

CRANFIELD UNIVERSITY

BERNARD CHUKWUDI TASHIE-LEWIS

CONCEPTUAL DESIGN AND ASSESSMENT OF
TURBOELECTRIC AND HYBRID ELECTRIC PROPULSION
SYSTEM ARCHITECTURES FOR CIVIL TRANSPORT AIRCRAFT

SCHOOL OF AEROSPACE, TRANSPORT AND
MANUFACTURING
Aerospace

PhD

Academic Year: 2013 - 2018

Supervisor: Dr. Panagiotis Laskaridis
Industrial Advisors: Paul Miller & Mark Husband
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This thesis is submitted in partial fulfilment of the requirements for
the degree of Doctorate of Philosophy

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ABSTRACT

To achieve ambitious future environmental targets for aircraft set out by organisations such as NASA and the European Union, turboelectric distributed propulsion (TeDP) has been proposed as a novel concept that has the potential to achieve these targets by significantly improving integrated propulsion-airframe performance. Realising TeDP as a technology option brings into play a number of design and development challenges due to the highly integrated nature of TeDP-airframe configurations, low technology-readiness-levels of key enabling technologies and new modes of operation opened up by shift to a more electric architecture. In tackling these challenges a multidisciplinary and integrated method to assess the benefits and challenges of turboelectric and hybrid-electric propulsion system configurations by considering the effect of aircraft size, mission specifications, airframe, electrical system, energy storage, propulsor architecture and gas turbine architecture was created.

The method created was used in the assessment of turboelectric and hybrid electric performance for a regional transport aircraft and a medium haul transport aircraft. For the regional role the employment of a DC hybrid superconducting turboelectric architecture managed to achieve 16.7% block fuel saving and 3.23% total energy consumption saving over a baseline turboprop aircraft at 600 n.mi range. Driving performance benefits was increased duration of mission time batteries spend discharging at relatively high battery power rating which overcomes weight penalties from installation of electric machinery. For medium haul role the employment of a geared hybrid electric architecture managed to achieve a 3.07% block fuel saving over a baseline turbofan aircraft at 900 n.mi range. Driving performance benefit for the mission was increased battery-operative-cruise time at relatively high battery power rating overcoming aircraft weight penalty and electric machinery installation weight penalty. Despite fuel burn reduction, hybrid electric aircraft consumes more energy than a baseline configuration primarily due to utilisation of additional energy from battery pack.

Keywords:

DC, TeDP, HeP, HeDP, HTS, Superconducting, Hydrogen, Battery, Energy Storage, Propulsor, Propeller, Gas Turbine, CADI, Ducted Fan

ACKNOWLEDGEMENTS

I would like to thank my supervisor Panagiotis Laskaridis for never allowing me to setline on past laurels and pushing me to develop my technical skills to produce good research. He also supported me during the tough moments I did not think it was possible to complete my thesis.

I would like to thank the EPSRC and Rolls-Royce for the funding they provided for the research. Without their contribution none of the work carried out would be possible and I would not have had the opportunity to undertake research in an exciting field.

I would like to thank my industrial advisors Paul Miller and Mark Husband for the guidance and direction they provided in keeping the project on track and the technical input they provided from an industrial perspective.

I would like to thank Dr. Luca Gallo whom at the beginning of my PhD helped me with my learning of Turboelectric Distributed Propulsion technology through providing me with relevant literature. He also supported my research through the provision of numerical tools which I used in carrying out preliminary studies.

I would like to thank Drewan Sanders for his assistance in helping me sanity-check my assumptions and for his aid in numerical tool development.

I would like to thank Dr. Christos Mourouzidis for the technical advice I received from him on the subjects of GT sizing, GT performance assessment and gearbox design. I also appreciate the many philosophical discussions we had the contributed to a more engaging PhD experience.

I would like to thank Alvise Pellegrini for the support I received from him in the utilisation of Turbomatch

I would like to thank the MSc students I worked with whose efforts were directly integrated into my research effort. Florent Grison and Sotirios Bumplinis aided my efforts in propeller sizing, propeller performance assessment and understanding propeller-wing interaction. Additionally the work of Francesc

Fuster Climent and Jaganmohana Rao Danda contributed to my understanding of hybrid electric propulsion systems and boundary layer ingestion.

I would like to thank the NHS, Fran Radcliffe and my counsellor Theresa Townsend whose efforts during the lowest point of my PhD experience when my health deteriorated. They helped improve my well-being and gave me the confidence to push through with research till the end.

I would like to thank my family and friends who have supported me emotionally, financially and physically from a young age to throughout my PhD. I would particularly like to thank my father, Aunt Antonia, Aunt Fidelia, Cynthia and Abimbola Adegbulu.

Last but not the least, I would like to thank God who through all the difficulties I have gone through during life and education gave me the strength to reach the end of my PhD.

