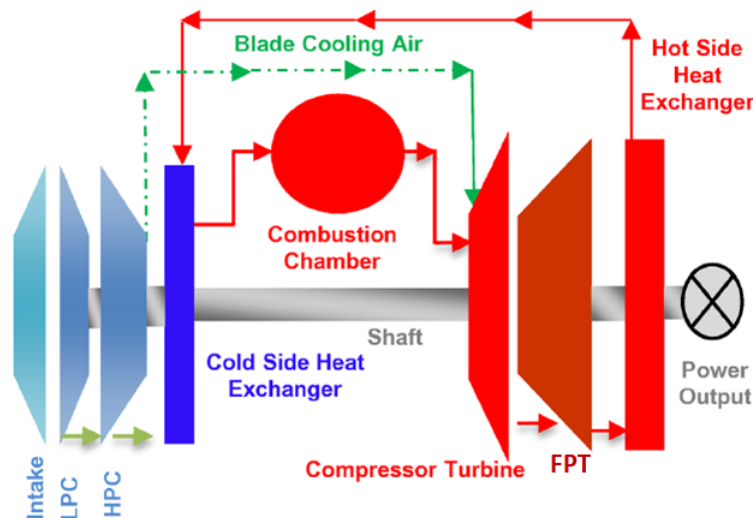


FIGURE 2.10: Single-spool recuperated turboshaft scheme [58]

One important aspect to notice is that the weight estimation of the *HE* is completed by means of the empirical correlation plotted in Fig. 2.11 and derived by Kailos [62]. Note that the expression correlates the specific weight of the *HE* with its effectiveness.

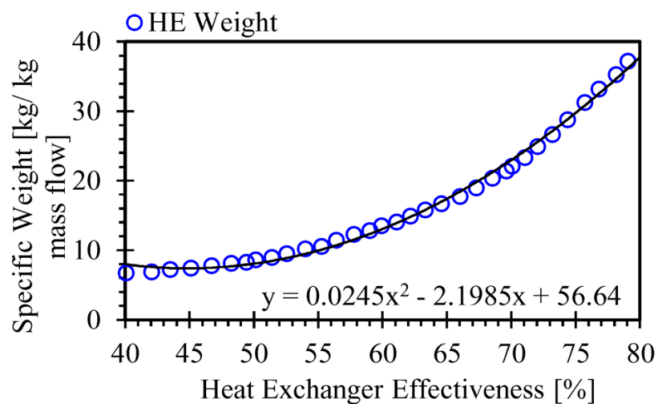


FIGURE 2.11: Correlation for heat exchanger weight estimation [58]

The added *HE* weight calculated this way introduces a slight penalty in the in-flight power required by the helicopter, specially in the low and medium speed segments of the flight. A similar result is observed by the author when the merits of the optimal conceptual engines are compared in Section 5.2.2.5.

Another aspect to highlight from this analysis is that the combustor model implemented in HEPHAESTUS for NO_x prediction was validated against empirical *FOCA* correlations, instead of actual experimental data which would be more recommendable for a better accuracy of the model.

After the trade-off analysis the authors continued the work by completing a series of single-objective optimisations for minimum mission fuel burn, minimum mission NO_x and minimum engine weight, respectively. The results showed that optimising for minimum engine weight leads to a configuration that offers greater operational improvements compared to benefits obtained when optimising for minimum mission fuel burn and minimum mission NO_x inventory, under the conditions simulated. For the powerplant weight optimisation, a *WEB* method was adopted to cater for the effect of the engine design parameters on engine mass. This *WEB* method is based on a *RSM* built after a sample of 48 different turboshaft engine models. Similar methods deployed in [34] were used for the design space exploration, and a constant technology level was also assumed (constant *TET*).

The methodology developed by these authors is later deployed in a multi-objective optimisation to quantify the trade-off between the weight of the regenerative engine and the SFC_{DP} for a *TEL* helicopter. A first sensitivity analysis between the regenerated engine design variables and the optimisation outputs (engine weight and SFC_{DP}) is first completed. This is similar to the parametric analysis completed by the author in Section 5.1 prior the optimisation of the conceptual engines.

Then, a Pareto front surface corresponding to minimum engine weight and minimum engine SFC_{DP} at constant technology level was derived. The engine cycle design variables corresponding to the points in the Pareto front proved that the *HE* effectiveness is the key factor affecting the engine weight and fuel efficiency, and that fuel efficiency can be substantially improved in comparison with the conventional baseline engine without any penalty in powerplant engine mass. A similar set of Pareto front surfaces comparing engine weight and SFC are to be produced by the author for the comparison of the optimal conceptual engine designs (see Section 5.2.2).

2.4 Reheated engine concept

As mentioned in the Introduction, reheating has been traditionally used for thrust augmentation in military gas turbine engines. In such applications, the sequential combustion takes place at a lower pressure compared with that at the main burner, and so, the increase in engine thrust is attained with a reduction in the overall thermal efficiency. However, if the additional heating process is moved within the

turbines the cycle efficiency may be potentially increased. In fact, the thermal efficiency of the gas turbine cycle (Brayton cycle), for a fixed maximum temperature (T_{max}), has a maximum at a certain OPR (Overall Pressure Ratio). If reheating is considered, the maximum thermal efficiency occurs at a higher OPR (as observed by Schobeiri *et al* [69]) and a higher value can be attained. As an example, a real Brayton cycle defined by the parameters listed in Table 2.4 is compared with a reheated cycle in Fig. 2.12. For the reheated cycle, two similar expansion stages are assumed, and a temperature of 1200 K at the inlet of the second expansion is considered.

TABLE 2.4: Real Brayton cycle reference data

Ambient pressure [Pa]	100000
Ambient temperature [K]	298
C_p [J/kgK]	1004.5
γ [—]	1.4
η_c [%]	85
η_{tur} [%]	90
T_{max} [K]	1400
OPR [—]	5-50

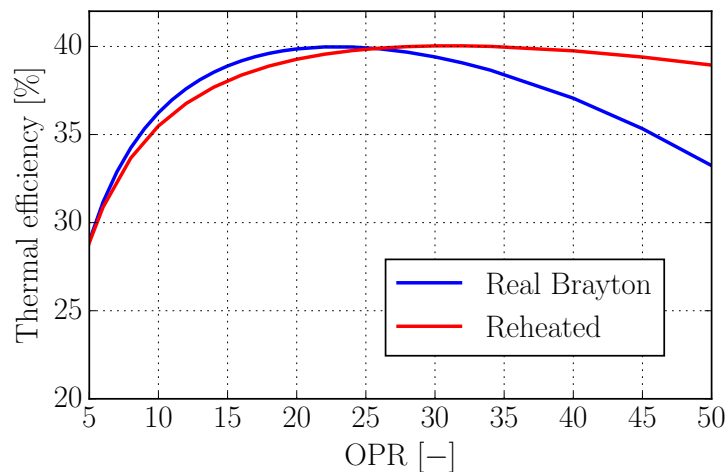


FIGURE 2.12: Thermal efficiency comparison

It is observed that the optimum OPR is higher for the reheated configuration and that a small thermal efficiency benefit is attained with this configuration under the set of assumptions considered. Therefore, the reheating concept appears as a potential solution to enhance thermal efficiency, specially considering the current technology context, where the cycle maximum temperature can be barely increased due to material restrictions.

Another promising application of reheating is its implementation in a regenerative intercooled cycle. This engine arrangement would approximate the Ericsson cycle efficiency. As observed in Fig 2.13, The Ericsson cycle consists of an isothermal compression (1-2), a heat addition process at constant pressure (2-3), an isothermal expansion (3-4), and a heat transfer at constant pressure (4-1). Besides, this is a regenerative cycle, where the heat required to increase the temperature during the isobaric heating (2-3) comes from the scape gases during the cooling process (4-1). Therefore, the only external heat addition occurs during the isothermal expansion (3-4).

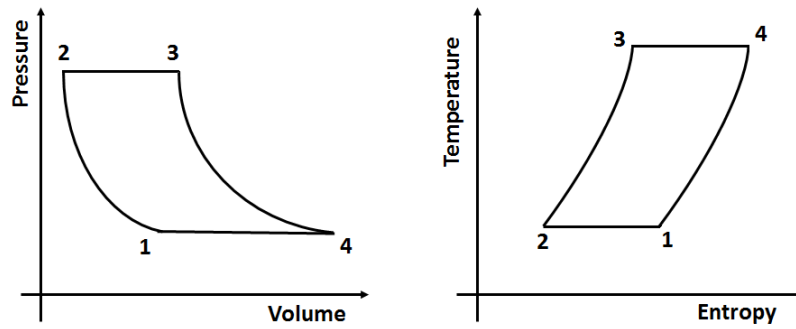


FIGURE 2.13: Ericsson cycle diagram

The Ericsson cycle can be considered as a regenerative Brayton cycle which employs infinite compression-cooling and expansion-heating stages. In fact, this cycle sets the upper limit for the thermal efficiency a Brayton cycle can achieve, which is in agreement with the Carnot cycle efficiency [63]:

$$\eta_{th} = 1 - \frac{T_{min}}{T_{max}} \quad (2.2)$$

Where T_{min} and T_{max} , represent the minimum and maximum temperatures of the cycle, respectively. This expression provides the theoretical maximum thermal efficiency attainable by any heat engine according to the second law of thermodynamics.

Piqueras *et al* [63] showed the potential of this reheated-regenerated-intercooled cycle with some simple calculations and assumptions. Assuming isentropic compressions and expansions, these authors calculated a thermal efficiency of 46.62% for the simple cycle, and 62.92% for the reheated-regenerated-intercooled cycle, which is closer to the thermal efficiency of 72.91% for the Ericsson cycle.

Several researchers have accomplished promising studies on reheating and turbine burners applied in turbojet and turbofan engines. Sirignano *et al* [64] performed a simple cycle analysis comparing a conventional turbojet configuration and a continuous turbine-burner configuration in which the total temperature is kept constant along the expansion process (Fig. 2.14). They found that the continuous turbine-burner can increase Specific Thrust (ST) about 20% with only 10% increase in SFC , whilst the afterburner design increases SFC by almost 50% for the same ST increment. They also showed that the performance gains of the turbine-burner design drastically increase with compressor pressure ratio and flight Mach number.

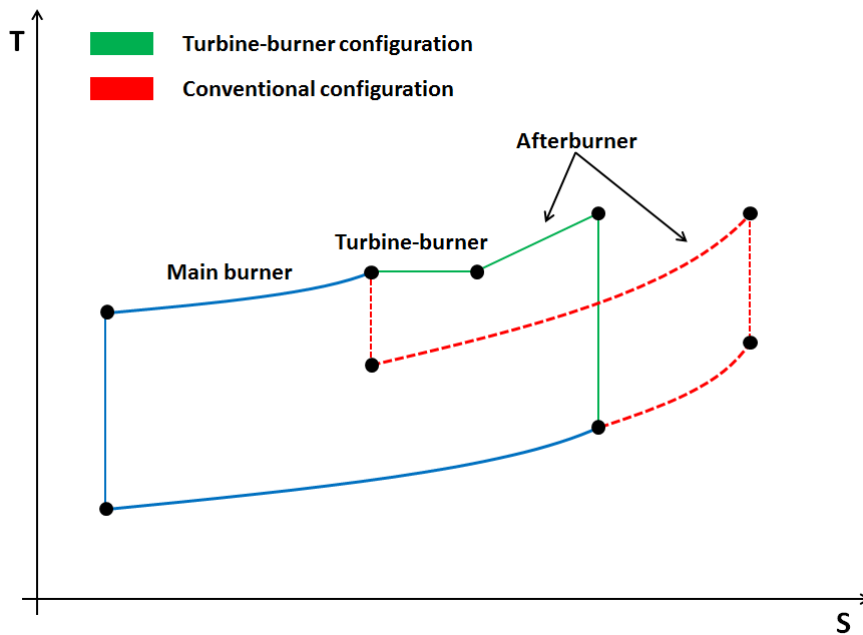


FIGURE 2.14: Comparison of thermodynamic cycles with and without turbine-burner, adapted from [64]

In a later publication [65], they discussed the performance benefits of a more feasible Inter-Turbine-Burner (ITB) in which the turbine stators are converted into combustors. In Fig. 2.15, different T - S diagrams are shown for a) continuous turbine-burner, b) 1 turbine-burner (1- ITB), c) 2 turbine-burners (2- ITB) and d) multiple turbine-burners (M- ITB).

It is noted that as the number of burners goes to infinity the cycle approaches the continuous turbine-burner cycle. This study also confirmed that the turbine-burner engine performance is improved with higher pressure ratios and flight Mach numbers. They also concluded that the power distribution among the segments of an ITB engine can be optimised for a given mission and configuration. To

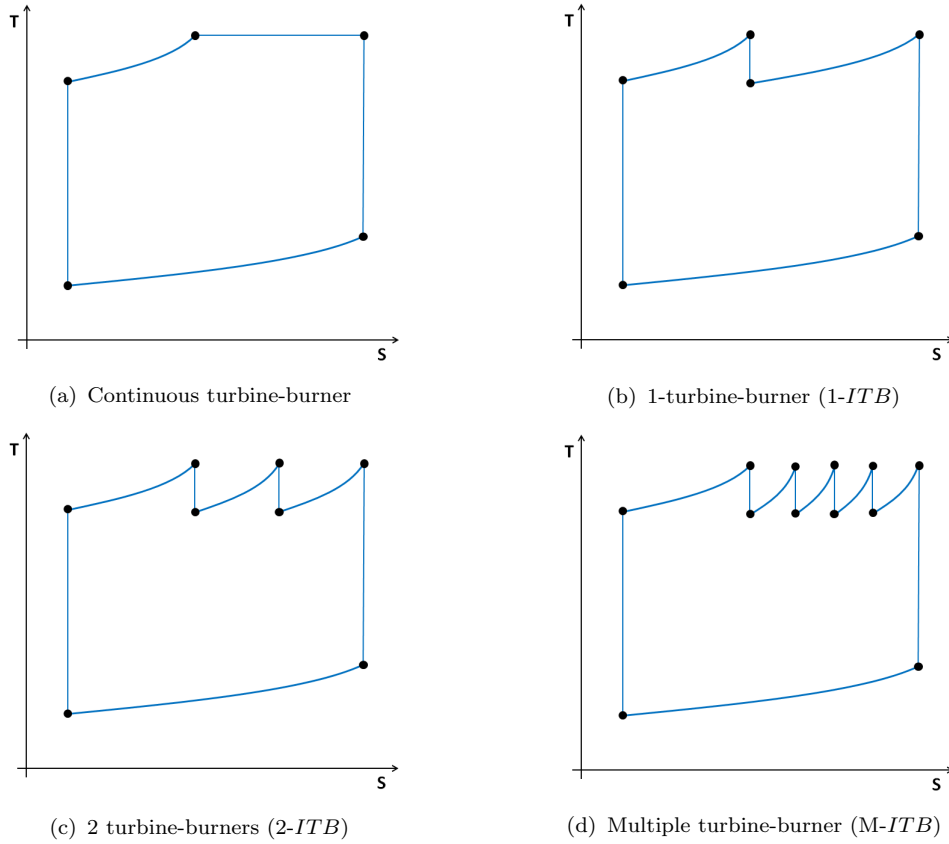


FIGURE 2.15: T - S diagrams for different ITB cycles, adapted from [65]

find this optimal distribution for the best thermal efficiency is one of the aspects to be investigated by the author (see Section 5.2.2). The potential of the ITB cycle is highlighted by the performance of the 1- ITB configuration which provided more than 50% increase in ST with a similar SFC level than the conventional baseline turbofan engine. Chen *et al* [66] developed an extension to the Sirignano *et al* research by including the turbine cooling effects into the calculations. This investigation showed minimal improvements for aircraft engines due to the large amount of cooling air required to keep turbine surface temperatures below critical limits. This result emphasises the need for a correct modelling of the engine cooling flows in order to perform a trustworthy assessment of the reheating concept potential.

Another contribution to the analysis of ITB turbofan engines was made by Liew *et al*. These researchers evaluated the performance at DP [67] and OD conditions of a dual-spool turbofan engine with a secondary burner located between the high-pressure and the low-pressure turbines [68]. They confirmed the relevance of the turbine cooling in the engine performance calculation and highlighted the

advantage of this technology: higher ST, less cooling air required, and possibly less NO_x production as a result of the lower peak temperatures inside the burners. Furthermore, they performed a mission study, which showed that the *ITB* engine used less fuel in every flight segment with a final fuel saving of 2.6%. Nevertheless, no engine weight penalties were considered in this mission analysis.

An advanced turbine-burner concept called *UHEGT* (Ultra-High Efficiency Gas Turbine) engine was presented by Schobeiri *et al* [69]. This concept distributes the combustion process in the turbine stators, thus eliminating the need for a combustion chamber and leading to a distributed combustion instead (Fig. 2.16). The combustion system of this design is based on a row of tubular fuel injectors placed in front of the stators which also acts as a vortex generator by creating Von Karman vortices that improve the fuel mixing. The configuration shown in this figure presents the most uniform temperature distribution and the lowest pressure losses amongst the numerous designs investigated by the authors. Besides, it is also the simplest configuration to be manufactured. Apart from the thermal efficiency benefits, the authors also claim that this concept results in a much shorter gas turbine which increases the thrust-power/weight ratio of the engine.

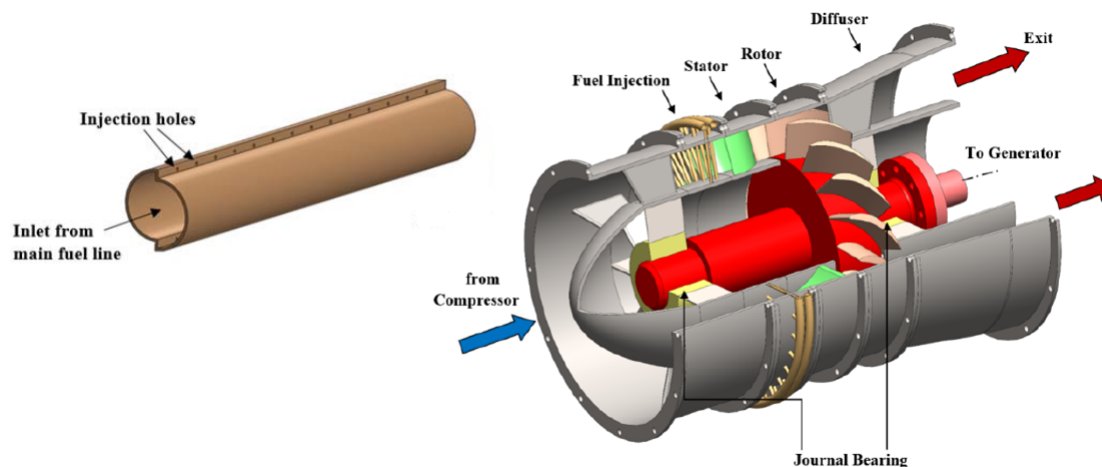


FIGURE 2.16: *UHEGT* concept configuration

A study is conducted comparing three different power generation gas turbine engines: a conventional gas turbine, an existing gas turbine with sequential combustion (Alstom GT24) and an *UHEGT* concept. The results are reproduced in Fig. 2.17.

It is seen that the *UHEGT* with four internal combustion stators exhibits a thermal efficiency above 47%, which means an increase of at least 7% above the GT24.

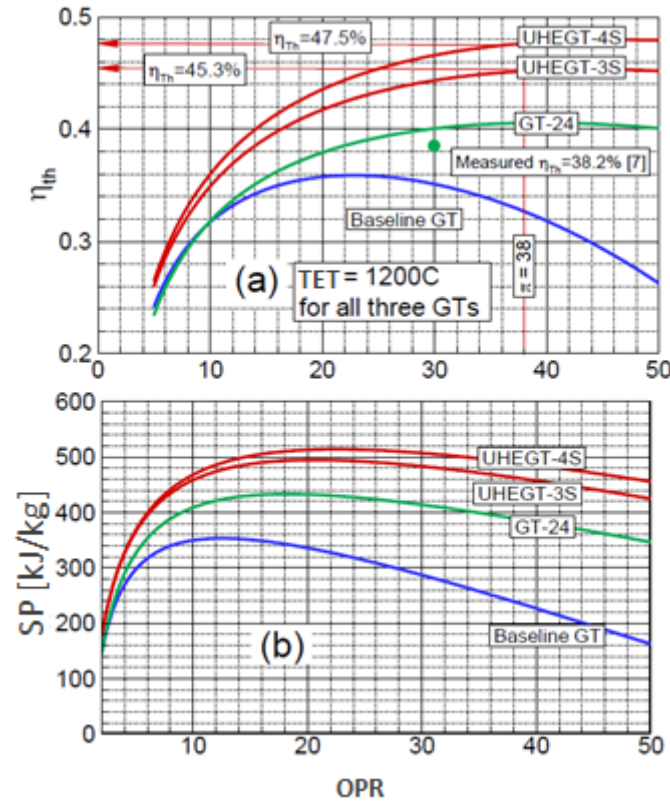


FIGURE 2.17: Baseline engine vs. Reheated architectures. Thermal efficiency and specific work comparison [69]

Regarding the specific work, about 20% higher specific work compared with the GT24 can be observed in Fig. 2.17(b). It is also noticed that the *UHEGT*-concept improves the thermal efficiency and the specific work at any given pressure ratio, which makes the *UHEGT* attractive for small and large size engine applications. However, the authors do not mention anything about the cooling requirements of the *UHEGT*, which could significantly affect the benefits presented.

Finally, Schmidt *et al* [25] conducted a thermodynamic assessment about novel engine concepts applied to civil turbofan powerplants. This assessment included not only the analysis of the sequential combustion concept but also the intercooled-sequential, recuperated-sequential and intercooled-recuperated-sequential concepts. The main findings of the study were that the sequential concept allows for more compact power units compared with the baseline configuration, and that the combination with intercooling further enhances the engine size reduction. In terms of efficiency, no improvement was found for the sequential concept alone for the particular application analysed. Only the intercooled-recuperated-sequential configuration allowed for an efficiency benefit about 5% compared with the optimised

baseline engine.

At the sight of the previous results, the actual potential of the reheated engines is not clearly defined, since the particular application and the assumptions considered during the analysis play a decisive role in the final performance gains observed. Apart from that, the extra weight added by the additional burners is systematically neglected in the aforementioned analysis, and the effect of cooling flows in engine performance is either ignored or vaguely modelled. Another fact that can be noticed is the general gap in knowledge that exists regarding the application of this technology in small turboshaft engines for rotorcraft applications. The research activities reviewed by the author focus on turbofan engines and power generation purposes solely.

Therefore, one of the main targets of the current project is to develop a thorough methodology to offer a representative and realistic insight about the actual potential of this technology in the rotorcraft propulsion field.

Chapter 3

Methodology

The main aim of this research activity is to assess the performance gains of conventional turboshaft engines, and the exploration of the potential benefits of novel engine configurations based on the reheated cycle principle. To that end, a performance parametric analysis and optimisation of the rotorcraft powerplant are to be completed.

In the initial parametric analysis, the sensitivity of the different engine architectures to changes in the cycle design variables is assessed. In particular, the variation of SFC at cruise (SFC_{cr}), Specific Power at Maximum Power (SP_{MP}) and engine weight (W_{eng}) is studied, (for the conventional engine $EINO_{x_{cr}}$ is also included in the parametric analysis). Once the parametric analysis is completed, an optimisation analysis is conducted in order to find the most optimal powerplant in terms of thermal efficiency and engine weight.

The most promising architectures derived from the powerplant optimisation stage will be integrated in a powerplant-rotorcraft system to undertake the final helicopter performance analysis at mission level. In this final assessment, mission block fuel figures corresponding to the conceptual powerplants will be benchmarked against the baseline powerplant-rotorcraft system results attained at a representative flight mission.

Thus, the tools and modules involved in the analysis has been integrated in a simulation framework in order to efficiently undertake the performance parametric assessment and optimisation of the novel configurations. The flow chart shown in

Fig. 3.1 summarises the integration of the engine performance modelling tools in the computational framework.

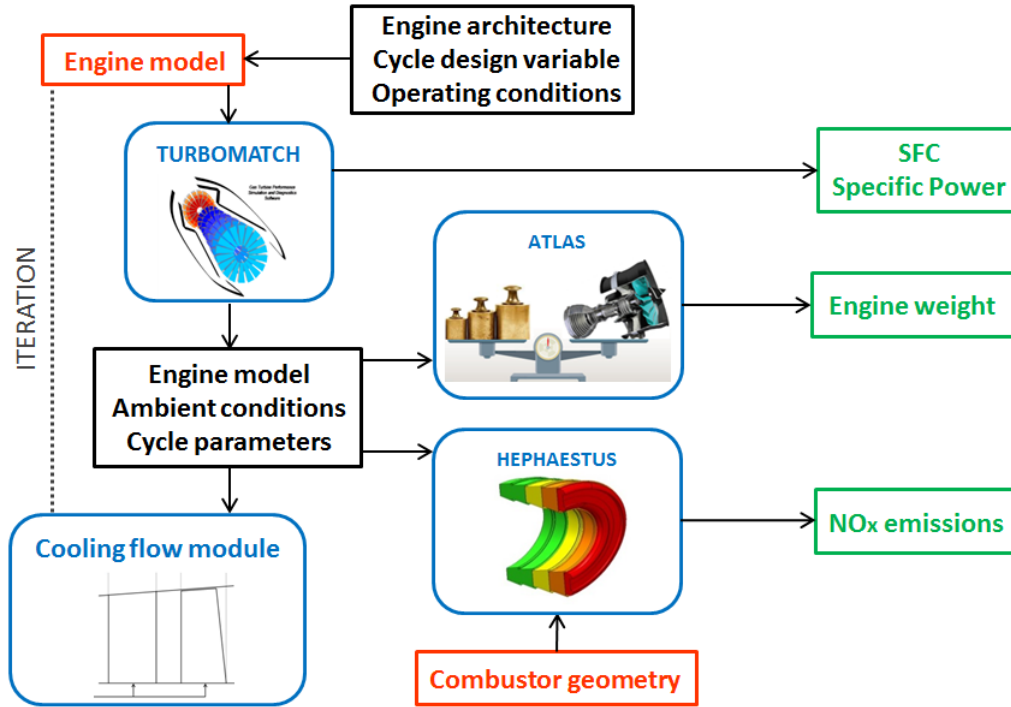


FIGURE 3.1: Engine performance analysis tool scheme

The engine architecture, the cycle design variables and the conditions of the operating point of interest are first introduced in the simulation framework. This latter information includes the shaft power, flight altitude and Mach number, and ambient conditions (temperature and pressure). Note that the size and weight of the turboshaft engine is independent of the operating conditions, thus, the power setting details used to size the engine have to be also introduced in the routine. These data is then introduced in the engine model, which is simulated in TURBOMATCH in order to evaluate its thermodynamic performance and obtain the cycle parameters (temperatures, pressures and mass flows at each engine station). Note that this is an iterative process which involves the computation of the cooling flows required by the turbine stages. This calculation process is detailed in Fig. 3.2. Once convergence is reached, the final engine *SFC* and *SP* values are obtained. The engine cycle parameters (at the sizing power setting), and the type of engine architecture are then transferred to ATLAS for the estimation of the engine weight. Uniquely for the conventional engine architecture (the analysis of the NO_x emissions is only undertaken for this particular configuration), the parameters calculated by TURBOMATCH are introduced in HEPHAESTUS, which

calculates the amount of NO_x emitted by the engine at the operating condition defined. Note that additional geometry information of the engine combustor is required by HEPHAESTUS.

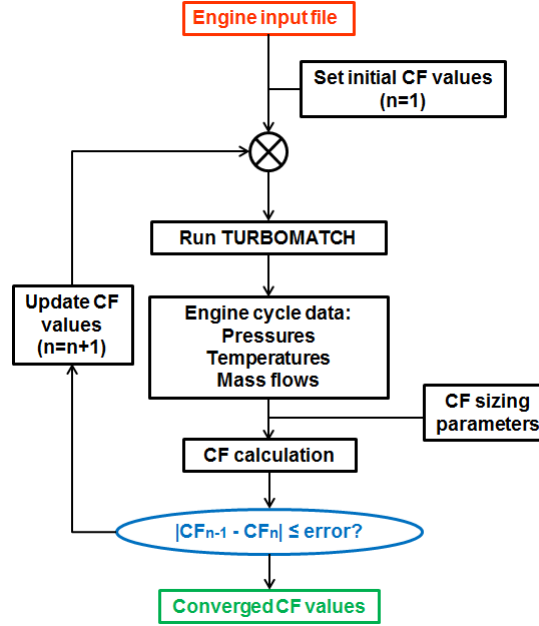


FIGURE 3.2: Cooling flowrate calculation scheme

Finally, for the assessment of the engine configurations at mission level, the tool *HECTOR*, described in Section 2.1.1.2, is used.

The elaboration of the different simulation models and tools required for the complete simulation of the conceptual engine architectures is detailed hereafter in this chapter. This description includes the architectural definition of the models, the numerical values employed, the list of constraints and assumptions applied, and the updates introduced in the simulation tools when necessary. In order to ease the understanding of each of the simulation aspects, the chapter is organised in four sections covering the modelling of the helicopter performance, powerplant thermodynamic performance, engine weight, and combustor NO_x emissions, respectively.

3.1 Helicopter performance modelling

The model of the reference *TEM* helicopter is implemented in *HECTOR*. The model to be used in the analysis is based on the configuration of the Sikorsky UH-60A Black Hawk helicopter displayed in Fig. 3.3.

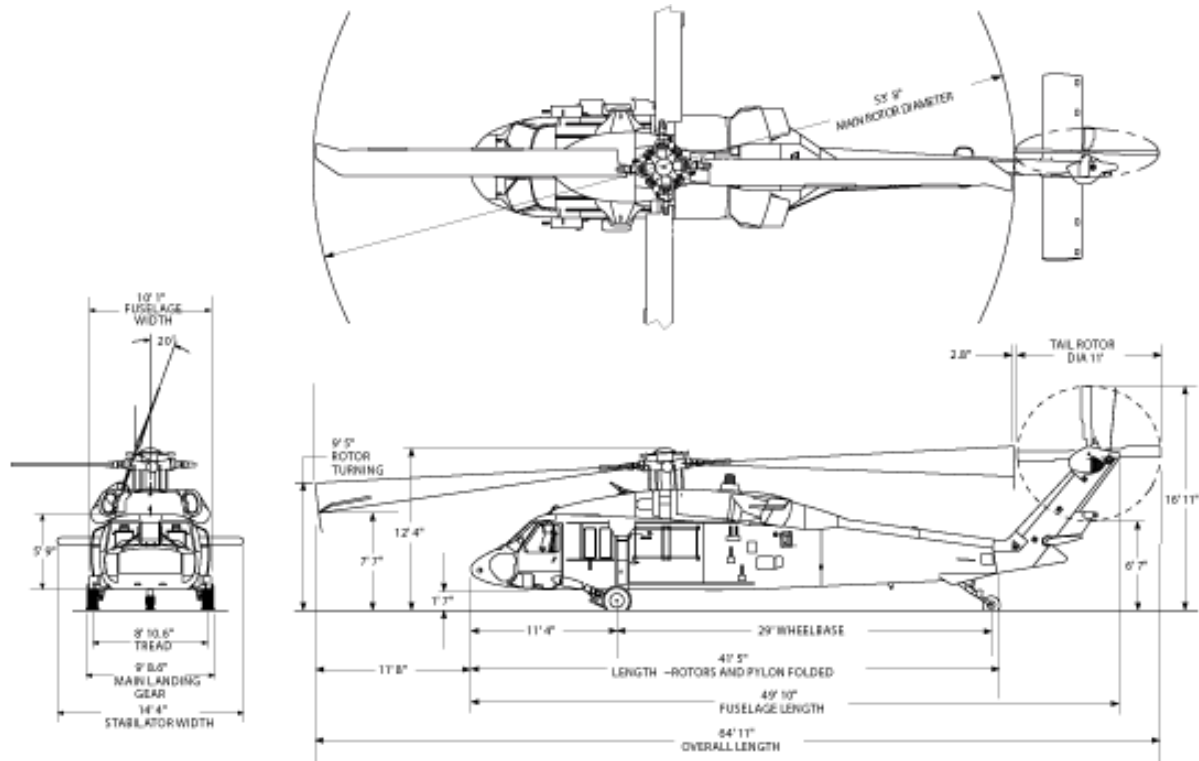


FIGURE 3.3: Sikorsky UH-60 Black Hawk helicopter 3-view drawing [70]

The Sikorsky UH-60 is the U.S. Army's primary medium lift utility transport and air assault aircraft with 2,300 units on operation. It is also employed by the armed forces of 26 other countries, which makes a total of more than 3,000 Black Hawk helicopters in service worldwide. Since its maiden flight in 1974, this multi-role helicopter has been operated successfully in a wide range of mission roles *i.e.* internal/external lift, combat assault, medevac, aerial firefighting, search and rescue, and VIP transport [71]. Apart from its extended use, the Sikorsky UH-60 was chosen as the reference helicopter due to the elevated number of documented experiments available in the public domain. This extensive information is paramount for the verification of the different simulations models to be developed.

The helicopter model described in this section serves the purpose of simulating the engine performance at mission level. This allows to compare the benefits of the different engine configurations against the performance of the baseline helicopter configuration over realistic mission scenarios. The main rotorcraft parameters used for the design of the model are presented in Table 3.1.

TABLE 3.1: Rotorcraft model design parameters [72, 73]

Empty weight [kg]	5,082
Maximum gross weight [kg]	9,185
Solidity, σ [—]	0.089
Number of blades [—]	4
Blade radius, R [m]	8.178
Blade chord [m]	0.527
Equivalent blade twist [deg]	18
Nominal rotor speed, Ω_{nom} [rad/s]	27.01
Blade airfoils	SC1095-SC1094 R8

3.2 Powerplant performance modelling

3.2.1 Baseline engine architecture

The engine integrated in the Sikorsky UH-60A Black Hawk helicopter is the General Electric T700-GE-700. The engine layout is shown in Fig. 3.4.

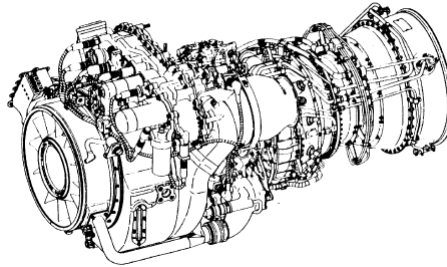


FIGURE 3.4: General Electric T700-GE-700 engine [74]

The T700-GE-700 is the first production model of the T700 family (first engine delivered in early 1978), and its civil derivatives, named as CT7 [75]. The T700 model is a small turboshaft engine which consists of a five-stage axial and a single-stage centrifugal flow compressor; a low-fuel-pressure, annular combustion chamber; a two-stage axial flow gas generator turbine; and a two-stage uncooled independent power turbine [76]. A schematic of the major engine components is shown in Fig. 3.5. This is the engine to be used as baseline in the present study, and its architecture will be used to simulate the performance of the conventional cycle (*CONV*) engine.

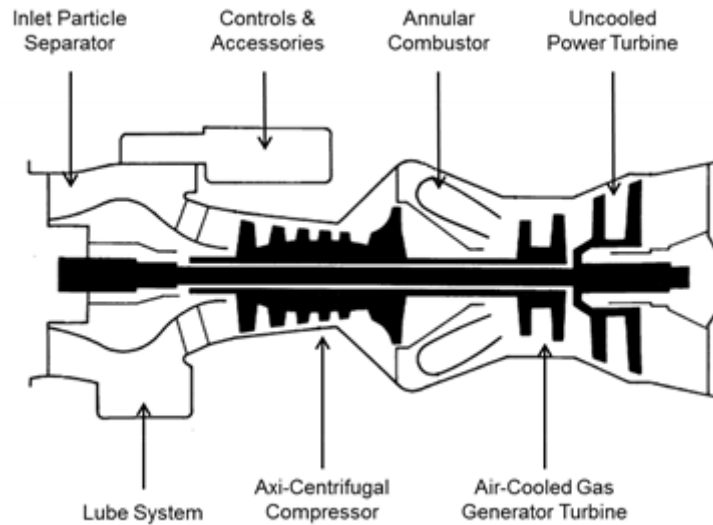


FIGURE 3.5: General Electric T700/CT7 engine family architecture, adapted from [74]

3.2.2 Reheated engine architectures

Apart from the analysis of the *CONV* engine performance, the potential of the reheated cycle is to be explored. For this purpose, three different architectures are to be analysed:

- Inter-Turbine Reheated (*ITR*) configuration
In this configuration a secondary burner is placed between the first and second stages of the Gas Generator Turbine (*GGT*).
- Sequential Combustor (*SEQ*) configuration
In this case, the additional burner is placed between the *GGT* and the *FT*.
- *ITR* + *SEQ* configuration
This latter architecture is a combination of the previous concepts, which means that two extra combustors are installed at the positions already indicated.

These three engine configurations were built by modifying the *CONV* engine architecture as shown in Fig. 3.6.

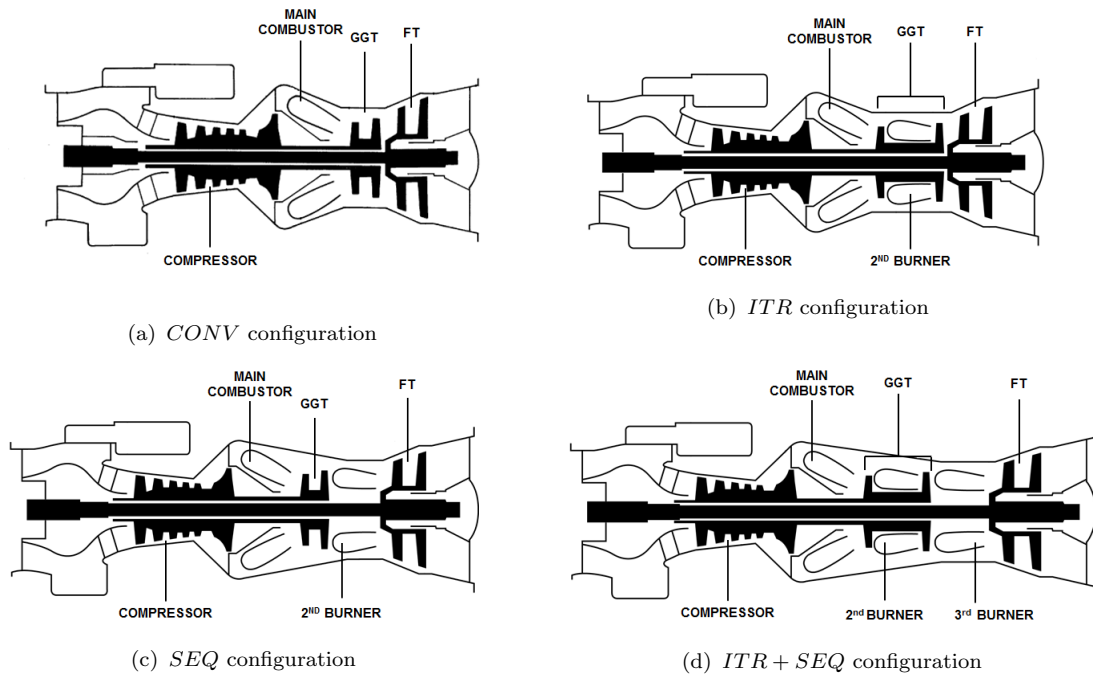


FIGURE 3.6: Turboshaft engine conceptual configurations

3.2.3 Engine model description

The engine performance simulation in TURBOMATCH relies on an engine input file detailing the engine components arrangement and their parameters. Instead of creating an individual input model for each engine architecture, a single versatile model was developed. This way, the performance of any of the mentioned configurations can be simulated by just switching some engine parameters in the input file. The bricks structure of the engine model is shown in Fig. 3.7.

The engine axi-centrifugal compressor is modelled with two COMPRE bricks which simulate the axial and centrifugal compressors, respectively. Despite the added complexity, this approach allows an increased fidelity and flexibility in the engine modelling and analysis:

- Accurate engine air bleed modelling

For the T700-GE-700 engine, bleed flow is extracted at stage 4 of the axial compressor for seal pressurisation and power turbine balance [74]. As the axi-centrifugal compressor is split into two bricks, this engine bleed can be modelled with a PREMAS brick placed between both components. Bleed extraction affects the engine performance by modifying the work and mass

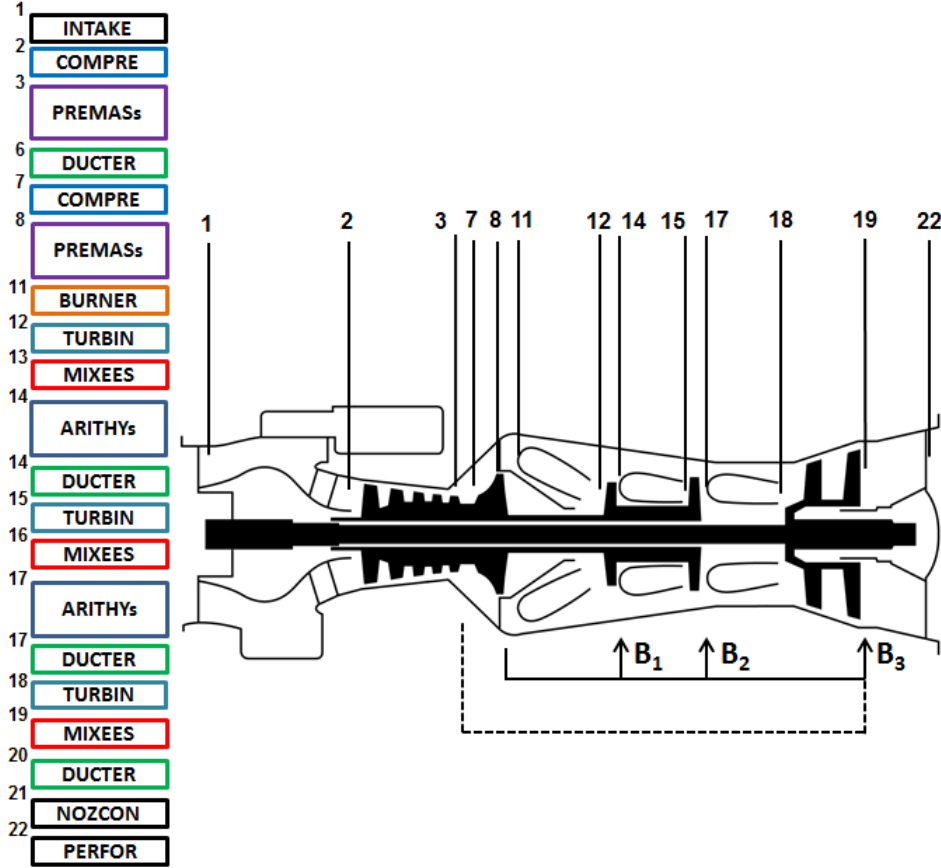


FIGURE 3.7: Engine model bricks structure in TURBOMATCH

flow balance between the engine components [77]. Therefore, a realistic implementation of the engine air bleeds is paramount to accurately account for their effects on the engine performance.