TABLE OF CONTENTS

ABSTRACT	i
ACKNOWLEDGEMENTS.....	iii
LIST OF FIGURES.....	ix
LIST OF TABLES	xv
NOMENCLATURE	xix
Abbreviations	xix
Greek Symbols	xxi
Latin Symbols	xxii
Subscripts	xxviii
1 INTRODUCTION.....	1
1.1 Motivation	1
1.2 Research Aim and Objectives.....	2
1.3 Research Contribution	3
1.4 Thesis Structure.....	4
References	6
2 LITERATURE REVIEW	7
2.1 Distributed Propulsion.....	7
2.1.1 Background	7
2.1.2 Distributed Propulsors Driven by a Few Engine Cores.....	9
2.2 Turboelectric Distributed Propulsion Technology.....	14
2.2.1 Superconducting Turboelectric Distributed Propulsion.....	14
2.2.2 Gas Turbine Performance	19
2.2.3 Distributed Propulsor Architecture.....	22
2.2.4 Electrical Systems & Architecture	25
2.2.5 Cooling Systems	33
2.3 Summary	33
References	36
3 METHODOLOGY	43
3.1 Introduction	43
3.2 Mission Role Selection.....	45
3.3 Technologies & Synergies Identification	46
3.4 Synthesis of Propulsion System Architectures.....	51
3.5 Multidisciplinary Modelling	52
3.6 Systems Integration	53
3.6.1 Propulsion System Characterisation	54
3.6.2 Modes of Operation.....	55
3.6.3 Propulsion System – Aircraft Integration	56
3.7 Configuration Assessment.....	58
References	59
4 PROPULSION SYSTEM MODELLING	63

4.1 Propulsion System Architecture Synthesis	63
4.2 Propulsors.....	66
4.2.1 Ducted Fans.....	66
4.2.2 Propellers.....	78
4.2.3 Propulsive Device Efficiency	83
4.3 Transmission	84
4.3.1 AC Turboelectric	86
4.3.2 DC Turboelectric	86
4.3.3 Geared	86
4.3.4 DC Hybrid Turboelectric.....	87
4.3.5 Geared Hybrid Electric	88
4.4 Gas Turbine	88
4.5 Energy Storage.....	90
4.5.1 Fuels	90
4.5.2 Electric Energy Storage.....	90
4.6 Cooling.....	95
4.6.1 Cryocooling	95
4.6.2 Cryogen Cooling	96
4.7 Propeller-Wing Interaction	97
4.8 Integrated Performance Assessment.....	98
4.8.1 Design Point Assessment	99
4.8.2 Off-Design Assessment.....	102
4.8.3 Sizing	114
4.8.4 Weight Estimation	114
References	114
5 AIRCRAFT MODELLING	119
5.1 Introduction	119
5.2 Airframe Sizing	122
5.2.1 Wing.....	122
5.2.2 Fuselage	123
5.2.3 Empennage.....	124
5.3 Aircraft Component Mass Estimation.....	125
5.4 Drag Assessment.....	128
5.4.1 Zero-Lift Drag	128
5.4.2 Lift-Induced Drag.....	130
5.4.3 Wave Drag	131
5.4.4 Trim Drag	131
5.5 Flight Performance.....	132
5.5.1 Main Mission	132
5.5.2 Reserve Mission.....	136
5.5.3 Take-Off	138
5.5.4 Climb	139

5.5.5 Cruise.....	142
5.5.6 Descent.....	144
5.5.7 Hold.....	146
5.5.8 Landing & Approach.....	146
5.6 Cryogenic Fuel Use	146
5.6.1 Modelling at the Tank Level	147
5.6.2 Modelling at the Aircraft Level	150
5.7 Hybrid Electric Flight Performance.....	156
5.7.1 DC Hybrid Turboelectric & Geared Hybrid Electric with Propellers Flight Performance.....	158
5.7.2 Geared Hybrid Electric with Ducted Fans Flight Performance	165
5.7.3 Energy Specific Air Range	168
References	168
6 REGIONAL TRANSPORT	171
6.1 Mission Requirements	171
6.2 Baseline Propulsion System	173
6.2.1 Gas Turbine and Gearbox.....	174
6.2.2 Propellers	179
6.2.3 Integrated Propulsion System Performance.....	182
6.3 Baseline Aircraft.....	184
6.4 Retrofit Turboelectric & Hybrid Electric Aircraft.....	190
6.5 Turboelectric Operation	190
6.5.1 Turboelectric Architecture Technology Levels.....	190
6.5.2 Cooling Requirement.....	192
6.5.3 Turboelectric Propeller Performance.....	194
6.5.4 Propulsion System Mass Breakdown	198
6.5.5 Aircraft Performance	199
6.6 Hybridising the Turboelectric Powerplant.....	207
6.6.1 Electric Architecture Technology Level Consideration	207
6.6.2 Battery Pack.....	207
6.6.3 Propulsion System Mass Breakdown	208
6.6.4 Hybrid Electric Aircraft Performance	209
References	234
7 MEDIUM HAUL TRANSPORT	239
7.1 Mission Requirements	239
7.2 Baseline Propulsion System	241
7.3 Baseline Aircraft.....	246
7.4 Assessing the use of Hydrogen as a Fuel.....	251
7.4.1 Cryotank Performance	251
7.4.2 Propulsion System Performance.....	253
7.4.3 Hydrogen Turbofan Powered Aircraft Sizing and Performance Assessment.....	254