- Flexible cooling air bleed modelling

Cooling flow is bled at the axi-centrifugal compressor outlet for the baseline engine *GGT* stages [74] and so, a *PREMAS* brick was introduced after the centrifugal compressor brick. However, for the reheated configurations, it may be also necessary to cool the *FT* stages. As the pressure at the *FT* inlet is lower than the pressure at the *GGT*, it may be possible to cool the *FT* stages with air extracted at the axial compressor exit. The condition imposed for that to happen is that the total pressure at the extraction point must be 1.3 times the pressure at the injection point to account for the losses through the air system. This approach is beneficial since the earlier the air is bled, the lower the loss in performance as less work has been done on the extracted air [78]. Therefore, a *PREMAS* brick was introduced at the outlet of the axial compressor to cater for the *FT* stages cooling flows. In case

the pressure at the outlet of the axial compressor is not sufficient, another PREMAS brick was added at the outlet of the centrifugal compressor, in order to bleed the *FT* cooling flow at this station. The distribution of the engine cooling flows arrangement is shown in Fig. 3.7, where B_1 and B_2 represents the cooling fractions for the 1st and 2nd stages of the *GGT*, and B_3 corresponds to the *FT* cooling fraction.

- Representative compressors sizing

The fact that one brick is used for each compressor allows TURBOMATCH to calculate the cycle parameters at the outlet of the axial compressor and at the inlet of the centrifugal compressor. These data can be introduced in ATLAS to size both components independently, increasing the accuracy of the engine weight estimation.

- Enhanced engine design exploration

As two bricks are available, the *PR* of each compressor instead of the *OPR* can be used in the design exploration. This extra variable offers a more accurate analysis, as the performance changes in response to design modifications of each compressor can be investigated.

In order to allow for the integration of an additional burner within the *GGT* for the *ITR* and *ITR + SEQ* configurations, this component was modelled using two TURBIN bricks, one for each *GGT* stage. As no auxiliary burner is to be placed within the *FT*, its two stages can be modelled with a unique brick. Note that a MIXEES brick was added after each TURBIN brick to simulate the effect of the turbine cooling flows injection. Following the approach established by Ballin [74], the cooling flows are considered to be introduced downstream the turbines for the thermodynamic calculations.

With regard to the engine turbomachinery, an additional parameter must be declared in order to calculate the engine performance at *OD* conditions: the component map number. TURBOMATCH thermodynamic *OD* calculation is based on mass and energy balances, computed iteratively using component maps [42]. For compressors, the map contains values of pressure ratio, corrected mass flow and efficiency for each non-dimensional speed; for turbines the map provides values of enthalpy drop ratio, corrected mass flow and efficiency for each non-dimensional speed. The code includes a set of standard maps (five compressor maps and six

turbine maps) which are sufficient for most purposes. However, the user can also implement customised maps if needed.

For the current engine model, an axi-centrifugal map containing the values of corrected mass flow and pressure ratio at different non-dimensional speeds corresponding to the T700-GE-700 was available in [74]. Therefore, a customised map was created using this information. This user-defined map was initially used to simulate the *OD* performance of both compressors (axial and centrifugal); however, during the validation process (see Section 4.2) it was observed that a better match with the experimental results was attained by using the standard map 3 for the centrifugal compressor. For the engine turbines, standard maps 4 and 5 were used for the *GGT* and the *FT*, respectively, since there were no data that could be used to generate customise maps for these components. The turbine maps combination was selected in order to minimise the deviation between TURBOMATCH and experimental results.

Finally, the two auxiliary burners were introduced in the engine model as ducts: stations 14-15 and 17-18. Other ducts were introduced in the model to simulate pressure losses between components. The DUCTER brick in TURBOMATCH allows to switch the component from duct to burner, and to define the duct pressure loss and the combustion efficiency. The outlet temperature of the auxiliary burners (when required), is set by the ARITHY bricks placed upstream of the mentioned DUCTER bricks as the ratio of the Turbine Entry Temperature (*TET*) to the first Turbine Entry Temperature (*TET*₁):

$$TR_1 = \frac{TET_2}{TET_1} \quad TR_2 = \frac{TET_3}{TET_1} \quad (3.1)$$

The same values assumed for the main combustor pressure losses and combustion efficiency were used for the auxiliary burners. This is the approach followed by Liew *et al* in their reheated cycle analysis [67, 68].

Apart from these design aspects, some assumptions are considered during the performance analysis of the conceptual turboshaft engines:

- **Constant polytropic efficiency for compressors and turbines**

In order to keep the technology level constant, the polytropic efficiencies of the axi-centrifugal compressor, and the *GGT* and *FT* were kept constant

and equal to the values employed in the baseline engine model. Although the injection of cooling flows in the turbine blades surface has a detrimental effect on polytropic efficiency, this effect was not considered in the present analysis. Hence, the polytropic efficiency of the turbines is not varied with the cooling flows fluctuation.

- **Fixed combustion efficiency and pressure loss in the engine burners**

Following with the assumption of constant technology level, the combustion efficiency and the pressure losses incurred in the engine burners are kept constant. Note also that, for the reheated cycle configurations, the pressure loss in the auxiliary burners is maintained even when they are off.

- **Constant total pressure at the outlet of the FT at DP**

The total pressure at which the FT expands at DP is an input of the TURBOMATCH model. The value initially introduced in the model of the baseline engine was selected using some engineering judgement and trying to get the closest match possible between the model predictions and the actual engine data. For the analysis of the conceptual powerplant designs the same value is utilised.

3.2.4 Cooling flow sizing module

The air bled in the compressor stages impairs the engine final performance to a point that the maximum TET attainable for modern engines is limited by the amount of air required to cool the turbine stages. Therefore, the sizing of the engine cooling flows (CF) must be correctly addressed if a proper evaluation and optimisation of the engine performance is pursued. In this section, the method implemented by the author is introduced.

The estimation of the CF required by the turbine stages is not a simple task as it depends not only on the thermodynamic cycle parameters (temperatures, pressures and mass flows), but also upon many other parameters such as: blade life required, technology level employed (including both materials and cooling implementation), blades centrifugal stress (it triggers the creep effect), and blade geometry [78].

Apart from the thermodynamic cycle parameters, most of these data are hardly available at an early design stage. Therefore, simple methods that correlate the amount of cooling required with thermodynamic cycle variables are available in

the open literature. These methods offer a first insight of the cooling requirements trend rather than an accurate prediction, so they are appropriate for a preliminary design phase. An example of this approach is the chart derived by Walsh [78] that shows typical values for blade cooling airflows versus stator outlet temperature. Individual trends are shown in this chart for rotor and stator blades at two different technology levels.

The simplicity and lack of generality of these methods discard them to be used in the present study. Instead, CF are sized using the more rigorous procedure developed by Young and Wilcock [79]. This method, which is based on the scheme used by Holland and Thake [80], is a simplified approach that keeps the fundamental aspects of the physics involved in the cooling process. In fact, this calculation scheme only requires a set of cycle related parameters, namely, cooling flow air temperature, inlet and outlet conditions at the engine turbines (total pressure, total temperature and mass flow), and fuel flow; together with the assumption of several empirical parameters. These empirical numbers avoid the complex calculation of the flow behaviour within the blade passages and in the external film, and the irregularities in the combustor outlet temperature pattern. The empirical parameters employed in the present analysis are listed in Table 3.2. In the absence of T700-GE-700 data, the values are based on those provided by Young and Wilcock, which were slightly adapted to match the actual engine CF . Some relevant thermodynamic cycle data are shown in Table 3.2, as well. These figures correspond to Maximum Continuous Power rating (MCP), which is chosen as the engine DP . The cooling air bleeds are sized at this power setting in order to ensure that the CF amount is sufficient throughout the entire engine power range (contingency situations are not considered).

The application of this CF sizing scheme provides the cooling flowrate required by each blade row separately, although the procedure for stator and rotor blades is fairly similar. The application of this method for a one-stage turbine is shown in Appendix A for clarification. The cooling flowrates calculated for the compressor turbine of the T700-GE-700 engine are compared with the actual engine cooling flows [74] in Table 3.3.

Note that the CF values in the table are defined as percentage of the engine inlet airflow. The actual engine cooling flowrate varies with compressor inlet airflow [74], so the value considered here corresponds to the maximum CF value attained by the engine. As expected, the cooling flow demanded by the turbine stages

TABLE 3.2: Cooling flows sizing parameters

Maximum metal temperature, T_{metal} [K]	1,104
Combustion pattern factor (rotor), $K_{b,rot}$ [—]	0.05
Combustion pattern factor (stator), $K_{b,sta}$ [—]	0.1
Cooling flow factor, K_{cool} [—]	0.039
Internal cooling efficiency, $\eta_{c,int}$ [—]	0.6
Film cooling effectiveness, ϵ_f [—]	0.2
Metal Biot number, Bi_{metal} [—]	0.2
Coolant supply total temperature [K]	703
Coolant supply total pressure [atm]	14.47
GGT 1 st stage inlet total temperature [K]	1,440
GGT 1 st stage inlet total pressure [atm]	13.89
GGT 2 nd stage inlet total temperature [K]	1,214
GGT 2 nd stage inlet total pressure [atm]	6.84

TABLE 3.3: General Electric T700-GE-700 GGT cooling flows distribution

	Cooling flow amount [%]	
	Module	Actual engine
1st stage		
Stator	4.976	—
Rotor	2.592	—
2nd stage		
Stator	0.889	—
Rotor	0.000	—
Total	8.457	8.510

reduces as the flow is expanded, and so its total temperature reduced. The overall cooling flowrate is comparable to the actual value of the GE T700-GE-700 engine with an acceptable error. In order to maintain the cooling technology level fixed through the current study, the cooling parameters listed in Table 3.2 are kept constant.

3.3 Powerplant weight modelling

The weight of the different turboshaft engine configurations is to be assessed by means of ATLAS. Since this software was originally developed to predict the weight of turbofan engines, some updates were introduced by the author in the code in order to expand the sizing and weight estimation to turboshaft engines. On

top of that, additional routines were also added to cater for the weight analysis of the auxiliary burners. Following ATLAS philosophy, care was taken during the upgrade process to reduce the number of inputs and assumptions required. Furthermore, the already developed subroutines, expressions and methods were used or adapted when possible. The main features added to the original ATLAS code deal with the following aspects:

- **Turboshaft intake sizing**
- **Turbomachinery disk temperature distribution**
- **Engine materials library expansion**
- **Centrifugal compressor weight prediction capability**
- **Free turbine shaft weight prediction capability**
- **Auxiliary reheaters sizing**

3.3.1 Turboshaft intake sizing

As observed in Fig. 3.5, turboshaft engines often incorporate an intake duct upstream of the first compressor stage. The design of this component is similar to the engine ducts already modelled in ATLAS. However, some distinct features of intake ducts justify the development of a separate routine:

- The engine duct routine in ATLAS retrieves the flow characteristics from the components placed upstream and downstream. In the case of the intake duct, since it is the first component of the engine, the velocity vector at the inlet must be defined by the author, whilst the thermodynamic state of the gas is obtained from the engine performance simulation tool.

- The fact that the foremost component is the intake duct implies that, unlike the rest of engine ducts, a dimension input (hub, tip or mean diameter) is mandatory, which serves as starting point of the gas path design.

Apart from the reasons just mentioned, the intake duct may present a wide range of complex shapes (such as the S-ducts in some supersonic fighters) and/or incorporate additional devices such as the inlet particle separator that prevents most of the air suspended dirt from being ingested by the engine. Therefore, the intake duct routine allows to include these particular features in future versions of the code and develop its design separately from the rest of engine ducts.

In the initial version of the intake duct routine, the design of the inlet particle separator is not attempted (its weight is considered, instead, within the *Controls and Accessories* category), and the component is modelled as two concentric conical parts. With the shape of the intake defined this way, its geometry is fully defined if the dimensions and flow characteristics at the engine inlet and the component downstream are known. The flow characteristics are defined by the ambient conditions, and the intake pressure losses, data which are provided by the engine performance tool. Apart from this information, the user must introduce some additional details:

a) Inlet velocity

Similar to compressors and turbines, a velocity vector at the intake inlet is required as input. Thus, one out of absolute Mach number, axial Mach number, absolute velocity or axial velocity, at intake inlet, together with the absolute velocity angle, must be provided.

b) Inlet reference dimension

A dimension input is necessary to define the intake inlet geometry. This dimension input may be the hub, tip or mean diameter at the inlet of the component.

c) Intake duct length

Regarding the length of the intake duct, it can be defined by three different ways:

(i) Intake duct angle

The angle the intake duct mean line forms with the horizontal plane is used to estimate the length of the component.

(ii) Intake duct length to inlet height ratio

The length to inlet height ratio can be also used as input, which is a non-dimensional parameter commonly used to define reasonable shaped ducts [78].

(iii) Intake duct projected length

The length of the intake duct projected on the horizontal direction is another possible option for the intake length estimation.

Finally, the thickness of the intake duct is calculated applying the pressure vessel theory, as it is done for the engine ducts:

$$t_{int} = \frac{0.101325 \cdot P_{st} \cdot D_{int}}{2 \cdot \sigma_y} \quad (3.2)$$

Where t_{int} is the intake thickness, P_{st} is the static pressure, D_{int} is the intake diameter and σ_y is the yield stress of the material. This expression indicates the theoretical minimum thickness required; however, for practical applications it would be recommendable to include a safety factor in Eq. 3.2 in a future version of ATLAS. This safety factor could be defined either internally in the code, or declared by the user as an input.

3.3.2 Turbomachinery disk temperature distribution

For the sizing of the compressor and turbine components, the stresses at every disk stage are computed to ensure that a feasible disk design is delivered by the code. Not only the mechanical properties of the materials are sensitive with temperature, but also the stress calculation includes temperature related terms due to thermal expansion phenomena. For these reasons, the temperature distribution along the disk cannot be disregarded during the disk design stage.

However, the description of this temperature field is not an easy task. The approach followed by the original version of the code bypasses this complexity by using a simple empirical curve to reproduce the temperature drop in the disk as a function of the radial position (Fig. 3.8). Whilst the mentioned correlation may produce realistic temperature values for turbine disks, it fails in the modelling of the temperature field for compressor disks, since the temperature drop in this

latter components is overestimated. As a result, unrealistic temperature values are computed by the code for the radial positions near the disk bore. Besides, the presence of the secondary air system (formed by the engine air bleeding) in contact with the disks bore may cause the temperature of the disk to increase rather than drop towards the bore. This is specially true for the foremost compressor stages, where the gas path temperature is lower than the temperature of the compressor bled air.

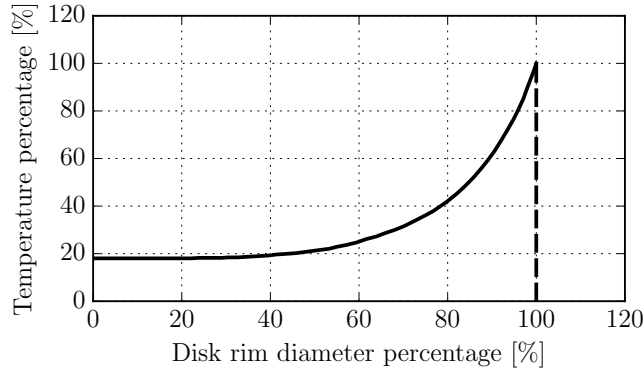


FIGURE 3.8: Original disk temperature distribution [81]

Therefore, towards a more thorough representation of the radial steady-state temperature distribution in the disks, a model based on Fourier's law of conduction [82] was implemented by the author. As opposed to the original empirical description, this new method accounts for the effect of the disk bore temperature in the disk temperature distribution:

$$T = T_{bore} + \frac{\Delta T}{\ln\left(\frac{r_{rim}}{r_{bore}}\right)} \ln\left(\frac{r}{r_{bore}}\right), \quad \text{where : } \Delta T = T_{rim} - T_{bore} \quad (3.3)$$

In previous expression, T_{bore} and r_{bore} refer to the temperature and radial distance at the disk bore, respectively; whilst T_{rim} and r_{rim} correspond to the temperature and radial distance at the disk rim. A similar expression has been used by Tong *et al* [83] to enhance the weight-calculation routines for the compressor and turbine disks implemented in the NASA's engine weight computer code *WATE*.

In the code, it has been assumed that the bore disk temperature, T_{bore} , is the same for all the engine disks, and equal to the static temperature of the air at the last compressor outlet. Thus, this assumption considers that the air circulating

by the secondary system is bled at the last compression stage, which is not far from the design practice followed on real engines. This premise is also adopted in the original version of ATLAS in order to estimate the temperature of the shafts material.

Regarding the rim disk temperature, T_{rim} , the original version of the code considered its value to be equal to the static temperature at the rotor station. This assumption, whilst reasonable for the cold section of the engine, leads to unrealistic high temperature values for the hot section, and so, to distorted temperature gradients along the disk. In fact, the blades operating at temperatures above the material limits are internally cooled so the material is kept within its operational limits. Therefore, in the updated version of the code, it is assumed that T_{rim} is equal to the material limit temperature of the blade for those stations where the gas static temperature exceeds this value. The limit value for the material temperature is declared by the user.

With these considerations, and in order to graphically explain the temperature distributions obtained with Eq. 3.3, the temperature fields for a compressor and a turbine disks operating at representative temperatures are reproduced in Fig. 3.9.

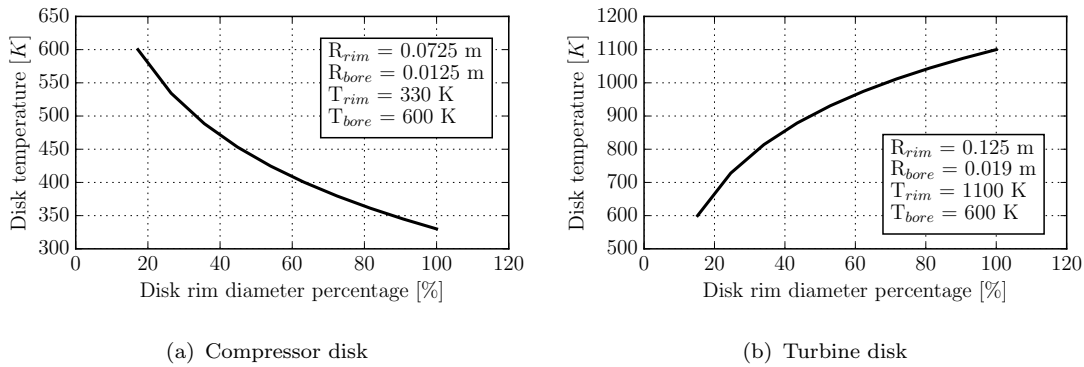


FIGURE 3.9: Updated disk temperature distribution

In this figure, the difference between the compressor and turbine disk temperature fields is evident. As explained before, the higher temperature at the disk bore, for the compressor case, causes an increase in disk temperature towards this section of the disk. This trend is opposed to that predicted by the empirical distribution of Fig. 3.8. Regarding the turbine disk, the behaviour of the trend is similar to the original correlation, although the temperature gradient predicted by Eq. 3.3 is more gradual. In fact, for the particular case reproduced, the disk temperature at 20% of the disk rim is about 60% the temperature at the disk rim according to

Fig. 3.9(b), whilst, the temperature predicted by the original correlation at the same position is circa 20%.

3.3.3 Engine materials library expansion

New materials were included in the ATLAS library to allow for the realistic weight estimation and stress prediction of turboshaft engines components. The new list of materials adds some light alloys typically used in turboshaft engine intakes and ducts, as well as, some steel and titanium alloys employed in the compressor section. The new material files kept the same structure of the original ones, listing the properties of the material *i.e.* yield stress, Ultimate Tensile Stress, Poisson's ratio, Young's Modulus and thermal expansion coefficient, as a function of temperature.

In particular, the properties of the materials employed in the production of the T700-GE-700 engine were added, such as, the AM355 steel (used to build the compressor 'blisk' stages [75]) and the Rene 95 superalloy (used in the turbine disks manufacture [84]). Apart from these two materials, a couple of titanium alloys, IMI 834 and Timet TI 1100, were also included in the list. These two materials are used in compressor blades, discs and casings, and offer an increased temperature capability than the Ti-6Al-4V alloy [85], which is the titanium alloy of use in the original ATLAS version. Finally, a light aluminium cast alloy, the Al 222, was included in order to model the light intakes/ducts some turboshaft engines incorporate. The properties of the cited materials were obtained from [86–92].

3.3.4 Centrifugal compressor weight prediction capability

Current turbofan engines work with axial compressors. The difficulties of obtaining a high pressure ratio at a high efficiency with centrifugal compressors, together with their low flow rate for given frontal area, make axial flow compressors much more suitable for current high-mass flow turbofan engines [85]. Therefore, no routines are included in the original version of ATLAS to account for centrifugal compressor geometries. However, the use of centrifugal or radial turbomachinery is extended for other applications such as small turboprops, turboshafts and auxiliary power units. Centrifugal flow compressors are attractive due to the ability to

handle small mass flows, reduced length compared with an equivalent axial compressor, improved resistance to foreign object ingestion damage, less susceptibility to loss of performance by debris deposits on the blade surfaces, and the capability to operate over a wider range of mass flow at a particular rotational speed [85]. As observed in Fig. 3.5, the baseline engine is equipped with a centrifugal compressor connected with a 5-stage axial compressor. Thus, the author implemented some routines to account for the sizing of the centrifugal compressor component.

The components to be included in the centrifugal compressor weight analysis are shown in Fig. 3.10. The parts considered by Lolis in the axial compressor weight estimation are included for comparison. It is worth to mention that the centrifugal disk is more commonly known as 'impeller' in the open literature, so this term will be used hereafter as well.

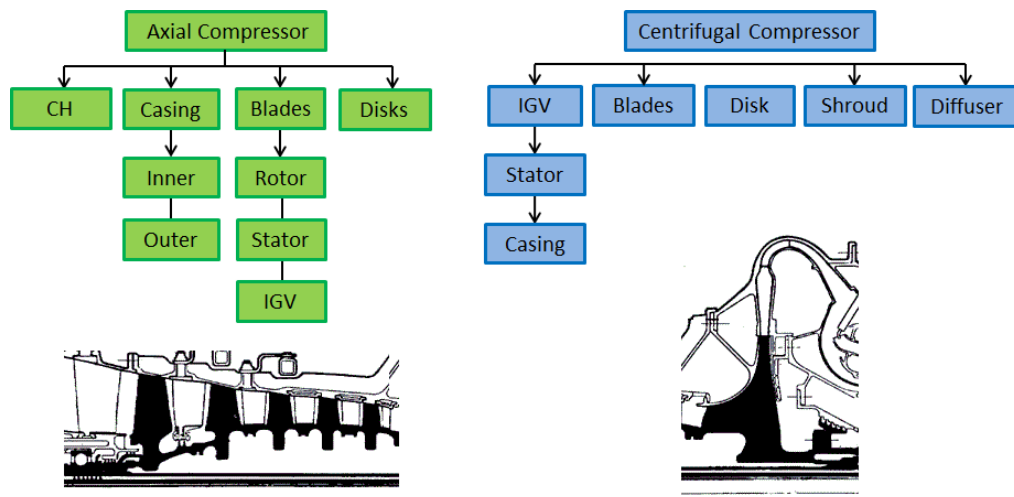


FIGURE 3.10: Axial and centrifugal compressor weight parts

The elements shown in this figure are the same employed by Hale in the WATE-S code and reported in [93]. In the case an Inlet Guide Vane (*IGV*) stage is placed upstream the centrifugal compressor, its weight is computed following the procedure used for *IGV* stages in axial compressors.

3.3.4.1 Centrifugal compressor gas path designation

Similar designation scheme followed by Lolis (Fig. 3.11(a)) was used for the centrifugal compressor. In this component, the rotor and stator blades have been replaced by a rotor blade and a diffuser, as shown in Fig. 3.11(b). However, the stations numbering is conserved, with the stage inlet and outlet being numbers

1 and 4 respectively, and the blade outlet and the diffuser inlet being numbers 2 and 3.

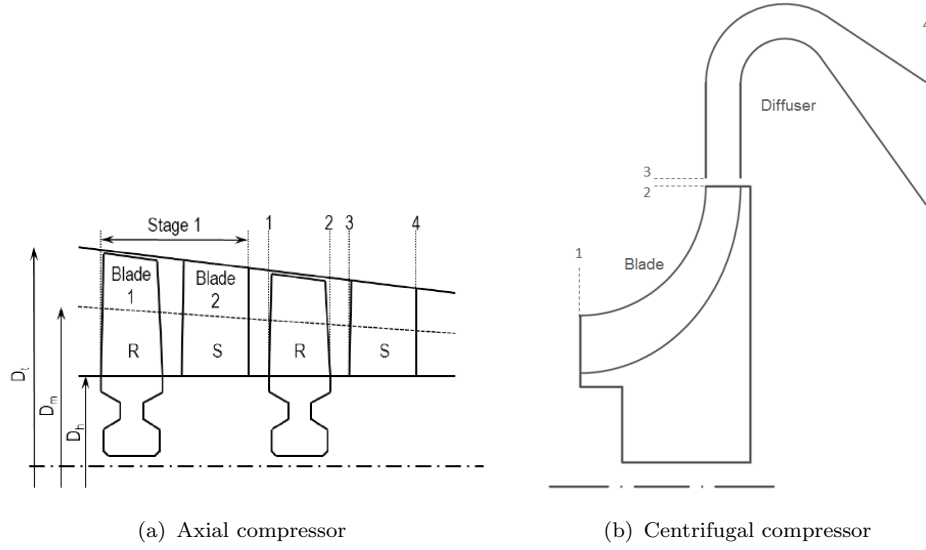


FIGURE 3.11: Gas path stations numbering

3.3.4.2 Annulus design

Similarly to the axial compressor, in order to estimate the weight of the centrifugal compressor blades, it is necessary to determine first their shape and dimensions. The principles used herein to design the flowpath are found in [85, 93, 94]. A simplistic method with the minimum amount of inputs required is sought, so complex aerodynamic calculations are to be reduced to a minimum following the approach set by Hale for centrifugal compressors sizing [93]. As for the axial compressor design, several input parameters are required to define the annulus shape. The list of input parameters can be divided in four groups:

a) Cycle Parameters

The cycle parameters required by the centrifugal compressor are the same than the ones required by the axial compressor i.e. mass flow at the component inlet, total pressure and total temperature at the inlet and outlet of the component, FAR at component inlet, and PCN . These data may be acquired from a performance simulation code, such as TURBOMATCH.

b) Velocities

As for the axial compressor, a velocity vector at the component inlet is

required, so the necessary annulus area of the compressor inlet can be estimated. To define the velocity vector, one out of absolute Mach number, axial Mach number, absolute velocity or axial velocity at the component inlet is needed, together with the absolute velocity angle at the component inlet. The velocity value is normally chosen based on existing engines or design guidance. A typical value of 150 m/s is suggested by Saravanamuttoo for the inlet axial velocity [85], this ensuring a suitable compromise between high flow per unit frontal area and low frictional losses at the inlet.

c) Annulus Dimensions

For the complete definition of the centrifugal compressor, a dimension input is required at the component inlet, namely, one out of hub diameter, tip diameter, mean diameter or hub to tip ratio (h/t). On top of that, the inlet hub to outlet tip radius ratio (h/t_{exit}) is required in order to obtain the outlet dimensions:

$$h/t_{exit} = \frac{R_{h1}}{R_{h2}} \quad (3.4)$$

Where R_{h1} and R_{h2} are the hub radii at the blade inlet and outlet, respectively. There exist some reference values available in the open literature for this parameter. In particular, Onat *et al* employed a value of 0.333 in the NASA *WATE* code [95], typical for wheels from 10 (0.254 *m*) to 20 (0.508 *m*) in diameter. With the outlet tip radius (R_{t2}), which is equal to R_{h2} , and the blade outlet area (A_2), the exit blade length (BL) can be calculated:

$$BL = \frac{A_2}{2\pi \cdot R_{t2}} \quad (3.5)$$

Once the blade inlet and outlet dimensions are known, the shape of the blade hub and shroud must be detailed in order to fully define the annulus layout. Following the approach set by Hale [93], the gas path is assumed to have an elliptical shape, as shown in Fig. 3.12. The shape of an ellipse is defined by the expression:

$$\frac{x^2}{a^2} + \frac{y^2}{b^2} = 1 \quad (3.6)$$

Where a and b represent the ellipse semi-axis on the x and y directions, respectively. The values of a and b for the blade hub and shroud are defined in Table 3.4:

TABLE 3.4: Ellipse semi-axis values for centrifugal blade sizing

	Blade Hub	Blade Shroud
Semi-axis a	$R_{t_2} - Rh_1$	DH
Semi-axis b	$DL + BL$	DL

The centrifugal blade radial (DH) and axial (DL) dimensions are defined as:

$$DH = R_{t_2} - R_{t_1} \quad DL = 2 \cdot R_{t_2} \cdot L/D - BL \quad (3.7)$$

Where L/D is defined as the ratio between the axial blade length and the outlet blade tip diameter. Its value is a trade-off between reduced skin friction (low L/D), and the need for an adequately gentle flow turning from axial to radial direction (high L/D). Came *et al* suggest values of 0.32-0.37 based on typical centrifugal impeller diameters [94].

d) Impeller blade parameters

The last input to be introduced is the number of blades of the centrifugal impeller (n_b). This parameter, which is calculated by ATLAS for every axial stage, is set as an input for the centrifugal compressor case. In the industry, cost and manufacturing aspects are considered in the selection of number of blades; however, aerodynamic performance is the dominant parameter. Some guidance is found in the literature about the impeller number of blades. For example, Saravanamuttoo [85] affirms that a common practise is to use a number of blades which gives a slip factor around 0.9, *i.e.* about 19 to 21 blades (see Eq. 3.20). This design decision is a compromise between the work capacity increase and the friction losses increment in the impeller. Came *et al* [94] introduced the definition of a parameter, Ψ , analogous to the space-chord ratio used in axial compressors:

$$\Psi = \frac{\pi \cdot R_{t_2}}{n_b \cdot (DL + BL)} \quad (3.8)$$

stage. These assumptions, that are summarised below, are based on common design practice.

- a) **Radial velocity at impeller outlet (V_{rad_2}) equal to inlet axial velocity (V_{ax_1})**

In order to calculate the area at the impeller outlet, the velocity at this station is required. As observed in the open literature [85], a suitable assumption is to make it equal to the axial velocity at the impeller inlet.

- b) **Constant total temperature through the centrifugal compressor diffuser ($T_{tot_2} = T_{tot_3} = T_{tot_4}$)**

The total temperature is also required to evaluate the density at the impeller outlet. An adiabatic expansion process is assumed to occur in the compressor diffuser, so the total temperature at the impeller outlet is equal to the total temperature at the outlet of the compressor, which is known from the engine cycle data.

- c) **Half of the compressor total loss assumed to occur in the impeller**

Apart from the total temperature, the total pressure at the impeller outlet (P_{tot_2}) is needed in order to evaluate the density at this station. To calculate this parameter, it is assumed that impeller and diffuser evenly contribute to the total compressor loss [85], so the total pressure at the impeller outlet can be evaluated through the impeller inlet conditions, the outlet total temperature, and the compressor isentropic efficiency (Eq. 3.16).

The mentioned limitations and assumptions are integrated in the design process described herein:

- a) **Compressor isentropic efficiency (η_c) evaluation**

The isentropic efficiency of the centrifugal compressor is evaluated with the following equation:

$$\eta_c = \frac{PR_c^{\frac{\gamma_{av}-1}{\gamma_{av}}} - 1}{TR_c - 1} \quad (3.9)$$

In previous expression, γ_{av} is the average specific heat ratio between the compressor inlet and outlet values, and PR_c and TR_c corresponds to the compressor total pressure and temperature ratios, respectively.

b) Component impeller inlet & outlet calculations

The velocities, annulus area and diameters are calculated for the compressor inlet and outlet using the available inputs. To exemplify the inlet calculation process, it is assumed that the inlet axial velocity (V_{ax_1}) is known, together with the total temperature (T_{tot}) and pressure (P_{tot}), FAR and absolute velocity angle at the component inlet. Inlet annulus area (A_1) can be estimated by using the equation of continuity:

$$w = \rho \cdot V_{ax_1} \cdot A_1 \quad (3.10)$$

The density (ρ) is calculated by the gas law equation (Eq. 3.13), which requires the static temperature (T_{st}) and pressure (P_{st}) values. These two parameters are obtained by using the isentropic flow equations:

$$\frac{T_{tot}}{T_{st}} = 1 + \frac{\gamma - 1}{2} \cdot M_{abs}^2 \quad (3.11)$$

$$\frac{P_{tot}}{P_{st}} = \left(1 + \frac{\gamma - 1}{2} \cdot M_{abs}^2\right)^{\frac{\gamma}{\gamma - 1}} \quad (3.12)$$

$$\rho = \frac{101325 \cdot P_{st}}{R_{air} \cdot T_{st}} \quad (3.13)$$

The absolute Mach number (M_{abs}) is obtained from its definition:

$$M_{abs} = \frac{V_{abs}}{\sqrt{\gamma \cdot R_{air} \cdot T_{st}}} = \frac{V_{ax}}{\cos \alpha \cdot \sqrt{\gamma \cdot R_{air} \cdot T_{st}}} \quad (3.14)$$

Once the inlet area is known, the diameters at this station can be easily calculated given one out of inlet hub diameter, tip diameter, mean diameter or hub to tip ratio, using the following expression:

$$A_1 = \pi \cdot (R_{t_1}^2 - R_{h_1}^2) \quad (3.15)$$

For the calculation of the outlet annulus area (A_2) a similar approach is followed. The continuity equation (Eq. 3.10) is applied at the outlet station, as well. Therefore, the static temperature and pressure at station 2 are required in order to calculate the density at this station. As previously mentioned,

the total pressure at this station (P_{tot2}) is first calculated assuming that half of the centrifugal compressor losses occurs in the impeller:

$$P_{tot2} = P_{tot1} \cdot \left[1 + \frac{[1 - 0.5 \cdot (1 - \eta_c)](T_{tot2} - T_{tot1})}{T_{tot1}} \right]^{\frac{\gamma_{av}}{\gamma_{av}-1}} \quad (3.16)$$

The total temperature at station 2 (T_{tot2}) is assumed to be equal to the total temperature at the diffuser outlet (T_{tot4}). With this value, the static temperature at station 2 (T_{st2}) is evaluated as:

$$T_{st2} = T_{tot2} - \frac{V_{abs2}^2}{2 \cdot C_p} \quad (3.17)$$

Where the absolute velocity at station 2 (V_{abs2}) is calculated using the following expressions:

$$\cos \alpha_2 = \frac{V_{rad2}}{V_{abs2}} \quad \tan \alpha_2 = \frac{V_{w2}}{V_{rad2}} \quad (3.18)$$

In previous equations, V_{w2} refers to the whirl (or tangential) component of the impeller outlet velocity. The velocity components at this station are graphically represented in Fig. 3.13. Under ideal conditions, the whirl component is equal to the impeller tip speed (U_2). However, in real wheels, the air trapped between the impeller vanes is reluctant to move round with the impeller due to its inertia. This phenomenon, known as slip, prevents the air from acquiring a whirl velocity equal to the impeller speed. Therefore, it is necessary to define a slip factor (s_f) to account for the slip phenomenon in the design of centrifugal impellers (Eq. 3.19). As suggested in [85], the definition to be used for the slip factor (Eq. 3.20) is the one derived by Stanitz [96] due to its good agreement with experimental data.

$$V_{w2} = s_f \cdot U_2 = s_f \cdot R_{t2} \cdot \omega_{bl} \quad (3.19)$$

$$s_f = 1 - \frac{0.63\pi}{n_b} \quad (3.20)$$

Once the static temperature at station 2 (T_{st2}) is known, the static pressure at this station is finally obtained (Eq. 3.21) and the density at station 2 can be calculated.

$$\frac{P_{tot}}{P_{st}} = \left(\frac{T_{tot}}{T_{st}} \right)^{\frac{\gamma}{\gamma-1}} \quad (3.21)$$

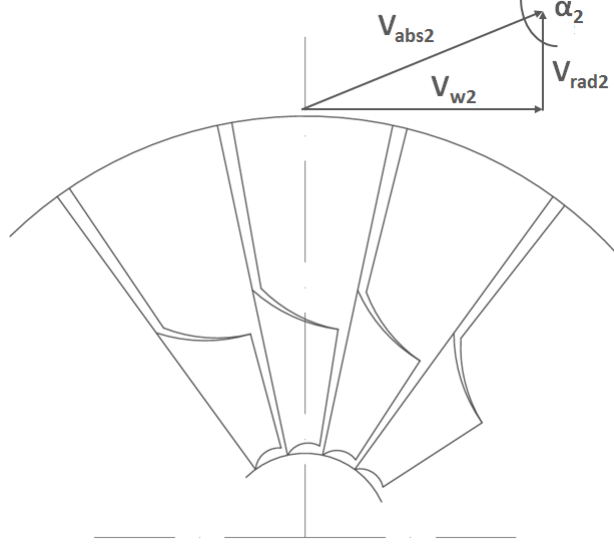


FIGURE 3.13: Velocity triangle at centrifugal impeller outlet

The procedure explained here to obtain the conditions at the inlet and outlet of the impeller is fairly similar to the method applied by Saravanamuttoo in his example design of a single-sided centrifugal compressor [85], with some clarifications:

- In the textbook example, the author uses A_1 , w_1 , and inlet total conditions (T_{tot1} , P_{tot1}) to derive V_{ax1} and ρ_1 , whilst, in the case of ATLAS, T_{tot1} , P_{tot1} , w_1 and V_{ax1} are used to derive ρ_1 and A_1 .
- For the estimation of the impeller outlet area (A_2), same set of assumptions is used, *i.e.* V_{rad2} equal to V_{ax1} , T_{tot2} equal to T_{tot4} , and half of the compressor total losses are assumed to occur in the impeller. Moreover, the absolute velocity at the impeller outlet (V_{abs2}) is also derived having the slip factor (s_f) into account (in the textbook example its value is an input, whilst in ATLAS its value is calculated as a function of the number of blades).

c) Blade shape definition

Based on the elliptical shape assumption, the parameters in Table 3.4 are used in Eq. 3.6 in order to define the geometry of the blades.

3.3.4.3 Blade weight

Similarly to the axial blades, the weight of the centrifugal blades is computed using basic physic principles:

$$W_{bl} = vol_{bl} \cdot \rho_{bl} \quad (3.22)$$

Blade density (ρ_{bl}) is known since the blade material is chosen by the user. As higher pressures, and so temperatures, are expected in the centrifugal compressor, materials with good creep resistance, high thermal conductivity and low thermal expansion are commonly used. Titanium alloys offer these properties together with a relatively low density. Therefore, the materials library was expanded with two titanium alloys, namely, IMI 834 and Timet TI 1100, which have a good temperature capability up to 873 K (600 °C) and high creep and corrosion resistance [85].

Blade volume is calculated as a product of the blade lateral area (A_{bl}) and blade thickness (t_{bl}). The blade lateral area is computed using Eq. 3.23, whilst for the blade thickness an average value between hub and shroud thickness is employed. The blade thickness at the hub is calculated using the expression provided by Hale [93] (Eq. 3.24), whilst the thickness at the shroud is computed following Came *et al* approach [94]. These authors state that the blade thickness at the shroud is fixed by manufacturing considerations and should be as thin as possible for optimum efficiency. They affirm that 0.3% of the impeller tip diameter is achievable (Eq. 3.25), with a lower limit of 0.35 mm. An upper limit of 0.76 mm is also introduced in the code based on the design guidance provided by Hale.

$$A_{bl} = \frac{\pi}{4} \cdot (a_h \cdot b_h - a_s \cdot b_s) \quad (3.23)$$

$$t_h = 0.15 \cdot \frac{2\pi \cdot R_{h1}}{n_b} \quad (3.24)$$

$$t_s = \frac{0.3}{100} \cdot 2 \cdot R_{t2} \quad 0.00035 < t_s < 0.00076 \text{ m} \quad (3.25)$$

3.3.4.4 Impeller disk weight

Same design philosophy followed in ATLAS for the axial stages disks is applied for the impeller disk design. It means that an optimisation process is implemented in order to find the minimum volume solution that satisfies the stress limitations imposed by its shape and its material. The disk design to be used in the sizing process is similar to the one shown in Fig. 3.12. Once the optimal disk geometry is found, the weight contribution of the impeller disk is calculated as:

$$W_{dsk} = vol_{dsk} \cdot \rho_{dsk} \quad (3.26)$$

Several aspects of the impeller disk design and geometry optimisation are detailed herein.

Impeller disk stress calculation

For the disk stress calculation, the *sum and difference* method implemented in ATLAS for the stress analysis of axial disks is employed. The description of this method and the set of equations included on it are explained by Lolis [54] and can be found as well in the open literature [81, 97–99], thus further elaboration is omitted. In fact, this finite difference method has been successfully used in centrifugal disk stress analysis in the past by Ray *et al* [100] and by Tumarkin [101]. This author made use of the *equivalent thickness* technique to account for the centrifugal forces induced by the laterally attached blades (see Fig. 3.12). This technique consist in adding to the disk contiguous element a certain nominal thickness, t_{nom} , of the same disk material which does not carry any stresses, but gives an added centrifugal force; t_{nom} is defined through the area of the cylindrical section of the blade, f , at the disk element mean height radius, R_{em} , as shown in Eq. 3.27.

$$t_{nom} = \frac{n_b \cdot f}{2\pi \cdot R_{em}} \quad (3.27)$$

$$f = h_e \cdot t_{be} \quad (3.28)$$

In Eq. 3.28, h_e is the disk element height and t_{be} is the blade element thickness. As a result of the increased thickness, the density of the disk element is increased

whilst the same loaded area is maintained. The definition of this equivalent density is shown in Eq. 3.29.

$$\rho_{eq} = \rho_e \cdot \left(1 + \frac{t_{nom}}{t_e} \right) \quad (3.29)$$

The discretisation of the impeller blades and disk for the stress analysis together with the dimensions aforementioned are reproduced in Fig. 3.14. The blade has been divided in twelve constant thickness elements for visualisation purposes. However, the number of elements used in the blade/disk discretisation is specified by the user as an input. This approach is different to the one applied to axial stages, where the disk geometry is simulated with a fixed number of five discrete elements. The choice of discretisation elements for the centrifugal stage is a trade-off between stress analysis accuracy (high number of elements) and moderate computational time (low number of elements).

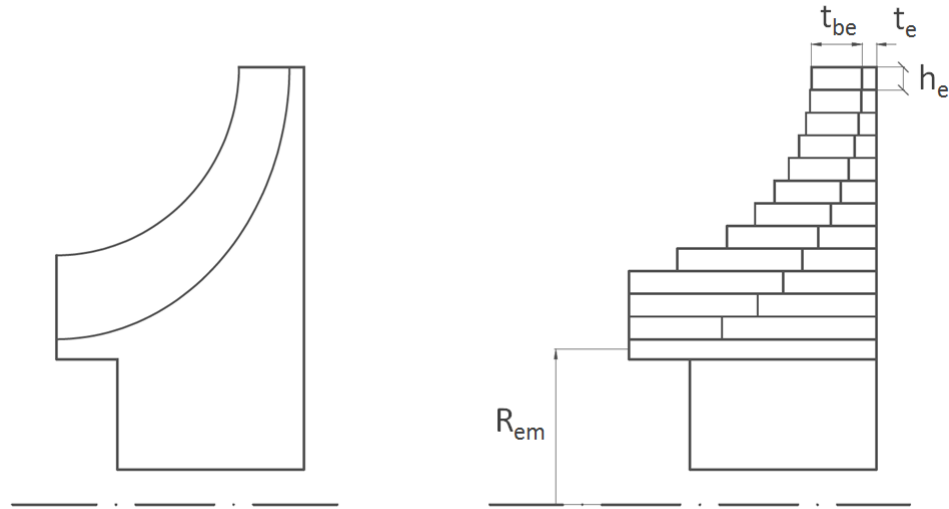


FIGURE 3.14: Impeller blade and disk discretisation for stress analysis

In a similar fashion to axial components, the stress values at the inner disk radius are required to calculate the stresses at the different sections of the disk. The radial stress at this position is assumed equal to zero, since there are no centrifugal loads at the inner diameter of the disk, whilst the tangential stress is selected through a trial and error method so that the calculated radial stress at the outer disk radius matches the blade centrifugal stress at that disk station, which for the centrifugal impeller is zero, as no blades are attached at the disk rim.