7.4.4 Hybridising the Powerplant.....	262
7.5 The Employment of a Geared Hybrid Electric Architecture.....	267
7.5.1 Retrofit Hybrid Electric Aircraft	268
7.5.2 Transmission Assumptions	268
7.5.3 Battery Pack.....	269
7.5.4 Propulsion System Performance.....	269
7.5.5 Propulsion System Mass Breakdown.....	275
7.5.6 Aircraft Performance	276
7.5.7 Comparison with Works from Public Literature	282
7.6 Configuration Summary	285
7.6.1 Summary at Design Range	285
7.6.2 Summary at 900 n.mi	287
References	289
8 CONCLUSIONS AND FUTURE WORK.....	293
8.1 Conclusions	293
8.1.1 Research Summary.....	293
8.1.2 Regional Transport Performance Concluding Remarks	295
8.1.3 Medium Haul Transport Performance Concluding Remarks	297
8.1.4 Hybrid Electric Operation for Civil Transport Aircraft.....	300
8.1.5 Limitations in Research Undertaken.....	301
8.1.6 Contribution to Knowledge	302
8.2 Future Work	302
APPENDICES	307
Appendix A	307
Appendix B	309
Appendix C	318
Appendix D	324
Appendix E	325

LIST OF FIGURES

Figure 2-1 Early DP concept by Griffith and Rolls-Royce [6]	8
Figure 2-2 ADAM III fighter configuration using gas driven multi-fans propulsion system [8]	10
Figure 2-3 STOL transport using LPC discharge tip-driven multi-fans [9]	10
Figure 2-4 SAX-40 Silent Aircraft Initiative Concept using gear driven three fans with single-engine core propulsion system [12], [13] & [14]	13
Figure 2-5 N3-X aircraft TeDP concept	16
Figure 2-6 Overall engine efficiency as a function of core thermal efficiency, propulsive efficiency and transmission efficiencies [28].....	20
Figure 2-7 Propulsive efficiency for a selection of GT configurations [35]	23
Figure 2-8 Generic TeDP system architecture [11]	26
Figure 2-9 Components in a superconducting TeDP system	26
Figure 2-10 General description of electric power generation, distribution and consumption in a TeDP system [44]	27
Figure 2-11 Candidate Electric Architectures for TeDP (a) DC architecture with protection system comprising of solid-state circuit breakers (SSCB) and SCFCLs (b) Hybrid AC-DC architecture (c) AC-AC architecture [46]	29
Figure 2-12 Block diagram representation of parallel HeP configuration/architecture considered in [51]	31
Figure 3-1 TeDP System Relationships	43
Figure 3-2 Top-Level Method	45
Figure 3-3 Web diagram showing direct interrelationships between TeDP sub-systems	49
Figure 3-4 Systems integration overview for assessment of TeDP & HeP performance at the aircraft level	58
Figure 3-5 Configuration Assessment	59
Figure 4-1 Synthesized PSAs (a) PSA#1 – AC Turboelectric (b) PSA#2 – DC Turboelectric (c) PSA#3 – Geared (d) PSA#5 – DC Hybrid Turboelectric (e) PSA#6 – Geared Hybrid Electric	65
Figure 4-2 Ducted fan schematic.....	66
Figure 4-3 Architectures used for ducted fan validation (a) Turboshaft with ducted fan (b) Separate stream turbofan.....	71
Figure 4-4 Ducted fan tool validation.....	73

Figure 4-5 Fan sizing procedure.....	75
Figure 4-6 Fan sizing procedure for an input angular speed	78
Figure 4-7 Procedure for propeller performance assessment throughout the flight envelope using QMIL and QPROP	80
Figure 4-8 Aerofoil drag polar characterisation for QMIL.....	81
Figure 4-9 Procedure to size propeller utilising QMIL for a specified activity factor.....	82
Figure 4-10 Electrical battery cell model proposed in [24].....	92
Figure 4-11 Design of a battery pack consisting of n_{cells} in series and $m_{modules}$ in parallel from [25].....	94
Figure 4-12 Design Point Assessment Procedure.....	100
Figure 4-13 Design point assessment procedure for a retrofitted ducted fan .	101
Figure 4-14 Off-design assessment procedure for AC Turboelectric architecture	103
Figure 4-15 Off-design assessment procedure for DC turboelectric architecture with ducted fans.....	105
Figure 4-16 Off-design assessment procedure for DC turboelectric architecture with propellers	107
Figure 4-17 Off-design assessment procedure for geared architecture with propellers.....	108
Figure 4-18 Third order polynomial curve fit between SFC and net thrust creation procedure for a given flight condition	109
Figure 4-19 Third order polynomial curve fit between SFC and net thrust creation procedure for a given flight condition, battery power rating and SOC.....	111
Figure 4-20 Third order polynomial curve fit creation procedure for hybrid-electric flight parameters for a given flight condition	113
Figure 5-1 Aircraft design point assessment procedure in CADI.....	122
Figure 5-2 Transport tube-and-wing half-span geometry (shaded area represents stored fuel in the wing).....	123
Figure 5-3 Fuselage sizing	124
Figure 5-4 Flight mission with stepped cruise.....	132
Figure 5-5 Overview of main mission performance assessment procedure ...	135
Figure 5-6 Overview of reserve mission performance assessment procedure	137

Figure 5-7 Vented gas boil-off estimation procedure.....	149
Figure 5-8 Tank with insulation gravimetric efficiency estimation procedure..	150
Figure 5-9 Typical payload-range diagram showing cryotanks sizing point....	151
Figure 5-10 Cryogenic tube-and-wing fuselage.....	152
Figure 5-11 Overview of procedure for sizing and performance assessment of aircraft with cryogenic fuel	156
Figure 5-12 Overview of hybrid electric flight performance assessment procedure for hybrid mission segments of DC hybrid turboelectric architecture and geared hybrid electric architecture with propellers	164
Figure 5-13 Overview of hybrid electric flight performance assessment procedure for hybrid mission segments of geared hybrid electric architecture with ducted fans	167
Figure 6-1 Regional GT configuration	174
Figure 6-2 Minimum induced loss blade geometry.....	180
Figure 6-3 Power coefficient vs advance ratio with contours of efficiency for 2.5° pitch angle intervals at 1020 RPM and sea-level ($\Delta\beta$ is the radially constant pitch angle change)	182
Figure 6-4 Long range cruise baseline propulsion system performance for regional aircraft at selected flight conditions.....	183
Figure 6-5 Comparison of propulsion system performance at cruise between propulsion system operating in long range cruise mode and high speed cruise mode	183
Figure 6-6 Payload-range diagram for baseline regional aircraft; digitised data obtained from [2].....	186
Figure 6-7 Geometry visualisation of baseline regional aircraft using OpenVSP	187
Figure 6-8 Cryocooler power requirements	193
Figure 6-9 Baseline and turboelectric running lines on propeller performance map at cruise flight condition for long range cruise mission (propeller pitch angles specified at 2.5° intervals)	196
Figure 6-10 Propeller efficiency vs thrust for long range cruise mission at cruise flight condition.....	196
Figure 6-11 Baseline and turboelectric powerplant TSFC performance for long range cruise mission at selected flight conditions for regional transport mission	198

Figure 6-12 Regional transport TET variation with cruise time elapsed for turboelectric aircraft configurations at design range for long range cruise mission	200
Figure 6-13 Regional transport propeller efficiency variation with time elapsed during cruise for turboelectric aircraft configurations at design range.....	201
Figure 6-14 Regional transport propeller operation at cruise for turboelectric aircraft configurations at design range for long range cruise mission on propeller performance map.....	202
Figure 6-15 Regional transport propulsion system TSFC variation with cruise time elapsed for turboelectric aircraft configurations at the design range	203
Figure 6-16 Effect of aircraft range on block fuel of turboelectric aircraft configurations for long-range cruise mission	205
Figure 6-17 Regional transport TET variation with cruise time elapsed for turboelectric aircraft compared to baseline aircraft at 600 n.mi for high speed cruise mission	206
Figure 6-18 Regional transport block fuel comparison for baseline and turboelectric aircraft for mission types considered at 600 n.mi	206
Figure 6-19 Regional transport block fuel comparison for baseline and turboelectric aircraft for mission types considered at 200 n.mi	206
Figure 6-20 Battery discharge efficiency versus state-of-charge.....	208
Figure 6-21 Battery SOC vs cruise time elapsed for different battery power ratings at the design range for batteries that weigh 500 kg and have specific energy of 1500 Wh/kg for regional transport.....	211
Figure 6-22 Energy Specific Air Range vs cruise time elapsed for different battery power ratings at the design range for batteries that weigh 500 kg and have specific energy of 1500 Wh/kg for regional transport.....	212
Figure 6-23 Battery SOC vs cruise time elapsed for different battery specific energies at the design range for batteries that weigh 500 kg and have power rating of 500 kW for regional transport.....	214
Figure 6-24 Energy Specific Air Range vs cruise time elapsed for different battery specific energies at the design range for batteries that weigh 500 kg and have power rating of 500 kW for regional transport.....	216
Figure 6-25 Battery SOC vs cruise time elapsed for different battery masses at the design range for batteries with power rating of 500 kW that have a specific energy of 1500 Wh/kg for regional transport.....	218
Figure 6-26 Energy Specific Air Range vs cruise time elapsed for different battery masses at the design range for batteries with power rating of 500 kW that have a specific energy of 1500 Wh/kg for regional transport.....	219

Figure 6-27 Power degree-of-hybridisation versus battery power rating	220
Figure 6-28 Deviation in block fuel from the baseline aircraft for hybrid electric aircraft configurations versus power hybridisation for a range of battery masses at the design range for 1500 Wh/kg batteries.....	221
Figure 6-29 Deviation in main mission total energy consumed from baseline aircraft for hybrid electric aircraft configurations versus power hybridisation for a range of battery masses at the design range for 1500 Wh/kg batteries	222
Figure 6-30 Deviation in block fuel from the baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 600 n.mi for 1500 Wh/kg batteries.....	223
Figure 6-31 Deviation in main mission total energy consumed from baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 600 n.mi for 1500 Wh/kg batteries	224
Figure 6-32 Deviation in block fuel from the baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 200 n.mi for 1500 Wh/kg batteries.....	225
Figure 6-33 Deviation in main mission total energy consumed from baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 200 n.mi for 1500 Wh/kg batteries	226
Figure 7-1 Medium Haul Turbofan Configuration	242
Figure 7-2 Medium haul baseline propulsion system performance at selected flight conditions.....	245
Figure 7-3 Payload-range diagram for baseline medium haul aircraft; digitised data obtained from [1].....	248
Figure 7-4 Geometry visualisation of baseline medium haul aircraft using OpenVSP.....	249
Figure 7-5 Insulated tank gravimetric efficiency versus number of LRMLI layers	252
Figure 7-6 Boil-off from cryotank versus number of LRMLI layers.....	252
Figure 7-7 Comparison of hydrogen turbofan performance to kerosene baseline for medium haul aircraft at selected flight conditions	254
Figure 7-8 Aircraft geometry comparison between baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2 using OpenVSP.....	255
Figure 7-9 Aerodynamic efficiency versus cruise time elapsed for baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2	256