In Fig. 3.15, the variation of the tangential stress at the disk bore, $\sigma_{\theta_{bore}}$, with number of discretisation elements is analysed. For the geometry assumed (Fig. 3.15(a)), it is observed that beyond 100 elements, increasing the number of elements does not offer a significant change in the stress estimation (Fig. 3.15(b)). Therefore, the default number of elements set in ATLAS for the discretisation of the centrifugal compressor is 100.

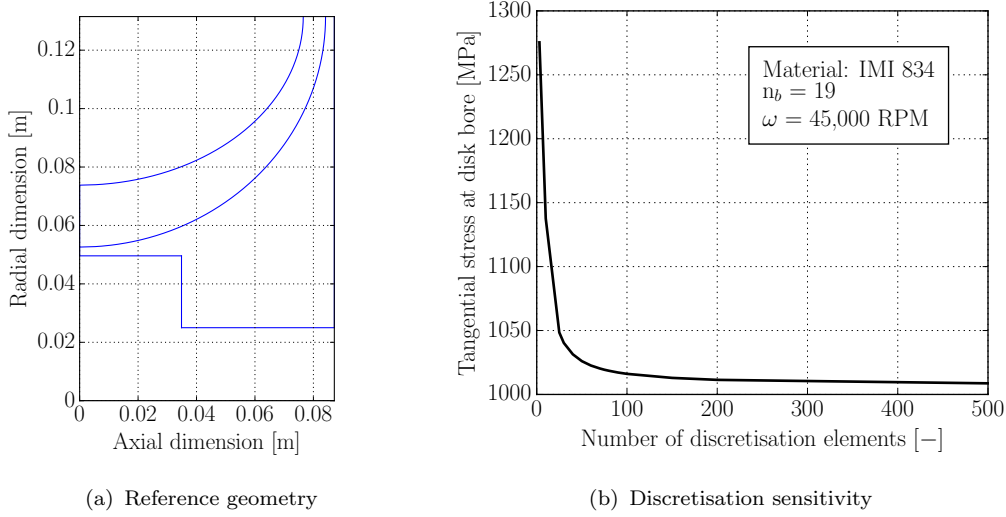


FIGURE 3.15: Analysis of the centrifugal compressor discretisation sensitivity

Once the stress analysis is complete, the calculated tangential and radial stresses are used to assess the disk design integrity. Same failure criteria employed by Tong *et al* [98] and already implemented by Lolis in ATLAS for the axial components analysis are used (Eq. 3.30).

$$\frac{\sigma_y}{SF \cdot \sigma_{VM}} - 1 > 0 \quad \frac{0.47 \cdot \sigma_{UTS}}{\sigma_{\theta_{av}}} - 1 > 0 \quad (3.30)$$

Impeller disk temperature distribution

Material properties are sensitive to the significant temperature gradients present in turbomachinery disks. Therefore, the temperature distribution along the disk must be considered for an accurate disk stress analysis and sizing. The temperature gradient in the centrifugal impeller is different to the axial disk due to the fact that there is a region where the hot air of the gas path is in direct contact with the disk walls as observed in Fig. 3.14. No information was found in the public domain about the modelling of the temperature distribution in centrifugal disks, so a linear gradient was considered for this region as an initial assumption (Fig.

3.16). This approach captures the progressive increase in temperature the gas experiences as it is compressed in the rotor blade stage. The derivation of a more exhaustive temperature distribution is considered as future work. On the other hand, the disk region that is not in direct contact with the gas path can be treated similarly to an axial disk with a given set of rim and bore temperatures. Hence, the same temperature distribution introduced in Section 3.3.2 is used to model the temperature gradient in this region (Fig. 3.16).

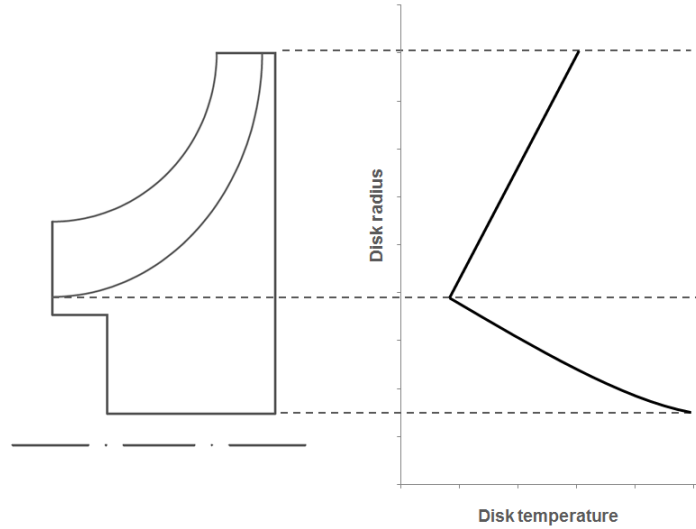


FIGURE 3.16: Centrifugal disk temperature distribution

3.3.4.5 Minimum volume impeller

The volume of the centrifugal impeller is a function of its geometry, which is fully defined by the dimensions shown in Fig. 3.12: R_{bore} , t_h , t_t and t_{rim} , and the ratios L/D and h/t_{exit} . Therefore, in the case of the centrifugal compressor, six variables are required to fully define the disk shape, compared with the twelve dimensions employed in the axial disk geometry. Another difference in the design of the centrifugal impeller disk is observed in Fig. 3.12. In this sketch it can be appreciated that, a change in the shape of the disk will alter the geometry of the impeller blades. In other words, the shape of the blades is coupled with the geometry of the impeller. This feature requires that, in order to effectively select the minimum weight configuration, the sum of the blades and impeller wheel volumes must be minimised. The sizing problem, similarly to axial disks, is solved iteratively by using an optimisation routine that minimises the total

volume, respecting the stress limits defined in Eq. 3.30. Finally, some constraints are introduced in the code to ensure a realistic impeller shape:

- a) The thickness of the disk at the blade hub (t_h) and tip (t_t), is limited between 1.5 mm and 6 mm.

$$0.0015 \text{ m} \leq t_h, t_t \leq 0.006 \text{ m} \quad (3.31)$$

- b) The bore radius (R_{bore}) has to be greater than the maximum shaft radius ($R_{sh_{max}}$) multiplied by a spacing factor, which is set as 1.1 as it is done for axial disks. Besides, the bore radius value is limited by the inlet hub radius (R_{h_1}) and the minimum disk thickness at the hub ($t_{h_{min}}$).

$$1.1R_{sh_{max}} \leq R_b \leq R_{h_1} - t_{h_{min}} \quad (3.32)$$

- c) The range of reasonable values for the inlet hub to outlet tip radius ratio, h/t_{exit} , is based on Onat & Klees criterium (0.33, for typical impeller wheels) [95], and the range of values observed by Takeda [102] for typical impeller designs (0.6-0.7).

$$0.33 \leq h/t_{exit} \leq 0.7 \quad (3.33)$$

- d) The value of the ratio between axial blade length and outlet blade tip diameter, L/D , is limited by the range of values suggested by Came *et al* [94].

$$0.32 \leq L/D \leq 0.37 \quad (3.34)$$

- e) The ratio of axial length to radial height of the blade, DL/DH is limited to a range of values based on the assumption considered by Hale [93], who sets a value of 2 for this parameter.

$$1.75 \leq DL/DH \leq 2.25 \quad (3.35)$$

3.3.4.6 Centrifugal compressor shroud and diffuser weight

The weight of the rotor blade shroud, W_s , and the diffuser, W_{diff} , complete the total weight estimation of the centrifugal component. Following the approach described by Hale [93], the weight of these parts is calculated empirically (Eqs. 3.36 and 3.37). Therefore, an exhaustive sizing process like the one applied to the impeller gas path and the disk is not required.

$$W_s = 0.1673 \cdot \rho_s \cdot \left(\frac{R_{t_1} + R_{t_2}}{2} \right)^3 \quad (3.36)$$

$$W_{diff} = 0.2845 \cdot \rho_{diff} \cdot R_{t_2}^3 \quad (3.37)$$

3.3.5 Free turbine shaft weight prediction capability

An exclusive feature of turboprop and turboshaft engines, and industrial gas turbines is the presence of a shaft that transmits the power extracted from the FT to an output device (propeller, rotor or electricity generator). The length of this output shaft cannot be properly estimated in the original ATLAS code due to the fact that the shaft is not connected with an engine compressor. In other words, the original code estimates the length of an engine shaft as the distance between the compressor and the turbine that shaft is connecting. Therefore, the code is not able to estimate the length of the shaft driven by the FT , and so, its weight cannot be properly predicted. Besides, the connection between the FT shaft and the output device may take place at the front of the engine, as it is the case of the T700-GE-700 turboshaft engine. This arrangement implies that the most inner shaft of the engine is the output shaft, thus, modifying the diameters of the rest of shafts and their weight, accordingly. For all the mentioned reasons, the capability to account for the presence of an output shaft was implemented by the author. This update will allow not only to predict the weight of turboshaft and turboprop engines, but also the weight of industrial gas turbines. Although, in this latter application the weight is not as relevant as it is in aviation, it may influence the final price of the power generation plant.

Therefore, in order to distinguish the type of shaft, another turbine category, the free power turbine (FT), was added to the list already existing in ATLAS:

- *HPT*: High Pressure Turbine
- *IPT*: Intermediate Pressure Turbine
- *LPT*: Low Pressure Turbine

The design of the *FT* is completed using the same principles applied in the *LPT* sizing. However, the new category allows to identify and properly size the shaft that transmits the power generated by the engine. For this kind of shafts, the user may choose between two options: front connection and rear connection, depending on the position where the shaft is connected to the output device. An example of each kind of arrangement is displayed in Fig. 3.17, where a front connected turboshaft engine and an industrial gas turbine connected to the electrical generator at the rear part of the engine are reproduced.

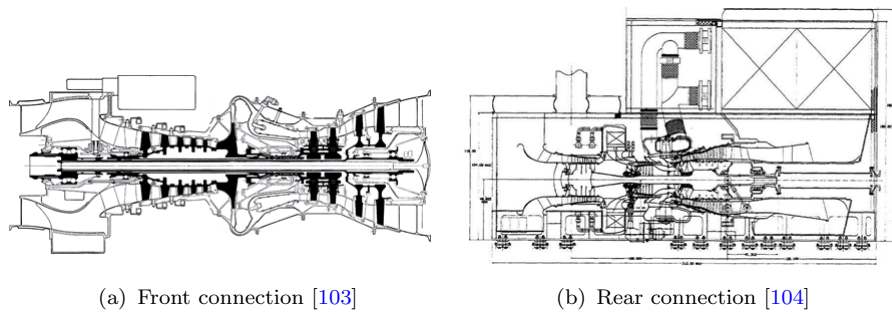
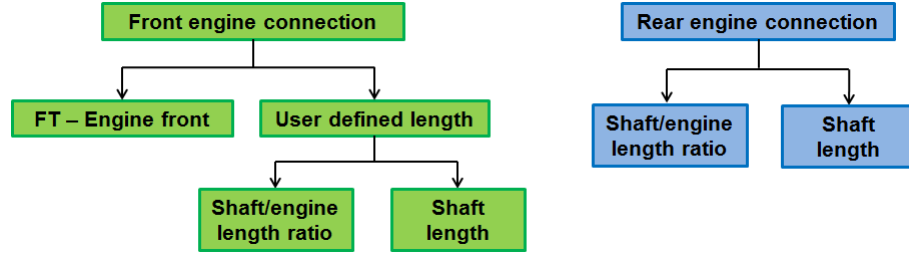


FIGURE 3.17: Types of power shaft connection

Note that in case (a), the output shaft is the most inner shaft, thus, affecting the diameter dimensions of the rest of shafts, whilst for case (b) the output shaft is decoupled from the rest of engine shafts. For both cases, the length of the shaft can be introduced by the user, either as the actual value or as a ratio between the length of the output shaft and the length of the engine. Besides, in case (a), the user can select the length of the shaft to be internally calculated by the code as the distance between the *FT* and the engine inlet. The different design choices are summarised in Fig. 3.18.

3.3.6 Auxiliary reheaters sizing

The additional heat addition present in the reheated engine architectures occurs in auxiliary burners placed between the turbine stages. The geometry and features

FIGURE 3.18: *FT* shaft length calculation options

of these components are not necessarily similar to those of the main combustor. Hence, a new brick was added to the code in order to tackle the geometrical design and sizing of the reheaters independently, as it was done for the engine intake.

However, the annular combustor structure was used as starting point for the initial design of the reheater geometry. As observed in Fig. 3.19, a diffuser is placed at the inlet of the reheater to lower the velocity of the gas, after which the flow is split into two streams: one led to the dome, and the other driven into the cooling passage. This constitutes the unique difference in comparison with the annular combustor design, where the flow undergoes two diffusion processes at the pre-diffuser and the dump diffuser, respectively.

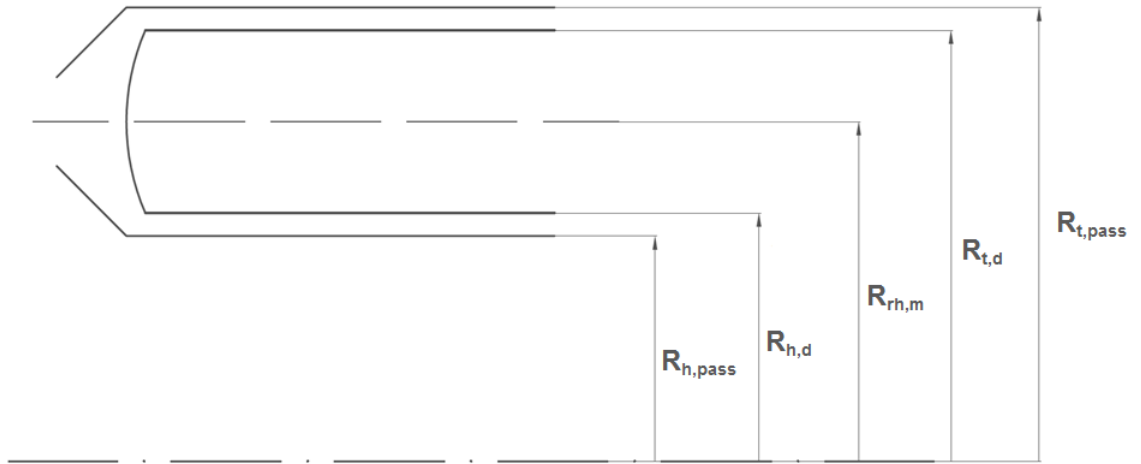


FIGURE 3.19: Reheater layout

These two reheater components, the dome and the cooling passage, have a duct shape with an outer and an inner ring. Therefore, the methodology followed to size the engine ducts is applied to the estimation of the reheater weight. A preliminary design process is followed in order to calculate the annulus dimensions and length

of each reheater component (Fig. 3.20). Note that as described before, the pre-diffuser and dump diffuser of the annular combustor are replaced by a unique diffuser component.

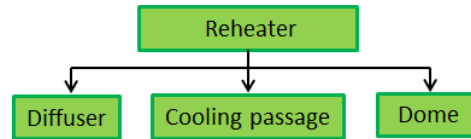


FIGURE 3.20: Reheater weight parts

3.3.6.1 Reheater design

The design procedure followed for the reheater sizing is based on the method already implemented in ATLAS for the combustor sizing. The inputs required to define the shape of the reheater are classified in four groups:

a) Cycle Parameters

The inputs listed in this category include the mass flow, total pressure, total temperature and FAR at the inlet and outlet of the component. These data may be obtained from a performance simulation code, such as TURBOMATCH.

b) Velocities

Similarly to other engine components, the velocity vectors at the inlet and outlet of the reheater are required. This means that, at these stations, one out of absolute Mach number, axial Mach number, absolute velocity or axial velocity is needed, together with the absolute velocity angle. These values can be retrieved from the components placed upstream and downstream of the reheater. Apart from these velocity vectors, the axial velocities at the dome and cooling passage inlet, respectively, have to be also provided.

c) Annulus Dimensions

The annulus dimensions required for the preliminary design of the reheater are the same than those needed for the combustor. Therefore, either the hub diameter, tip diameter, mean diameter or the hub to tip ratio is required for the inlet and outlet stations, respectively. Similarly to the inlet and outlet velocities, the annulus dimensions can be either supplied by the user or taken from the upstream and downstream components.

d) Reheater dimension parameters

Finally, some additional input parameters are required from the user to complete the reheater design which are listed as follows:

- Atomising Flow Ratio (AFR)
- Dome Cooling Flow Rate ($DCFR$)
- Stoichiometric Fuel to Air Ratio (FAR_{stoi})
- Primary Zone Equivalence Ratio ($PZER$)
- Dome length to height ratio (l/h_d)

These parameters have been already used in the design of the engine combustor, and so, their description can be found in [54]. Together with these input data, the diffuser length to height ratio (l/h_{diff}) is required for the sizing of the diffuser placed at the inlet of the reheater.

Some assumptions are required at this point to compensate the limited amount of parameters available at the preliminary design phase, and to reduce the complexity of the reheater sizing and weight estimation. The assumptions listed below are similar to those considered in the combustor design, and so, further description is avoided.

- a) **Straight walls for each reheater part**
- b) **Axial flow assumed at all reheater stations**
- c) **Symmetry around the reheater mean diameter line**
- d) **Passage mass flow introduced in the dome flow at reheater outlet**
- e) **Constant distance between the dome and passage annuli**

The mentioned input parameters and assumptions are applied in the design process described herein:

a) **Reheater mass flows estimation**

This process is similar to that followed for the combustor mass flows calculation. As shown in Eq. 3.38, the dome mass flow (w_d) is the sum of the mass flow used to atomise the fuel (w_{ato}), the one employed to cool the dome (w_{dc}) and the mass flow used to create a stabilising swirl flow (w_{swirl}):

$$w_d = w_{ato} + w_{dc} + w_{swirl} \quad (3.38)$$

The fuel atomising mass flow is calculated using Eq. 3.39:

$$w_{ato} = w_{f_{rh}} \cdot FAR \quad (3.39)$$

Where, $w_{f_{rh}}$ is the fuel flow injected in the reheater. This amount is calculated with Eq. 3.40, since the reheater inlet mass flow (w_{rh}) and the FAR at the component inlet and outlet (FAR_{rh_I} and FAR_{rh_O} , respectively) are known from TURBOMATCH:

$$w_{f_{rh}} = \frac{(FAR_{rh_O} - FAR_{rh_I}) \cdot w_{rh}}{1 + FAR_{rh_I}} \quad (3.40)$$

Note that this expression is different from the one employed to estimate the fuel flow in the combustor. In this case, as reheating occurs after a previous heat addition, the value of FAR_{rh_I} is also required to derive the fuel flow injected.

Regarding the dome cooling mass flow (w_{dc}), it is calculated as:

$$w_{dc} = w_{rh} \cdot DCFR \quad (3.41)$$

The stabilising swirl flow (w_{swirl}) is computed using the expression below:

$$w_{swirl} = \frac{w_{f_{rh}}}{FAR_{stoi} \cdot PZER} - w_{ato} \quad (3.42)$$

With the dome mass flow evaluated this way, the passage mass flow (w_{pass}) can be obtained as:

$$w_{pass} = w_{rh} - w_d \quad (3.43)$$

b) **Reheater inlet parameters calculation**

The annulus area, dimensions and velocities at the reheater inlet station are calculated following the method described in Section 3.3.4.2 for the centrifugal compressor inlet station (Eqs. 3.10 to 3.15). These parameters are used to define the inlet station of the reheater diffuser.

c) **Diffuser length evaluation**

With the l/h_{diff} introduced, the axial length of the diffuser is calculated, since the inlet height is known at this stage.

d) **Reheater axial length assumption**

Similarly to the combustor design process, the reheater axial length is initially guessed. This way, the mean diameter at the different stations can be calculated by interpolation between the inlet and outlet mean diameters. This is necessary for the next design steps.

e) **Reheater outlet parameters calculation**

The annulus area, dimensions and the velocity vector at the outlet of the reheater are calculated by using the data retrieved from the downstream component, and the method applied for the calculation of the reheater inlet parameters. Note that the reheater outlet parameters are common to the dome outlet station, since the passage mass flow is assumed to be zero at this station.

f) **Dome inlet parameters estimation**

The velocity vector, area and annulus dimensions are calculated at the dome inlet. The dome length can be then estimated as well, since the inlet height is known and the dome length to height ratio (l/h_d) is an input.

g) **Passage inlet parameters calculation**

The parameters at the inlet of the passage are calculated, since all the required inputs are known at this point.

h) **Diffuser outlet parameters estimation**

The outlet of the diffuser is connected to the dome inlet and the passage inlet. Thus, the diffuser outlet area is calculated as the sum of the dome inlet and the passage inlet areas. With this area value, the annulus dimensions and the velocity vector at the diffuser outlet are computed.

i) **Reheater axial length check**

At this point, the final reheater length (computed as the sum of diffuser and dome length) is compared with the value previously guessed. If they do not match, a new value is assumed and a new iteration is started from step (d).

j) **Passage outlet parameters calculation**

The dimensions at the outlet of the passage are calculated by using the passage outlet area (A_{passO}), which is estimated with Eq. 3.44. Note that the mean diameter of the passage is equal to the mean diameter of the dome and equal to the reheater mean diameter, R_{rh_m} (see Fig. 3.19).

$$A_{passO} = A_{dO} + A_{passI} \cdot \frac{A_{dO}}{A_{dI}} \quad (3.44)$$

Finally, the material selection was driven by the same properties required by the combustor, *i.e.* high temperature resistance, reduced weight, good thermal conductivity and low thermal expansion, and good oxidation and corrosion properties. Therefore, nickel-based alloys were considered the most appropriate materials for the reheater design. In particular, the Inconel 718 alloy was used, although any other nickel-based alloy could be employed, since only the material density is used to estimate the component weight.

3.3.7 Reheated engines weight modelling

Once the baseline engine model is implemented in ATLAS, the models of the different reheated cycle configurations are built using the baseline engine as reference. In Section 3.2.3, it was shown how the performance of the different configurations can be simulated using just one TURBOMATCH model. However, ATLAS is not as flexible as TURBOMATCH, since the duct and the reheater bricks do not share the same input scheme. Therefore, an independent model has to be created for each configuration. Figure 3.21 shows the general layout of the schemes produced by ATLAS for the three reheated architectures analysed. In this figure, the particular location of the auxiliary burners for each configuration is clearly displayed.

For the weight analysis of the conventional and the reheated cycle engines the following assumptions and constraints are introduced in the models:

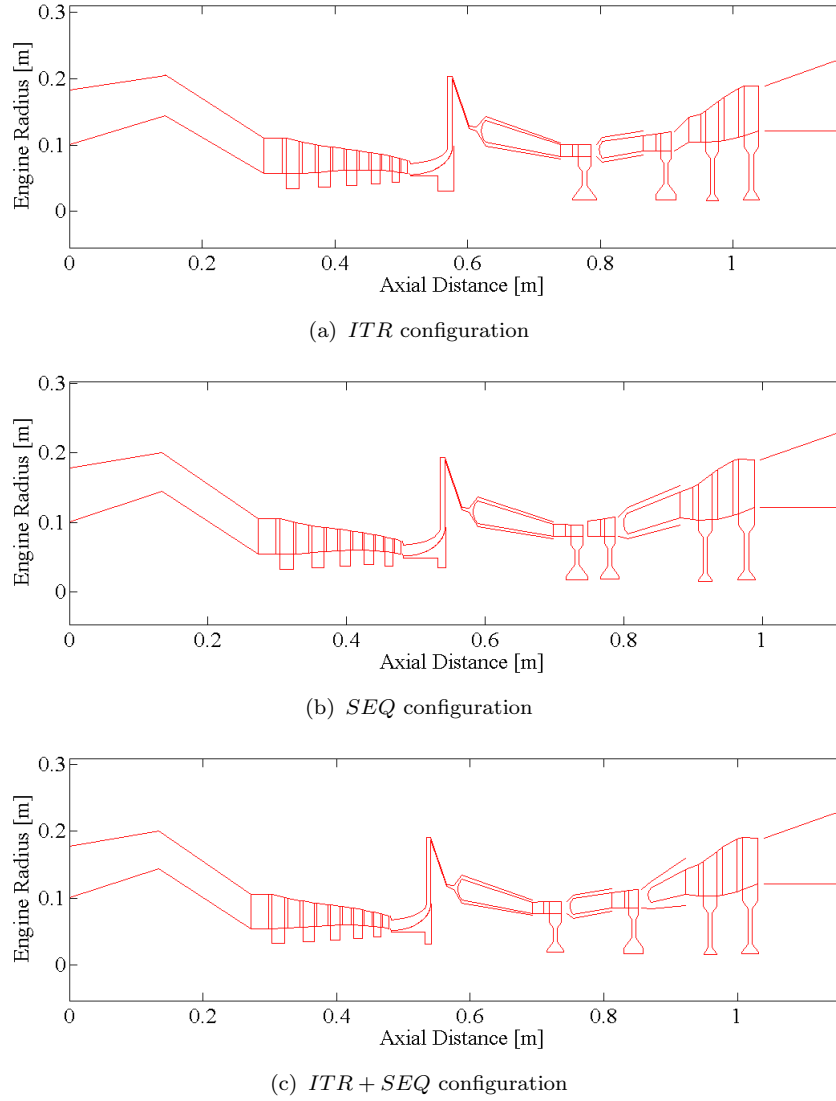


FIGURE 3.21: Reheated engines cutaways generated by ATLAS

- **Constant D_h at inlet of the engine intake**

The calculation of the turboshaft engine geometry requires the definition of an inlet dimension which can be either D_m , D_h or D_t . In the baseline engine employed in this study, the *FT* shaft outlet is placed at the front of the engine (Fig. 3.5). Therefore, in order to ensure that enough space is available for the coupling of the *FT* with the power transmission system, the value of D_h of the original engine is maintained for all the engine architectures.

- **Fixed number of stages**

To minimise the re-design of the engine turbomachinery, the stages count of the compressors and turbines is kept fixed and equal to the number of the original engine. However, the rotational speed of the *GGT* is calculated

by ATLAS for each candidate engine as part of the turbomachinery design process. Note that the rotational speed of the FT is fixed by the rotational speed of the main rotor and the power transmission system. As these parts of the helicopter are not modified in the analysis, the rotational speed of the FT remains constant.

- **Constant h/t at inlet and outlet of the main components**

The changes in engine design inputs will necessarily modify the dimensions of the engine components. However, the layout and proportion of the engine parts are intended to be maintained. The design of the engine components in ATLAS requires the definition of an inlet and outlet dimension variable. Therefore, in order to accommodate the model to changes in the gas path size, the non-dimensional h/t is used as the sizing variable. In order to preserve the engine proportions, the value of this parameter at the inlet and outlet of the main engine components is kept constant for all the engine architectures analysed.

- **Constant U_{ax} at inlet and outlet of the main components**

The design of the engine components also needs the definition of an inlet and outlet velocity variable. In order to clearly reflect the impact of engine mass flow fluctuation on the gas path size, it was decided to use U_{ax} as the velocity variable, which is kept constant for all the engine configurations. If a constant M_{ax} were used instead, U_{ax} would change for each candidate engine, due to the effect of T_{st} on the speed of sound.

- **Constant centrifugal compressor number of blades**

As it has been already mentioned, the number of blades in the centrifugal compressor is selected by the user. Since the change in the impeller wheel size is expected to be small, the number of blades used in the design of the baseline engine is employed for all the engine architectures.

- **Constant l/h_d for main combustor and reheaters**

By keeping this parameter constant, the dimensions of the main and the auxiliary burners will change for each candidate engine, but the proportion of the combustors will be conserved.

- **Fixed configuration for compressors and turbines**

Three different design choices can be selected in ATLAS for the shape of the turbomachinery gas path *i.e.* constant D_m , constant D_h or constant D_t

(Fig. 3.22). Following the principle of minimum turbomachinery re-design, the configuration chosen for the baseline engine is kept for all the engine configurations analysed.

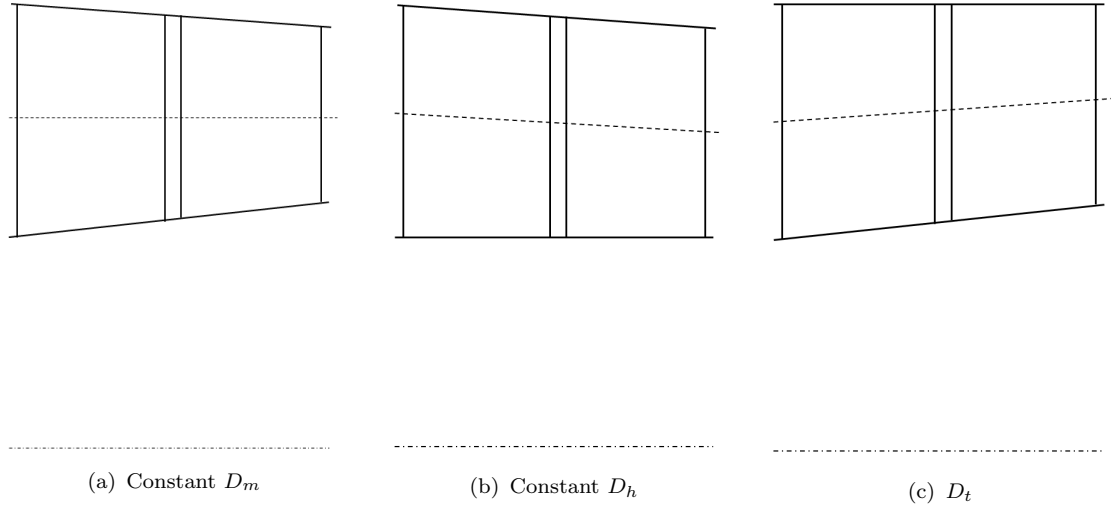


FIGURE 3.22: Turbomachinery configuration options in ATLAS

- **Similar materials selection**

The materials originally employed in the weight estimation of the baseline engine are maintained for the weight analysis of the different architectures. This way, the variation in engine weight is uniquely due to changes in gas path dimensions and its impact on the rest of engine components size. For the weight estimation of the reheaters, the same material employed in the design of the combustor is used.

3.4 Combustor emissions modelling

In order to estimate the NO_x emissions produced during the helicopter operation, a combustor model was created in HEPHAESTUS after the combustor installed in the T700/CT7 engine family. These engines use a straight-through annular combustion system of 12.5 cm in length with a machined ring film cooled construction and 12 low-pressure air blast fuel injectors [105]. A representation of the T700/CT7 combustor is shown in Fig. 3.23.

The combustor geometry is implemented in HEPHAESTUS through the definition of the volume and air mass flow distribution in the four combustor regions utilised

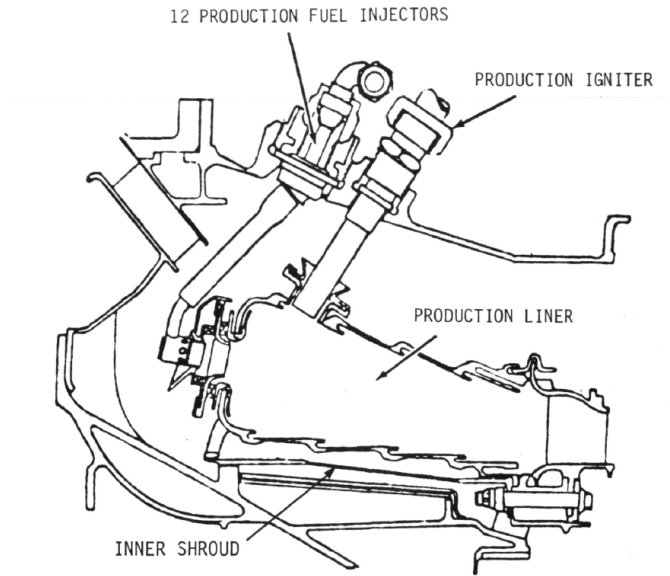


FIGURE 3.23: General Electric T700/CT7 combustor cross section [106]

by the code to model the combustor *i.e.* Flame Front (FF), Primary Zone (PZ), Intermediate Zone (IZ) and Dilution Zone (DZ). A scheme of the HEPHAESTUS combustor layout is shown in Fig. 3.24.

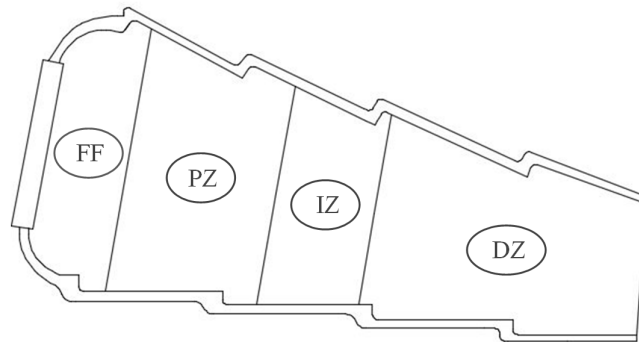


FIGURE 3.24: Engine combustor model layout

The combustor dimensions, listed in Table 3.5, were obtained from combustor cross section plots provided by Monty *et al* [106], whilst the air flow distribution along the combustor is based on the values employed by Fakhre in the modelling of different turboshaft engines combustors [107, 108].

The model described is to be used for the assessment of NO_x emissions produced by the *CONV* engine architecture. During the re-design and optimisation of this engine, the dimensions of the main combustor will necessarily vary with changes in the cycle design variables. However, in order to not over complicate the emissions analysis, the same combustor geometry of Table 3.5 is employed for all the *CONV*

TABLE 3.5: Combustor model design parameters

	FF	PZ	IZ	DZ
Inlet area [m^2]	0.0230	0.0293	0.0165	0.0136
Outlet area [m^2]	0.0293	0.0165	0.0136	0.0098
Length [m]	0.0117	0.0217	0.0143	0.0391
Air flow fraction [%]	22.0	10.0	10.0	58.0

candidate engines. Apart from the combustor geometry, the air flow fraction scheme used in the baseline engine is also kept for the rest of designs. This set of assumptions was also considered in the past by Fakhre *et al* [32, 35, 60, 61].

The modelling of the NO_x formation in the auxiliary burners requires an update of the HEPHAESTUS code that allows the user to modify the composition of the air reacting with the fuel in the combustion chamber. The stirred reactor scheme employed to simulate the combustion in the reheaters has to be also adapted to the physics events occurring during the reheating process, which do not have to be necessarily similar to the phenomena observed in the main combustor. For these reasons, the simulation of the NO_x emissions produced by the reheated cycle configurations is not attempted in this study. Instead, this task is recommended for future work.

Chapter 4

Models Validation

This section is devoted to the validation of the main models integrated in the simulation framework *i.e.* the helicopter and powerplant performance models, the model for the powerplant weight estimation, and the emissions prediction model. Since the software ATLAS was extensively upgraded, the validation of its new capabilities is also covered within this section. For all the models developed, actual data extracted from public sources were compared with the models predictions in order to measure their adequacy to be used in the powerplant performance analysis.

4.1 Helicopter performance model validation

The rotorcraft model implemented in *HECTOR* has been validated against experimental and flight test data available in the public domain in order to be confident about its capacity to reproduce the actual helicopter behaviour. Figure 4.1 presents the resonance chart calculated for the UH-60A main rotor. The main rotor blade has been modelled as a spring articulated rotor blade with coincident flap and lag hinges and an elastomeric root in torsion. The natural frequencies of the flap, F , lag, L , and torsional, T , vibration modes, and the rotor speed have been normalized with the nominal rotor speed (Ω_{nom}). The solid lines represent the calculated natural frequencies, whilst the discrete points correspond to the experimental data. The procedure used to obtain the latter data is described by Hamade *et al* [109].

Overall good agreement between calculated and experimental data is observed for all the natural modes analysed, with relative errors below 5%. This favourable comparison between calculated and experimental modal characteristics is paramount for the prediction of rotor hub moments and forces, as they determine the rigid and elastic displacements of the blade.

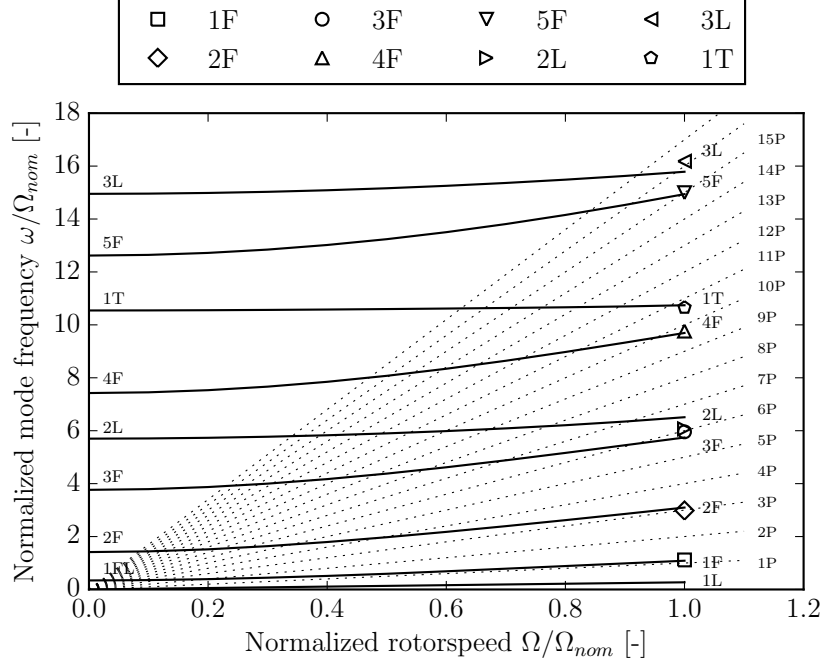


FIGURE 4.1: Resonance chart of the Sikorsky UH-60A main rotor

Figure 4.2 presents the calculated dominant mode shapes for the first nine modes of vibration of the UH-60A main rotor at nominal rotorspeed plotted in Fig. 4.1. Two types of modes are distinguished in this figure: rigid and elastic. Rigid modes of vibration, $1F$ and $1L$, are associated with rigid rotations around the hinges with negligible elastic deformation. Certain elastic curvature is observed next to the hinge due to the presence of discrete flap and lead-lag springs. Note that the second torsional mode is also presented for completeness.

Figure 4.3 shows the variation of the power coefficient, C_P , of main rotor (MR), tail rotor (TR), and total power coefficient with the advance ratio ($V/\Omega_{nom}R$) for a given blade loading ($C_T/\sigma = 0.08$). The power consumed by the helicopter is initially large due to the high induced losses occurring at high induced rotor velocities near hover. As the advance ratio increases, C_P drops since mean induced velocity decreases. A minimum is reached at which the power demand rises again due to the contribution of the parasitic drag at high flight velocities.

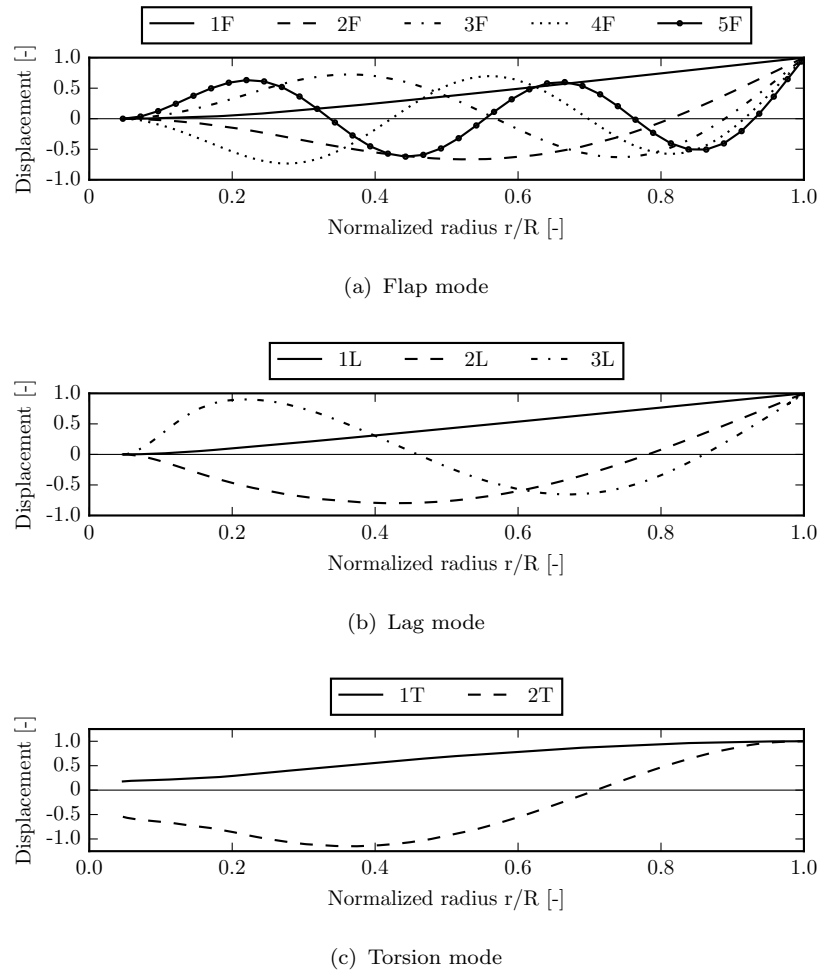
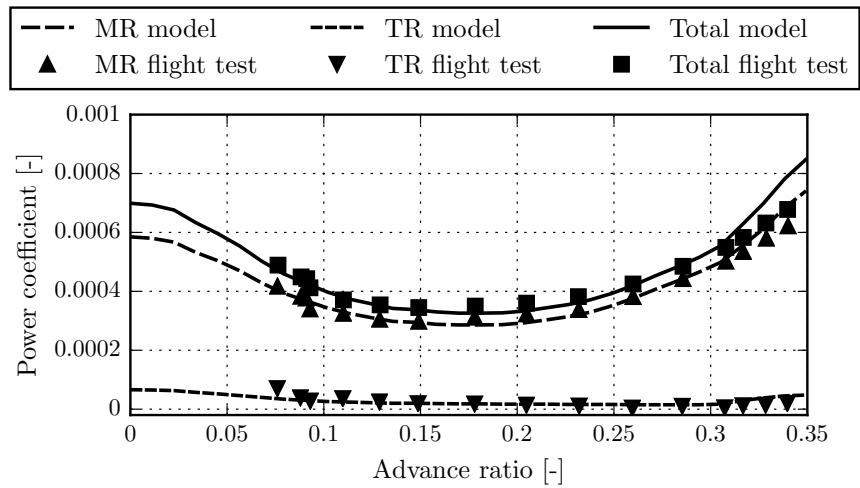
FIGURE 4.2: Shape of the main rotor vibration modes, $\Omega_{nom} = 27.01 \text{ rad/s}$ 

FIGURE 4.3: Rotorcraft model flight dynamics trim results

The power requirements estimated with *HECTOR* exhibit good correlation with the set of flight test data documented by Bousman *et al* [110]. The largest discrepancies are observed at high speed for both, main and tail rotor. This can be attributed to the uncertainty of the estimated fuselage aerodynamic maps used in *HECTOR* to calculate the parasitic drag, which is magnified with flight speed. This uncertainty can lead to a different flight trim state, and thus, to a different overall power at high-speed conditions. However, the maximum advance ratio employed in the simulations never exceeds 0.3 (66 *m/s*), and therefore these discrepancies have no effect on the predicted results. The average error incurred by the model in the total power prediction within this range is below 4% on average.