Figure 7-10 Overall thrust requirement versus cruise time elapsed for baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2	257
Figure 7-11 TET versus cruise time elapsed for baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2.....	258
Figure 7-12 Deviation in main mission total energy consumption from baseline for hydrogen aircraft at different fuselage structural penalties	262
Figure 7-13 Medium haul transport deviation in block fuel from the baseline aircraft for minimum fuel hybrid electric configurations with 1500 Wh/kg batteries and hydrogen turbofan configurations versus aircraft range	266
Figure 7-14 Medium haul transport deviation in total energy consumed from the baseline aircraft for minimum fuel hybrid electric configurations with 1500 Wh/kg batteries and hydrogen turbofan configurations versus aircraft range	267
Figure 7-15 Comparison of overall system efficiency between geared hybrid electric powerplant at a SOC of 1.0 and baseline powerplant (power supplied to GT by battery is half of battery power rating since it is assumed that two turbofans power the aircraft)	271
Figure 7-16 Zoomed in LPC map for geared turbofan showing hybrid electric running line for 6000 kW battery power rating at SOC = 1.0 at initial cruise flight condition.....	272
Figure 7-17 TSFC versus thrust comparison at initial cruise flight condition between baseline turbofan, geared turbofan with power input from 6000 kW battery and variable nozzle geared turbofan with power input from 6000 kW battery.....	275
Figure 7-18 Deviation in block fuel from baseline aircraft for geared hybrid electric aircraft configurations versus battery power and mass at the design range for 1500 Wh/kg batteries	280
Figure 7-19 Deviation in block fuel from baseline aircraft for geared hybrid electric aircraft configurations versus battery power and mass at 900 n.mi for 1500 Wh/kg batteries.....	281

LIST OF TABLES

Table 2-1 NASA's Technology Goals for Future SFW Vehicles [11]	11
Table 2-2 Integrated Aircraft-Engine comparison [24]	18
Table 2-3 Summary of synergies of decoupled distributed propulsors and disciplines affected	24
Table 2-4 Comparison of mass and efficiencies of candidate architectures at optimal operating points without protection [46]	30
Table 2-5 Electrical machinery comparison	32
Table 4-1 Propulsion system sub-systems technology options	63
Table 4-2 Ducted fan tool validation engine performance parameters	71
Table 4-3 Typical values for K_B	76
Table 4-4 Overview of battery cell parameters taken from [24]	92
Table 6-1 General mission requirements for regional aircraft obtained from public literature	171
Table 6-2 Mission Requirements as a function of cruise type	172
Table 6-3 Characteristics of regional GT obtained from public literature	175
Table 6-4 Supplied data by Rolls-Royce to be used for simulation of GT performance for regional aircraft	176
Table 6-5 Regional GT performance parameters set based on typical values	176
Table 6-6 Comparison of simulated engine with values obtained from public literature of engine with similar architecture	177
Table 6-7 Values of iterated GT performance parameters used to simulate regional GT performance	178
Table 6-8 Propeller characteristics for regional aircraft from obtained public literature	179
Table 6-9 Regional transport aircraft design parameters and characteristics	184
Table 6-10 Baseline aircraft simulation results compared with values obtained for similar aircraft from public literature at the design mission for regional transport	187
Table 6-11 Baseline aircraft weights comparison with values obtained for similar aircraft from public literature	188
Table 6-12 Aircraft top-level performance summary in long-range cruise mode at different aircraft ranges for regional aircraft	188

Table 6-13 High Speed Cruise Mission Top-Level Performance Summary for Regional Aircraft	189
Table 6-14 Non-fuel aircraft masses for baseline regional aircraft	189
Table 6-15 Power densities of electric components in propulsion system.....	191
Table 6-16 Transmission efficiencies of electric components in propulsion system	191
Table 6-17 Turboelectric propulsion system mass breakdown.....	199
Table 6-18 Regional transport aircraft top-level performance summary for turboelectric aircraft configurations at the design range	204
Table 6-19 Utilised battery pack performance parameters.....	207
Table 6-20 Regional transport hybrid turboelectric propulsion system mass breakdown	209
Table 6-21 Regional transport deviation in block fuel and main mission total energy consumption from baseline values for hybrid turboelectric aircraft configurations at the design range for batteries that weigh 500 kg and have specific energy of 1500 Wh/kg.....	213
Table 6-22 Regional transport deviation in block fuel and main mission total energy consumption from baseline values for hybrid turboelectric aircraft configurations at the design range for batteries that weigh 500 kg and have power rating of 500 kW.....	217
Table 6-23 Top-level performance summary for hybrid electric aircraft in long range cruise mode compared to baseline and superconducting turboelectric aircraft at the design range	227
Table 6-24 Top-level performance summary for hybrid electric aircraft in long range cruise mode compared to baseline and superconducting turboelectric aircraft at 600 n.mi.....	228
Table 6-25 Top-level performance summary for hybrid electric aircraft in high speed cruise mode compared to baseline and superconducting turboelectric aircraft at 200 n.mi	230
Table 6-26 Comparison of minimum fuel hybrid electric configuration simulated at design range with results from relevant propeller-powered regional hybrid electric aircraft simulations carried out in public literature.....	232
Table 7-1 General mission requirements for medium haul aircraft.....	239
Table 7-2 Reserve mission requirements	240
Table 7-3 Characteristics of medium haul GT obtained from public literature	242
Table 7-4 Medium haul aircraft gas turbine parameters set based on typical values	243

Table 7-5 Comparison of simulated engine for medium haul transport mission with values obtained from public literature of engine for with similar architecture	243
Table 7-6 Values of iterated GT performance parameters used to simulate medium haul GT performance. Design point flight condition is M0.78/37,000 ft.	244
Table 7-7 Medium haul aircraft design parameters and characteristics	246
Table 7-8 Medium haul baseline aircraft simulation results compared with values obtained for similar aircraft from public literature at the design mission	249
Table 7-9 Aircraft top-level performance summary at different aircraft ranges for medium haul aircraft	250
Table 7-10 Non fuel aircraft masses for baseline medium haul aircraft.....	250
Table 7-11 Mass breakdown of hydrogen aircraft compared to baseline aircraft at the design range	259
Table 7-12 Top-level sizing and performance summary of hydrogen aircraft compared to baseline aircraft at the design range	260
Table 7-13 Medium haul transport electric transmission component power densities and transmission efficiencies for hydrogen DC turboelectric architecture	263
Table 7-14 Medium haul transport hydrogen DC hybrid turboelectric propulsion system mass breakdown	264
Table 7-15 Medium haul transport transmission power densities and efficiencies for geared hybrid electric architecture	269
Table 7-16 LPC Surge TET at cruise flight conditions for 4500 kW and 6000 kW battery power ratings at SOC = 1.0	273
Table 7-17 Bypass and core nozzle area reductions at cruise flight conditions for power intake into geared turbofan from 45000 kW battery and 6000 kW battery to achieve a nominal minimum surge TET of 1000 K	274
Table 7-18 Medium haul transport geared hybrid electric propulsion system mass breakdown compared to baseline	276
Table 7-19 Block fuel and total energy consumption for hybrid electric aircraft operating in a (1) “ideal” scenario and (2) scenario with operational penalties accounted for. Battery mass for hybrid electric aircraft is fixed at 6000 kg and battery power rating is fixed at 6000 kW	278
Table 7-20 Comparison of minimum fuel hybrid electric configuration simulated at 900 n.mi economic mission with results from similar configurations assessed in public literature	284

Table 7-21 Medium haul transport top-level performance summary for aircraft configurations considered at the design range	286
Table 7-22 Medium haul transport top-level performance summary for aircraft configurations considered at 900 n.mi	288

NOMENCLATURE

Abbreviations

AC	Alternate Current
ADAM	Air-Deflection and Modulation
ADP	Aerodynamic Design Point
AF	Activity Factor
BFL	Balanced Field Length
BL	Boundary Layer
BLI	Boundary Layer Ingestion
BPR	Bypass Ratio
BSCCO	Bismuth Strontium Calcium Copper Oxide
BWB	Blended Wing Body
CADI	Conceptual Aircraft Design Integration
CAEP	Committee on Aviation Environmental Protection
CAS	Calibrated Airspeed
CB	Circuit Breaker
CESTOL	Cruise Efficient Short Take-Off and Landing
CO ₂	Carbon Dioxide
CTE	Conventional Turboelectric
DC	Direct Current
DP	Distributed Propulsion
eBPR	Effective Bypass Ratio
eDP	Electric Distributed Propulsion
EIS	Entry Into Service
EoR	End-of-Runway
ESAR	Energy Specific Air Range
FPR	Fan Pressure Ratio
GT	Gas Turbine
HALE	High Altitude Long Endurance
HeDP	Hybrid Electric Distributed Propulsion
HeP	Hybrid Electric Propulsion
HSC	High Speed Cruise
HTS	High Temperature Superconducting

IBF	Internally Blown Flap
IOC	Initial Operating Capability
ISA	International Standard Atmosphere
LFL	Landing Field Length
LH ₂	Liquid Hydrogen
LHV	Lower Heating Value
LP	Low Pressure
LPC	Low Pressure Compressor
LRC	Long Range Cruise
LRMLI	Load-Responsive Multilayer Insulation
LTO	Long Term Operation
MgB ₂	Magnesium Diboride
MLW	Maximum Landing Weight
MLM	Maximum Landing Mass
MTOM	Maximum Take-Off Mass
MTOW	Maximum Take-Off Weight
MZFW	Maximum Zero Fuel Weight
MZFM	Maximum Zero Fuel Mass
NO _x	Nitrogen Oxides
OEI	One-Engine-Inoperative
OEM	Operating Empty Mass
OEW	Operating Empty Weight
OPR	Overall Pressure Ratio
PMAD	Power Management and Distribution
PR	Pressure Ratio
PSA	Propulsion System Architecture
SAR	Specific Air Range
SCFCL	Superconducting Fault Current Limiter
SFC	Specific Fuel Consumption
SFW	Subsonic Fixed Wing
SL	Sea-Level
SSCB	Solid-State Circuit Breaker
STE	Superconducting Turboelectric