4.2 Powerplant performance model validation

In order to validate the engine model developed in TURBOMATCH, its capability to predict the performance of the baseline GE T700-GE-700 was assessed. Due to confidentiality reasons, engine manufacturers do not tend to disclose detailed information regarding the design of on-service engines. However, as this engine is being currently replaced by a new T700 version (the T700-701D) [73], more information has been publicly disclosed regarding the design of engine components, and experimental performance results.

The T700-GE-700 engine parameters listed in Table 4.1 correspond to the *MCP* rating at *SL-ISA+0* conditions [75]. The turbine cooling and engine bleeding mass flows are defined relative to the inlet mass flow rate, and the values are based on mass flow functions specific of this particular engine employed by Ballin [74]. Inlet mass flow and pressure recovery, and components efficiency values are based on the T700-GE-700 model built by Garavello *et al* [26]. Combustion pressure drop datum was extracted from experiments completed by Monty *et al* on T700/CT7 combustors [106]. For the *PR* of each compressor, only the *OPR* was known, therefore the compression distribution of the axial compressor (PR_1) was estimated based on engineering judgement (*PR* per axial stage around 1.5), and trying to get the closest match to the fuel flow at *DP*. Finally, for TET_1 there was no information available, as per usual. Hence, this parameter is a result calculated by the code.

TABLE 4.1: General Electric T700-GE-700 *DP* at *MCP SL-ISA+0* conditions

Inlet pressure recovery $[-]$	0.986	Garavello <i>et al</i>
Inlet mass flow $[kg/s]$	4.17	Garavello <i>et al</i>
Compressor section		
Axial compressor PR_1 $[-]$	5.32	Garavello <i>et al</i>
Centrifugal compressor PR_2 $[-]$	2.80	Garavello <i>et al</i>
Axial compressor η_{poly} $[-]$	0.857	Garavello <i>et al</i>
Centrifugal compressor η_{poly} $[-]$	0.845	Garavello <i>et al</i>
Cooling flow bleed percentage $[\%]$	8.46	Ballin
Engine bleed percentage $[\%]$	1.06	Ballin
Combustor section		
Combustion efficiency $[-]$	0.984	Garavello <i>et al</i>
Combustor pressure drop $[\%]$	4.0	Monty <i>et al</i>
TET_1 $[K]$	1,440	—
Fuel flow $[kg/s]$	0.0807	Garavello <i>et al</i>
Turbine section		
Gas generator turbine η_{poly} $[-]$	0.840	Garavello <i>et al</i>
Free turbine η_{poly} $[-]$	0.827	Garavello <i>et al</i>
Free turbine outlet total pressure $[atm]$	1.052	Ballin
Shaft power $[kW]$	987	Jane's Aero-Engines

Experimental data available in [74] at six different operating conditions were used to validate the model defined by the parameters in Table 4.1. The ambient conditions and engine loading for the six cases simulated are listed in Table 4.2. These conditions have been reproduced in TURBOMATCH and the obtained results have been compared with the experimental data through the charts in Fig. 4.4. The first test case was neglected as it corresponds to an operating point with a FT rotational speed abnormally low (52.6%), and therefore, not representative of the normal turboshaft operating conditions. For the sake of completion, the results obtained by the real-time simulation model described by Ballin [74], and the ones derived by Garavello *et al* [26] by using *GSP* have been also included.

TABLE 4.2: General Electric T700-GE-700 experimental test conditions [74]

Test case	P_{amb} $[atm]$	T_{amb} $[K]$	Shaft power $[kW]$
1	0.978	287.05	46.98
2	0.964	286.44	255.87
3	0.963	282.38	421.14
4	0.958	282.22	586.42
5	0.954	281.78	778.95
6	0.947	281.78	1024.6

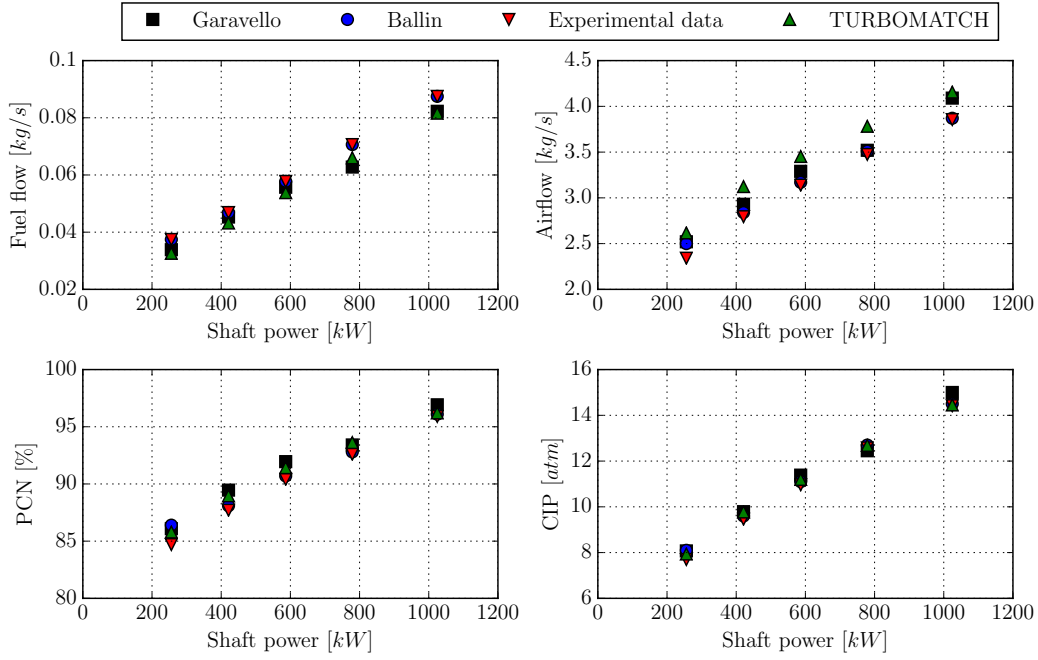


FIGURE 4.4: General Electric T700-GE-700 experimental results comparison

The agreement between the TURBOMATCH model and the experimental results is good for all the shaft power range simulated in the experiment. The high-fidelity model developed by Ballin was conceived to simulate the performance of this engine in particular, so it offers the most accurate results. However, it is observed that the more versatile *GSP* and TURBOMATCH tools are able to capture the trend of the engine performance and produce very close results. In fact, the level of accuracy obtained with the TURBOMATCH model is similar to the one attained by the *GSP* model developed by Garavello *et al.*

A further verification of the steady-state simulation capability of the TURBOMATCH model was completed by comparison with data supplied by GE status-81. This software is a comprehensive analysis-oriented model used for detailed representation of the engine thermodynamic cycle developed by the engine manufacturer [74]. Performance data generated by this software together with the results obtained by the TURBOMATCH model are reproduced in Fig. 4.5. The results obtained by Ballin are included again to enhance the comparison.

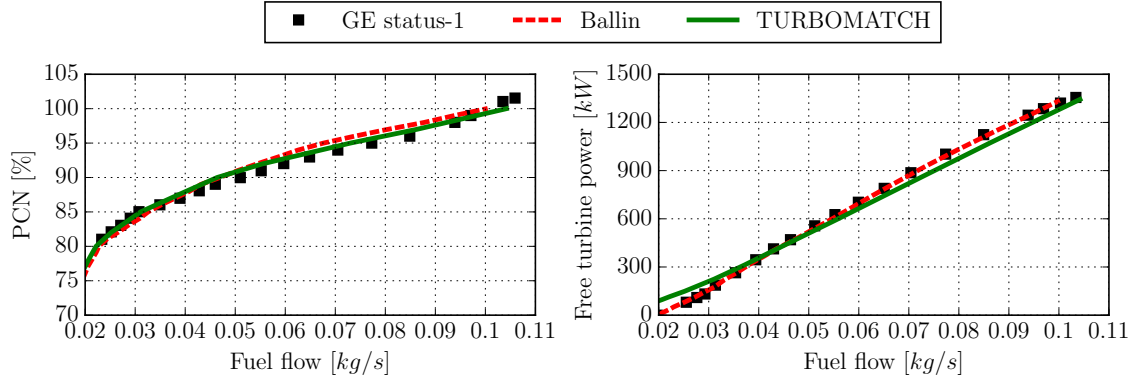


FIGURE 4.5: General Electric T700-GE-700 simulation results comparison

4.3 Weight prediction validation

The new capabilities added to the original version of ATLAS went through a validation process before it was used to model the weight of the conventional and reheated cycle architectures. For the validation of the upgrades introduced in the code, the geometry and size of two actual turboshaft engines were reproduced in ATLAS. On purpose, the engines selected come from distinct manufacturers and have a slightly different architecture and components arrangement, which made the validation assessment more rigorous. Once the updated version of the code was validated, the baseline model of the conventional engine was built, and the dimensions and size predicted by the code were compared with real engine data.

4.3.1 ATLAS updates validation

Due to the confidential policies followed by the engine manufacturers, there is a general lack of turboshaft engine weight breakdowns in the open literature to perform a component-by-component comparison. Therefore, the new updates can only be verified by whole engine weight comparisons. In this process, not only the total weight of the engine is cross-checked with the one predicted by the code, but also the shape and dimensions of the engine flowpath.

Following this approach, two different turboshaft engines were compared against the ATLAS output to account for different design philosophies and architectures, namely: the General Electric CT7-8A and the Turbomeca Makila 1A1. Apart from the different performance parameters (see Tables 4.3 and 4.6), these engines have also some peculiarities that can be observed in Fig. 4.6. Firstly, the connection

between the output shaft and the main rotor gearbox is different for each engine. Whilst this connection takes place at the front of the CT7-8A engine, the Makila 1A1 is connected to the output device at the rear end. Secondly, the combustor geometry is slightly different; the Makila 1A1 has an annular combustor chamber with centrifugal fuel injection from rotary atomiser on the main shaft [75], which differs from the more conventional injection system installed in the CT7-8A engine. The number of stages of the axial compressor is also different for each engine, and so the length of this component. Finally, the shape of the Makila 1A1 intake is more simplistic since it does not incorporate an anti-iced separator device.

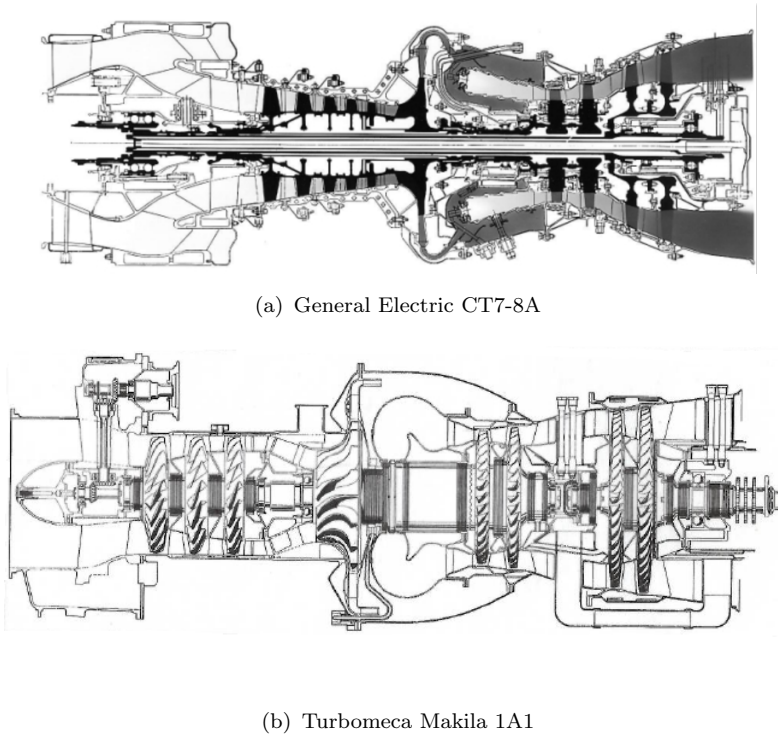


FIGURE 4.6: Cutaways of the turboshaft engines for validation [75]

4.3.1.1 General Electric CT7-8A weight model verification

The thermodynamic cycle inputs required by ATLAS were calculated using a model similar to the one described in Section 3.2.3 and correspond to the performance simulation results provided by TURBOMATCH at the 30-Second *OEI* (One Engine Inoperative) rating, which is defined by the parameters shown in Table 4.3. This condition is chosen to size the engine since it is the most demanding operating point from the point of view of gas path dimensions and gas temperature.

TABLE 4.3: General Electric CT7-8A performance parameters at 30-Second *OEI SL - ISA + 0* conditions

Shaft power [kW]	2,043
Inlet mass flow [kg/s]	6.05
Axial compressor PR_1 [-]	4.680
Centrifugal compressor PR_2 [-]	3.939
TET_1 [K]	1,699
Fuel flow [kg/s]	0.1490

The required input dimensions were obtained from the engine cross section shown in Fig. 4.6(a), using the engine length, 1.24 m, as reference. Other geometry related inputs, such as rotational speeds or materials, were obtained from open literature [75, 111]. The 2D engine cutaway generated by ATLAS with the set of inputs provided is plotted in Fig. 4.7 overlapped with the original engine layout.

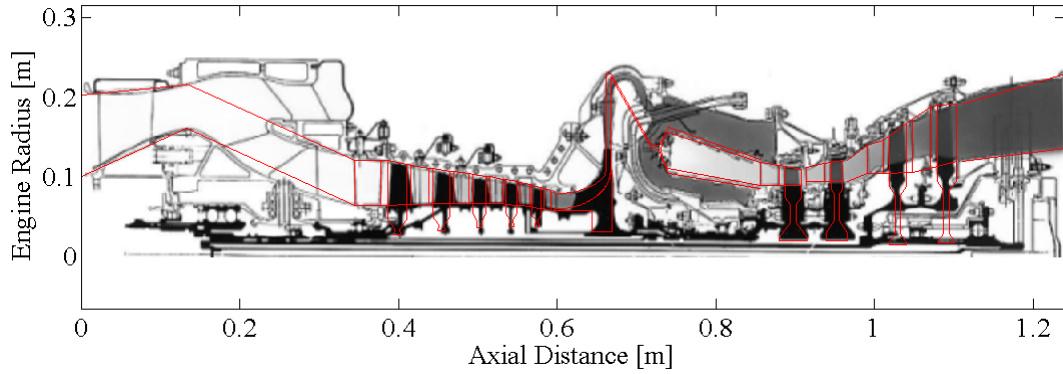


FIGURE 4.7: General Electric CT7-8A engine cutaway comparison

The cross section obtained with ATLAS is similar to the actual engine with minor differences. The shape of the engine intake is accurately simulated through the combination of an intake and a duct brick. Besides, the geometry of the centrifugal compressor stage is captured by the new compressor sizing routine. The most remarkable difference is observed at the free turbine disks geometry. This discrepancy is due to the fact that the design of the disks-shaft coupling is not considered in ATLAS, which implies that the geometry of this component cannot be fully captured.

In addition to the engine cutaway produced by the code, the estimated final mass is shown in Table 4.4, together with the actual mass of the engine for comparison. A discrepancy of 2.8% is obtained, which lies below the acceptable error limit

of $\pm 10\%$ adopted by Lolis [54] based on the criteria followed by other existing preliminary weight estimation methods [112].

TABLE 4.4: General Electric CT7-8A engine mass comparison

	ATLAS	Actual	Error [%]
Engine mass W_{eng} [kg]	250.53	243.60	2.8

Finally, although a component-by-component weight comparison is not possible, the engine weight breakdown is listed in Table 4.5 for completeness.

TABLE 4.5: General Electric CT7-8A engine mass breakdown

Intake	6.7%
Axial compressor	7.7%
Centrifugal compressor	2.1%
Combustor	4.5%
Compressor turbine	6.1%
Free power turbine	8.9%
Ducts	3.5%
Shafts	2.4%
Frames	48.1%
Controls & Accesories	10.0%

4.3.1.2 Turbomeca Makila 1A1 weight model verification

The same design validation proccess used for the CT7-8A engine was applied to the Makila 1A1 engine. A performance model was built for this particular engine in TURBOMATCH and validated against publicly available data [75, 113, 114]. Using this model, the performance parameters corresponding to Take-Off (*TO*) conditions (Table 4.6) were introduced in ATLAS. *TO* is considered in order to ensure the engine is adequately sized for the most demanding operating point.

The input dimensions required by the code were obtained from the engine cross section shown in Fig. 4.6(b), using the engine length, 2.103 *m*, as reference. As with the CT7-8A engine, the geometry input data, such as rotational speeds or materials, were obtained from open literature [75]. The 2D engine cutaway generated by ATLAS with this set of inputs is plotted in Fig. 4.8 together with the original engine layout.

TABLE 4.6: Turbomeca Makila 1A1 performance parameters at *TO SL-ISA+0* conditions

Shaft power [kW]	1,357
Inlet mass flow [kg/s]	5.45
Axial compressor PR_1 [–]	4.261
Centrifugal compressor PR_2 [–]	2.441
TET_1 [K]	1,394
Fuel flow [kg/s]	0.1102

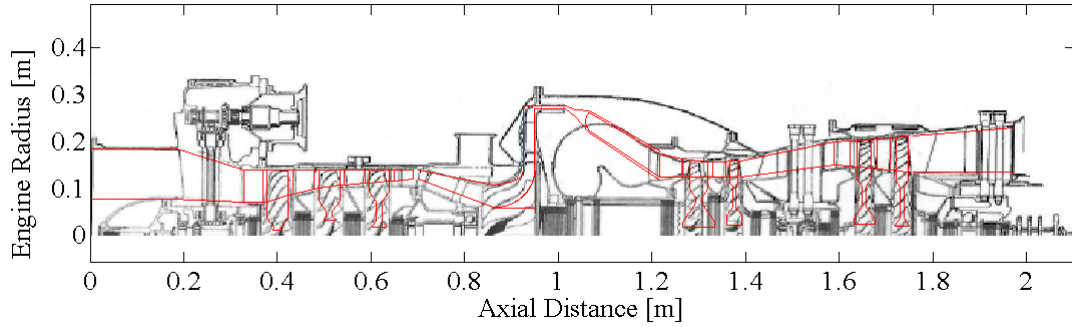


FIGURE 4.8: Turbomeca Makila 1A1 engine cutaway comparison

The cross-section generated by the code reproduces accurately the gas path of the original engine with the exception of the the centrifugal fuel injection arrangement that cannot be modelled by the current version of ATLAS. However, despite the inaccuracy of the combustor geometry model, the weight estimation offered by the code is still within the error target of 10%, as observed in Table 4.7. As before, an engine weight breakdown of the engine is included for completeness (Table 4.8).

TABLE 4.7: Turbomeca Makila 1A1 engine mass comparison

	ATLAS	Actual	Error [%]
Engine mass W_{eng} [kg]	255.71	241.00	6.1

4.3.2 Powerplant weight model validation

Once the updates implemented in ATLAS have been validated and its capability to simulate the weight of actual turboshaft engines has been proven, the model of the baseline engine was created.

The CT7-8A engine is a civil derivative of the T700-GE-700, and so, the layout of the latter engine (Fig. 4.9) is comparable to the one of the CT7-8A (Fig. 4.6(a)).

TABLE 4.8: Turbomeca Makila 1A1 engine mass breakdown

Intake	1.8%
Axial compressor	19.0%
Centrifugal compressor	3.6%
Combustor	11.3%
Compressor turbine	17.3%
Free power turbine	16.0%
Ducts	5.8%
Shafts	1.8%
Frames	13.3%
Controls & Accesories	10.0%

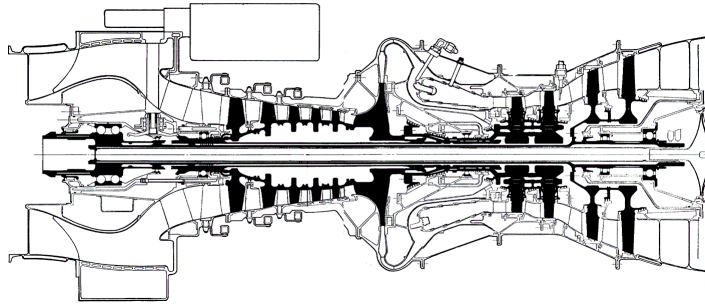


FIGURE 4.9: General Electric T700-GE-700 engine cutaway [75]

The thermodynamic data employed to build the ATLAS model correspond to Maximum Power (MP) rating to guarantee that the engine is sized for the most severe operating point. The MP performance parameters are summarised in Table 4.9.

TABLE 4.9: General Electric T700-GE-700 performance parameters at MP
 $SL - ISA + 0$ conditions

Shaft power [kW]	1,344
Inlet mass flow [kg/s]	4.69
Axial compressor PR_1 [–]	5.898
Centrifugal compressor PR_2 [–]	2.966
TET_1 [K]	1,567
Fuel flow [kg/s]	0.1041

The engine cross section in Fig. 4.9, together with the engine length, 1.194 m , were used to obtain the main input dimensions needed. Other design details, such as shafts rotational speeds or materials, were extracted from an open source [75]. The 2D engine cutaway generated by ATLAS with the described set of inputs is plotted in Fig. 4.10. The original engine layout is shown as well for comparison.

The geometry obtained with ATLAS is comparable to the actual engine layout, with the most remarkable difference being the design of the free turbine disks. This discrepancy was also observed in the case of the CT7-8A engine and is due to the lack of a thorough sizing routine in the code for the disks-shaft coupling.

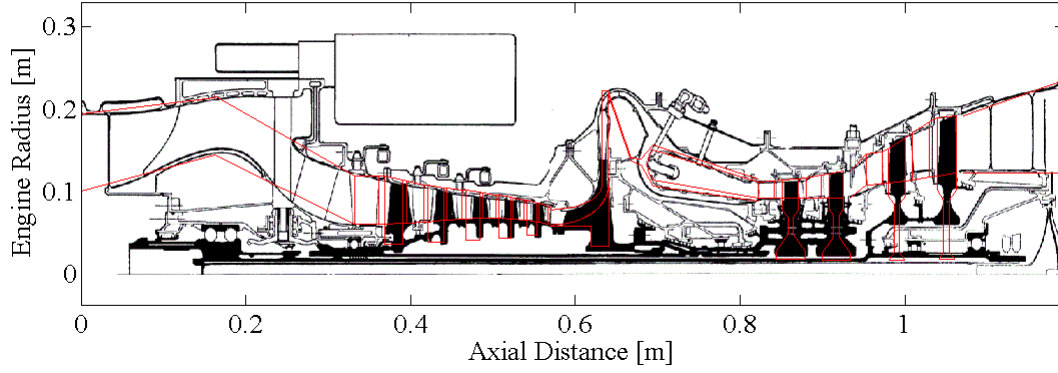


FIGURE 4.10: General Electric T00-GE-700 engine cutaway comparison

To complete the validation of the baseline engine model, the mass estimated by ATLAS is compared with the actual engine mass in Table 4.10. An error of -5.1% is incurred, which lies within the error margin assumed, and proves its validity to be used in the current study. The engine weight breakdown is listed in Table 4.11 for completeness.

TABLE 4.10: General Electric T700-GE-700 engine mass comparison

	ATLAS	Actual	Error [%]
Engine mass W_{eng} [kg]	187.81	198.00	-5.1

TABLE 4.11: General Electric T700-GE-700 engine mass breakdown

Intake	8.3%
Axial compressor	10.6%
Centrifugal compressor	2.6%
Combustor	6.4%
Compressor turbine	8.7%
Free power turbine	11.3%
Ducts	4.4%
Shafts	2.3%
Frames	35.4%
Controls & Accesories	10.0%

If the mass breakdowns of the three engines modelled are examined (Tables 4.5, 4.8 and 4.11), some differences appear. Firstly, after comparing the two General

Electric engines (CT7-8A and T700-GE-700), it is observed that the contribution of each component to the total weight is similar, with the exception of the frames. Since the cutaways of both engines are similar, the same frames arrangement was considered in ATLAS, with one placed at the inlet of the compressor and another at the outlet of the *FT*. However, due to the augmented shaft power of the CT7-8A (around 50% higher than the T700-GE-700), a heavier bearing frame type at the *FT* outlet was considered for this engine. This decision explains why the contribution of the frames to the total engine weight is higher for the CT7-8A engine.

With regard to the Makila 1A1 engine, its weight distribution differs from the General Electric engines since the architecture is different. However, some aspects can be commented:

- The contribution of the intake weight in the Makila 1A1 is significantly lower since it represents only 15% of the total length of the engine compared to the 30% of the General Electric turboshaft engines.
- The weight contribution of the frames is also significantly lower. The shaft power of this engine is comparable to the T700-GE-700, and so the type of frame employed at the *FT* is the same. However, the frame mounted in front of the axial compressor in the Makila 1A1 engine (see Fig. 4.8) seems lighter than the one mounted in the T700-GE-700 (single bearing frame versus double bearing frame). For this reason, a lighter kind of frame was selected for the Makila 1A1 engine, which explains why the overall frame contribution for this engine is lower compared to the T700-GE-700.

Once the baseline engine model was validated, a further verification was completed with regard to the effect of the new temperature distribution on the turbomachinery disks weight. In fact, for the T700-GE-700 engine, it was observed that a reduction around 10% and 3% in weight for the compressors and turbines, respectively, is attained if the original temperature distribution is employed. As it was commented in Section 3.3.2, this temperature distribution shows particularly ineffective for the simulation of the temperature fields in compressors, which explains the larger weight deviation observed in this latter component. The effect of the temperature distributions on the turbomachinery disks weight translates in an overall 2.5% reduction in the engine mass estimation.

4.4 Combustor emissions model validation

The ability of the model created in HEPHAESTUS to predict the NO_x emissions at different operating conditions must be verified in order to assess the adequacy of the T700-GE-700 combustor model. To that end, the operating conditions of the experiments conducted by Cohen for the T700/CT7 engine family combustor [105] were simulated in the combustor model. These experiments were performed for JP-4 and JP-5 fuels. The experimental results for the JP-5 fuel are used for validation, since this fuel has a hydrogen content similar to the Jet A fuel, which is the aviation fuel utilised in HEPHAESTUS. The comparison between the experimental results and the values predicted by the code under the same conditions is reproduced in Fig. 4.11. In this figure, not only the point-to-point comparison is shown, but also the trend generated by the code within the range of CIT simulated in the experiment.

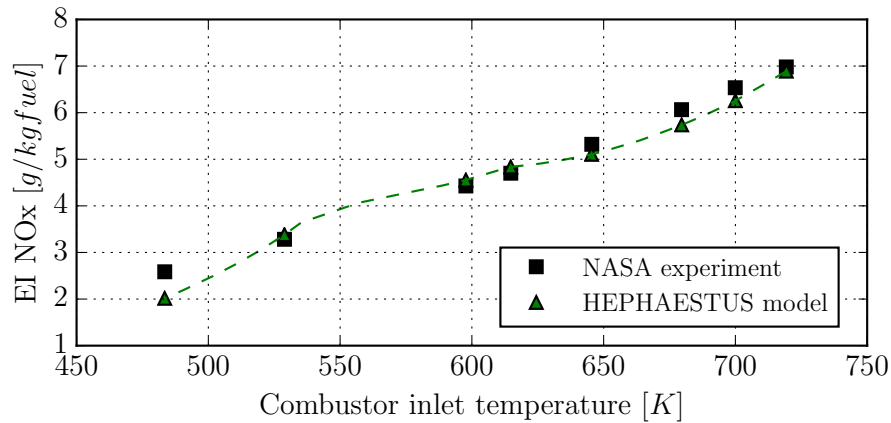


FIGURE 4.11: Combustor model NO_x results comparison

Overall, a good agreement with the experimental results is observed since the trend is accurately captured by the simulation model. An error of 22% is observed at the lowest CIT end (483 K) which corresponds to a shaft power of 35 kW. Not only the engine rarely operates at this low power rating conditions, but also the emissions are quite low at that point, so the model discrepancy can be considered acceptable. For the rest of CIT (shaft power) range, the error in the NO_x prediction lies below 10%, which is considered accurate enough for the present analysis.

It is also observed that the trend of the $EI NO_x$ values predicted by HEPHAESTUS is not constant. Since the NO_x production is a temperature dependant

process, its trend is governed by the variation of the equivalent temperature in the FF and PZ regions plotted in Fig. 4.12. Under the conditions simulated, it can be observed that the Equivalent Temperature (ET) in the FF starts to reduce at $CIT = 535\text{ K}$ up to $CIT = 650\text{ K}$, where it reaches a minimum. The ET in the PZ region, on the other hand, continuously increases with CIT . As a result of this ET distribution, it is observed in Fig. 4.11 that the NO_x production rate is mitigated in the CIT region where the ET in the FF region decreases ($CIT = 535 - 650\text{ K}$). The behaviour of the ET in the FF region is explained by the distribution of the mixing parameter, S , which has been plotted in the same graph for better visualisation. This parameter, first introduced by Fletcher and Heywood [115], was introduced in the code by Celis [48] to measure the uniformity of turbulent mixing within the combustor region; in such a way that the lower the value of S , the better the mixing. The unmixedness of a combustor reaction is difficult to estimate, so a generic correlation between S and the equivalence ratio of the combustor zone was implemented by Celis (see [48] for more information). Therefore, with the equivalence ratio distribution in the FF , the unmixedness trend obtained is the one shown in Fig. 4.12. A good level of mixing translates in a high ET , and *vice versa*, which explains the trend observed for the ET in the FF region. Note that the mixing parameter concept is not applied in the PZ , as complete mixing between fuel and air is assumed in this region.

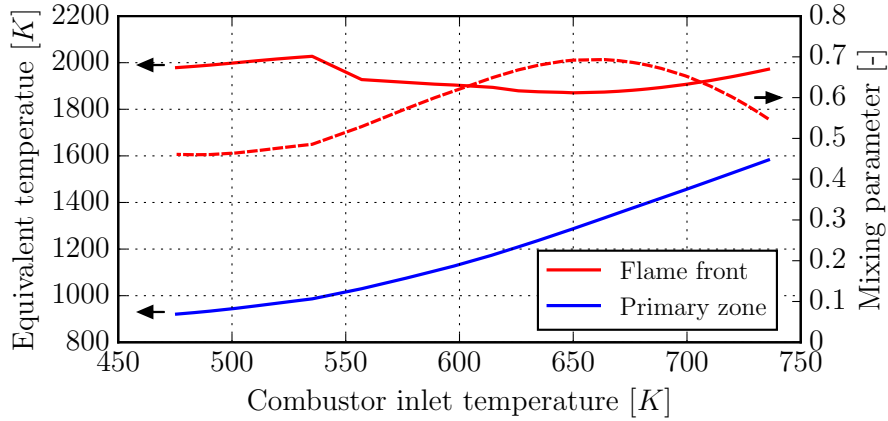


FIGURE 4.12: ET variation at FF and PZ

Chapter 5

Results & Discussion

This section presents the core results of the study which demonstrate the capabilities and merits of the developed methodology. The simulation framework shown in Fig. 3.1 is first used to conduct a parametric analysis of the different engine configurations. This study shows the trend of the performance parameters of interest with changes in the engine cycle design variables, and is also useful to check that the methodology developed is able to properly capture the physics phenomena involved in the engine design and performance. The results derived from this analysis constitute a contribution to knowledge itself, since they enhance the understanding of the performance behaviour of reheated cycle turboshaft engines.

Subsequently, the simulation performance tool is coupled with an optimiser routine in order to identify the optimal engine designs in terms of thermal efficiency and weight. This study showcases the existing trade-off between high thermal efficiency and reduced powerplant size. For the completion of the optimisation analysis, the development of surrogate models for the rapid estimation of the engine weight was required. This approach constitutes another novelty of the project, since such a 'hybrid' method has not been applied before for turboshaft weight estimation.

The best candidate engines derived from the optimisation analysis are then evaluated at mission level. This allows to quantify the potential benefits in mission block fuel that can be obtained by the reheated cycle architectures, and also, to analyse the performance of these engines at the different mission phases.

Finally, the author includes an assessment of the NO_x emissions for the *CONV* engine, which consists in a multi-objective optimisation of the mission fuel and

NO_x . This analysis highlights the trade-off existing between these two parameters that impedes the simultaneous minimisation of fuel consumption (and CO_2 emissions), and NO_x production.

5.1 Parametric analysis of powerplant performance

Prior to the numerical optimisation of the different engine architectures, it is beneficial to observe the sensitivity of the performance parameters of interest to changes in the engine design variables. This initial analysis is not only useful to capture the isolated effect of each engine design variable, but also to foresee the combination of design inputs that would yield the optimal results.

The performance parameters of interest are the SFC at cruise conditions, SFC_{cr} , the Specific Power at MP rating, SP_{MP} , and the engine weight, W_{eng} . Therefore, the effect of the engine design variables on these outcomes is to be assessed. For the $CONV$ architecture, the NO_x emissions are also investigated; hence, the $EINO_{x_{cr}}$ fluctuation with the $CONV$ engine design inputs is also included in this section.

MP rating and helicopter cruise operational conditions are defined by the parameters listed in Table 5.1.

TABLE 5.1: Engine design space exploration operating points

	Design Point	Maximum Power	Cruise Flight
Altitude [m]	0.0	0.0	400.0
Ambient temperature [K]	288.15	288.15	285.55
Mach number [–]	0.0	0.0	0.192
Shaft power [kW]	987	1,344	680

Every engine design must comply with the airworthiness specification requirements in terms of power rating. Therefore, the MCP (considered as the DP condition), and the MP ratings are constrained to the values of the baseline engine. The latter power rating is the condition used to size the engine as it is regarded as the most critical operating point. The condition considered to be representative of the cruise mode corresponds to the UH-60A Black Hawk helicopter flying 400 m over SL at 65 m/s and with a TOW of 7,700 kg, which corresponds to the primary mission gross weight defined for this helicopter [73]. The power required to fly at

this condition is computed by *HECTOR* as 1,360 *kW*, which means 680 *kW* per engine.

The variables employed in the parametric analysis and its lower and upper boundaries are presented in Table 5.2 . The limits of the two compressor pressure ratios at *DP*, $PR_{1_{DP}}$ and $PR_{2_{DP}}$, and the *GGT* working split, *WS*, are selected assuming a moderate engine redesign that does not alter the baseline turbomachinery layout. The *DP* total temperature at the first turbine entry, $TET_{1_{DP}}$, is also handled, but in order to maintain the reference technology level, the blade material limit temperature, and the cooling technology and effectiveness are kept constant and equal to the baseline engine values (see Table 3.2). This way, any increment in $TET_{1_{DP}}$ or in the *DP* temperature ratios of the auxiliary burners, $TR_{1_{DP}}$ and $TR_{2_{DP}}$, will translate in an increased cooling flows demand, which will impair the engine cycle efficiency. Note that the $TET_{1_{DP}}$ range is shifted towards values higher than the baseline temperature, since the optimum configurations are expected to be located in this hotter region. Regarding the reheaters temperature ratios, the low limit is set to 0.5 since a lower value would imply a negligible, if any, temperature increase in the auxiliary burner; on the other hand, the upper bound is restricted by the maximum reheating attainable that ensures that the limit imposed for $TET_{1_{DP}}$ is not exceeded:

$$TR_{max} = \frac{TET_{high\ limit}}{TET_{low\ limit}} = \frac{1800}{1400} = 1.29 \quad (5.1)$$

TABLE 5.2: Design parameter boundaries for space exploration

Design parameter	Low limit	Baseline value	High limit
$PR_{1_{DP}}$ [—]	4.00	5.32	7.00
$PR_{2_{DP}}$ [—]	2.00	2.80	4.00
$TET_{1_{DP}}$ [K]	1,400	1,440	1,800
<i>WS</i> [—]	0.3	0.5	0.7
$TR_{1_{DP}}$ [—]	0.5	—	1.3
$TR_{2_{DP}}$ [—]	0.5	—	1.3
$TR_{1_{MP}}$ [—]	0.5	—	1.3
$TR_{2_{MP}}$ [—]	0.5	—	1.3
$TR_{1_{cr}}$ [—]	0.5	—	1.3
$TR_{2_{cr}}$ [—]	0.5	—	1.3

5.1.1 *CONV* engine

For the *CONV* engine, the estimation of SFC_{cr} , $EINO_{x_{cr}}$, SP_{MP} and W_{eng} relies on three variables:

$$SFC_{cr}, EINO_{x_{cr}}, SP_{MP}, W_{eng} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}) \quad (5.2)$$

The effect of these design variables on the performance parameters is described herein. The reference cycle parameters employed to complete this parametric analysis are listed in Table 5.3, and are based on the parameters used to build the model of the baseline engine (Table 4.1). Note that the variables employed in the parametric analysis are highlighted in blue.

TABLE 5.3: *CONV* engine reference data

Inlet pressure recovery [–]	0.986
Compressor section	
Axial compressor PR_1 at DP [–]	5.32
Centrifugal compressor PR_2 at DP [–]	2.80
Axial compressor η_{poly} [–]	0.857
WS [–]	0.5
Centrifugal compressor η_{poly} [–]	0.845
Engine bleed percentage [%]	1.06
Combustor section	
Combustion efficiency [–]	0.984
Combustor pressure drop [%]	4.0
TET_1 at DP [K]	1,440
Turbine section	
Gas generator turbine η_{poly} [–]	0.840
Free turbine η_{poly} [–]	0.827
Free turbine outlet total pressure at DP [atm]	1.052

5.1.1.1 Effect on SFC_{cr}

For the SFC_{cr} (Fig.5.1) it is observed that there is a combination of $TET_{1_{DP}}$ and OPR_{DP} that maximises the thermal efficiency, and so, minimises SFC_{cr} . For this particular engine, this combination corresponds to an OPR_{DP} around 18, and a $TET_{1_{DP}}$ circa 1700 K. Note that the compression work distribution between compressors has little effect on the SFC_{cr} . Uniquely, a slightly lower SFC_{cr} is observed in the left hand chart. In this plot, the value of $PR_{1_{DP}}$ at the optimum

OPR_{DP} is higher than in the right hand plot, which implies that more compression is occurring in the more efficient axial compressor. As a result, a slightly higher thermal efficiency is attained.

A gas turbine working under the Brayton real cycle is expected to present an optimum OPR , for each value of TET , that maximises the thermal efficiency; this maximum efficiency increasing with TET . However, due to the increment in cooling flows demand with combustor outlet temperature, there is a TET value upon which no longer benefits in efficiency can be obtained. This fact explains why the thermal efficiency is limited by both OPR and TET , and so, the appearance of an absolute minimum for SFC_{cr} .

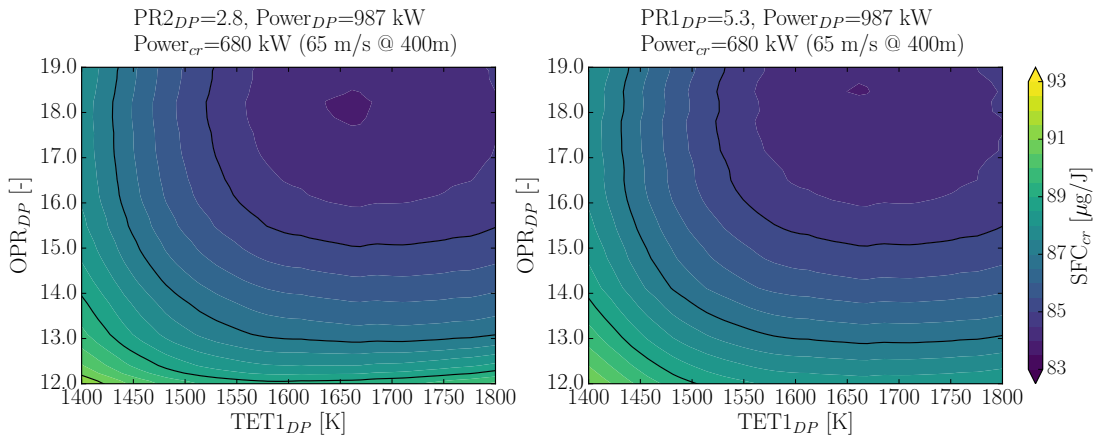


FIGURE 5.1: Effect of *CONV* engine design inputs on SFC_{cr}

5.1.1.2 Effect on $EINO_{x_{cr}}$

The trend of $EINO_{x_{cr}}$ is shown in Fig. 5.2. It is observed that the dominant effect on the production of NO_x is OPR_{DP} . In fact, combustor inlet temperature and pressure have a significant effect on thermal NO_x formation as proven by the experiment results shown by Lefebvre [4]. The variation of combustor inlet total pressure and temperature (T_b , P_b) with the design parameters at cruise are plotted in Fig. 5.3. These graphs show how the zone of higher production of NO_x lays over the region with higher pressure and temperature. In this figure, the equivalence ratio at the FF (ϕ_{FF}) is also displayed. The FF is the region of the combustor with the highest flame temperatures, and thus, the responsible for most of the NO_x formation. If the ϕ_{FF} distribution is analysed, it is observed that its value increases with $TET_{1_{DP}}$, and that for almost all the design space it is well above the stoichiometric value ($\phi_{stoi} = 1$). Uniquely, at the region with

high OPR_{DP} and low TET_{1DP} , the fuel-to-air ratio is close to the stoichiometric. It is at this condition at which the temperature of the flame peaks, and so does the NO_x formation. The trend observed in Fig. 5.2 is in agreement with this phenomenon. This graph showcases that the maximum value of $EINO_x$ occurs at high OPR_{DP} and low TET_{1DP} , where ϕ_{FF} is close to stoichiometric. Note that the compression work distribution between compressors has also little effect on $EINO_x$.