STOL	Short Take-off and Landing
T&W	Tube-and-Wing
TAS	True Air Speed
TeDP	Turboelectric Distributed Propulsion
TET	Turbine Entry Temperature
TOFL	Take-Off Field Length
TOGW	Take-Off Gross Weight
TOM	Take-Off Mass
TOW	Take-Off Weight
TRL	Technology-Readiness-Level
TS	Thrust Split
TSFC	Thrust Specific Fuel Consumption
TSPC	Thrust Specific Power Consumption
UAV	Unmanned Aerial Vehicle
UCAP	Unmanned Aerial Vehicle Common Aircraft Platform
UHBPR	Ultra-High Bypass Ratio
VTOL	Vertical Take-Off and Landing
WAC	Corrected non-dimensional mass flow
YBCO	Yttrium Barium Copper Oxide

Greek Symbols

α	Angle of attack
β	Pitch angle
γ	Ratio of specific heats
$\gamma_{flight,climb}$	Climb flight path angle
$\gamma_{flight,min}$	Minimum flight path angle
ϵ	Layer emittance
η	Efficiency
$\eta_{discharge}$	Discharge efficiency of electric energy storage device
η_{fc}	Cryocooler efficiency as a fraction of Carnot efficiency
$\eta_{grav,clean}$	Clean tank gravimetric efficiency (see Eq. 5-53)
$\eta_{grav,ins}$	Insulated tank gravimetric efficiency (see Eq. 5-54)

η_{PD}	Propulsive device efficiency (see Eq. 4-40)
$\eta_{recharge}$	Recharge efficiency of electric energy storage device
K_A	Korn factor
$\Lambda_{0.25}$	Quarter chord sweep angle
$\Lambda_{0.5}$	Half chord sweep angle
Λ_{LE}	Leading edge sweep angle
μ	Dynamic viscosity
ρ	Density
ρ_{energy}	Specific energy
ω	Angular speed

Latin Symbols

a	Speed of sound
A	Area
$A_{i \rightarrow n}$	Constants for determination of gas thermodynamic properties
AR	Aspect ratio
AR_{eff}	Effective aspect ratio
A_{surf}	Surface area
b	Wing span (see Eq. 5-2)
B_1	First balanced field length coefficient
B_2	Second balanced field length coefficient
b_{CS}	Wing centre section span
bf	Boil-off fraction
b_{fuel}	Fuel tank span
b_{kink}	Wing centre section to kink distance
C_2	Fuselage mass coefficient
C_3	Powerplant installation factor
C_{3E}	Electric machinery installation factor
C_4	Systems factor
C_{batt}	Battery pack capacity (see Eq. 4-58)
C_d	2D drag coefficient
C_{d0}	2D zero lift drag coefficient

C_{D0}	3D zero lift drag coefficient
C_{d2}	2D drag coefficient function
C_f	Skin friction coefficient (see Eq. 5-18)
C_g	Interstitial gas coefficient
C_{HT}	Horizontal tail volume coefficient
C_l	2D lift coefficient
C_L	3D lift coefficient
$C_{l,Cd0}$	2D lift coefficient for zero lift drag
$C_{L,climb}$	3D lift coefficient at 1.2 times stall speed
C_{l0}	2D lift coefficient at zero angle of attack
$C_{l\alpha}$	2D lift curve slope
C_P	Isobaric specific heat capacity
C_{t1}	1 st tank insulation material coefficient
C_{t2}	2 nd tank insulation material coefficient
C_{t3}	3 rd tank insulation material coefficient
C_{VT}	Vertical tail volume coefficient
C_X	Nozzle thrust coefficient (see Eq. 4-15)
d	Diameter
DPL	Duct pressure loss
e	Oswald span efficiency factor (see Eq. 5-25 & Eq. 5-26)
E	Energy
$ESAR$	Energy Specific Air Range (see Eq. 5-65)
$etadisA$	Third order polynomial curve fitting coefficient for electric energy storage discharge efficiency as a function of state-of-charge
$etadisB$	Third order polynomial curve fitting coefficient for electric energy storage discharge efficiency as a function of state-of-charge
$etadisC$	Third order polynomial curve fitting coefficient for electric energy storage discharge efficiency as a function of state-of-charge
$etadisD$	Third order polynomial curve fitting coefficient for electric energy storage discharge efficiency as a function of state-of-charge
$etaPDA$	Third order polynomial curve fitting coefficient for propulsive device efficiency as a function of propulsor thrust
$etaPDB$	Third order polynomial curve fitting coefficient for propulsive device efficiency as a function of propulsor thrust
$etaPDC$	Third order polynomial curve fitting coefficient for propulsive device efficiency as a function of propulsor thrust