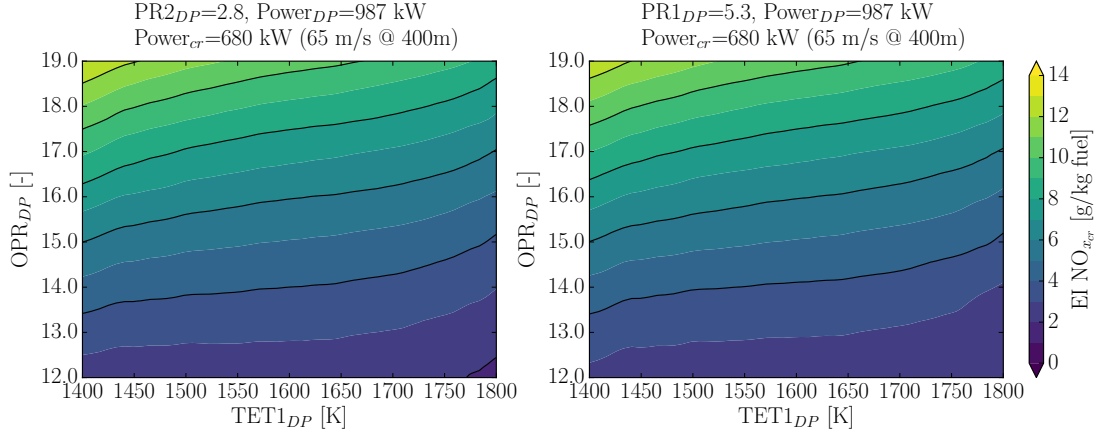


FIGURE 5.2: Effect of *CONV* engine design inputs on $EINO_{x_{cr}}$

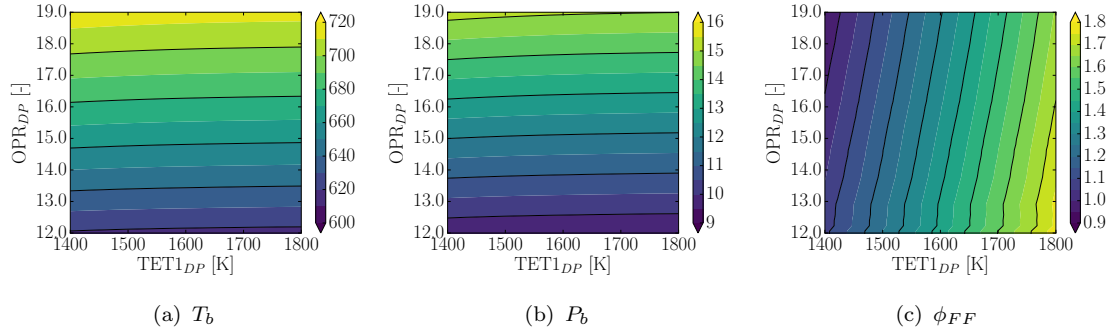


FIGURE 5.3: Effect of design inputs on combustor parameters at cruise

5.1.1.3 Effect on SP_{MP}

Similarly to SFC_{cr} , it can be observed in Fig. 5.4 that a combination of TET_{1DP} and OPR_{DP} exists that offers a maximum of SP_{MP} . For the particular conditions simulated, TET_{1DP} that maximises the SP_{MP} lays out of the design space explored, but the OPR seems to be around 12. The location of this maximum, that would not exist without the presence of turbine cooling flows, indicates that the penalty introduced by these bleeds is more relevant for SFC than for SP . Note that the

maximum SP_{MP} occurs at a lower OPR_{DP} than the minimum SFC_{cr} does, which is also the expected behaviour of the Brayton real cycle.

It has to be also mentioned that the OPR_{DP} that maximises SP_{MP} depends on the relative work done by each compressor. In fact, it is observed that at low TET_{1DP} values, the shape of the contours in Fig. 5.4 is similar. However, at high TET_{1DP} , the free power turbine starts to require cooling. This cooling air is set in the cooling flow routine to be bled at the axial compressor outlet, provided that the pressure of the air at this station is high enough. In the left hand plot, where PR_{2DP} is held constant, as OPR_{DP} reduces, a point is reached below which PR_{1DP} is not sufficient and, thus, the cooling air required by the power turbine has to be bled at the centrifugal compressor outlet. This implies a lost in engine performance which limits the maximum SP_{MP} and alters the value of OPR_{DP} at which it is attained.

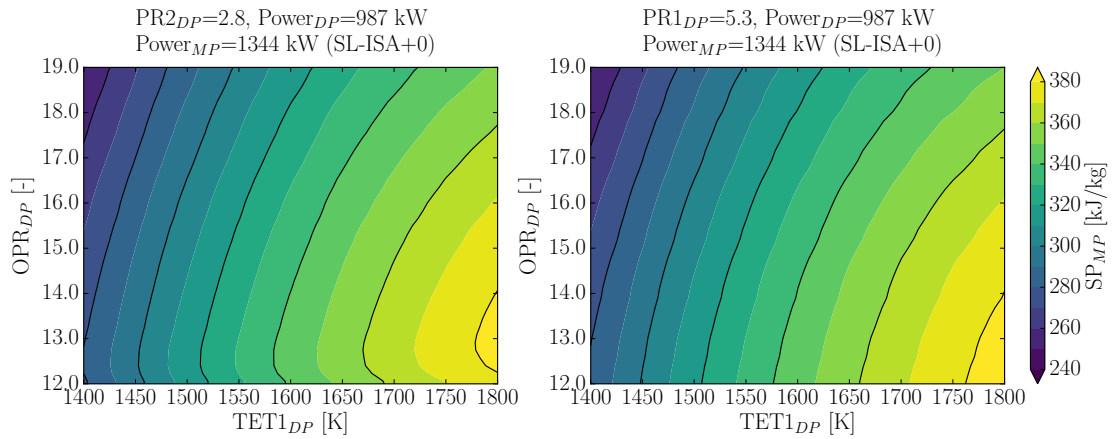
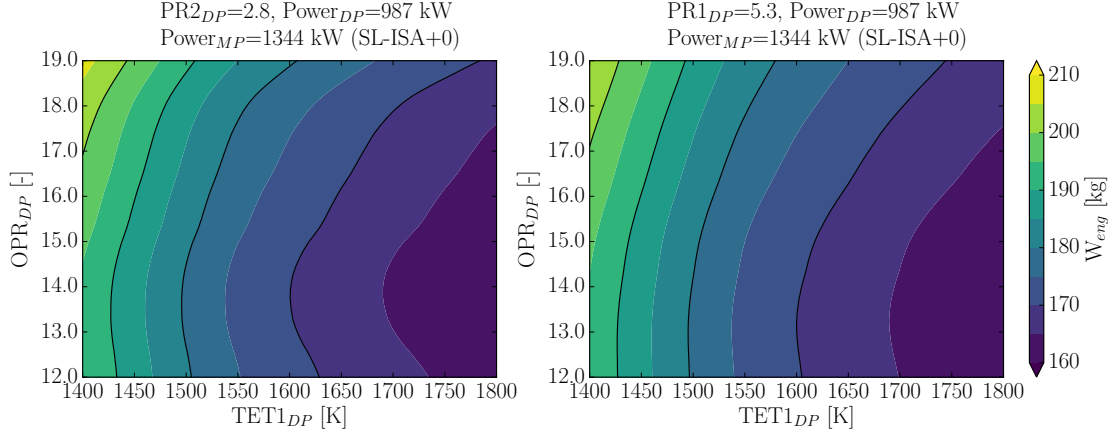


FIGURE 5.4: Effect of *CONV* engine design inputs on SP_{MP}

5.1.1.4 Effect on W_{eng}

Finally, the response of W_{eng} to changes in the design variables is shown in Fig. 5.5. For a fixed shaft power output, the dimensions of the engine are highly linked to SP . The higher the SP , the lower the mass flow required to provide the desired power, and *vice versa*. An increase in the engine mass flow will translate in an increment in the engine gas path dimensions and ultimately, an increment in size and engine weight. Therefore, it is expected for W_{eng} to follow the inverse trend of SP_{MP} , (MP is the operating condition that sizes the engine), as Fig. 5.5 demonstrates.

FIGURE 5.5: Effect of *CONV* engine design inputs on W_{eng}

5.1.2 *ITR* engine

For the *ITR* engine we have that SFC_{cr} , SP_{MP} and W_{eng} are a function of six input variables:

$$SFC_{cr} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{1_{DP}}, TR_{1_{cr}}) \quad (5.3)$$

$$SP_{MP}, W_{eng} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{1_{DP}}, TR_{1_{MP}}) \quad (5.4)$$

Note that the auxiliary burner can be operated at a different *TR* at cruise and at *MP* ratings. The variation of the engine performance parameters with the design variables is plotted in Figs. 5.6-5.8. To build the curves, only one design input is varied at a time, while the rest are kept fixed to the reference values of Table 5.4, which are based on the parameters used to build the model of the baseline engine. The data listed in this table is used for the parametric analysis of all the reheated configurations. Note that the variables involved in the parametric analysis are highlighted in blue. The reference values of the baseline engine were used to normalise the output results.

5.1.2.1 Effect on SFC_{cr}

First, the effect of the design inputs on SFC_{cr} is analysed (Fig. 5.6). It is observed that there exists a optimum OPR_{DP} circa 17 ($x \approx 0.7$) that minimises SFC_{cr} . Note that the operation of the auxiliary burner at *DP* and at cruise, reduces the

TABLE 5.4: Reheated engines reference data

Inlet pressure recovery $[-]$	0.986
Compressor section	
Axial compressor PR_1 at DP $[-]$	5.32
Centrifugal compressor PR_2 at DP $[-]$	2.80
Axial compressor η_{poly} $[-]$	0.857
WS $[-]$	0.5
Centrifugal compressor η_{poly} $[-]$	0.845
Engine bleed percentage $[\%]$	1.06
Combustors section	
Combustion efficiency $[-]$	0.984
Combustors pressure drop $[\%]$	4.0
TET_1 at DP $[K]$	1,440
TR_1 at DP $[K]$	1.0
TR_1 at <i>cruise</i> $[K]$	1.0
TR_1 at <i>MP</i> $[K]$	1.0
TR_2 at DP $[K]$	1.0
TR_2 at <i>cruise</i> $[K]$	1.0
TR_2 at <i>MP</i> $[K]$	1.0
Turbine section	
Gas generator turbine η_{poly} $[-]$	0.840
Free turbine η_{poly} $[-]$	0.827
Free turbine outlet total pressure at DP $[atm]$	1.052

value of the optimum OPR_{DP} in comparison with the conventional engine. It is also worth to mention, that the compression work distribution between compressors has no effect on SFC_{cr} . Similarly to the conventional engine, there is a $TET_{1_{DP}}$ that maximises the thermal efficiency. For the particular conditions simulated, the value of $TET_{1_{DP}}$ that minimises SFC_{cr} is around 1450 K ($x \approx 0.125$), which is much lower than the value for the no reheater case. WS has a strong influence in thermal efficiency, since it determines the pressure at which the reheating process takes place. From the thermal efficiency point of view it is desired for the additional combustion process to occur at the highest possible pressure. Therefore, the lower the pressure after the $GGT 1^{st}$ stage (higher WS), the worse the SFC_{cr} . For low values of TR_1 , the 2^{nd} burner is switched off, so no changes in thermal efficiency are appreciated. It is observed that for the set of conditions simulated, there is a $TR_{1_{DP}}$ circa 0.9 ($x \approx 0.5$) that minimises SFC_{cr} . Beyond this point, cooling requirements at $GGT 2^{nd}$ stage (B_2), and FT (B_3) increase significantly with $TET_{2_{DP}}$ which impairs the cycle thermal efficiency. The operation of the auxiliary reheater a cruise, does not provide any benefit in SFC_{cr} , for the

set of conditions simulated. Note that SFC_{cr} remains above the baseline reference value during all the sensitivity analysis.

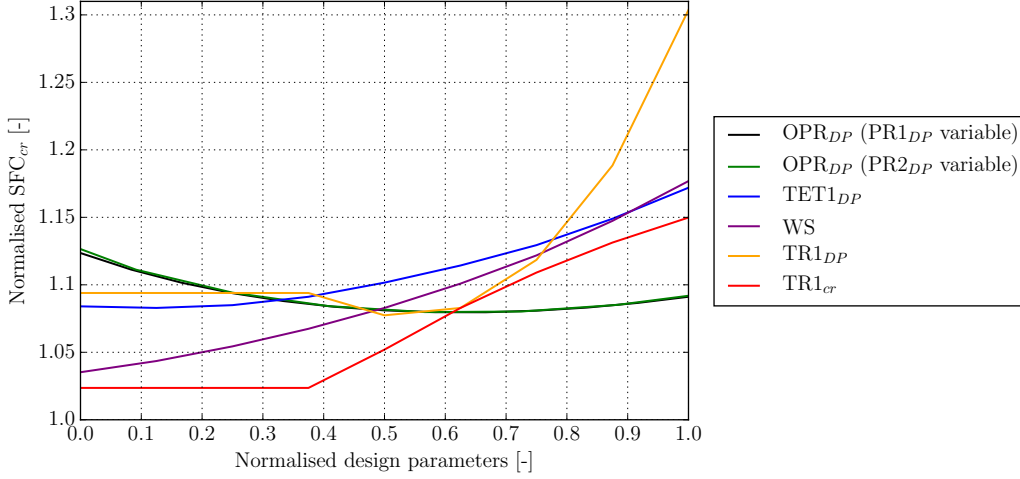


FIGURE 5.6: Effect of *ITR* engine design inputs on SFC_{cr}

5.1.2.2 Effect on SP_{MP}

The fluctuation of SP_{MP} with changes in the input variables is displayed in Fig. 5.7. An OPR_{DP} for maximum SP_{MP} is identified at around 12.5 ($x \approx 0.15$), which is lower than the optimum OPR_{DP} for cruise thermal efficiency. Note that the compression work distribution has no influence either on SP_{MP} ; thus, it is concluded that for the *ITR* architecture the compression work distribution does not affect the engine performance. A maximum in SP_{MP} is attained for a TET_{1DP} around 1750 K ($x \approx 0.875$). The double penalty introduced in B_2 and B_3 by the *ITR* burner limits the gain in SP_{MP} obtained with TET_{1DP} . For WS , if the effect of cooling flows is disregarded, the higher the WS , the higher will be the SP_{MP} since the temperature at the entry of the $GGT 2^{st}$ is lower, and thus, the heat addition in the *ITR* combustor is higher. However, as the work in the 1^{st} stage increases (higher WS), the temperature of the gas at the GGT outlet rises, which causes the cooling fractions B_2 and B_3 to escalate. Therefore, the final result observed is that SP_{MP} presents a maximum for WS around 0.6 ($x \approx 0.75$). Cooling requirements also affect the trend of SP_{MP} with TR_{1DP} . With no cooling, SP_{MP} would increase continuously with TR_{1DP} , however, at some point the combined effect of increased TET_{2DP} first, and increased TET_{3DP} later (B_3 is not required until TR_{1DP} around 1), causes the SP_{MP} to drop with TR_{1DP} . For TR_{1MP} , it is observed that once its value is high enough to turn the

auxiliary burner on, it boosts the value of SP_{MP} . No cooling effect is observed at MP rating as the cooling fractions are sized at DP , and they are kept constant all over the engine envelope.

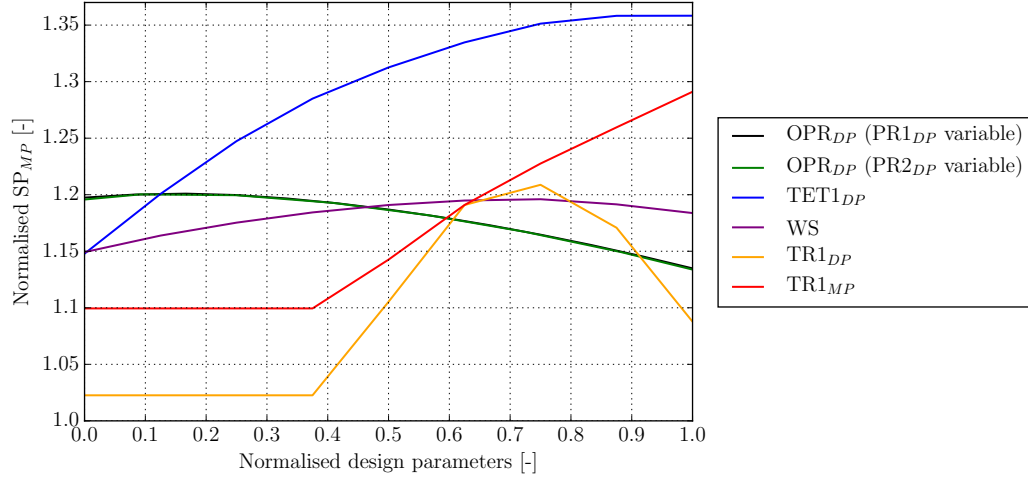


FIGURE 5.7: Effect of ITR engine design inputs on SP_{MP}

5.1.2.3 Effect on W_{eng}

The effect of the engine design variables on W_{eng} is presented in Fig. 5.8. A minimum in W_{eng} is attained with OPR_{DP} , which is the result of two opposed trends. On one hand, the higher the OPR_{DP} , the higher the gas pressure in the hot section of the engine, and the higher the gas density. Under the assumption of constant U_{ax} at the inlet and outlet of the main engine components, the increased gas density translates in a smaller gas path, which finally causes the engine weight to drop. On the other hand, the weight of the engine components not affected by the augmented gas pressure, *i.e.* the engine intake and the axial compressor (also the centrifugal compressor when PR_2 is the variable), fluctuates with the mass flow at MP , w_{MP} . This quantity is inversely proportional to SP_{MP} , thus, the drop in SP_{MP} with OPR_{DP} observed in Fig. 5.7, causes an increase in w_{MP} , and finally, an increase in the weight of the mentioned engine components. At low OPR_{DP} , the effect of higher gas density prevails, up to a point where the effect of increased w_{MP} becomes dominant. It must be also mentioned that compression work distribution has certain effect on W_{eng} . In fact, it is observed that at low OPR_{DP} , W_{eng} is lower for the trend with variable PR_2 (green line); whilst at high OPR_{DP} , the lighter engine is achieved with variable PR_1 (black line). The reason is that, for a fixed OPR_{DP} , the higher the PR_1 , the lower the W_{eng} , since the

annulus dimensions of the centrifugal compressor will be smaller. At the intersection point both lines have similar PR_1 ($PR_1 = 5.32$); below this point, green line has higher PR_1 , and *vice versa*. For $TET_{1_{DP}}$, the W_{eng} trend is similar but opposed to the SP_{MP} trend in Fig. 5.7. Indeed, the effect of mass flow reduction is dominant and it affects the size of all the major engine components. The effect of WS in SP_{MP} is not strong, so neither is the effect on w_{MP} . Therefore, the effect that is driving the trend of W_{eng} is the increase in weight of the GGT and the auxiliary burner with WS . As the expansion work in the 1st stage rises (higher WS), the pressure at the outlet of this stage decreases. This pressure drop affects accordingly the pressure of the gas at the reheater placed between the stages, and the pressure at the inlet of the 2nd stage. This pressure drop in the GGT section translates in a reduction of the gas density and, finally, and increase in the gas path dimensions. For the TRs note that no effect in weight is observed when the auxiliary burner is switched off. When the burner is working at DP , the W_{eng} trend is the inverse of the SP_{MP} trend. When the auxiliary burner is switched on at MP it is observed that W_{eng} reduces with $TR_{1_{MP}}$; the effect of mass flow reduction dominates over the increased size of the components placed downstream the auxiliary burner when the gas temperature rises.

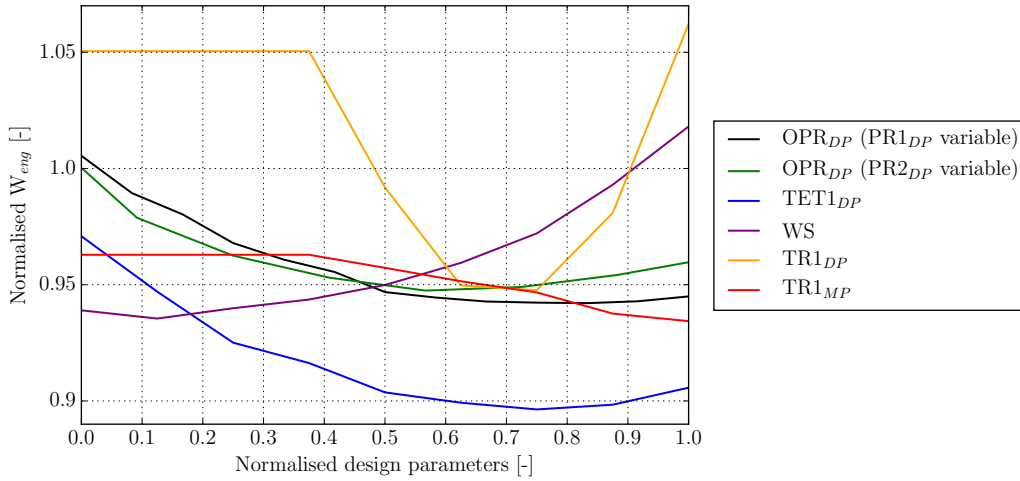


FIGURE 5.8: Effect of *ITR* engine design inputs on W_{eng}

5.1.3 *SEQ* engine

Similarly to the *ITR* engine, the *SEQ* engine performance depends on six design variables. In this configuration, the *TRs* of the inter-turbine reheater are replaced

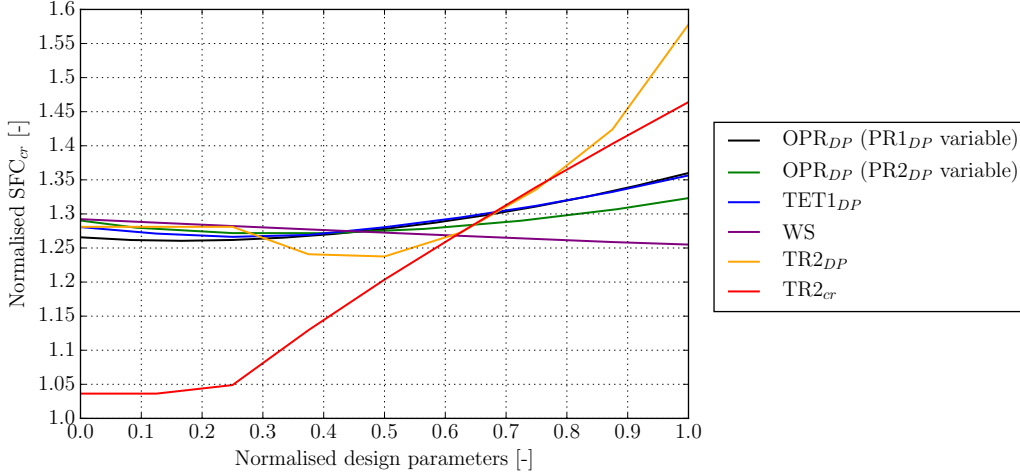
by those of the sequential combustor.

$$SFC_{cr} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{2_{DP}}, TR_{2_{cr}}) \quad (5.5)$$

$$SP_{MP}, W_{eng} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{2_{DP}}, TR_{2_{MP}}) \quad (5.6)$$

5.1.3.1 Effect on SFC_{cr}

Figure 5.9 shows the variation of SFC_{cr} with the design variables. It is observed that it has a minimum with OPR_{DP} , similarly to the conventional engine. In this case, the OPR_{DP} for maximum cruise thermal efficiency is reduced to a value around 13.5 ($x \approx 0.27$) due to the operation of the auxiliary burner at DP and cruise. Contrary to the ITR engine, the compression work distribution has a certain effect on SFC_{cr} for the SEQ architecture. This is due to the difference in the FT cooling flow fraction (B_3). This air is bled at the exit of the axial compressor, and thus, B_3 will be highly affected by the fluctuations of $PR_{1_{DP}}$. Therefore for $PR_{1_{DP}}$ variable (black line), B_3 increases with OPR_{DP} , whilst it does not change for the case $PR_{1_{DP}}$ is constant (green line). As a result, while $PR_{1_{DP}}$ is lower than the baseline value ($PR_{1_{DP}}=5.32$), B_3 will be also lower and so, the SFC_{cr} , and *vice versa*. For $TET_{1_{DP}}$, we have that a minimum SFC_{cr} is attained at a value around 1525 K ($x \approx 0.31$), which is also lower than the value observed for the conventional engine. The operation of the reheater increases the cooling flow penalty introduced to the cycle, which causes the optimum $TET_{1_{DP}}$ to drop. Regarding WS , it does not have a large impact on SFC_{cr} ; the thermal efficiency of the engine slightly increases with WS as a result of the reduced temperature at the GGT 2nd stage inlet, which reduces the cooling fraction required at this station (B_2). The TR s are the design variables with the higher impact on engine efficiency. Note that, for low values of TR_2 , the reheater is switched off, and no changes in thermal efficiency are introduced. It is observed that for these particular conditions, there is a $TR_{2_{DP}}$ around 0.9 ($x \approx 0.5$) that minimises SFC_{cr} . Beyond this point, cooling starts to be required at the FT and so no further benefit is obtained in increasing $TET_{3_{DP}}$, due to the cooling penalty. Finally, for $TR_{2_{cr}}$ the graph shows the big impact this parameter has on SFC_{cr} and that, for the conditions simulated, there is no gain in thermal efficiency at cruise by increasing $TET_{3_{cr}}$. Note that no benefits in SFC_{cr} are attained for the range of conditions simulated.

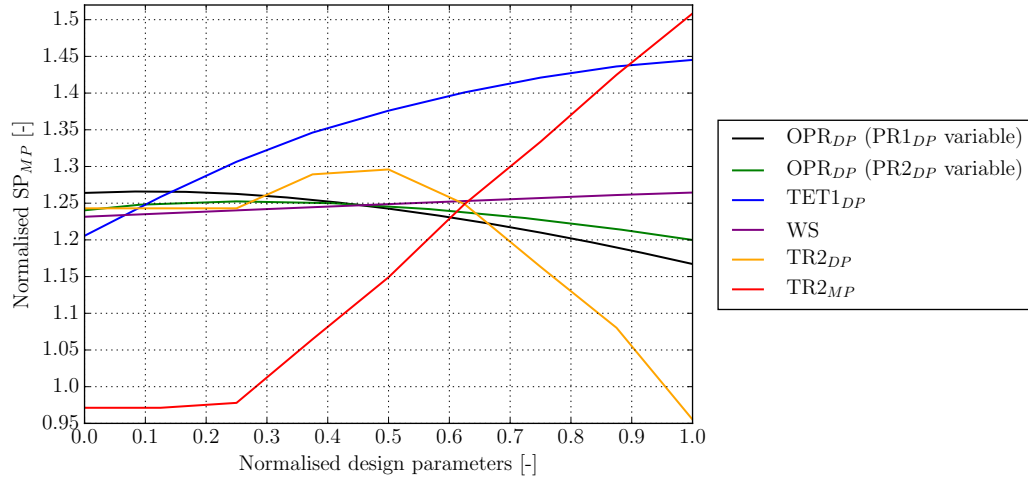
FIGURE 5.9: Effect of *SEQ* engine design inputs on SFC_{cr}

5.1.3.2 Effect on SP_{MP}

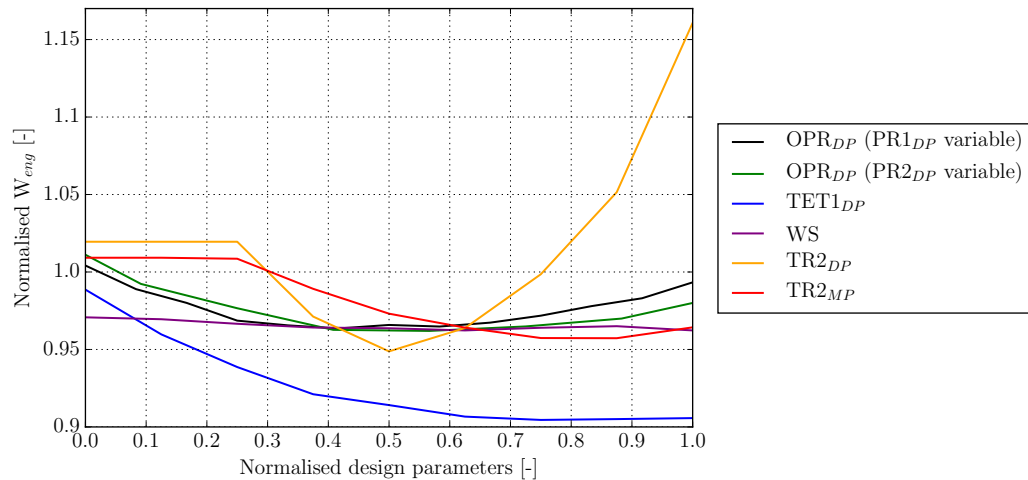
Figure 5.10 exhibits the sensitivity of SP_{MP} to changes in the input variables. There exists an OPR_{DP} around 12 ($x \approx 0.09$) that maximises the SP_{MP} , which is lower than the OPR_{DP} that minimises SFC_{cr} . The effect of the compression work share, already discussed, also affects the values of SP_{MP} . Regarding TET_{1DP} it seems that there is an optimum value, which, for the conditions simulated is higher than the upper limit imposed. This fact was also observed for the conventional engine, where it was proven that the negative effect of cooling flows was more relevant for SFC than for SP . A slightly benefit in SP_{MP} is attained with the increase in WS due to the mentioned effect this parameter has on B_2 . The trend of SP_{MP} with TR_{2DP} is similar to the SFC_{cr} , but opposed. It is also observed the penalty introduced in SP_{MP} when B_3 starts to be required. In the same way that TR_{2cr} reduced the thermal efficiency at cruise, TR_{2MP} boosts SP_{MP} once its value is high enough to switch the reheater on.

5.1.3.3 Effect on W_{eng}

The fluctuation of W_{eng} with the design variables is plotted in Fig. 5.11. The effect of OPR_{DP} on W_{eng} is similar to the one identified in the *ITR* engine. The two opposed trends described justify the presence of a minimum. However, note that for the *SEQ* architecture the compression work distribution does have an impact on SP_{MP} (Fig. 5.10), that is also appreciated in the W_{eng} pattern. Regarding TET_{1DP} , its effect on W_{eng} is dictated by the reduction in w_{MP} , which affects all

FIGURE 5.10: Effect of *SEQ* engine design inputs on SP_{MP}

the major components of the engine. Note that this is the same result observed for the *ITR* engine. WS does not have a significant effect on SP_{MP} (Fig. 5.10), and neither does on W_{eng} . The changes in size of the main engine components with TR_{2DP} are dictated by the fluctuation of w_{MP} , which defines the trend of W_{eng} . The variation of W_{eng} with TR_{2MP} also copies the pattern of w_{MP} ; however, a minimum W_{eng} is observed at a high TR_{2MP} . The presence of this minimum is justified by the continuous increase in size of the reheater and the *FT*, due to the gas temperature rise with TR_{2MP} , that eventually offsets the effect of the drop in w_{MP} .

FIGURE 5.11: Effect of *SEQ* engine design inputs on W_{eng}

5.1.4 *ITR* + *SEQ* engine

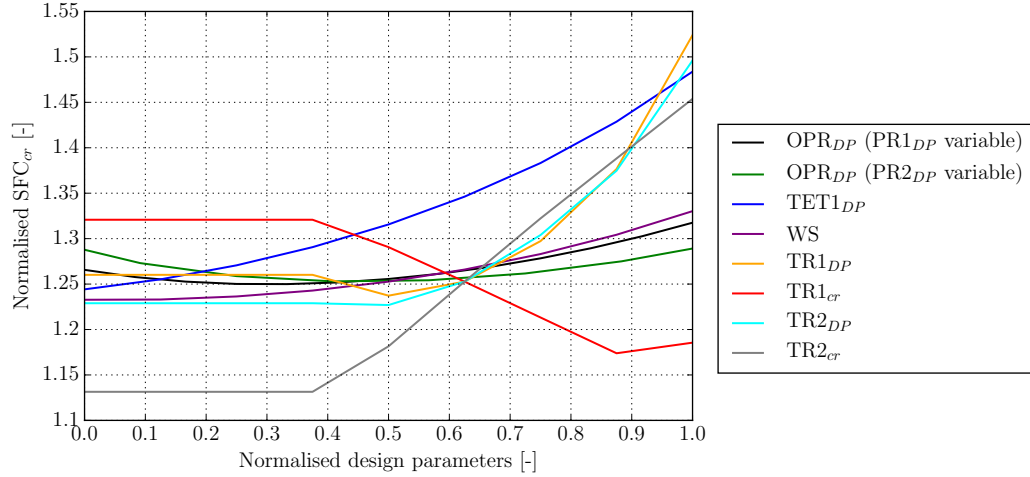
Since this latter engine architecture counts with two auxiliary burners, the total number of input variables increases to ten:

$$SFC_{cr} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{1_{DP}}, TR_{2_{DP}}, TR_{1_{cr}}, TR_{2_{cr}}) \quad (5.7)$$

$$SP_{MP}, W_{eng} = f(PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{1_{DP}}, TR_{2_{DP}}, TR_{1_{MP}}, TR_{2_{MP}}) \quad (5.8)$$

5.1.4.1 Effect on SFC_{cr}

The fluctuation of SFC_{cr} is presented in Fig. 5.12. It is seen that there exists an optimum OPR_{DP} between 13-15 ($x = 0.21 - 0.45$) that maximises the thermal efficiency at cruise. Note that same effect of compression work distribution detected in the *SEQ* architecture also appears in the *ITR* + *SEQ* engine. For the $TET_{1_{DP}}$, the double operation of the auxiliary reheaters, with the cooling flows penalty associated, further reduces its optimal value, which lies below 1400 K ($x < 0$). The effect of WS in SFC_{cr} is similar to the effect already observed in the *ITR* engine (Fig. 5.6): the pressure drop at the inlet of the 1st reheater impairs the cycle efficiency. The effects of the TR s at DP on cruise thermal efficiency is comparable to the separate effect of $TR_{1_{DP}}$ and $TR_{2_{DP}}$ on the *ITR* and *SEQ* engines, respectively. Note that, due to the operation of the 1st reheater, the 2nd reheater is switched on at a higher TR than in the *SEQ* engine. An interesting trend is observed for $TR_{1_{cr}}$: once the 1st reheater is operating at cruise, the temperature at the 2nd reheater inlet rises and, since $TR_{2_{cr}}$ is fixed, the heat addition (at a lower pressure) in this burner reduces. As a result, SFC_{cr} drops up to the point where the 2nd reheater switches off since the temperature at the inlet is higher than $TET_{1_{cr}}$. At this point, it is observed in Fig. 5.12 that, as expected, the *ITR* + *SEQ* architecture starts to behave as an *ITR* engine. Regarding the effect of $TR_{2_{cr}}$, it can be stated that SFC_{cr} varies similarly as it does in the *SEQ* engine. Note that, due to the operation of the 1st reheater at cruise, the 2nd reheater is activated at a higher TR in comparison with the *SEQ* engine (Fig. 5.9).

FIGURE 5.12: Effect of $ITR + SEQ$ engine design inputs on SFC_{cr}

5.1.4.2 Effect on SP_{MP}

The behaviour of SP_{MP} is reproduced in Fig. 5.13. For the conditions simulated, an optimum SP_{MP} is reached at OPR_{DP} similar to the optimal value for cruise thermal efficiency. Note that, as expected, the compression work distribution also affects SP_{MP} . For $TET_{1_{DP}}$, a maximum SP_{MP} is observed at 1700 K ($x = 0.75$). The double penalty introduced in B_2 and B_3 by the auxiliary burners limits the maximum SP_{MP} attainable. Note that the optimum $TET_{1_{DP}}$ for the $ITR + SEQ$ engine is lower than for the ITR architecture (Fig. 5.7). For the $ITR + SEQ$ engine, WS controls the heat addition in each auxiliary combustor. The higher WS , the higher the heat addition in the 1st reheater, and the lower the heat addition in the 2nd reheater. The combustion process at the 2nd burner, placed before the FT , has a higher effect on SP_{MP} augmentation, which explains the distribution of SP_{MP} with WS . The influence of the TR s at DP on SP_{MP} is similar to the separate effect of $TR_{1_{DP}}$ and $TR_{2_{DP}}$ described for the ITR and SEQ configurations. Note that, due to the operation of the 1st reheater, the 2nd burner is activated at a higher TR than in the SEQ engine. The effect of $TR_{1_{MP}}$ on SP_{MP} is similar to the effect of $TR_{1_{cr}}$ on SFC_{cr} : a reduction in the heat addition in the 2nd reheater. As already mentioned, the operation of the 2nd burner has the highest effect on SP augmentation. Thus, a reduction in the heat addition in this combustor will reduce the gain in SP_{MP} . At a certain $TR_{1_{MP}}$, the 2nd burner is switched off, and the $ITR + SEQ$ engine begins to behave as an ITR design (Fig. 5.7). The effect of $TR_{2_{MP}}$ in SP_{MP} is the same as in the SEQ engine (Fig. 5.10). The only difference observed is that, due to the operation of the 1st reheater, the 2nd one is

switched on at a higher TR .

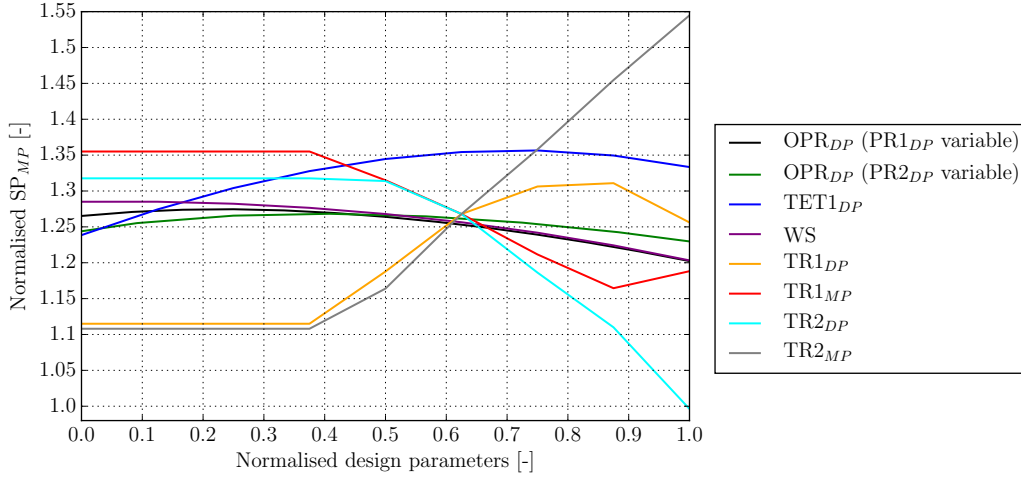


FIGURE 5.13: Effect of $ITR + SEQ$ engine design inputs on SP_{MP}

5.1.4.3 Effect on W_{eng}

Figure 5.14 presents the fluctuation of W_{eng} with the engine design variables. The effect of OPR_{DP} on engine weight is similar to the effect observed in the ITR and SEQ configurations. Note that the effect of compression work distribution that appeared in the SEQ engine, slightly affects the pattern of W_{eng} in the $ITR+SEQ$ engine. As for the ITR and the SEQ architectures, the W_{eng} trend follows the variation of w_{MP} with $TET_{1_{DP}}$; however, note that the minimum W_{eng} occurs at a lower $TET_{1_{DP}}$ than the one for maximum SP_{MP} . This is due to the combined increase in size of the auxiliary burners with $TET_{1_{DP}}$: the higher this temperature, the higher the temperature at the outlet of the reheaters ($TR_1 = TR_2 = 1$), and the lower the density of the gas. WS has a strong influence on W_{eng} due to the combination of two effects: the rise in w_{MP} with WS , and the increase of the GGT section weight which is similar to the one observed in the ITR engine (Fig. 5.8). Regarding the effect of TR_1 at DP and MP , its impact on W_{eng} is defined by the trend of w_{MP} (Fig. 5.13), as it occurs for the ITR engine. Note that the increase in SP_{MP} at the highest value of $TR_{1_{MP}}$ in the $ITR + SEQ$ configuration is well captured by ATLAS, and W_{eng} is reduced accordingly. The behaviour of W_{eng} with TR_2 is also controlled by the fluctuation of w_{MP} , similarly to the SEQ engine. Note that the minimum in W_{eng} at high $TR_{2_{MP}}$ identified in the SEQ configuration, also appears in the $ITR + SEQ$ engine.

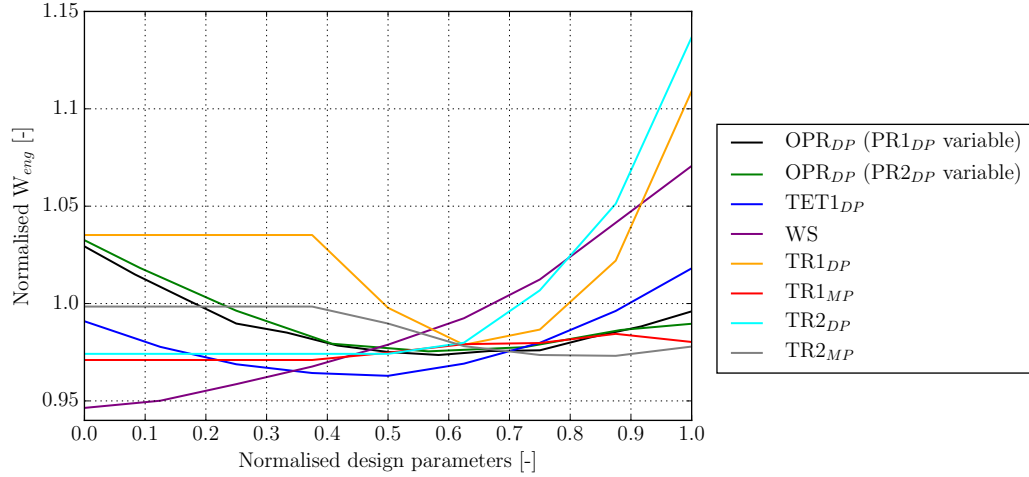


FIGURE 5.14: Effect of $ITR + SEQ$ engine design inputs on W_{eng}

Finally, Figs. 5.12 and 5.13 also allows to compare the merits of each auxiliary burner at a given operating condition. It can be observed that at $TR = 1$ ($x = 0.625$), there is no difference in SFC_{cr} nor SP_{MP} . Regardless the TR used as variable, the engine is working at the same temperatures setting ($TET2 = TET3 = TET1$). However, at the left of this point it is noted that SFC_{cr} is lower for those curves with TR_{2cr} variable (grey lines), whilst SFC_{cr} is lower for the curves with TR_{1cr} variable (red lines) at the right of this point. The reason is that at a TR below 1, $TET_2 > TET_3$ for the curves with variable TR_{2cr} , whilst at a TR above 1, $TET_2 > TET_3$ for the curves with variable TR_{1cr} . $TET_2 > TET_3$ implies that more heat addition is occurring at the 1st reheater, where the pressure is higher and so, the process is more thermally efficient.

Similar comments apply for the relative distribution of SP_{MP} (red and grey lines); in this case, SP_{MP} is higher when $TET_3 > TET_2$, which means that higher SP_{MP} gains are obtained when most of the combustion occurs in the 2nd burner. Therefore, it can be affirmed that for higher thermal efficiency is better to operate the engine at $TET_2 > TET_3$, whilst for the sake of SP augmentation it is preferred to have $TET_3 > TET_2$.

5.2 Powerplant optimisation

In this section, the optimisation of the conceptual engine configurations is described. This analysis consists of a multi-objective approach targeting the minimisation of SFC_{cr} and W_{eng} . An individual optimisation was completed for each

of the engine architectures shown in Fig. 3.6.

SFC_{cr} is used in the optimisation, since this parameter is considered to be representative of the helicopter fuel consumption over a typical mission, where cruise is the dominant flight phase. Alternatively, the mission block fuel could be used instead, but that would imply a higher number of variables involved in the calculation, since the TRs at the different missions phases should be added. This would result in an increased computational time. As observed in Table 5.5, the number of variables utilised in the optimisation is already high (specially for the $ITR + SEQ$ case).

Therefore, under the assumption considered, the optimal thermal efficiency at cruise is to be sought. At a later stage, the most optimal configurations will be assessed over a complete mission to evaluate the final benefits on mission block fuel.

TABLE 5.5: Cycle variables used in the optimisation of each engine architecture

Engine architecture	Cycle design variables	
CONV	$PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}$	[3]
ITR	$PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS$ $TR_{1_{DP}}, TR_{1_{cr}}, TR_{1_{MP}}$	[7]
SEQ	$PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS$ $TR_{2_{DP}}, TR_{2_{cr}}, TR_{2_{MP}}$	[7]
ITR+SEQ	$PR_{1_{DP}}, PR_{2_{DP}}, TET_{1_{DP}}, WS, TR_{1_{DP}}$ $TR_{1_{cr}}, TR_{1_{MP}}, TR_{2_{DP}}, TR_{2_{cr}}, TR_{2_{MP}}$	[10]

The method followed to complete the multi-objective optimisation of the conceptual engines is graphically explained in Fig. 5.15. As observed, the optimiser produces the combination of cycle design variables for each engine candidate. With this information, the thermodynamic performance of the engine is calculated iteratively by the combination of TURBOMATCH and the turbine cooling flows sizing routine. This process is similar to the one shown in Fig. 3.1. For the engine weight estimation, uniquely the set of cycle design variables of the candidate engine is required by the engine weight RSM . The calculated fitness values for each candidate (SFC_{cr} and W_{eng}) are then used by the optimiser routine to built the final Pareto front containing the most attractive designs. The process followed to generate the engine weight RSM for each engine architecture is explained in the next subsection.

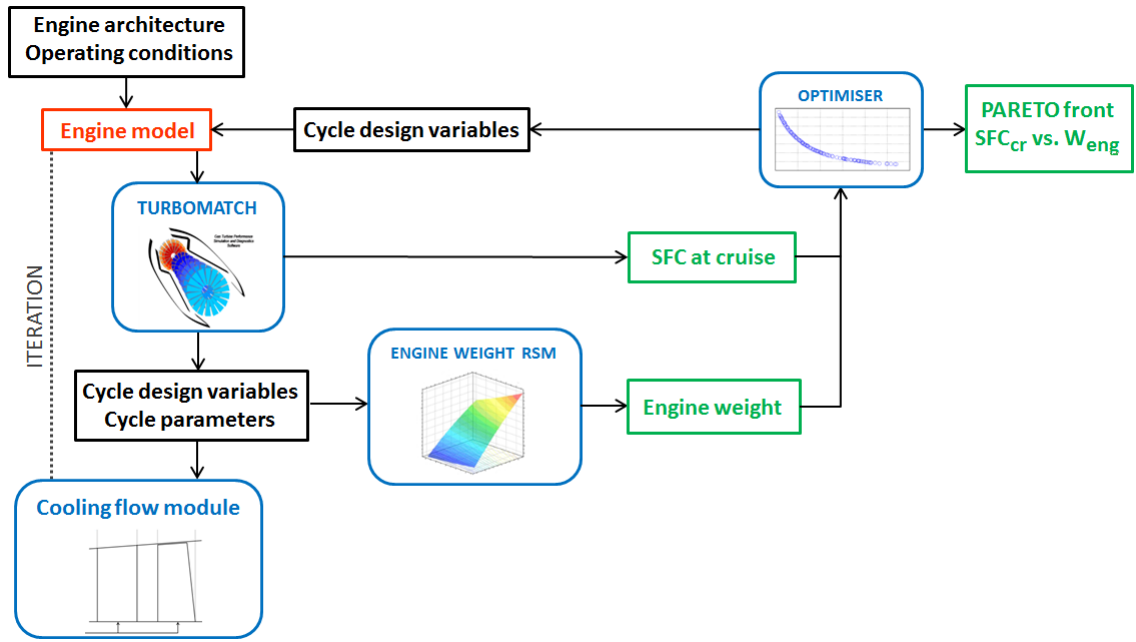


FIGURE 5.15: Powerplant multi-objective optimisation scheme

The optimiser employed is based on Genetic Algorithms (*GAs*), which constitute a stochastic approach based on the mechanisms of natural selection and its main principle: survival of the fittest. *GAs* based optimisers start with the definition of the mathematical model (or objective function) that is used to evaluate the fitness of each candidate. Next step consists on the evolution of the solutions, which starts by creating an initial population. Beginning with this population, the genetic operators process the population iteratively until a given stopping criteria is satisfied. The most common operators used in *GAs* are: selection (allocation of more offspring to better individuals), crossover (combination of parental solutions parts to form new and potential better offspring), and mutation (similar to crossover, but uniquely a single individual is modified). The fact that *GAs* methods have a widespread use in engineering design, together with the advantages listed below, motivated the author to use this kind of optimiser approach:

- Concept easy to understand.
- Capability to support multi-objective optimisation.
- Suitable for optimisations involving large number of parameters.
- Ability to avoid being trapped in local optimal solutions.
- Availability of developed *GAs* modules in the public domain.

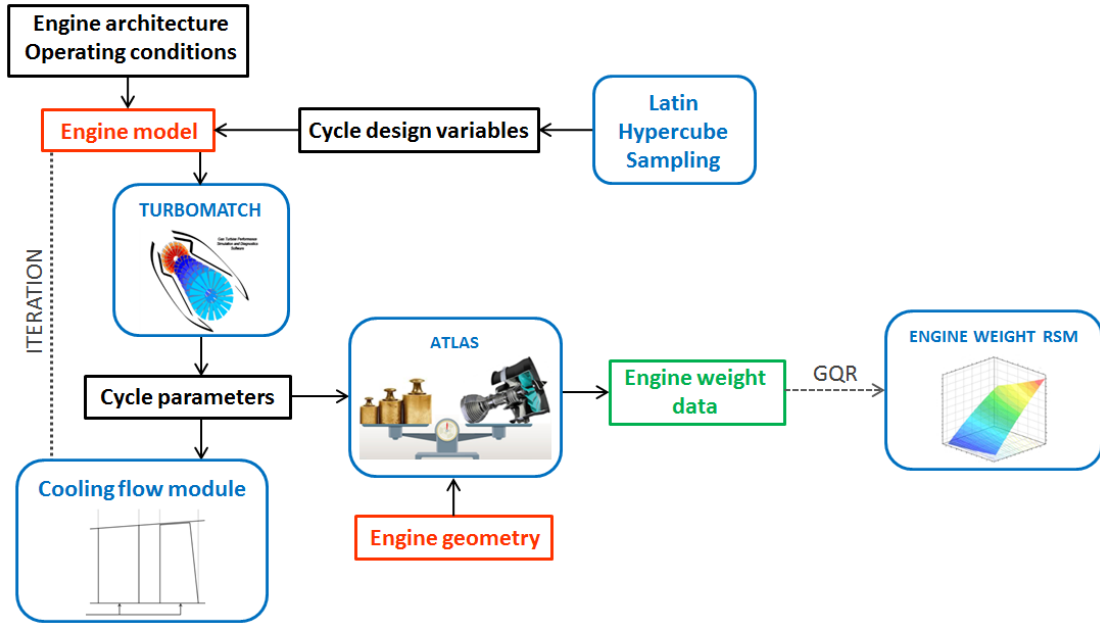
5.2.1 Surrogate engine weight modelling

The powerplant optimisation process involves the evaluation of a large amount of potential candidates. The high computational cost of each ATLAS simulation precludes its direct application in the optimisation analysis. Therefore, surrogate models (or meta-models) are used instead to alleviate the execution time.

The development of the surrogate models for the engine weight estimation requires an initial design space exploration which is later employed to build a *RSM* capable to predict the outputs of the system to a given set of inputs. For the design space mapping it is important to use an appropriate *DOE* technique. In this study, the *LHS* approach is chosen to fill the design space. This method is able to capture the maximum possible information of a system response in a systematic way, and has proven its efficacy as space-filling technique in previous rotorcraft powerplant optimisation analysis [32]. An exhaustive description of this method can be found in [57].

Finally, the mathematical relation between the engine design inputs and outputs is established by a Gaussian Quadratic Regression (*GQR*) method. The Gaussian Regression (or Kriging regression) is a probabilistic-based method that offers a better fitting of the intermediate values than other interpolation methods based on smoothness of the fitted values, such as simple polynomial regression. As it will be observed, the quadratic regression offers a high level of accuracy for this particular application, so cubic or higher order regressions were not attempted. The resulting engine weight *RSMs*, one for each engine architecture, will be used in the optimisation case undertaken in this study. The application of the described surrogate modelling technique to the current design process is shown in Fig. 5.16.

The *LHS* module generates a list of candidate engines that populate the design space, each of the engine designs consisting of a list of cycle design variables based on Eqs. 5.2, 5.4, 5.6 and 5.8. Since there is no need to evaluate cruise performance for the generation of the engine weight *RSM*, the *TRs* at cruise are not included in the process. Note that the design space explored by the *LHS* routine is limited by the boundaries listed in Table 5.2. The engine architecture and the operating conditions at *DP* (for turbine cooling flows sizing) and *MP* (power setting that sizes the engine) have to be declared. The details of these two operating points are similar to those listed in Table 5.1. Once the calculation of the *DP* performance converges, TURBOMATCH calculates the engine cycle parameters at *MP* setting,

FIGURE 5.16: Engine weight *RSM* generation

which are then used by ATLAS to calculate the engine dimensions and weight. For each engine candidate, the set of cycle variables and its weight are saved in a database. The application of the *GQR* method over this 'cloud' of data constitutes the engine weight *RSM* to be integrated in the optimisation routine. The generated *RSM* is then able to quickly estimate the weight of the engine, by simply introducing the design variables.

Finally, some constraints are also applied in the design space exploration to ensure that unrealistic or unfeasible engine configurations are not included in the system response map:

- The entry temperatures at the second and third turbines at *DP*, $TET_{2_{DP}}$ and $TET_{3_{DP}}$, have to be lower than the upper limit set to $TET_{1_{DP}}$ (Eq. 5.9). This way, the limit *TET* value is never exceeded in any burner, regardless the combination of temperature ratios.

$$TET_{2_{DP}}, TET_{3_{DP}} < TET_{1_{DP,u}} \quad (5.9)$$

- The entry temperatures at all turbines at cruise have to be lower than their value at *DP* (Eq. 5.10). This limitation is adopted to ensure that the cooling flows, calculated at *MCP* rating, are sufficient for the engine operation at

cruise.

$$TET_{1_{cr}} < TET_{1_{DP}}, \quad TET_{2_{cr}} < TET_{2_{DP}}, \quad TET_{3_{cr}} < TET_{3_{DP}} \quad (5.10)$$

- The entry temperatures at all turbines at *MP* rating is limited to guarantee a reliable engine operation. The *MP* rating is a power level that the engine must be able to deliver during a very short period of time in an emergency situation. Generally, the engine needs an overhaul after such contingency since certain damage in the hot section blades is expected as a result of the extremely high gas path temperatures. However, to reduce the severity of this damage, the temperature by which the *TETs* at *MP* can exceed the *TETs* at *DP* is restricted in the design space exploration (Eq. 5.11). *DP* is adopted as reference condition since the turbines cooling flows are sized at this operating point.

$$\frac{TET_{1_{MP}}}{TET_{1_{DP}}}, \frac{TET_{2_{MP}}}{TET_{2_{DP}}}, \frac{TET_{3_{MP}}}{TET_{3_{DP}}} < 1.1 \quad (5.11)$$

A margin of 10% is chosen after inspection of the temperature ratio in the baseline engine, where the TET_1 at *MP* (1,567 K) is around 9% higher than the TET_1 at *DP* (1,344 K).

- The axi-centrifugal compressor *PCN* at *MP* rating is limited to ensure the turbomachinery does not operate far from its normal working range (Eq. 5.12). The TURBOMATCH model of the baseline engine dictates that the engine is operating at $PCN = 96.2\%$ at *DP*, and at $PCN = 100\%$ at *MP*. For the design exploration a higher *PCN* limit is desired. The GE CT7-8A engine, which is a civil derivative of the T700, has a gas generator speed 4% higher at 30-Second *OEI* (maximum power setting) than at *MCP* rating, where the *PCN* is considered to be 100% for this particular engine [111]. Therefore, for the design space exploration a *PCN* limit of 105% is considered as a realistic value.

$$PCN_{MP} < 105\% \quad (5.12)$$

Following the described approach, a *RSM* is created for each engine architecture to efficiently predict the engine weight from a given set of design inputs. The verification of the *RSMs* generated is covered in the next section.

5.2.1.1 Engine weight surrogate models validation

In order to validate the surrogate models, the weight prediction of the software ATLAS is compared with the prediction of the *RSM* for each engine candidate. The validation method applied herein is the extensively-used Leave-One-Out Cross Validation (*LOOCV*). This technique is a kind of cross-validation where the training algorithm (in this case, the quadratic regression) is applied once for each population item, using all other items as a training set and using the selected item as test set [116]. Around 500 items were used for the validation of each weight *RSM*. The results of the *LOOCV* are displayed in Fig. 5.17.

The quality of the regression is then assessed in terms of the Pearson correlation coefficient and the gradient of its associated linear regression line. This way, a Pearson coefficient of unity and a linear gradient of 45° indicate a perfectly linear correlation. Finally, the error incurred by the surrogate model at each population item is averaged to provide the mean error of the *RSM*.

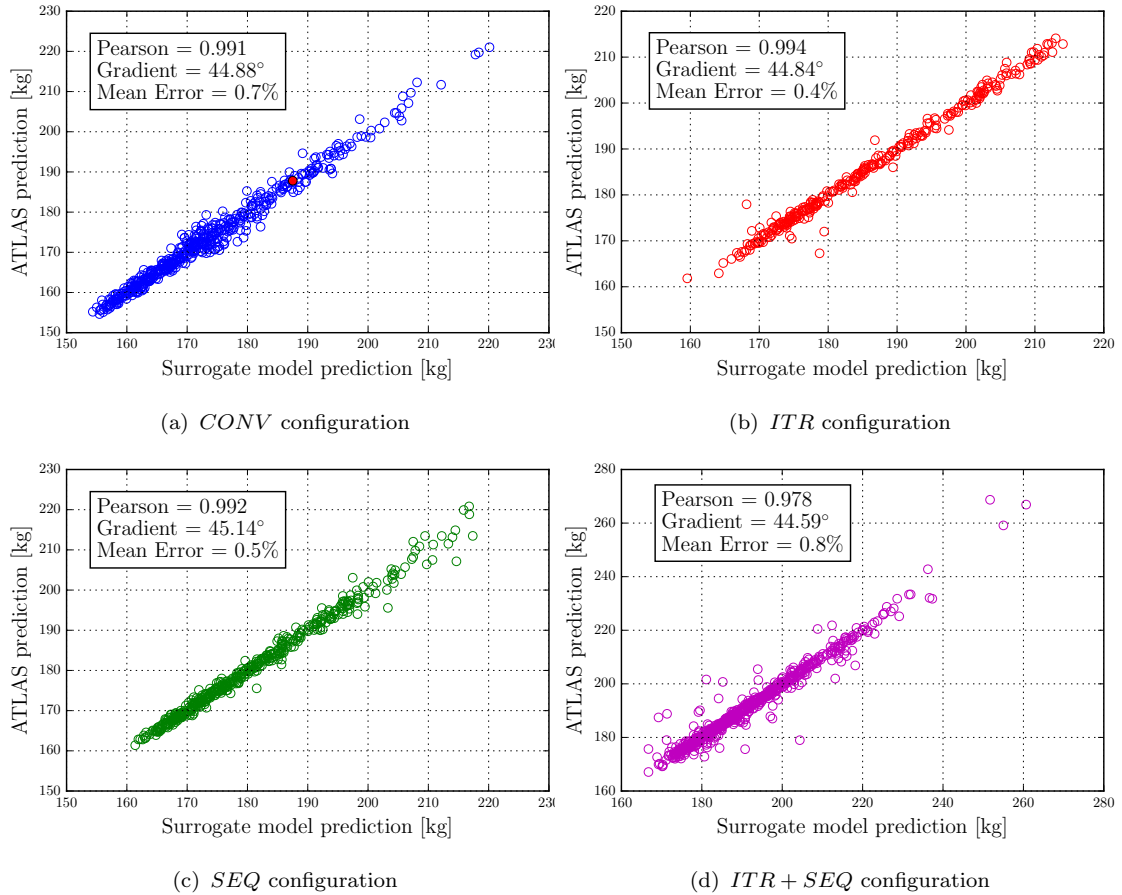


FIGURE 5.17: *LOOCV* results for the engine weight *RSM*s

As observed in Fig. 5.17, the Pearson coefficients and the linear gradients indicate a high level of correlation between the software results and the surrogate predictions for all the engine architectures. Furthermore, it is noted that the mean error is also below 1%. For the *CONV* engine, the comparison between the weight predicted by ATLAS and the weight predicted by the surrogate model for the baseline engine (red dot) shows a good level of correlation, as well.

5.2.2 Optimised conceptual engines

In this section the optimisation results for each engine configuration are presented. The data are shown in the shape of a Pareto front surface highlighting the trade-off that exists between thermal efficiency and engine size. It is worth to mention that the same constraints introduced in the previous design exploration (see Section 5.2.1) were also applied during the optimisation process.

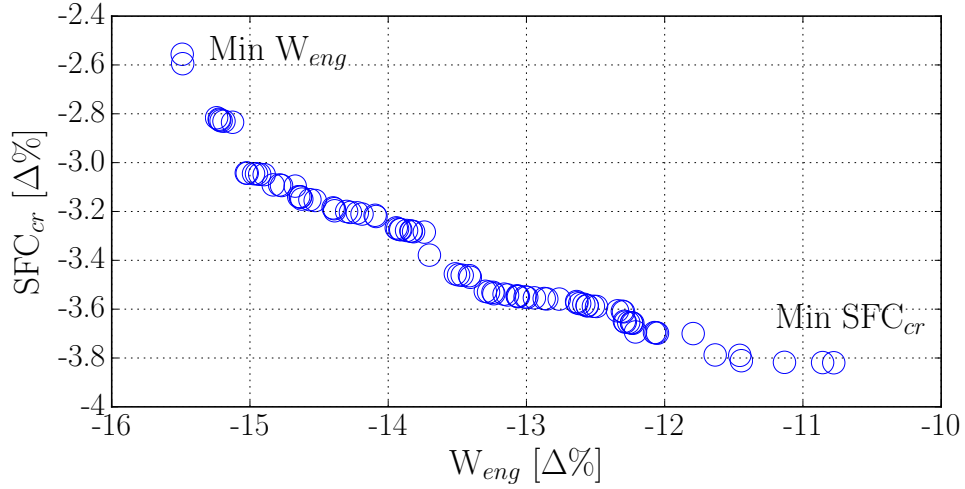
A final comparison is established between the optimal solutions of each engine architecture, that will serve to select the most promising engine candidates to be assessed at mission level.

5.2.2.1 Optimised *CONV* engine

The results of the multi-objective optimisation of SFC_{cr} and W_{eng} for the *CONV* engine are presented in Fig. 5.18. The fitness parameters are shown in terms of percentage increments with respect to the baseline engine values. Therefore, a negative value indicates a reduction in the fitness variable and *vice versa*. Note that, within the design space explored, the optimiser routine was able to find engine candidates with a better combination of SFC_{cr} and W_{eng} than the baseline engine.

Two points from the previous Pareto front are selected as the most optimal engine candidates, namely the $Min W_{eng}$ and the $Min SFC_{cr}$ solutions. The cycle design variables of these engines are listed in Table 5.6.

As expected, the design variables for the $Min SFC_{cr}$ and the $Min W_{eng}$ solutions respond to the trends identified in Figs. 5.1 and 5.5, respectively. The benefits in SFC_{cr} and W_{eng} attained by these two optimal configurations are summarised in

FIGURE 5.18: Pareto front for *CONV* engine optimisation. SFC_{cr} vs. W_{eng} TABLE 5.6: Optimal *CONV* engines design variables

	Baseline	Min W_{eng}	Min SFC_{cr}
PR_{1DP} [—]	5.32	5.19	6.51
PR_{2DP} [—]	2.80	2.92	2.82
OPR_{DP} [—]	14.90	15.14	18.33
TET_{1DP} [K]	1,440	1,789	1,692

Table 5.7. Note that a maximum reduction of 3.82% and 15.49% in SFC_{cr} and W_{eng} , respectively, can be obtained by the optimisation of the *CONV* engine.

TABLE 5.7: Optimal *CONV* engine results

	SFC_{cr}		W_{eng}	
	[$\mu\text{g}/\text{J}$]	% Δ	[kg]	% Δ
Baseline	87.429	-	187.81	-
Min W_{eng}	85.194	-2.56	158.72	-15.49
Min SFC_{cr}	84.090	-3.82	167.57	-10.78

5.2.2.2 Optimised *ITR* engine

The results of the multi-objective optimisation of SFC_{cr} and W_{eng} for the *ITR* engine are shown in Fig. 5.19. Similarly to the *CONV* engine case, the fitness parameters are presented as percentage increments with respect to the baseline engine values. It can be observed, that there exist optimal *ITR* candidates that improve the baseline engine design within the design space explored.

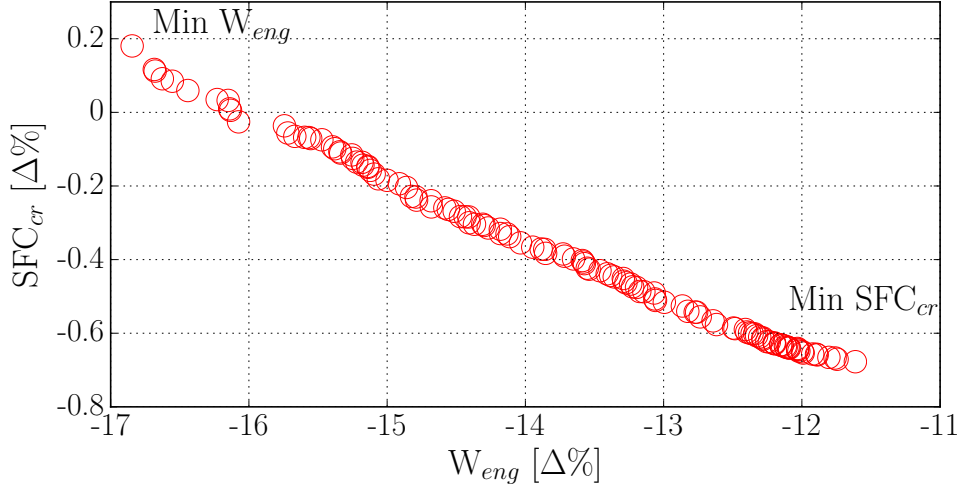


FIGURE 5.19: Pareto front for *ITR* engine optimisation. SFC_{cr} vs. W_{eng}

As before, to examine the optimisation results obtained for the *ITR* configuration, two points from the previous Pareto front are selected, the $Min W_{eng}$ and the $Min SFC_{cr}$ candidates. The cycle design variables of these two engines are listed in Table 5.8.

TABLE 5.8: Optimal *ITR* engines design variables

	Baseline	Min W_{eng}	Min SFC_{cr}
$PR_{1_{DP}}$ [—]	5.32	6.25	6.50
$PR_{2_{DP}}$ [—]	2.80	2.68	2.75
OPR_{DP} [—]	14.90	16.73	17.86
$TET_{1_{DP}}$ [K]	1,440	1,743	1,763
WS [—]	0.50	0.30	0.30
$TR_{1_{DP}}$ [—]	—	0.92	0.84
$TR_{1_{MP}}$ [—]	—	0.91	—
$TR_{1_{cr}}$ [—]	—	—	—

For the $Min W_{eng}$ solution the design parameters obtained are in agreement with the trends shown in Fig. 5.8. The OPR_{DP} and the $TET_{1_{DP}}$ obtained are close to the values that minimise W_{eng} , and the optimal value of WS is located in the lower range. Regarding the optimal values of TR_1 , it is observed that its value at *DP* is similar to the optimal value in Fig. 5.8, and that its value at *MP* rating is as high as possible (to minimise W_{eng}) within the limitation imposed for $TET_{2_{MP}}$.

On the other hand, for the $Min SFC_{cr}$ solution it is seen that the optimal OPR_{DP} is around the optimal value of 17 spotted in Fig. 5.6. However, as the optimal

$TET_{1_{DP}}$ is higher than the baseline one, the optimal OPR_{DP} slightly increases accordingly. The optimal WS value is the lowest possible, which is also in agreement with the trend of Fig. 5.6. For the values of TR_1 , it is observed that the optimal value at DP is limited by the impact on cooling flows, and that the operation of the inter-turbine reheater is not required neither at cruise nor at MP rating. At cruise, no benefit in SFC is attained by the operation of the auxiliary burner as seen in Fig. 5.6. With regard to $TR_{1_{MP}}$, the reheater is switched off at this condition in order to not exceed the limitation imposed for $TET_{2_{MP}}$.

Finally, the SFC_{cr} and W_{eng} results of the optimal ITR solutions are compared with the figures of the baseline engine in Table 5.9. It is observed that both optimal configurations have a lower weight than the baseline engine, but only the $Min SFC_{cr}$ offers a benefit in SFC_{cr} . In fact, the thermal efficiency of the optimal ITR engines is impaired by their higher cooling flows demand.

TABLE 5.9: Optimal ITR engine results

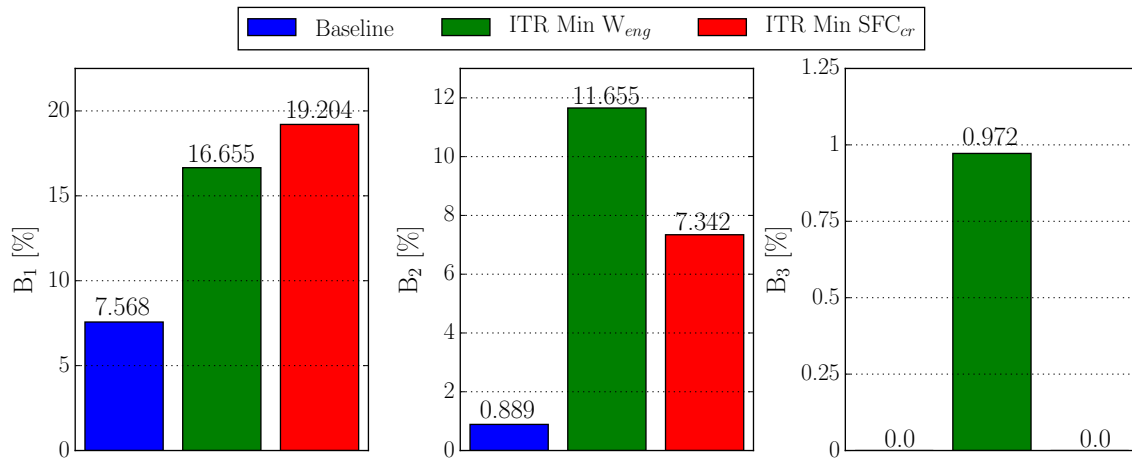
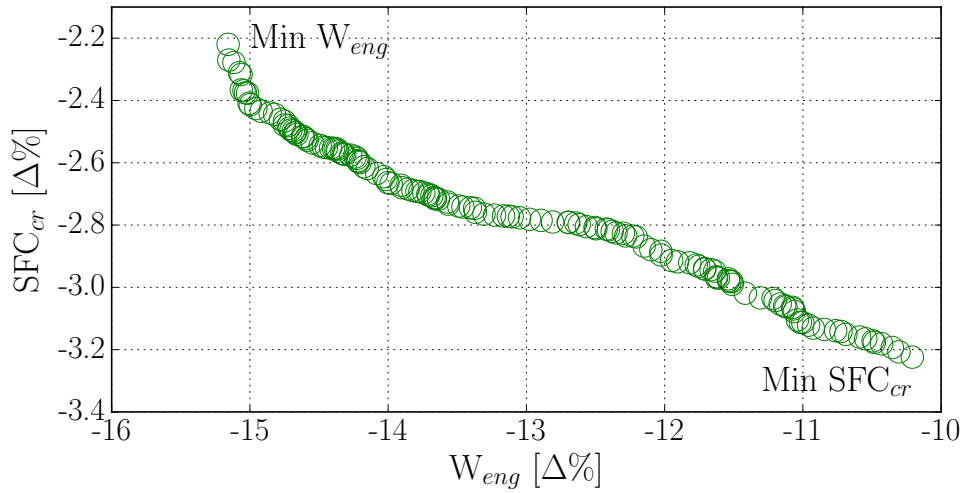
	SFC_{cr}		W_{eng}	
	$[\mu g/J]$	$\% \Delta$	$[kg]$	$\% \Delta$
Baseline	87.429	-	187.81	-
Min W_{eng}	87.587	0.18	156.17	-16.84
Min SFC_{cr}	86.837	-0.68	166.00	-11.61

The cooling flow fractions of each turbine for the different engines analysed are presented in Fig. 5.20. The bar diagrams show that due to the increase in TET_1 , the fraction B_1 is more than doubled. However, the biggest impact is observed in the value of B_2 , which drastically rises as a result of the reheating process that occurs between the GGT stages. Apart from the cooling effect, the increased pressure loss introduced by the auxiliary burner also penalises the thermal efficiency of the engine.

5.2.2.3 Optimised SEQ engine

Figure 5.21 presents the results of the multi-objective optimisation for the SEQ engine configuration.

From this Pareto front two candidate engines are selected as the most representative optimal designs: the $Min W_{eng}$ and the $Min SFC_{cr}$ solutions. The design variables of these two engines are listed in Table 5.10.

FIGURE 5.20: Cooling flow fractions comparison for *ITR* optimal enginesFIGURE 5.21: Pareto front for *SEQ* engine optimisation. SFC_{cr} vs. W_{eng} TABLE 5.10: Optimal *SEQ* engines design variables

	Baseline	Min W_{eng}	Min SFC_{cr}
$PR_{1_{DP}}$ [—]	5.32	5.37	5.52
$PR_{2_{DP}}$ [—]	2.80	3.19	3.49
OPR_{DP} [—]	14.90	17.16	19.23
$TET_{1_{DP}}$ [K]	1,440	1,741	1,605
WS [—]	0.50	0.34	0.67
$TR_{2_{DP}}$ [—]	—	0.75	0.78
$TR_{2_{MP}}$ [—]	—	0.76	0.78
$TR_{2_{cr}}$ [—]	—	—	—

For the $Min W_{eng}$ engine the value of $TET_{1_{DP}}$, that has a great impact on mass reduction, is close to the upper boundary as expected from the trend observed in Fig. 5.11. The WS variable barely affects W_{eng} (Fig. 5.11), so its value is defined by the thermal efficiency at cruise. It has been already seen that the higher the WS , the lower the cooling flows for the GGT . However, for the $Min W_{eng}$ engine, despite the cooling flow penalty, a lower WS proved to be beneficial due to the fact that at the OD condition of cruise, the engine is operating with a higher combination of OPR and TET that offsets that penalty. Finally for the optimal values of TR_2 , it is seen that the value at design point is limited by the increase in FT cooling requirements, and that at MP its value is also limited by the restriction imposed for $TET_{3_{MP}}$. At cruise there is no efficiency gain with the sequential combustor (Fig. 5.9), so it is off.

For the $Min SFC_{cr}$ solution it is seen that the optimal values of OPR_{DP} and $TET_{1_{DP}}$ are higher than those of the baseline engine. Due to the action of the sequential combustor, the optimal $TET_{1_{DP}}$ is reduced with respect to the $Min SFC_{cr}$ $CONV$ engine, which is in agreement with the parametric analysis results (see Fig. 5.1 and Fig. 5.9). The optimal WS value is close to the upper boundary which also coincides with the trend in Fig. 5.9. Regarding the optimal values of TR_2 , it is observed that, similarly to the ITR engine, the optimal value at DP is limited by the penalty introduced in the cooling flows. At MP , the higher $TR_{2_{MP}}$, the lower the W_{eng} (Fig. 5.11) with no impact in engine efficiency, so the optimal value is the highest possible without exceeding the limitation imposed for $TET_{3_{MP}}$.

Table 5.11 presents the results of the optimal SEQ solutions together with the figures of the baseline engine. It is highlighted that the two optimal configurations have a reduced W_{eng} and a lower SFC_{cr} than the baseline engine.

TABLE 5.11: Optimal SEQ engine results

	SFC_{cr}		W_{eng}	
	$[\mu g/J]$	$\% \Delta$	$[kg]$	$\% \Delta$
Baseline	87.429	-	187.81	-
Min W_{eng}	85.488	-2.22	159.34	-15.16
Min SFC_{cr}	84.610	-3.22	168.64	-10.21

Similarly to the ITR configuration, the improvement in thermal efficiency is limited by the cooling flow penalty introduced by the operation of the auxiliary burner

(Fig. 5.22). Cooling fractions B_1 and B_2 are increased with respect to the baseline engine, as a result of the higher TET_{1DP} . It is worth to mention that for the $Min SFC_{cr}$ the increase in B_2 is not as relevant due to its high WS value, which translates into a lower TET_{2DP} . For the FT cooling fraction, B_3 , the reheating process causes an increase in TET_{3DP} over the material temperature limit. However, since the FT cooling flow can be bled at the outlet of the axial compressor, the cooling flow penalty is softened due to the lower temperature of the coolant air.

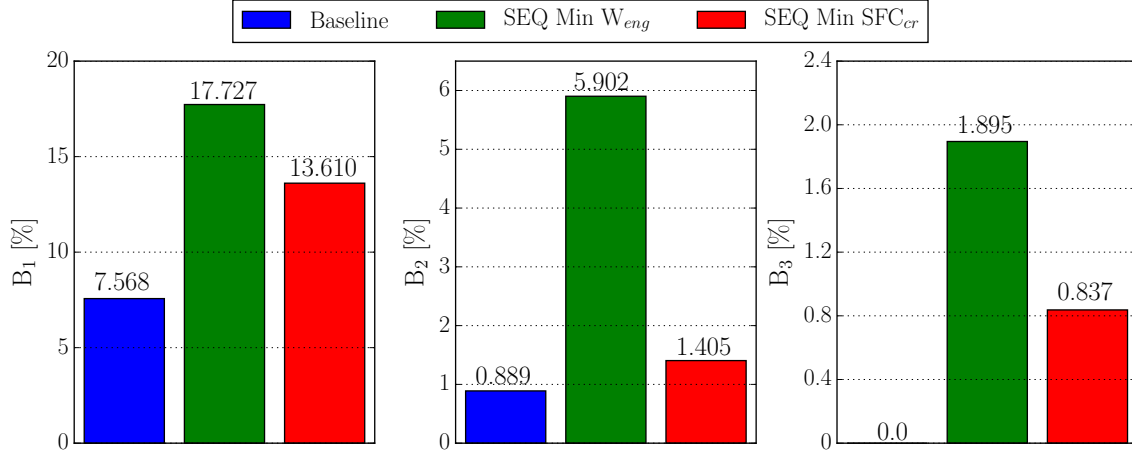


FIGURE 5.22: Cooling flow fractions comparison for SEQ optimal engines

5.2.2.4 Optimised $ITR + SEQ$ engine

Finally, the multi-objective optimisation results in terms of SFC_{cr} and W_{eng} for the $ITR + SEQ$ configuration are shown in the Pareto front of Fig. 5.23.

As it has been done for the rest of engine configurations, the optimal candidate engines with the lowest W_{eng} ($Min W_{eng}$), and lowest SFC_{cr} ($Min SFC_{cr}$) has been chosen as the most representative solutions of the optimisation analysis. The cycle design variables of these two points are summarised in Table 5.12.

The most important feature to notice is the fact that, for both optimal engines, the inter-turbine reheater is not operated at any working condition. This means that the optimal $ITR + SEQ$ behaves similarly to the SEQ engine with the additional drawback of the extra pressure loss introduced in the first auxiliary burner. This fact, explains that the design variables of the $ITR + SEQ$ optimal engines (Table 5.12) are close to the values of the optimal SEQ engines (Table 5.10).

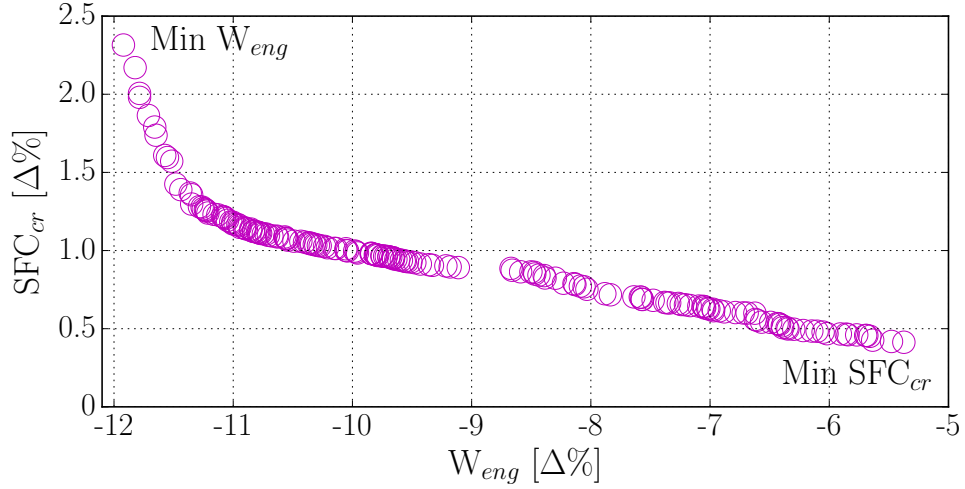


FIGURE 5.23: Pareto front for $ITR + SEQ$ engine optimisation. SFC_{cr} vs. W_{eng}

TABLE 5.12: Optimal $ITR + SEQ$ engines design variables

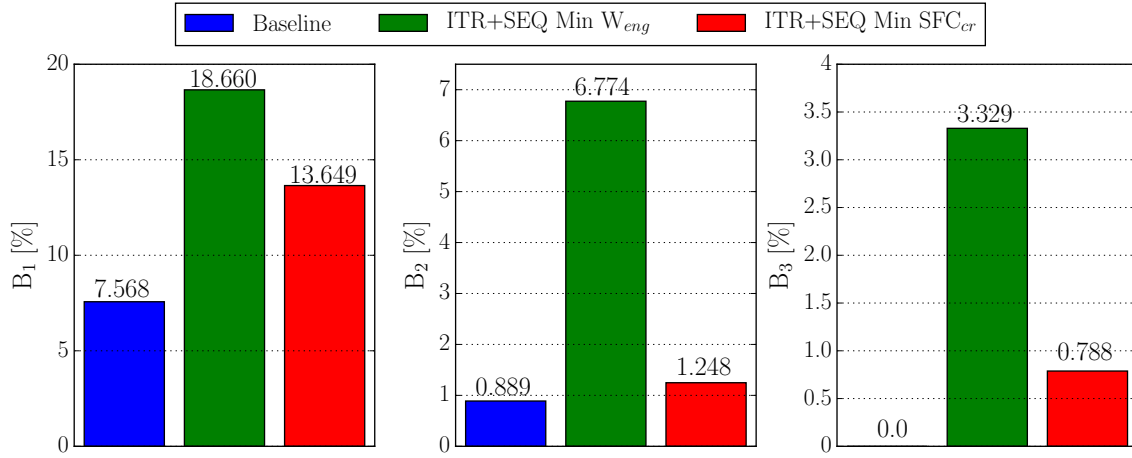
	Baseline	Min W_{eng}	Min SFC_{cr}
$PR_{1_{DP}}$ [—]	5.32	5.30	6.06
$PR_{2_{DP}}$ [—]	2.80	3.07	3.19
OPR_{DP} [—]	14.90	16.25	19.31
$TET_{1_{DP}}$ [K]	1,440	1,800	1,609
WS [—]	0.50	0.32	0.70
$TR_{1_{DP}}$ [—]	—	—	—
$TR_{1_{MP}}$ [—]	—	—	—
$TR_{1_{cr}}$ [—]	—	—	—
$TR_{2_{DP}}$ [—]	—	0.76	0.77
$TR_{2_{MP}}$ [—]	—	0.77	0.77
$TR_{2_{cr}}$ [—]	—	—	—

The SFC_{cr} and W_{eng} results of the optimal $ITR + SEQ$ engines are compared with the figures of the baseline engine in Table 5.13. Note that despite the extra weight added by the two auxiliary burners, the optimal $ITR + SEQ$ solutions are still lighter than the baseline engine. However, it is observed the thermal efficiency of the baseline engine cannot be improved by the $ITR + SEQ$ optimal designs.

Although the cooling flows of the optimal $ITR + SEQ$ engines (Fig. 5.24) are comparable to the values of the optimal SEQ solutions (Fig. 5.22), the additional pressure loss introduced by the presence of the inter-turbine reheater impairs the thermal efficiency of the engine, and ultimately causes that no benefits in SFC_{cr} can be obtained with the optimal $ITR + SEQ$ configurations.

TABLE 5.13: Optimal *ITR* + *SEQ* engine results

	SFC_{cr}		W_{eng}	
	$[\mu\text{g}/\text{J}]$	$\%\Delta$	$[\text{kg}]$	$\%\Delta$
Baseline	87.429	-	187.81	-
Min W_{eng}	89.453	2.31	165.42	-11.92
Min SFC_{cr}	87.790	0.41	177.72	-5.37

FIGURE 5.24: Cooling flow fractions comparison for *ITR* + *SEQ* optimal engines

5.2.2.5 Optimal configurations comparison

The merits of each conceptual engine configuration have been addressed in previous sections individually. The next step is to compare them in order to identify the most promising configurations. For that purpose, the Pareto fronts obtained for each engine are presented together in Fig. 5.25.

In terms of W_{eng} it is observed that all the optimal configurations are lighter than the baseline engine. As expected, the *ITR*+*SEQ* engine is the one with the lowest weight improvement due to the presence of two additional burners. Between the *ITR* and the *SEQ* configurations, the *ITR* is the one that offers the biggest weight reduction. This is explained by the fact that the inter-turbine reheater is generally lighter than the sequential reheater due to the higher pressure (and density) of the gases at this engine station. Compared to the *CONV* engine, both configurations have a similar or slightly better potential for powerplant weight saving. This means, that the additional weight of the auxiliary burner is completely offset by the reduction of the engine gas path dimensions.

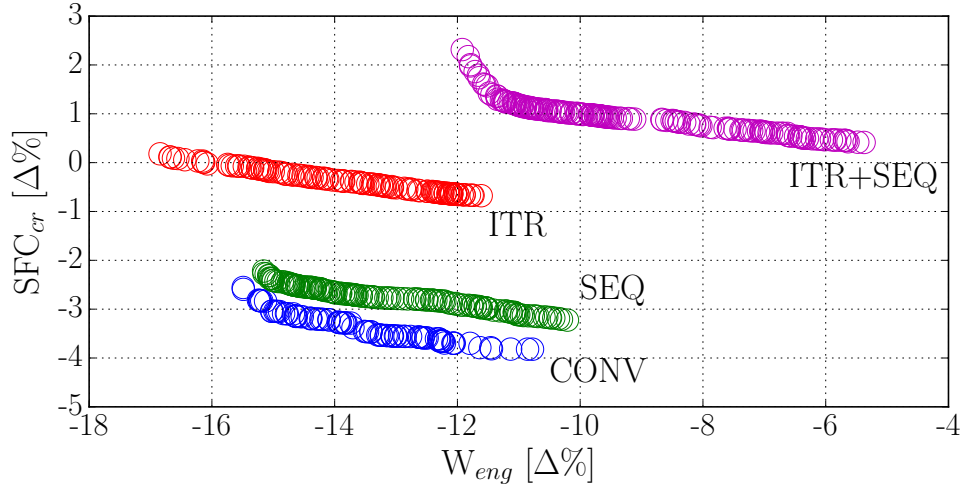


FIGURE 5.25: Comparison of optimal engine configurations. SFC_{cr} vs. W_{eng}

With regard to SFC_{cr} , except from the $ITR + SEQ$ architecture, the rest of configurations analysed offer a better efficiency at cruise than the baseline engine. In general, all the reheated cycle architectures are penalised by the additional cooling requirements and the added pressure loss in the auxiliary burners. Uniquely the SEQ optimal solutions show a thermal efficiency at cruise close to the $CONV$ engine. This is explained by the lower cooling flow and burner pressure loss penalties these configurations have in comparison with the rest of reheated architectures.

For better visualisation, the bar diagrams of Fig. 5.26 show the benefits in W_{eng} and SFC_{cr} achieved by the $Min W_{eng}$ and the $Min SFC_{cr}$ optimal solutions, respectively.

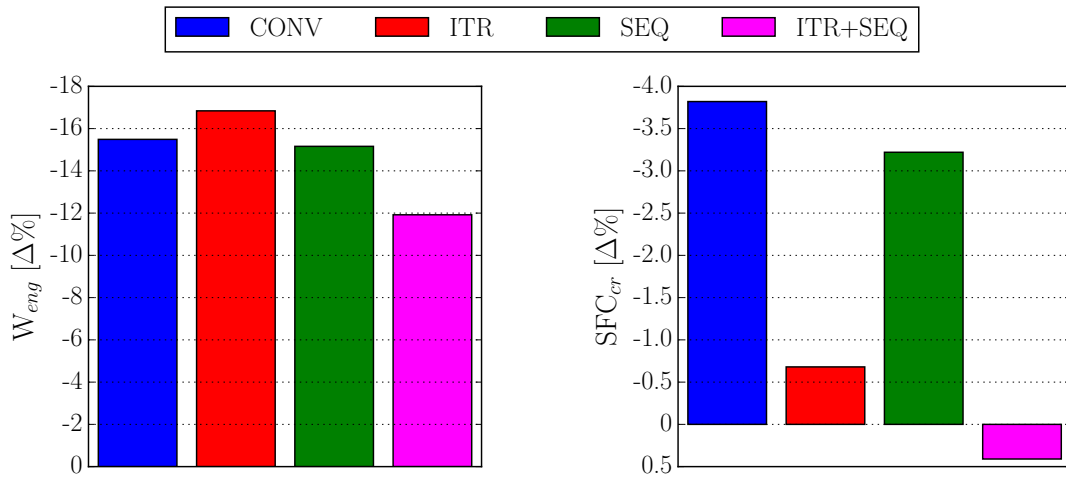


FIGURE 5.26: Optimal engines percentage savings in W_{eng} and SFC_{cr}

In sum, the most attractive configurations, at sight of the parameters assessed and under the assumptions considered, are the *CONV* and the *SEQ* engines. The *ITR* shows a slightly higher potential for weight reduction but at the cost of a worse thermal efficiency. Therefore, the *CONV* and the *SEQ* are the two configurations that are going to be analysed at mission level. In particular, the $Min SFC_{cr}$ candidate of each architecture is to be assessed. This decision is supported by Fig. 5.27, which shows the effect of a weight fluctuation of $\pm 250\text{ kg}$ in the reference TOW (TOW_{ref}) on the engine shaft power required by the engine at forward flight. It is observed that, even though the weight fluctuation range is much higher than the variation introduced by the optimal engines, the change in power requirement is not relevant. The maximum power variation (around 5%) appears at hover, where the effect of helicopter mass is dominant. From this point, the impact of weight in helicopter power diminishes with airspeed, and so does the power fluctuation. In fact, at the cruise speed considered (65 m/s), the impact of helicopter mass on shaft power is almost non-existent. Hence, as helicopter power is barely affected, the effect of total weight saving on mission fuel can be considered as negligible compared with the effect of thermal efficiency at cruise.

The reduction of powerplant weight has a great impact on the payload-range capabilities of the helicopter which are not assessed in this project. This analysis is suggested as future work.

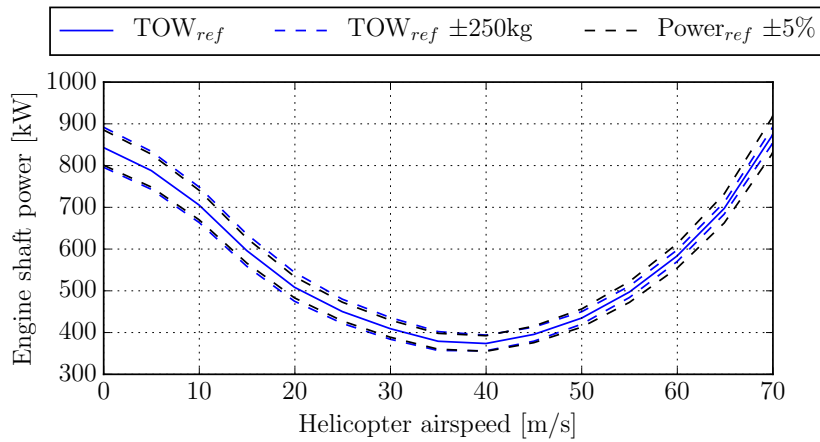


FIGURE 5.27: Engine shaft power at forward flight
(SL , $TOW_{ref} = 7700\text{ kg}$)

An additional aspect that has not been considered in the present analysis is the effect of the gas temperature in the hot section of the engine on components life, and the associated maintenance issues. In this regard, the results showed

that TET_{1DP} increases between 11% and 25% for the different optimal solutions in comparison with the baseline value. This rise in gas temperature will accordingly reduce the life of the engine components and the operational time between overhauls. As a consequence, the operational cost of the whole helicopter may increase if the benefits in fuel flow depletion are completely offset. Note that, if the $Min SFC_{cr}$ engines are compared, the SEQ architecture shows a TET_{1DP} 100 K lower than the $CONV$ engine. Therefore there exists a potential engine lifing benefit that may be achieved by the reheated configurations.

5.3 Engine performance assessment at mission level

As part of the optimisation process described in previous section, the most optimal engine architectures were identified. The next step is to assess the performance of the most suitable candidates over a realistic helicopter flight mission. The aim of this analysis is to foresee how the benefits in SFC_{cr} and W_{eng} translate into final mission fuel savings.

The procedure followed to complete the engine performance assessment at mission level is graphically described in the flowchart of Fig. 5.28. The starting point of the method is the information of the engine model (type of architecture, values of cycle design variables, cooling flows and W_{eng}) which is known from the previous optimisation process. For the reheated configurations, the values of the TRs at the different flight segments have to be also declared. The optimal TR at cruise is already known from the optimisation, but the optimal TRs at idle, hover, climb and descent have to be calculated, as well.

The engine weight information is used to update the helicopter fuselage mass, which modifies the helicopter TOW accordingly. *HECTOR* is then used to simulate the performance of the helicopter over the mission. As a result of this simulation, a list of the engine operating conditions at each mission sampling point is obtained. Each item of this list contains the information required to simulate the OD performance of the engine. Thus, by running *TURBOMATCH* iteratively at each sampling point and integrating the fuel flow data over time, the total mission block fuel is obtained.

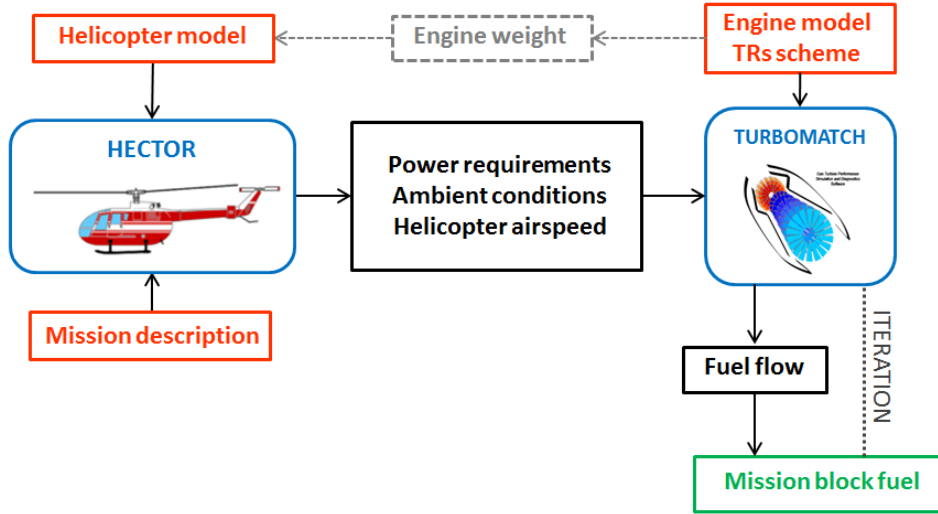


FIGURE 5.28: Assessment of conceptual powerplants at mission level flowchart

The routine to simulate the performance of each mission sampling point is already integrated in *HECTOR* (see Fig. 2.3). However, the current version of the code does not allow to declare the values of the *TRs* at each flight phase. This is the reason why *TURBOMATCH* has to be executed outside the main *HECTOR* routine for the assessment of the reheated cycle engines at mission level. The need to slightly adapt *HECTOR* to the novel reheated architectures is suggested as future work.

5.3.1 Flight mission description

The scenario chosen for the hypothetical rotorcraft mission is a simple one-leg mission. This flight path encompasses all the phases of the helicopter operation, *i.e.* idle, hover, climb, cruise, and descent, which permits to evaluate the relative performance of the engine at each flight segment. The geographical trajectory of the mission in global coordinates, and the altitude and airspeed profiles are reproduced in Fig. 5.29.

The flight altitude of the cruise leg is 400 *m* above *SL*, and the flight speed is set as 65 *m/s*. Climb and descent rates are held fixed at 5 *m/s*, and the initial (5 *min*) and final (1 *min*) idle segments performed to warm up and cool down the engines, respectively, are assumed to be operated at 20% of *DP* shaft power. A hover segment of 60 *s* duration is completed before the initial climb and after the final descent manoeuvre. The baseline *TOW* is set as 7,700 *kg*, which is the

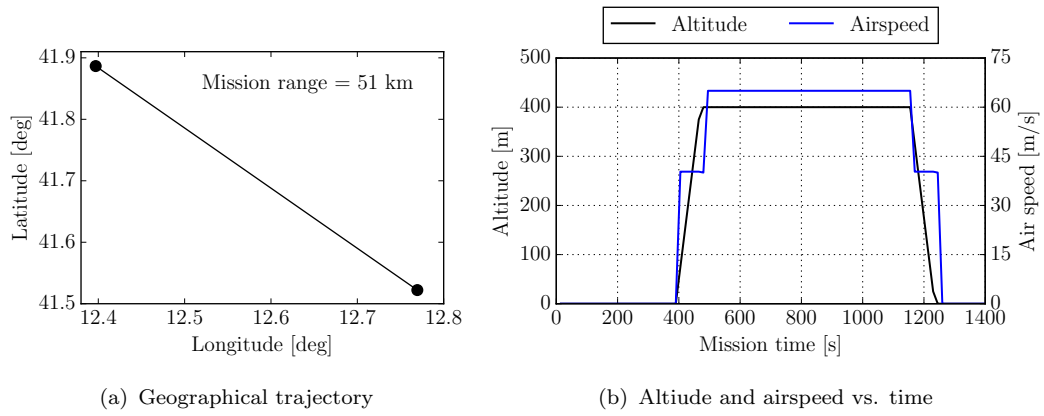


FIGURE 5.29: Flight mission definition

primary mission gross weight of the UH-60A Black Hawk helicopter. Intentionally, the helicopter mass, and cruise altitude and speed chosen are similar to those used to define the cruise power employed in the powerplant optimisation.

5.3.2 Calculation of flight segment optimal TRs

To simulate the performance of the optimal *SEQ* engine at mission level, the combination of TRs at each flight segment must be established first. If the operation of the auxiliary burner is needed to provide the required power, the TR value that minimises the *SFC* at each particular flight phase will be chosen as the most suitable value. For this initial assessment, the operating conditions at each mission segment are considered to be defined by the values listed in Table 5.14.

TABLE 5.14: Operating conditions of the mission flight segments

	Idle	Hover	Climb	Cruise	Descent
Flight altitude [m]	0	0	200	400	200
Flight airspeed [m/s]	0	0	40	65	40
Flight Mach number [–]	0	0	0.12	0.19	0.12
Ambient temperature [ISA dev.]	+0	+0	+0	+0	+0
Engine shaft power [kW]	197	825	644	680	116

Note that the flight altitude during climb and descent manoeuvres is variable. Hence, the altitude at these two segments was assumed to be the average of the initial and final values.

With regard to the engine shaft power values, they were obtained by simulating in *HECTOR* the helicopter performance over the mission path shown in Fig. 5.29. The baseline engine was used as the engine powerplant. The fuselage mass was modified according to the W_{eng} of the optimal *CONV* engine (167.57 *kg*), and the optimal *SEQ* engine (168.64 *kg*) indicated in Tables 5.7 and 5.11. For simplification, it is assumed that both engines has similiar mass and equal to 168 *kg*, which translates in a total weight saving of 40 *kg*. The effect of this mass reduction on the helicopter power requirements is shown in Fig. 5.30.

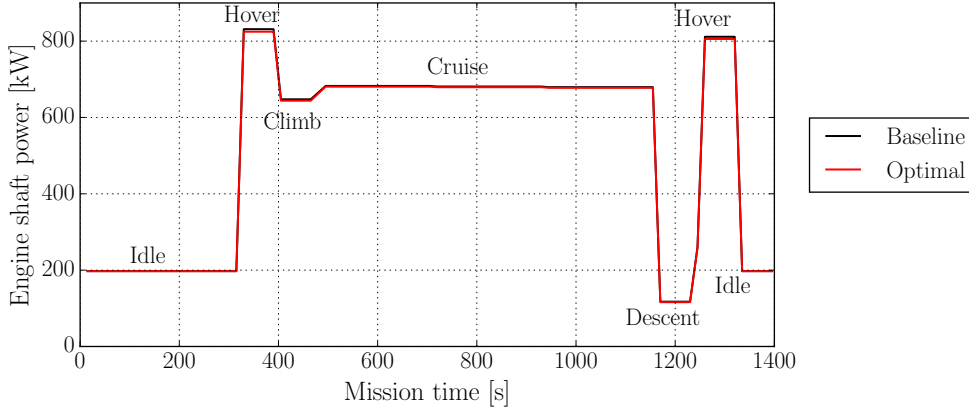
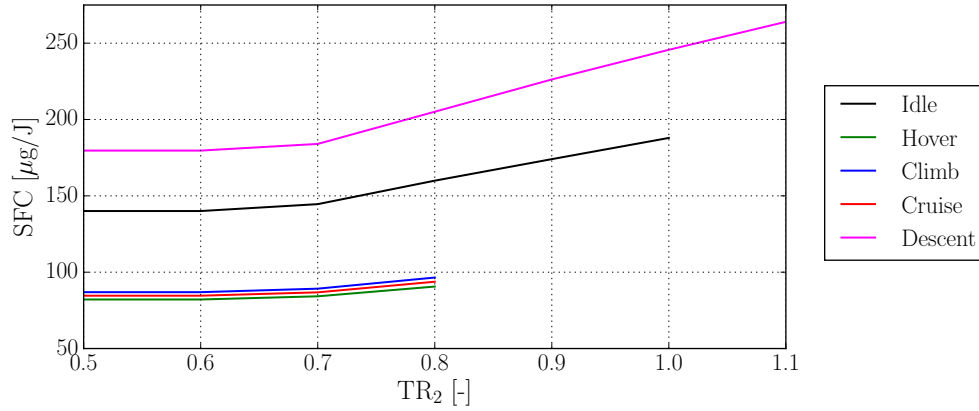


FIGURE 5.30: Engine power vs. time. Comparison between baseline and optimal engines

It is observed that the weight saving introduced by the optimal engine configurations has little effect on the power required by the helicopter over the mission. The power pattern obtained is in agreement with the trend shown in Fig. 5.27. In fact, uniquely at hover and climb (where the contribution of helicopter mass to the helicopter power is more relevant) there exists a small variation in the engine shaft power.

Finally, the response of SFC to changes in TR_2 at each flight condition is reproduced in Fig. 5.31. The operation of the additional burner is not required at any operating point to deliver the power required by the helicopter (not even at hover which is the condition with the highest power demand). In fact, it is observed that when the sequential reheater is switched on, a penalty in SFC is introduced. Note that there is a limit for the TR_2 value at each operating condition due to the restriction imposed that $TET_{3_{OD}}$ has to be lower than $TET_{3_{DP}}$. Therefore, the sequential combustor will remain off throughout the whole mission.

FIGURE 5.31: Effect of TR_2 on SFC at each mission flight segment

The scope of the optimisation is to derive relevant conclusions applicable to the normal helicopter operation, which is represented by the reference mission proposed. Therefore, even though the reheater is not required, this mission role is maintained for an initial optimisation analysis. In future analysis, when the NO_x contribution of the reheaters is accounted, the same optimisation can be repeated over the same scenario. If the reheater is still not needed, more atypical conditions (higher altitude and/or cruise speed, and increased TOW) could be employed to derive the potential benefits of the reheating concept at these extreme operating parameters.

5.3.3 Performance assessment at mission level results

Taking into account the considerations described, the performance of both optimal configurations at mission level was finally simulated. The distribution of engine fuel flow and SFC over the mission are presented in Fig. 5.32.

The fuel burn at every flight stage is reduced by the implementation of the optimal engine architectures (Fig. 5.32(a)). Therefore, the optimal engines that were optimised for SFC_{cr} show also fuel savings at the rest of mission segments.

The difference in thermal efficiency between the optimal *CONV* and *SEQ* architectures is better observed in Fig. 5.32(b). The optimal *CONV* engine has slightly better SFC than the optimal *SEQ* configuration at every segment of the mission. To explain that, the variation of the cycle variables (OPR and TET) over the entire mission are plotted in Fig. 5.33. It is highlighted that both optimal configurations have a combination of higher OPR and TET than the baseline

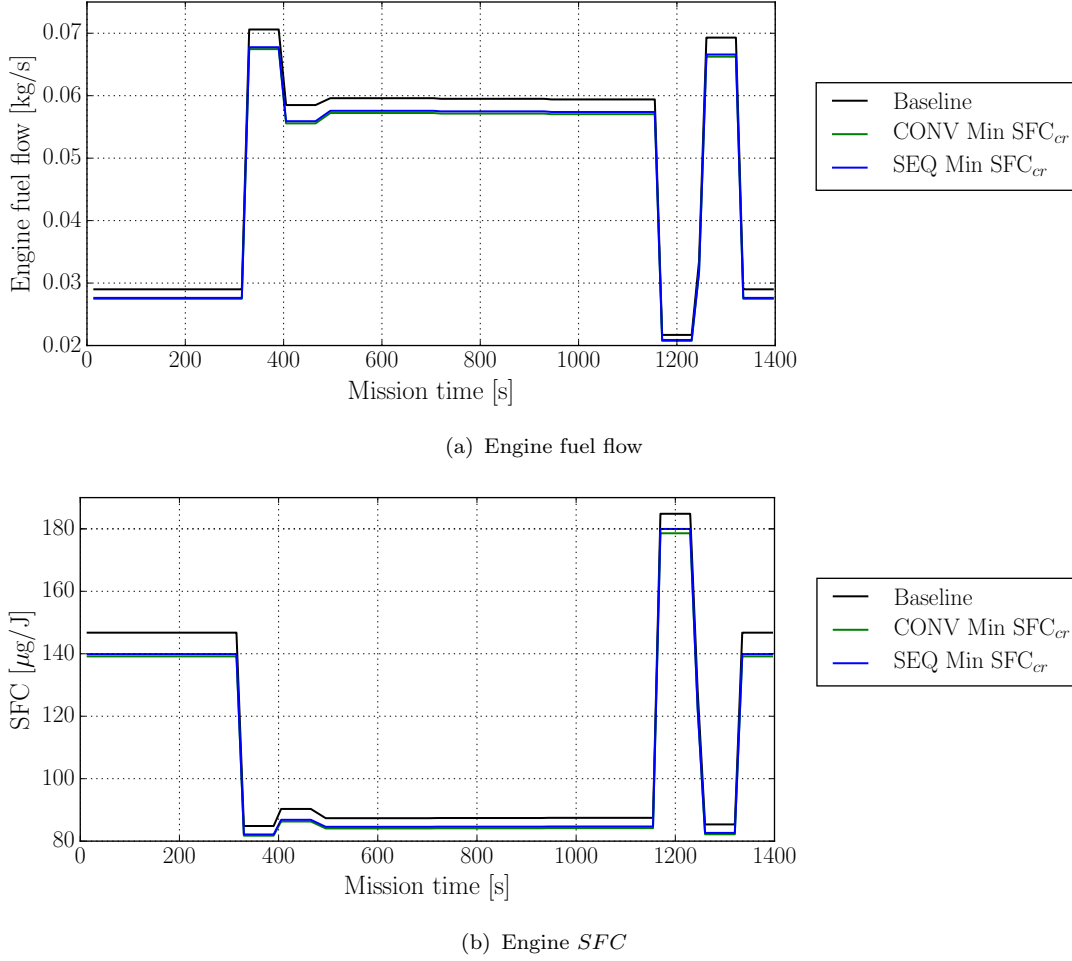


FIGURE 5.32: Optimal engines performance at mission level

engine, that translates into a better thermal efficiency, despite of the increased turbine cooling bleeding. If the optimal configurations are compared, it is observed that the values of OPR are higher for the SEQ engine than for the $CONV$ one, whilst the opposite happens with TET . These differences contribute to the slight variation in mission SFC . However, the contribution of the relative cooling requirement (the optimal $CONV$ engine has higher cooling flows), and the augmented pressure loss in the auxiliary burner of the SEQ architecture have also an important impact in the final thermal efficiency of the optimal engines.

Finally, the total mission fuel corresponding to each engine design is listed in Table 5.15. The trends reproduced in Fig. 5.32 explain the results observed in the table. Both optimal configurations show a fuel consumption improvement compared with the baseline engine and, from the two optimal solutions, the $CONV$ design is the one with the highest fuel reduction (-3.50%).

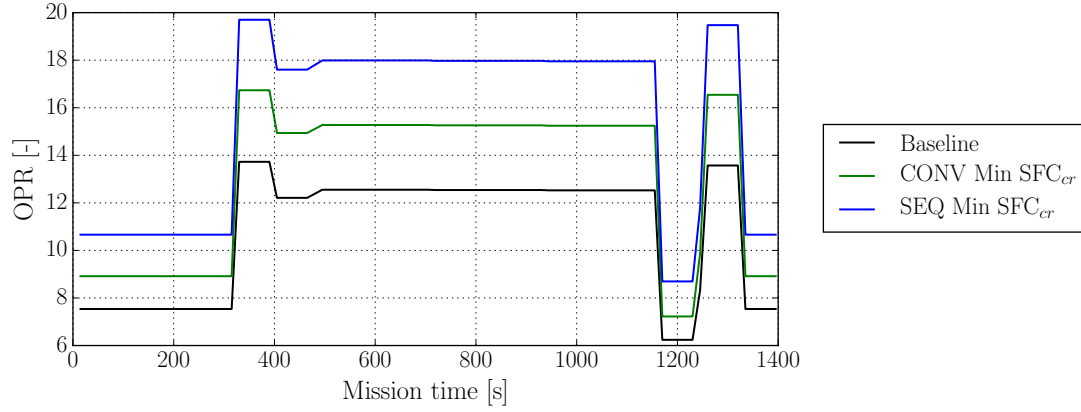
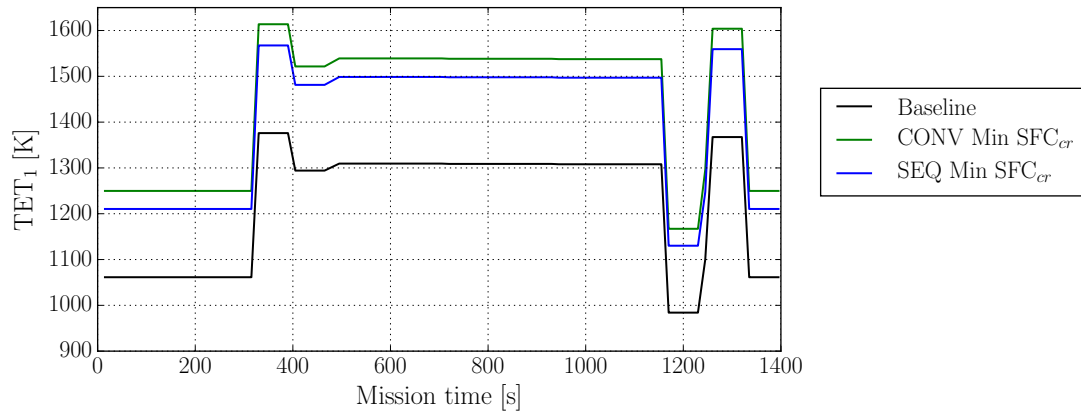
(a) Engine OPR (b) Engine TET_1

FIGURE 5.33: Optimal engines cycle parameters at mission level

TABLE 5.15: Optimal engines results at mission level

	Mission fuel	
	[kg]	% Δ
Baseline	137.13	-
CONV Min SFC_{cr}	132.33	-3.50
SEQ Min SFC_{cr}	133.11	-2.93

Therefore, it can be affirmed that, despite the fact that a mission fuel saving can be attained with the *SEQ* engine, the optimised *CONV* engine is still better than the optimum reheated configuration in terms of fuel and CO_2 emissions abatement. In terms of weight it has been proven that a similar powerplant weight saving is achieved by the two optimal solutions. Thus, the added complexity of the reheated cycle engine shows no benefits, neither in terms of fuel nor engine weight savings, in comparison with the *CONV* cycle engines for this particular application and under the assumptions considered in the analysis. However, the NO_x emissions

of the reheated cycle architectures have to be assessed before any decision about the reheated engines viability can be made. Besides, the engine lifing aspects should be considered, as well, to have the whole picture of the engine performance benefits.

5.3.4 Mission level optimisation for *CONV* engine. W_{fuel} vs. W_{NO_x}

For the *CONV* engine, an additional multi-objective optimisation targeting the minimisation of the overall mission fuel burn (W_{fuel}), and NO_x emissions (W_{NO_x}) is completed. The design variables and boundaries employed in the optimisation (Table 5.16) are the same used in Section 5.1.1 for the parametric analysis of the *CONV* engine.

TABLE 5.16: Design parameters boundaries for *CONV* engine optimisation

Design parameter	Low limit	Baseline value	High limit
$PR_{1_{DP}}$ [–]	4.00	5.32	7.00
$PR_{2_{DP}}$ [–]	2.00	2.80	4.00
$TET_{1_{DP}}$ [K]	1,400	1,440	1,800

The computational time required by *HECTOR* to simulate the flight mission described (40-50 *min*) is large enough to preclude its use to complete the mission optimisation. Instead, it is assumed that the fluctuation in helicopter weight introduced by each engine candidate, *i.e.* the change in powerplant weight and mission fuel, is negligible in comparison with the total mass of the helicopter, and so, the impact on the power required by the helicopter is not relevant. The validity of this assumption is proven by Fig. 5.27 which shows that a fluctuation of 250 *kg* in helicopter mass has an impact lower than 5% in power required.

This assumption implies that, regardless of the engine installed, the power requirement of the helicopter during the mission do not vary. Thus, a unique *HECTOR* computation is required to get the power consumed by the helicopter at each sampling point of the mission. These data were obtained using the baseline engine as powerplant (Fig. 5.34). Note that this mission power pattern is the same as the one shown in Fig. 5.30 for the baseline engine.

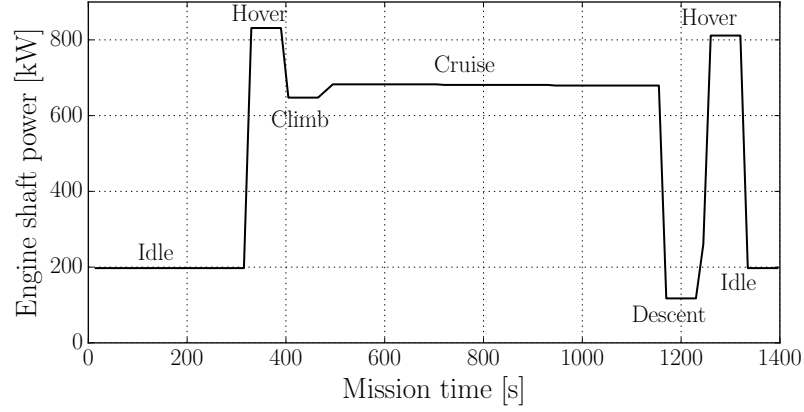
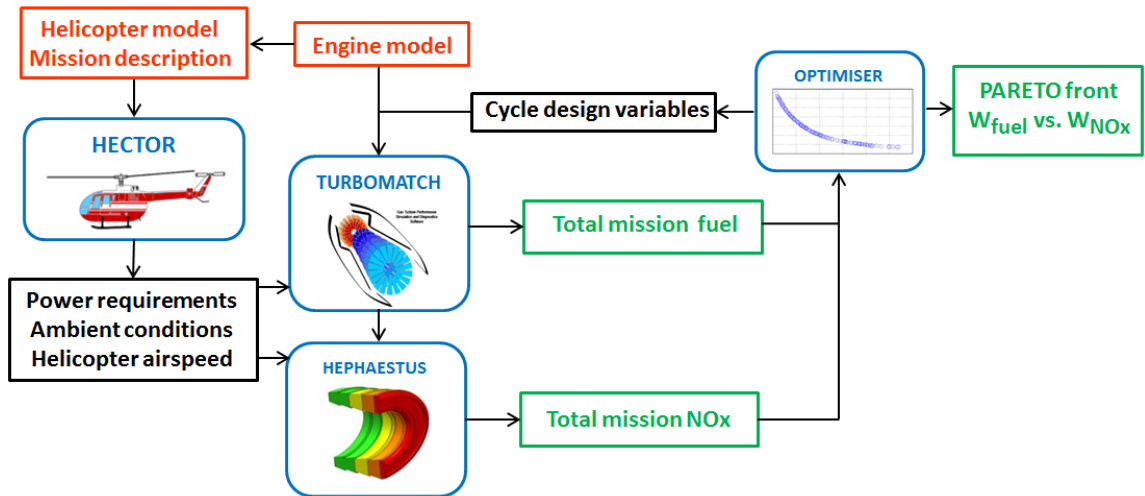


FIGURE 5.34: Engine power vs. time (Baseline engine)

Following this approach, the overall W_{fuel} and W_{NO_x} are computed by simply running TURBOMATCH and HEPHAESTUS at each mission sampling point (engine power, altitude, temperature and helicopter airspeed at each point are known). As a result, the time to evaluate the merits of each engine candidate is now reduced to less than a minute which is low enough to complete the mission optimisation. The whole optimisation process is summarised in Fig. 5.35.

FIGURE 5.35: *CONV* engine optimisation at mission level scheme

The results of the conventional engine multi-objective optimisation at mission level are presented as a Pareto front surface (Fig. 5.36). The plot shows that, within the design space explored, there exist feasible solutions which offer fuel and NO_x emissions savings. The layout of the points in the Pareto front highlights the trade-off between overall mission W_{fuel} and W_{NO_x} , since it is not possible to optimise the powerplant for minimum W_{fuel} and W_{NO_x} at the same time.

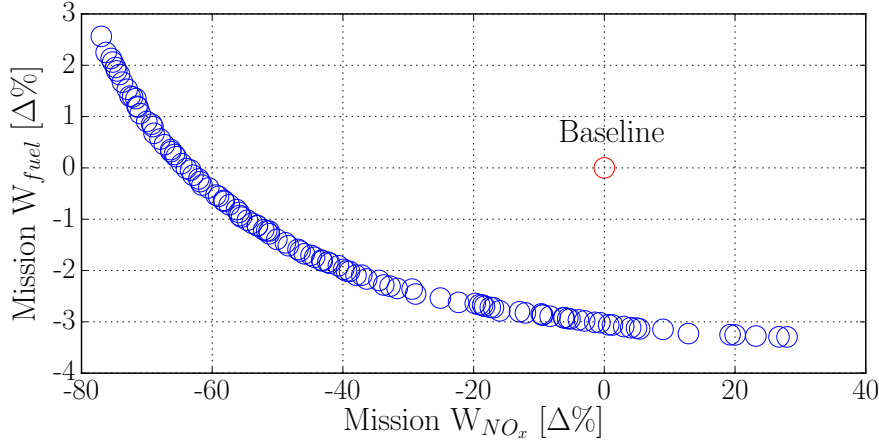


FIGURE 5.36: Pareto front for *CONV* engine optimisation at mission level.
 W_{fuel} vs. W_{NO_x}

From the previous Pareto front, three engine designs are chosen to be representative of the whole optimisation process; namely, the engine with lowest W_{fuel} , the one with lowest W_{NO_x} , and a third engine representing the balance between fuel efficiency and NO_x formation. To select this engine from the Pareto front, a Trade-Off Parameter (*TOP*) is evaluated for each optimal design (Eq. 5.13), and the engine with the lowest *TOP* is chosen.

$$TOP(x, y) = \sqrt{\left(\frac{x - x_{min}}{x_{max} - x_{min}}\right)^2 + \left(\frac{y - y_{min}}{y_{max} - y_{min}}\right)^2} \quad (5.13)$$

Symbols x , y correspond to the fitness parameters (mission W_{eng} and W_{NO_x}) of each optimal design and the subscripts *max* and *min* refer to the maximum and minimum values of the fitness results. The design parameters of the baseline engine and the selected optimal designs are presented in Table 5.17. It is worth to mention that the values of $PR_{1_{DP}}$, $PR_{2_{DP}}$ and $TET_{1_{DP}}$ for the *Min* W_{fuel} engine coincide with the values for minimum SFC_{cr} observed in Fig. 5.1. Likewise the design inputs that minimise mission W_{NO_x} also correspond with the optimal values for EI_{NO_x} at cruise (Fig. 5.2). These results evidence that, for this particular mission, the cruise flight is the most relevant segment, and thus, the one with the biggest impact on the helicopter environmental footprint.

The mission performance of the three optimal candidates is evaluated with *HECTOR*, and compared with the results obtained for the baseline engine. Note that the actual weight of the engine was used to adapt the helicopter *TOW* accordingly. The

TABLE 5.17: Optimal *CONV* engines data at mission level

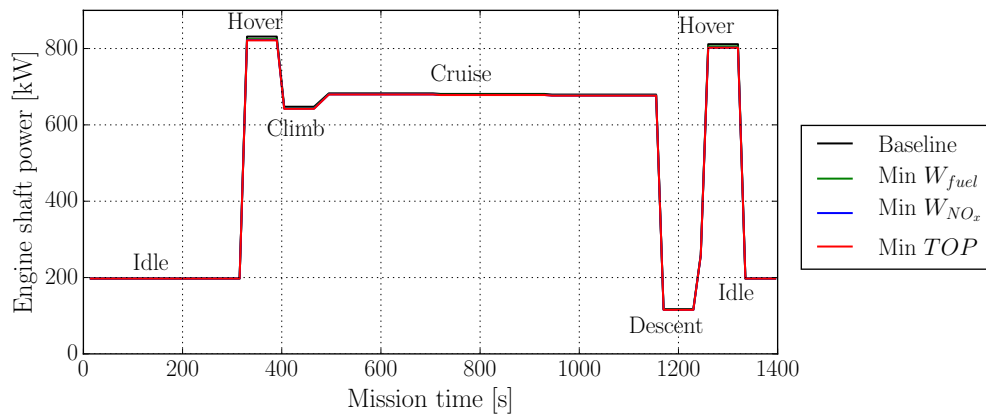
	$PR_{1_{DP}}$ [-]	$PR_{2_{DP}}$ [-]	OPR_{DP} [-]	$TET_{1_{DP}}$ [K]	W_{eng} [kg]
Baseline	5.32	2.80	14.90	1,440	187.81
Min W_{fuel}	5.30	3.30	17.47	1,701	169.13
Min W_{NO_x}	5.01	2.27	11.39	1,789	163.71
Min TOP	4.89	2.92	14.29	1,788	160.57

mission results are presented in Table 5.18 and Fig. 5.37. The $Min W_{fuel}$ solution offers a reduction of 4.4% in W_{fuel} with a penalty in NO_x formation around 25.9%. On the other hand, the $Min W_{NO_x}$ design cuts the NO_x emissions by a 76.7% with a penalty in W_{fuel} of 1.3%. Therefore, a trade-off decision is in place for the engine designer that will have to choose between higher energy efficiency (and less CO_2 formation) or lower NO_x emissions. In this context, the $Min TOP$ design turns out as an interesting solution since it offers a reduction of 2.9% in W_{fuel} and 46.0% in W_{NO_x} , at the same time.

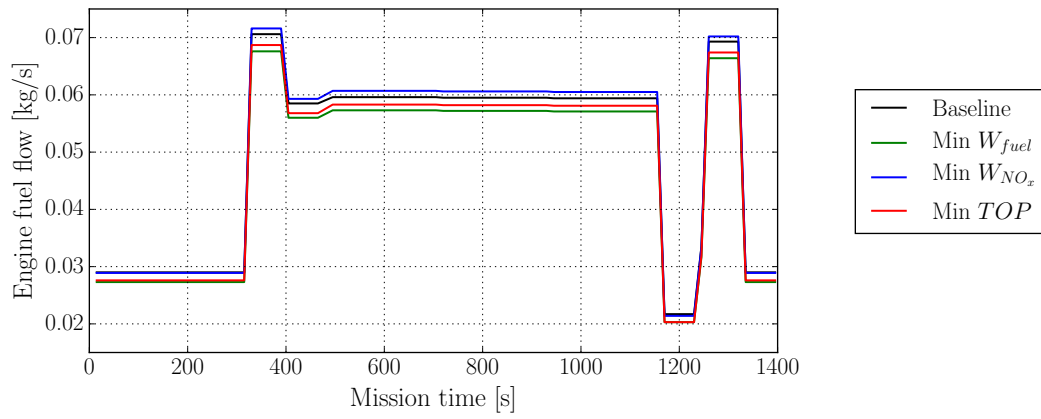
TABLE 5.18: Optimal *CONV* engines results at mission level

	Mission fuel		Mission NO_x	
	[kg]	% Δ	[kg]	% Δ
Baseline	137.13	-	0.722	-
Min W_{fuel}	131.15	-4.36	0.909	25.85
Min NO_x	138.88	1.27	0.168	-76.72
Min TOP	133.17	-2.89	0.390	-46.00

The engine power pattern over the mission presented in Fig. 5.37(a) confirms the adequacy of the assumption of constant power requirements, since the deviation in shaft power between the different engine designs is negligible. The engine fuel flow distribution during the mission is shown in Fig. 5.37(b), and supports the results presented in Table 5.18. The fuel flow of the $Min W_{NO_x}$ (blue line) is higher than the fuel flow of the baseline engine (black line), whilst the fuel flow of the $Min W_{fuel}$ and $Min TOP$ (green and red lines, respectively) are below the trend of the baseline engine; from this two designs the $Min W_{fuel}$ is the one with the lowest fuel flow, as expected. The differences in the NO_x production rate between the distinct engines are clearly observed in Fig. 5.37(c), and also confirm the results in Table 5.18, with the $Min W_{fuel}$ (green line) and the $Min W_{NO_x}$ (blue line) having the highest and lowest NO_x rates, respectively.



(a) Engine shaft power



(b) Engine fuel flow

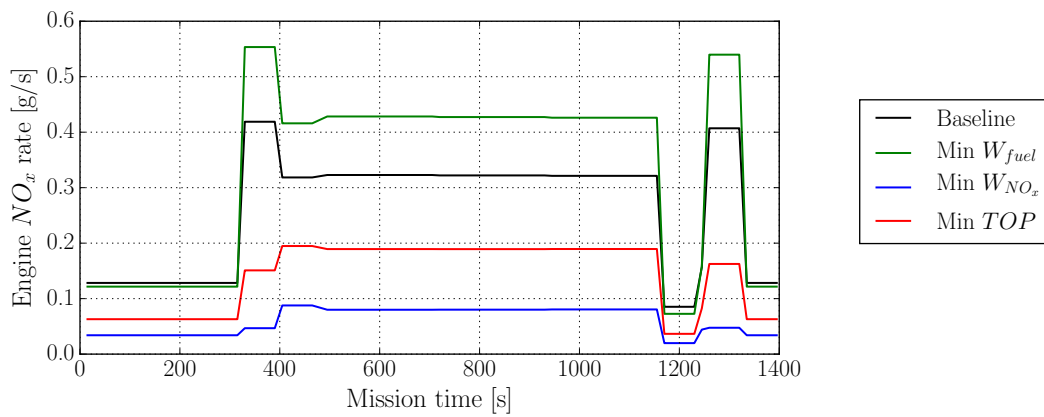
(c) Engine NO_x rate

FIGURE 5.37: Optimal conventional engines performance at mission level

Chapter 6

Conclusions & Future Work

6.1 Summary & conclusions

This final section of the Thesis summarises the work completed by the author to accomplish the set of objectives listed in Chapter 1. To that end, the main points of this work are reviewed and the main findings are highlighted. A list of suggestions for future work is included at the end of the chapter.

6.1.1 Helicopter aerodynamic performance

For the assessment of the conceptual engines performance at mission level a reliable tool that solves the flight mechanics of the helicopter is needed. For that purpose, the tool *HECTOR* was employed. A model after the Sikorsky UH-60 Black Hawk helicopter was built in *HECTOR* to be used as the reference platform. An exhaustive validation of the main rotor frequencies and the power consumed by the helicopter in forward flight was completed successfully with an error in the total power prediction below 4%.

6.1.2 Turboshaft engine performance

For the performance simulation of the conceptual engines the software TURBO-MATCH was utilised. A model after the General Electric T700-GE-700 (the engine installed in the Sikorsky UH-60 helicopter) was first developed to be used as the

starting point to generate a versatile model for the simulation of the *CONV* and the reheated cycle engines performance. A cooling flow sizing module was coupled with TURBOMATCH in order to thoroughly assess the effect of cycle parameters fluctuation on the engine cooling requirements, and its impact on final engine performance. The engine performance model was validated satisfactorily against experimental and simulation data calculated by other performance tools for the General Electric T700 engine.

6.1.3 Turboshift engine weight

The exhaustive weight evaluation of the conceptual engines requires the application of a *CB* method. Therefore, the *CB* weight prediction tool ATLAS was used. However, this software was initially conceived for the weight analysis of turbofan engines, and thus, several upgrades were introduced in the code by the author to cover the analysis of the conceptual turboshaft engines, as well. The mean features added to the original version of ATLAS deals with the following aspects:

- Turboshaft intake sizing
- Turbomachinery disk temperature distribution
- Engine materials library expansion
- Centrifugal compressor weight prediction capability
- Free turbine shaft weight prediction capability
- Auxiliary reheaters sizing

These updates were validated by comparing the weight prediction of ATLAS with the actual weight of two turboshaft engines from different manufacturers and slightly different architectures *i.e.* the General Electric CT7-8A and the Turbomeca Makila 1A1. The geometry of both engines was reasonably well captured and the average error on the weight estimation was below 5%.

Once validated, the tool was used to model the weight of the *CONV* engine and the reheated cycle configurations. As a further validation, the weight prediction of the baseline engine, the General Electric T700, was completed with a 5% error.

However, ATLAS was not used directly in the engine optimisation routine due to its high computational cost. Instead, it was used to generate *RSMs*, through the exhaustive exploration of the design space, that provide a faster weight estimation. This 'hybrid method' was also validated by comparing the weight predictions of ATLAS with the predictions of the *RSMs* developed for each engine configuration.

With the implementation of the mentioned improvements and with the methodology followed to optimise the weight of novel conceptual engines, the author accomplished some of the recommendations suggested by Lolis for future work [54], namely:

- The applicability of the ATLAS method has been expanded to include a wider variety of engines with different characteristics.
- The calculation time of the weight estimation has been reduced by the introduction of a 'hybrid' method. ATLAS results over a design space were used to build a *RSM* employed in the engine optimisation study.
- By the introduction of the reheater sizing routine, the study of a novel engine configuration was possible, namely, the reheated cycle engine. However, some other features such as a propeller, a contra-rotating turbine, an inter-cooler and a recuperator could be added to expand the analysis to other novel engine designs.

6.1.4 Turboshift engine NO_x emissions

The physics-based emissions prediction tool HEPHAESTUS was used for the analysis of the NO_x emissions of the *CONV* engine exclusively. The emissions analysis for the reheated configurations was not attempted since a set of updates are needed to be introduced in the software first. Instead, this task is suggested as future work. The model used in the simulations was built after the combustor installed in the General Electric T700/CT7 engines, and was successfully validated against experimental data. The error incurred in the NO_x prediction within the normal operating range of the engine was below 10%.

6.1.5 Integration of a global engine optimisation tool

The module for the engine performance calculation and the *RSMs* generated for the fast engine weight estimation were finally coupled with an optimiser routine. A multi-objective Genetic Algorithm was used for the simultaneous optimisation of SFC_{cr} and W_{eng} . The trade-offs between these two parameters were highlighted.

The optimiser routine was also used for the multi-objective optimisation of the *CONV* engine at mission level in terms of total W_{fuel} and W_{NO_x} . The trade-off existing between these two parameters was investigated.

6.1.6 Key findings

Firstly, the combined work done in ATLAS and the generation of the weight *RSMs* constitutes the first attempt to create a 'hybrid' model for the rapid weight estimation of turboshaft engines. This model was successfully deployed in the engine optimisation analysis. The results of this assessment showed that, under the assumptions considered in the present study, it is possible to obtain lighter engines than the baseline one. In particular, for the reheated engine configurations, it was found that the penalty in weight and length introduced by the additional burners is largely offset by the reduction in the gas path dimensions, which ultimately translates in weight savings higher than 10%.

With regard to the engine efficiency at cruise, the optimisation proved that it is possible to reduce SFC_{cr} with the appropriate choice of engine design variables for all the engines analysed with the exception of the *ITR + SEQ* engine. However, the *CONV* engine is the one that shows the biggest benefit in thermal efficiency with a reduction of 4% in SFC_{cr} compared with the baseline engine. The increased cooling requirements of the reheated designs added to the pressure losses incurred in the auxiliary burners (specially for the *ITR + SEQ* engine which has two additional combustors) limit the reduction in SFC_{cr} that can be attained with these novel cycle engines. The assessment of the *SEQ* and *CONV* designs with minimum SFC_{cr} at mission level confirmed the superior fuel efficiency of the *CONV* engine.

Due to time restrictions, the NO_x emissions of the reheated cycle architectures were not studied. Therefore, the potential benefits in pollutant depletion that

could justify the implementation of the reheating technology were not quantified. However, NO_x emissions were considered for the *CONV* engine. In fact, a final multi-objective optimisation completed at mission level for the *CONV* showed the possibility to simultaneously reduce total mission fuel and NO_x by 3% and 46%, respectively.

6.2 Future work

Due to the time availability, the amount of work and research that can be completed by the author is limited. Hence, some tasks are listed herein as suggestions for future work. These tasks not only include some upgrades to be introduced in the simulation tools, but also some potential research routes oriented to enhance the performance of turboshaft engines for rotorcraft propulsion applications.

6.2.1 ATLAS code updating

A reasonable amount of work was completed by the author in order to expand the ATLAS weight prediction capabilities to also include free-turbine turboshaft engines. However, there are still some features that must be added to the code to cover all the turboshaft engine design spectrum.

Regarding the routine for the design of the auxiliary burners, it was conceived as the first step towards a more exhaustive sizing analysis. Therefore, a future update of the brick is suggested by the author in order to get a closer prediction of the actual reheaters geometry.

According to these lines, the aspects that could be further improved are listed below:

- **Reverse-flow combustor design and sizing capability**

Helicopter turboshaft engines are very compact machines that often incorporate reverse-flow combustors. As observed in (Fig. 6.1), this device can be installed surrounding the turbines, which allows to shorten the engine shafts. As a result, the length of the engine and its total weight are reduced, and the power/weight ratio of the machine is increased, accordingly.

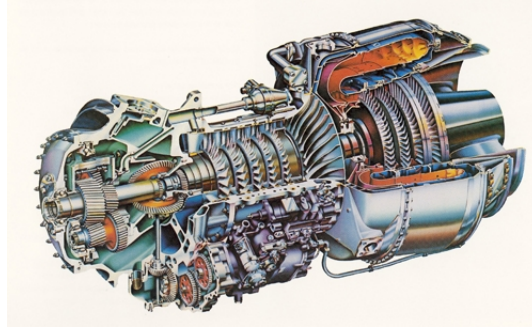


FIGURE 6.1: Lycoming T53 turboshaft layout [117]

Although the basic geometry of the reverse-flow combustor can be already modelled by the existing combustor routine, the calculation of the engine shafts length has to be adapted, and the design of the combustor and the turbines must be linked, since a change in turbines diameter directly affects the radial location of the reverse-flow combustor.

- **Design allowance for the coupling between turbomachinery disks and shafts**

Figure 6.2 shows the actual connection between the turbine disks and the engine shafts in the General Electric T700 engine, and how this connection is modelled in ATLAS. For the *FT* of this particular engine, it is observed that the coupling between this component and the free shaft affects the dimensions of the disks. The assumption of relating the minimum bore disk diameter with the outer shaft diameter is no longer valid herein, as observed in the figure. As a result, the geometry of the disks is not well captured, and a considerable error may be incurred in the weight calculation of the component, since the weight of the disks is the main contributor.

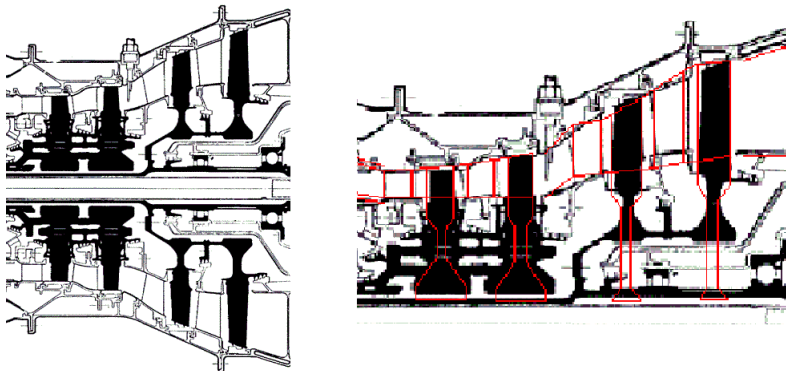


FIGURE 6.2: Detail of the disk-shaft coupling in the General Electric T700 engine [75]

Therefore in a future update of the code, an option could be included to allow the user to specify the minimum disk bore diameter of the turbomachinery component.

- **Improved design of the centrifugal compressor shroud and diffuser**

In the routine developed by the author for the weight calculation of centrifugal compressors, and following Hale's approach [93], the weight of the compressor shroud and diffuser is simply evaluated by means of two empirical expressions. This means that the calculation of the dimensions of these two components is not attempted. Instead, they are left as input variables in the code.

Thus, it would be desirable to introduce a more exhaustive routine (specially for the diffuser) that would adapt the length of this element, and so the total length of the centrifugal compressor, to changes in the thermodynamic state of the gas.

- **Consideration of the stress sources in the blades**

In the current version of ATLAS, uniquely the mechanical integrity of the disks is analysed in detail during the turbomachinery sizing. However, there exist some major sources of stress that directly affect the mechanical integrity of the blades [81]:

- Centrifugal loads produced by the rotating mass of the blade.
- Gas bending moments caused by the change in momentum and pressure of the gas passing across the blade.
- Bending moments acting when the centre of gravity of an aerofoil does not lie on the radial line from the axis of rotation through the centroid of the root section.
- Shear loads caused by the gas pressure and/or the centrifugal untwisting of the blade.
- Thermal loads due to important temperature gradients in the blade.

The combined action of these stress sources may compromise the mechanical integrity of the blades designed by ATLAS. Hence, a routine could be implemented in the code to calculate these stresses, and ensure that the safety limits are not exceeded.

- **Enhance the temperature gradient modelling for centrifugal disks**

In the context of this analysis a linear temperature gradient was assumed in the part of the centrifugal disk which is in contact with the blades (see Fig. 3.16). However, the actual temperature field may be more complex and so, it would be recommendable to investigate the effects of an alternative temperature gradient modelling on the final impeller mass.

- **Automatic definition of the blade material limit temperature**

As explained in Section 3.3.2, the blade material limit temperature is used in the calculation of the disk temperature gradient in some cases. In the latest version of the code, this parameter is manually introduced by the user. However, it would be desirable for the blade material limit temperature to be automatically linked with the blade material selection.

- **Upgrade the design and sizing routine for the reheater**

The inter-turbine reheating is a novel concept that has not been implemented yet in real turbofan or turboshaft engines. Thus, there is no information available about the practical features of inter-turbine reheaters. The simplistic routine that is integrated now in the code constitutes an initial attempt to size the dimensions of these auxiliary burners, but it should be updated once more information about the reheaters layout is available. Apart from this, the weight of other devices that may be integrated within the auxiliary burner such as fuel injectors or flameholders is not included in the analysis. Neither the weight of the fuel injectors is considered in the weight calculation of the main combustor. Therefore, the capability to estimate the weight of such devices must be introduced in the code, as well, for a more rigorous weight analysis.

These recommendations add to the unexplored elements of the list included by Lolis in his PhD Thesis [54].

6.2.2 HEPHAESTUS code updating

The limited time allocated for the project precluded any effort to include the required updates in HEPHAESTUS to examine the contribution of the auxiliary burners to the total NO_x emissions of the engine.

The main change to be introduced in the code to allow the NO_x emissions analysis of the reheated cycle configurations is related with the composition of the reacting air. The current version of the code assumes that the air reacting with the fuel in the combustion chamber is fresh air. This assumption, totally valid for the main burner combustion, is no longer applicable for the additional combustion reactions that take place in the reheated cycle engines. In fact, the reacting air in this auxiliary combustion has a lower content of O_2 which necessarily affects the formation of NO_x . Therefore, a new routine could be added to HEPHAESTUS in order to calculate the composition of the air after the combustion is completed, and then use this information to update the levels of O_2 , CO_2 and H_2O of the air reacting in the subsequent combustions.

6.2.3 *HECTOR* code updating

A simple modification is required in *HECTOR* in order to ease the simulation of the reheated cycle concepts at mission level. This modification consists in the addition of a new selector that allows to choose the desired reheater TR at each flight segment. The current engine performance simulation routine implemented in *HECTOR* allows to modify four parameters at each operating point *i.e.* flight altitude and Mach number, ambient temperature and engine power. Therefore, this routine should be modified in order to include also the value of the TR of each auxiliary burner. Ideally, the TR values to be used by the code at each flight segment (idle, hover, climb, cruise and descent) should be declared as inputs by the user.

In the case that the emissions of NO_x over the mission are to be evaluated, some modifications should be introduced in the routine implemented in *HECTOR* for the computation of pollutant emissions. In the current version of the code, an unique combustor model per engine is expected and so, HEPHAESTUS is only executed once for each mission sampling point. Therefore, for the simulation of the reheated concepts, the routine should be re-designed to ensure that it is able to deal with more than one combustor model, and that it executes HEPHAESTUS as many times as combustors has the engine.

6.2.4 Turboshaft performance analysis and optimisation

Apart from the suggested upgrades in the simulation tools, the performance analysis described in this report could be extended to further explore the potential benefits that can be attained by the application of the reheated cycle principle to rotorcraft turboshaft engines or by the design optimisation of the *CONV* engine. Some possible research routes are listed herein:

- **Evaluate the effect of powerplant weight reduction on the payload-range capabilities of the helicopter**

The savings in powerplant weight attained by the optimal engine designs offer three different helicopter design solutions:

- The empty weight of the helicopter can be reduced accordingly, which translates in a slightly reduction in the power required by the helicopter in flight (see Fig. 5.27).
- The empty weight of the helicopter can be maintained, so the weight saving is used to increase the payload capability of the helicopter.
- The empty weight of the helicopter can be maintained, but instead of increasing its payload capability, the weight saving is invested in a larger fuel capacity. This way, the maximum range of the helicopter is enhanced.

Therefore, a study can be conducted to measure the extend by which the maximum range of the helicopter can be augmented considering not only the extended fuel capacity, but also the increased thermal efficiency of the optimal configurations.

- **Explore the effect of the design assumptions on the engine performance**

A set of assumptions based on constant technology level and technical feasibility was considered to complete the performance analysis of this project. However, the effect of pushing the design boundaries on the engine final performance could be assessed to foresee if the potential benefits derived from the reheated cycle are limited by the technology status. In this context, the impact of improved cooling technologies, reduced pressure losses on the auxiliary burners and/or a larger reheat margin at contingency conditions could be quantified.

- **Include the computation of NO_x emissions in the optimisation of reheated cycle engines**

If the tool HEPHAESTUS is upgraded to allow the calculation of the NO_x formation in the auxiliary reheaters, the production of NO_x could be introduced in the optimisation analysis. This way, all the aspects of the engine performance (fuel consumption, powerplant mass and NO_x emissions) could be optimised simultaneously, which would provide the designer a better picture about the reheated cycle engines viability.

- **Analyse the performance benefits of a reheated-recuperated cycle configuration**

The operation of the auxiliary burners of the reheated cycle engines results in an elevated exhaust gas temperature. This high temperature could be employed to pre-heat the gas before it enters the main combustion chamber, which in turns, would reduce the amount of fuel required to elevate the temperature of the gas to a given TET_1 .

The analysis of the reheated-recuperated cycle would complement the work accomplished by Fakre *et al* on the recuperated engine for rotorcraft propulsion applications [32, 35, 58–61]. One of the challenge of the mentioned analysis is the weight estimation of the required recuperator/heat exchanger (*HE*). For that task, Fakhre *et al* initially made use of the correlation presented in Fig. 2.11, which simply correlates the specific weight of the *HE* with its effectiveness. However, it would be desirable that a sizing routine based on physic principles to be implemented in ATLAS for the *HE* sizing. This is a challenging task since, not only there exist a plethora of *HE* architectures, but also the design of these devices requires a large amount of geometrical and thermodynamic data, that may be not known at the preliminary design phase of the engine.

Appendix A

Cooling flow sizing for a turbine stage

In order to quantify the amount of cooling required by a turbine stage, the cooling flow requirements of the stator and rotor are calculated separately. The procedure for both components is similar except that the latter is carried out with respect to the rotating co-ordinate system [79].

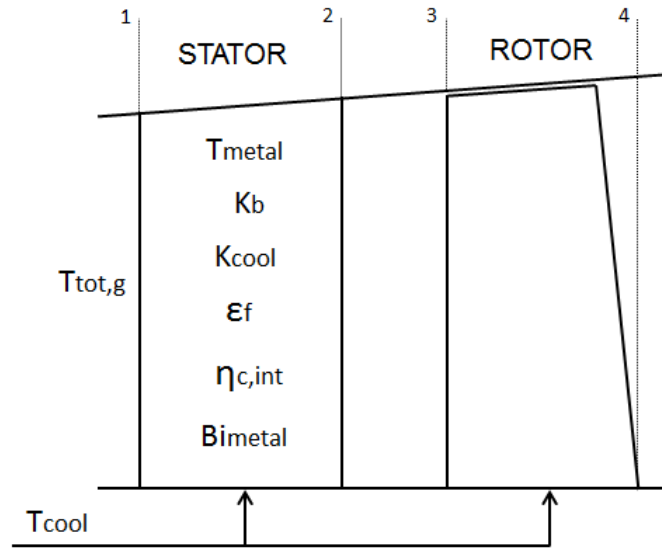


FIGURE A.1: Turbine stage cooling schematic diagram

- **Stator cooling flowrate**

First, the total gas temperature at the stator station inlet, $T_{tot,g}$, is corrected in order to allow for the possibility of local hot spots in the temperature pattern of the flow exiting the burner:

$$T'_{tot,g} = T_{tot,g} + K_{b,sta} \cdot \Delta T_b \quad (A.1)$$

Where ΔT_b is the total temperature rise through the burner. The temperature fluctuations at the outlet of the combustor are assumed to affect only the first turbine stage, so the value of $K_{b,sta}$ is set to zero after the first row of turbine stators.

The value of $T'_{tot,g}$ is compared with the maximum metal temperature of the stator blades, T_{metal} . If the corrected gas temperature is higher than the limit temperature of the material, then the cooling requirements for the stator row are calculated with the following set of expressions:

$$w_{cool}^* = \frac{\epsilon_0 - \epsilon_f + \eta_{c,int} \cdot (1 - \epsilon_0) \cdot \epsilon_f}{\eta_{c,int} \cdot [Bi_{metal} \cdot (\epsilon_f - \epsilon_0) + 1 - \epsilon_0]} \quad (A.2)$$

$$\frac{w_{cool}}{w_g} = K_{cool} \cdot w_{cool}^* \quad (A.3)$$

Where ϵ_0 is the blade cooling effectiveness defined as:

$$\epsilon_0 = \frac{T'_{tot,g} - T_{metal}}{T'_{tot,g} - T_{tot,cool}} \quad (A.4)$$

• Rotor cooling flowrate

For the rotor calculations, the value of the gas total temperature at the rotor inlet must be defined relative to the rotating co-ordinate system. Therefore, the value of the relative total enthalpy of the gas at rotor inlet, h_{rel3} , is first required. This parameter is calculated using the following expression [79]:

$$h_{rel3} = \frac{1}{2 \cdot \psi} \cdot h_{tot3} + \left(1 - \frac{1}{2 \cdot \psi}\right) \cdot h_{tot4} \quad (A.5)$$

Where ψ represents the stage loading coefficient, and h_{tot3} and h_{tot4} correspond to the total enthalpy at the inlet and outlet of the rotor, respectively. Once the relative total enthalpy at the rotor inlet is known, the relative total gas temperature, $T_{rel,g}$, corresponding to that value of enthalpy is finally obtained. This is the gas temperature value to be used in the rotor cooling flowrate calculation. As before, this gas temperature must be first corrected to allow for local hot spots at combustor outlet:

$$T'_{rel,g} = T_{rel,g} + K_{b,rot} \cdot \Delta T_b \quad (\text{A.6})$$

Similar to the stator flowrate calculation, the temperature fluctuations at the outlet of the combustor are assumed to affect only the first rotor row, so the value of $K_{b,rot}$ is set to zero for the rest of turbine stages. The steps followed for the rest of the sizing process are similar to those used for the stator row.

Bibliography

- [1] Air Transport Action Group (ATAG). Facts & Figures, 2017. URL <http://www.atag.org/facts-figures.html>.
- [2] Imperial College London. Green Aviation Research Network. URL <http://www.imperial.ac.uk/green-aviation/about-us/challenge/>.
- [3] Sustainable Aviation CO_2 Roadmap. *Sustainable Aviation*, (December): 1–12, 2008. URL <http://www.nats.aero/wp-content/uploads/2011/06/sa-road-map-final-dec-08.pdf>.
- [4] Lefebvre A. H. and Ballal D. R. *Gas Turbine Combustion: Alternative Fuels and Emissions*. CRC Press, 3rd edition, 2010.
- [5] Clarke J. P. The Role of Advanced Air Traffic Management in Reducing the Impact of Aircraft Noise and Enabling Aviation Growth. *Journal of Air Transport Management*, 9(3):161–165, 2003. ISSN 09696997. doi: 10.1016/S0969-6997(02)00080-7.
- [6] D’Ippolito R., Stevens J., Pachidis V., Berta A., Goulos I., and Rizzi C. A Multidisciplinary Simulation Framework for Optimization of Rotorcraft Operations and Environmental Impact. *2nd International Conference on Engineering Optimization*, (September):1–11, 2010.
- [7] ACARE. Vision 2020, ACARE, What did we achieve?, 2010. URL <http://www.cdti.es/recursos/doc/eventoscdti/aerodays2011/3h1.pdf>.
- [8] Flightpath 2050 Europe’s Vision for Aviation, 2011. URL <http://ec.europa.eu/transport/modes/air/doc/flightpath2050.pdf>.
- [9] FriendCopter Publishable Final Activity Report, 2009. URL https://trimis.ec.europa.eu/sites/default/files/project/documents/20121101_141501_81757_124772131EN6.pdf.

- [10] Novel Innovative Competitive Effective Tilt Rotor Integrated Project. URL <http://www.transport-research.info/project/novel-innovative-competitive-effective-tilt-rotor-integrated-project>.
- [11] Stabellini A. The European Tilt Rotor-Status of ERICA Design and Test Activities. In *Aerodays 2011*, Madrid.
- [12] Clean Sky. Clean Sky at a Glance, 2016. URL <http://www.cleansky.eu/sites/default/files/inline-files/Clean%2520Sky%2520at%2520a%2520Glance%2520FINAL.pdf>.
- [13] Clean Sky. URL <http://cleansky.eu/>.
- [14] European Commission. Clean Sky 2: Developing New Generations of Greener Aircraft, 2014. URL https://ec.europa.eu/research/press/jti/factsheet_cs2-web.pdf.
- [15] Clean Sky 2. Clean Sky 2 Joint Technical Programme. Technical report, Clean Sky 2, 2015. URL http://ec.europa.eu/research/participants/data/ref/h2020/other/guide-appl/jti/h2020-guide-techprog-cleansky-ju_en.pdf.
- [16] Clean Sky 2. Fast Rotorcraft. URL <http://www.cleansky.eu/fast-rotorcraft>.
- [17] Clean Sky 2. Clean Sky 2 Information Day dedicated to the 1st Call for Proposals (CFP01), 2015. URL http://cleansky.eu/sites/default/files/documents/events/20150203/4-{_}cs2{_-}cfp01{_-}info{_-}day{_-}slides{_-}-{_-}frc{_-}lifercraft.pdf.
- [18] Johnson W., Yamauchi G. K., and Watts M. E. Designs and Technology Requirements for Civil Heavy Lift Rotorcraft. In *AHS Vertical Lift Aircraft Design Conference*, pages 1–26, San Francisco, California, 2005.
- [19] Johnson W., Yeo H., and Acree C. W. Performance of Advanced Heavy-Lift, High-Speed Rotorcraft Configurations. *Aeromechanics Branch, NASA*, pages 1–28, 2007.
- [20] Snyder C. A. Exploring Advanced Technology Gas Turbine Engine Design and Performance for the Large Civil Tiltrotor (LCTR). *American Institute of Aeronautics and Astronautics*, pages 1–18, 2014.

- [21] Datta A. and Johnson W. Requirements for a Hydrogen Powered All-Electric Manned Helicopter. In *AIAA Aviation Technology, Integration and Operation Conference*, pages 1–28, Indianapolis, Indiana, 2012. doi: doi:10.2514/6.2012-5405.
- [22] Epstein A. H. Aeropropulsion for Commercial Aviation in the Twenty-First Century and Research Directions Needed. *AIAA Journal*, 52(May):901–911, 2014.
- [23] Sehra A. K. and Whitlow Jr. W. Propulsion and Power for 21st Century Aviation. *Progress in Aerospace Sciences*, 40(May-July):199–235, 2004.
- [24] Cumpsty N. A. Preparing for the Future: Reducing Gas Turbine Environmental Impact-IGTI Scholar Lecture. *Journal of Turbomachinery*, 132: 041017–1–041017–17, 2010.
- [25] Schmidt F. and Staudacher S. Generalized Thermodynamic Assessment of Concepts for Increasing the Efficiency of Civil Aircraft Propulsion Systems. In *Proceedings of ASME Turbo Expo, Montreal, Canada*, 2015.
- [26] Garavello A. and Benini E. Preliminary Study on a Wide-Speed-Range Helicopter Rotor/Turboshaft System. *Journal of Aircraft*, 49(4):1032–1038, 2012. ISSN 0021-8669. doi: 10.2514/1.C031526.
- [27] Padfield G. D. *Helicopter Flight Dynamics*. Blackwell Science, London, 2nd edition, 2007.
- [28] Goulos I., Pachidis V., D’Ippolito R., Stevens J., and Smith C. An Integrated Approach for the Multidisciplinary Design of Optimum Rotorcraft Operations. *Journal of Engineering for Gas Turbines and Power*, 134(9): 349–361, 2012. ISSN 0742-4795. doi: 10.1115/1.4006982.
- [29] Ortiz-Carretero J., Enconniere J., Pachidis V., Stevens J., Smith C., Thevenot L., and D’Ippolito R. Clean Sky Technology Evaluator Green Rotorcraft Mission Environmental Performance Analysis. In *AEGATS 2016*, Paris, 2016.
- [30] Smith C., Pachidis V., Castillo Pardo A., Ezanee G., Stevens J., Thevenot L., and d’Ippolito R. Achieving Rotorcraft Noise and Emissions Reduction for ‘Clean Sky’-The Measurement of Success. In *5th CEAS Air & Space Conference*, number 41, pages 1–14, 2015. ISBN 9781510802568.

-
- [31] Goulos I. *Simulation Framework Development for the Multidisciplinary Optimisation of Rotorcraft*. PhD thesis, Cranfield University, 2012.
- [32] Fakhre A., Tzanidakis K., Goulos I., Pachidis V., and D'Ippolito R. Design Space Exploration and Optimization of Conceptual Rotorcraft Powerplants. *Journal of Engineering for Gas Turbines and Power*, 137(12):121701, 2015. ISSN 0742-4795. doi: 10.1115/1.4031113.
- [33] Goulos I., Fakhre A., Tzanidakis K., Pachidis V., and D'Ippolito R. A Multidisciplinary Approach for the Comprehensive Assessment of Integrated Rotorcraft-Powerplant Systems at Mission Level. *ASME Journal of Engineering for Gas Turbines and Power*, 137(1):012603–1–012603–11, 2015. ISSN 0742-4795. doi: 10.1115/1.4028181.
- [34] Goulos I., Hempert F., Sethi V., Pachidis V., D'Ippolito R., and D'Auria M. Rotorcraft Engine Cycle Optimization at Mission Level. *ASME Journal of Engineering for Gas Turbines and Power*, 135(9):091202–1–091202–10, 2013. ISSN 0742-4795. doi: 10.1115/1.4024870.
- [35] Fakhre A., Tzanidakis K., Goulos I., Pachidis V., and D'Ippolito R. Multi-Objective Optimization of Conceptual Rotorcraft Powerplants: Trade-off Between Rotorcraft Fuel Efficiency and Environmental Impact. *ASME Journal of Engineering for Gas Turbines and Power*, 137(7):071201–1–071201–10, 2015. ISSN 0742-4795. doi: 10.1115/1.4029103.
- [36] Fakhre A. *Design and Optimisation of Rotorcraft Powerplants*. PhD thesis, Cranfield University, 2014.
- [37] Goulos I., Giannakakis P., Pachidis V., and Pilidis P. Mission Performance Simulation of Integrated Helicopter-Engine Systems Using an Aeroelastic Rotor Model. *ASME Journal of Engineering for Gas Turbines and Power*, 135(9):091201–1–091201–11, 2013. doi: 10.1115/1.4024869.
- [38] Performance Prediction and Simulation of Gas Turbine Engine Operation for Aircraft, Marine, Vehicular, and Power Generation. Technical report, Research and Technology Organisation of NATO, 2007.
- [39] NLR GSP. URL <http://www.gspteam.com>.
- [40] GasTurb GmbH. GasTurb. URL <http://www.gasturb.de/index.html>.

- [41] MacMillan W. L. *Development of a Modular Type Computer Program for the Calculation of Gas Turbine Off Design Performance*. PhD thesis, Cranfield University, 1974.
- [42] Nikolaidis T. *The Turbomatch Scheme*. Cranfield University, 2015.
- [43] Pellegrini A., Nikolaidis T., Pachidis V., and Köhler S. On the Performance Simulation of Inter-Stage Turbine Reheat. *Applied Thermal Engineering*, 113:544–553, 2017. ISSN 13594311. doi: 10.1016/j.applthermaleng.2016.10.034.
- [44] Pachidis V., Pilidis P., Marinai L., and Templalexis I. Towards a Full Two Dimensional Gas Turbine Performance Simulator. *Aeronautical Journal*, 111 (1121):433–442, 2007. ISSN 00019240.
- [45] Li Y., Marinai L., Pachidis V., Lo Gatto E., and Pilidis P. Multiple-Point Adaptive Performance Simulation Tuned to Aeroengine Test-Bed Data. *Journal of Propulsion and Power*, 25(3):635–641, 2009. ISSN 0748-4658. doi: 10.2514/1.38823.
- [46] Odgers J. and Kretschmer D. The Prediction of Thermal NO_x in Gas Turbines. *International Journal of Turbo and Jet Engines*, 5(1-4), 1988. ISSN 0334-0082. doi: 10.1515/TJJ.1988.5.1-4.225.
- [47] Rindlisbacher T. Division Aviation Policy and Strategy Guidance on the Determination of Helicopter Emissions. Technical report, Federal Office of Civil Aviation (FOCA), 2009.
- [48] Celis C. *Evaluation and Optimisation of Environmentally Friendly Aircraft Propulsion Systems*. PhD thesis, Cranfield University, 2010.
- [49] Celis C., Moss B., and Pilidis P. Emissions Modelling for the Optimisation of Greener Aircraft Operations. *Proceedings of the Asme Turbo Expo 2009*, pages 167–178, 2009. doi: 10.1115/GT2009-59211.
- [50] Sadanandan R., Lückcrath R., Meier W., and Wahl C. Flame Characteristics and Emissions in Flameless Combustion Under Gas Turbine Relevant Conditions. *Journal of Propulsion and Power*, 27(5):970–980, 2011. ISSN 0748-4658. doi: 10.2514/1.50302.

- [51] Lolis P., Giannakakis P., Sethi V., Jackson A. J. B., and Pilidis P. Evaluation of Aero Gas Turbine Preliminary Weight Estimation Methods. *Aeronautical Journal*, 118(1204):625–641, 2014. ISSN 00019240. doi: 10.1017/S0001924000009404.
- [52] Sagerser D. A., Lieblein S., and Krebs R. P. Empirical Expressions for Estimating Length and Weight of Axial-Flow Components of VTOL powerplants. Technical report, NASA Lewis Research Center, Cleveland, OH, 1971.
- [53] Pera R. J., Onat E., Klees G. W., and Tjonneland E. A Method to Estimate Weight and Dimensions of Aircraft Gas Turbine Engines. Volume 1: Method of Analysis. Technical Report CR-135170, NASA Lewis Research Center, Cleveland, OH, 1977.
- [54] Lolis P. *Development of a Preliminary Weight Estimation Method for Advanced Turbofan Engines*. PhD thesis, Cranfield University, 2014.
- [55] Mourouzidis C. *Cycle Optimisation & Preliminary Design of Very Low Specific Thrust Turbofan Engines*. PhD thesis, Cranfield University, 2016.
- [56] Linares C., Lawson C. P., and Smith H. Multidisciplinary Optimisation Framework For Minimum Rotorcraft Fuel and Air Pollutants at Mission Level. *Aeronautical Journal*, 117(1193):749–767, 2013. ISSN 00019240.
- [57] Olsson A., Sandberg G., and Dahlblom O. On Latin Hypercube Sampling for Structural Reliability Analysis. *Structural Safety*, 25(1):47–68, 2003. ISSN 01674730. doi: 10.1016/S0167-4730(02)00039-5.
- [58] Fakhre A., Goulos I., and Pachidis V. An Integrated Methodology to Assess the Operational and Environmental Performance of a Conceptual Regenerative Helicopter. *The Aeronautical Journal*, 119(1211):67–90, 2015.
- [59] Fakhre A., Goulos I., and Pachidis V. A Preliminary Design Tradeoff Study for an Advanced Propulsion Technology Rotorcraft at Mission Level. *Journal of Engineering for Gas Turbines and Power*, 138(1):012602, 2015. ISSN 0742-4795. doi: 10.1115/1.4031204.

- [60] Fakhre A., Tzanidakis K., Goulos I., and Pachidis V. Multidisciplinary Design and Optimisation of Conceptual Rotorcraft Powerplants for Operational Performance and Environmental Impact. *The Aeronautical Journal*, 119(4215):891–914, 2015.
- [61] Fakhre A., Tzanidakis K., Goulos I., Pachidis V., and D’Ippolito R. Multi-Objective Optimization of a Regenerative Rotorcraft Powerplant: Trade-Off between Overall Engine Weight and Fuel Economy. *Journal of Engineering for Gas Turbines and Power*, 137(12):121201, 2015. ISSN 0742-4795. doi: 10.1115/1.4030634.
- [62] Kailos N. C. Increased Helicopter Capability Through Advanced Powerplant Technology. *Journal of American Helicopter Society*, 12(3):1–15.
- [63] Piqueras P., Gimeno J., Tiseira A., and Gil A. *Ejercicios Resueltos de Motores a Reaccion y Turbinas de Gas*. Universitat Politecnica de Valencia, Valencia, 1st edition, 2011.
- [64] Sirignano W. A. and Liu F. Performance Increases for Gas-Turbine Engines Through Combustion Inside the Turbine. *Journal of Propulsion and Power*, 15(1):111–118, 1999. ISSN 0748-4658. doi: 10.2514/2.5398.
- [65] Liu F. and Sirignano W. A. Turbojet and Turbofan Engine Performance Increases Through Turbine Burners. *Journal of Propulsion and Power*, 17(3):695–705, 2001. ISSN 0748-4658. doi: 10.2514/2.5797.
- [66] Chen G., Hoffman M. A., and Davis R. L. Gas-Turbine Performance Improvements Through the use of Multiple Turbine Interstage Burners. *Journal of Propulsion and Power*, 20(5):828–834, 2004.
- [67] Liew K. H., Urip E., and Yang S. L. Parametric Cycle Analysis of a Turbofan Engine with an Interstage Turbine Burner. *Journal of Propulsion and Power*, 21(3):546–551, 2005.
- [68] Liew K. H., Urip E., Yang S. L., Mattingly J. D., and Marek C. J. Performance Cycle Analysis of Turbofan Engine with Interstage Turbine Burner. *Journal of Propulsion and Power*, 22(No.2), 2006.
- [69] Schobeiri M. T. and Ghoreyshi S. M. UHEGT, The Ultra-High Efficiency Gas Turbine Engine with Stator Internal Combustion. *Proceedings of ASME Turbo Expo, Montreal, Canada*, pages 1–17, 2015.

-
- [70] Igor I. Sikorsky Historical Archives. Sikorsky Product History, 2016. URL [http://www.sikorskyarchives.com/images/S-70A\(UH-60M\)/S-70A\(UH-60M\)-6M.jpg](http://www.sikorskyarchives.com/images/S-70A(UH-60M)/S-70A(UH-60M)-6M.jpg).
- [71] Lockheed Martin. Sikorsky Black Hawk helicopter, 2018. URL <https://lockheedmartin.com/us/products/h-60-black-hawk-helicopter.html>.
- [72] Johnson W. NDARC-NASA Design and Analysis of Rotorcraft Validation and Demonstration. *AHS Aeromechanics Specialists Conference 2010*, (February):804–837, 2010.
- [73] Sikorsky UH-60M Black Hawk Helicopter brochure. URL <http://lockheedmartin.com/content/dam/lockheed/data/ms2/photo/Sikorsky/black-hawk/sikorsky-UH60M-brochure.pdf>.
- [74] Ballin M. G. A High Fidelity Real-Time Simulation of a Small Turboshaft Engine. Technical report, NASA Ames Research Center, 1988.
- [75] Gunston B. *Jane’s Aero-Engines*. 1996.
- [76] General Electric Aircraft Engine Business Group. Model Specification for T700-GE-701 Turboshaft Engine, Part I. Technical report, 1983.
- [77] Pilidis P. and Palmer J. R. *”Gas Turbine Theory and Performance”*. Cranfield University, 2012.
- [78] Walsh P. and Fletcher P. *Gas Turbine Performance*. Blackwell Science, 2nd edition, 2004. ISBN 9780387202488. doi: 10.1007/978-0-387-77674-3.
- [79] Young J. B. and Wilcock R. C. Modeling the Air-Cooled Gas Turbine: Part 2-Coolant Flows and Losses. *Journal of Turbomachinery*, 124(April 2002): 214, 2002. ISSN 0889504X. doi: 10.1115/1.1415038.
- [80] Holland M. J. and Thake T. F. Rotor Blade Cooling in High Pressure Turbines. *Journal of Aircraft*, 17(6):412–418, 1980.
- [81] Cookson R. A. and Haslam A. S. *Mechanical Design of Turbomachinery*. Cranfield University, 2011.
- [82] Holman J. P. *Heat Transfer*. McGraw-Hill, 9th edition, 2002.

- [83] Tong M. T., Halliwell I., and Ghosn L. J. A Computer Code for Gas Turbine Engine Weight and Disk Life Estimation. *Journal of Engineering for Gas Turbines and Power*, 126(2):265, 2004. ISSN 07424795. doi: 10.1115/1.1691980.
- [84] Donachie M. J. and Donachie S. J. *Superalloys. A Technical Guide*. ASM International, 2nd edition, 2002.
- [85] Saravanamuttoo H. I. H., Rogers G. F. C., Cohen H., and Straznicky P. V. *Gas Turbine Theory*. Pearson Education, 6th edition, 2009.
- [86] High Temp Metals. AM 355 Technical Data. URL <http://www.hightempmetals.com/techdata/hitempAM355data.php>.
- [87] Davis J. R. *Heat-Resistant Materials*. ASM International, 1997.
- [88] NDT Resource Center. Powdered Metal and Materials. URL <https://www.nde-ed.org/EducationResources/CommunityCollege/Ultrasonics/Reference%20Information/matproperties.htm>.
- [89] Yaang. Rene 95. URL <https://www.yaang.com/data-center/High-alloy/rene-95.html>.
- [90] AZO Materials. Stainless Steel 355 Alloy (UNS S35500), . URL <https://www.azom.com/article.aspx?ArticleID=6851>.
- [91] AZO Materials. Titanium Alloys - Physical Properties, . URL <https://www.azom.com/article.aspx?ArticleID=1341>.
- [92] Welsch G., Boyer R., and Collings E. W. *Materials Properties Handbook: Titanium Alloys*. ASM International, 1993.
- [93] Hale P. L. A Method to Estimate Weight and Dimensions of Small Aircraft Propulsion Gas Turbine Engines. Technical report, NASA Lewis Research Center, 1982.
- [94] Came P. M. and Robinson C. J. Centrifugal Compressor Design. *Journal of Mechanical Engineering Science*, 213(2):139–155, 1998. ISSN 0954-4062. doi: 10.1243/0954406991522239.
- [95] Onat E. and Klees G. W. A Method to Estimate Weight and Dimensions of Large and Small Gas Turbine Engines. Technical report, NASA Lewis Research Center, 1979.

- [96] Stanitz J. D. Some Theoretical Aerodynamic Investigations of Impellers in Radial and Mixed-flow Centrifugal Compressors. *Transactions of the American Society of Mechanical Engineers*, 74:473–497, 1952.
- [97] Manson S. S. Determination of Elastic Stresses in Gas-Turbine Disks. Technical report, National Advisory Committee for Aeronautics (NACA), TR-871, 1947.
- [98] Tong M. T., Halliwell I., and Ghosn L. J. A Computer Code for Gas Turbine Engine Weight and Disk Life Estimation. *Journal of Engineering for Gas Turbines and Power*, 126(2):265, 2004. ISSN 07424795. doi: 10.1115/1.1691980.
- [99] Armand S. C. Structural Optimization Methodology for Rotating Disks of Aircraft Engines. Technical report, NASA Lewis Research Center, TM-4693, 1995.
- [100] Ray G. S. and Sinha B. K. Computation of Centrifugal Stresses in a Radial-Flow Impeller. *Computers and Structures*, 40(3):731–740, 1991. ISSN 00457949. doi: 10.1016/0045-7949(91)90240-M.
- [101] Tumarkin S. Methods of Stress Calculation in Rotating Disks. Technical report, National Advisory Committee for Aeronautics (NACA), TM-1064, 1944.
- [102] Takeda A. S. *A Centrifugal Compressor Design Assessment for a Small Gas Turbine*. PhD thesis, Cranfield Institute of Technology, 1987.
- [103] Global Security. URL http://www.globalsecurity.org/military/library/policy/army/accp/al0993/fig10{_}1.htm.
- [104] Crisalli A. J. and Parker M. L. Overview of the WR-21 Intercooled Recuperated Gas Turbine Engine System A Modern Engine for a Modern Fleet. In *International Gas Turbine and Aeroengine Congress and Exposition*, Cincinnati, Ohio, USA, 1993.
- [105] Cohen J. D. Analytical Fuel Property Effects - Small Combustors Phase I. Technical report, NASA-CR 168138, NASA Lewis Research Center, 1983.
- [106] Monty J. and Scott T. Analytical Fuel Property Effects - Small Combustors Phase II. Technical report, NASA CR-174848, NASA Lewis Research Center, 1985.

- [107] Fakhre A., Vassilios P., and Pervier H. NO_x Emissions Prediction for GE T700-T6A Type Combustor Using a Physics-Based Multi-Reactor Model. Technical report, Department of Power and Propulsion, Cranfield University.
- [108] Clean Sky (Technology Evaluator). Rotorcraft Turboshift Engine NOx Modelling: Progress Update. Technical report, Clean Sky, 2008.
- [109] Hamade K. S. and Kufeld R. M. Modal Analysis of UH-60A Instrumented Rotor Blades. Technical report, NASA TM 4239, NASA Ames Research Center, 1990.
- [110] Bousman W. G. and Kufeld R. M. UH-60A Airloads Catalog. Technical report, TM-2005-212827, NASA Ames Research Center, 2005.
- [111] European Aviation Safety Agency (EASA). EASA.IM.E.010 General Electric CT7 series engines. Technical report, EASA, Cologne, Germany, 2015. URL <https://www.easa.europa.eu/system/files/dfu/TCDS.IM{ }.E.010{ }issue05{ }20150612{ }1.0.pdf>.
- [112] Jackson A. J. B. *Optimisation of Aero and Industrial Gas Turbine Design for the Environment*. PhD thesis, Cranfield University, 2009.
- [113] Safran. MAKILA 1 & 2. URL <https://www.safran-helicopter-engines.com/sites/turbomeca/files/makila{ }-{ }sale{ }brochure{ }0.pdf>.
- [114] Meier N. Jet Engine Specification Database, 2005. URL <http://www.jet-engine.net/miltsspec.html>.
- [115] Fletcher R. S. and Heywood J. B. A Model for Nitric Oxide Emission from Aircraft Gas Turbine Engines. *9th Aerospace Sciences Meeting*, (71-123), 1971. doi: 10.2514/6.1971-123.
- [116] Sammut C. and Webb G. I. *Encyclopedia of Machine Learning*. Springer, 2010. ISBN 978-0-387-30768-8.
- [117] Connecticut Air & Space Center. Lycoming T53. URL <https://cascstratford.wordpress.com/exhibits/engines/lycoming-t53/>.