η_{PDD}	Third order polynomial curve fitting coefficient for propulsive device efficiency as a function of propulsor thrust
F	Thrust
F_{cryo}	Cryogenic fuselage structural factor
F_D	Drag
FF	Form Factor
$fflowA$	Third order polynomial curve fitting coefficient for fuel flow rate as a function of GT shaft power output
$fflowB$	Third order polynomial curve fitting coefficient for fuel flow rate as a function of GT shaft power output
$fflowC$	Third order polynomial curve fitting coefficient for fuel flow rate as a function of GT shaft power output
$fflowD$	Third order polynomial curve fitting coefficient for fuel flow rate as a function of GT shaft power output
$FGTA$	Third order polynomial curve fitting coefficient for gas turbine thrust as a function of GT shaft power output
$FGTB$	Third order polynomial curve fitting coefficient for gas turbine thrust as a function of GT shaft power output
$FGTC$	Third order polynomial curve fitting coefficient for gas turbine thrust as a function of GT shaft power output
$FGTD$	Third order polynomial curve fitting coefficient for gas turbine thrust as a function of GT shaft power output
F_L	Lift
$F_{M,fus}$	Fuselage material factor
$F_{M,tail}$	Tail/empennage material factor
F_{OP}	Operating items factor
F_{propA}	Third order polynomial curve fitting coefficient for propulsor thrust as a function of propulsor shaft power
F_{propB}	Third order polynomial curve fitting coefficient for propulsor thrust as a function of propulsor shaft power
F_{propC}	Third order polynomial curve fitting coefficient for propulsor thrust as a function of propulsor shaft power
F_{propD}	Third order polynomial curve fitting coefficient for propulsor thrust as a function of propulsor shaft power
F_T	Entropy polynomial temperature function
$F_{takeoff,static}$	Take-off static thrust
F_W	Weight
g	Gravitational acceleration
GR	Gear ratio

H	Specific enthalpy
H_E	Degree-of-hybridisation for energy (see Eq. 3-2)
h_{fus}	Fuselage height
$h_{obstacle}$	Obstacle height in feet
H_P	Degree-of-hybridisation for power (see Eq. 3-3)
H_T	Total specific enthalpy
htr	Hub-to-tip ratio
I	Current
l_{arm}	Tail moment arm
IPR	Inlet Pressure Recovery
K	Drag-due-to-lift factor (see Eq. 5-24)
K_1	Propeller mass curve fitting coefficient
k_{12}	Vertical tail mass factor
K_A	Acceleration factor (see Eq. 5-35)
K_B	Blockage factor
L	Length
L_{fus}	Fuselage length
m	Mass
M	Mach number
MAC	Mean Aerodynamic Chord
m_{cryo}	Cryocooler mass (see Eq. 4-63)
m_{elec}	Mass of electric transmission plus cryocooler
m_{ENG}	Basic engine mass
MFR	Mass flow ratio
$m_{fuel,boil,vent}$	Vented gas boil-off (see Eq. 5-55)
m_H	Horizontal tail mass (see Eq. 5-11)
$m_{modules}$	Number of battery modules (see Eq. 4-55)
m_{OP}	Operational items mass (see Eq. 5-6)
m_{PP}	Powerplant mass (see Eq. 5-13 & Eq. 5-14)
m_{prim}	Mass of primary wing structure
m_{sec}	Mass of secondary wing structure
MSL	Mean stage loading (see Eq. 4-20)
m_{sys}	Equipment, systems and landing gear mass (see Eq. 5-7)

m_{tank}	Tank mass
m_{TO}	Aircraft take-off mass
m_V	Vertical tail mass (see Eq. 5-12)
n	Number of units
n_c	Number of crew
n_{cells}	Number of battery cells (see Eq. 4-54)
n_g	Interstitial gas exponent
n_{layer}	Number of layers
n_p	Number of passengers
n_{poles}	Number of poles
NPR	Nozzle Pressure Ratio
N_t	Insulation layer density in layers/cm
p	Cabin maximum working pressure differential in bar
P	Pressure
PCN	Relative rotational speed
P_{int}	Interstitial gas pressure in torr
P_S	Static pressure
P_T	Total pressure
PW	Power
PW_{CC}	Cryocooler power demand (see Eq. 4-62)
q	Heat flux (see Eq. 5-56)
Q	Compressible flow
Q_H	Heat
$Q_{H,cold}$	Thermal load at cold temperature
Q_{int}	Interference factor
R	Specific gas constant
R_{elec}	Electrical resistance
$R_{elec,i}$	Internal electrical resistance
$R_{elec,series}$	Step response series resistance
$R_{elec,TL}$	Long time step transient resistance
$R_{elec,TS}$	Short time step transient resistance
Re	Reynold's number
Re_{exp}	Reynold's number scaling factor

Re_{ref}	Reference Reynold's number
RoC	Rate-of-climb (see Eq. 5-33)
RoC_0	Rate-of-climb with no acceleration (see Eq. 5-34)
RoC_a	Rate-of-climb with acceleration (see Eq. 5-36)
RoD_0	Rate-of-descent with no acceleration (see Eq. 5-47)
RoD_a	Rate-of-descent with acceleration (see Eq. 5-48)
R_{relief}	Bending relief factor
s	Horizontal distance travelled
S	Wing area (see Eq. 5-1)
$SFCA$	Third order polynomial curve fitting coefficient for SFC as a function of thrust
$SFCB$	Third order polynomial curve fitting coefficient for SFC as a function of thrust
$SFCC$	Third order polynomial curve fitting coefficient for SFC as a function of thrust
$SFCD$	Third order polynomial curve fitting coefficient for SFC as a function of thrust
$SFChybridA_i$	Third order polynomial curve fitting coefficient for hybrid powerplant SFC as a function of thrust
$SFChybridB_i$	Third order polynomial curve fitting coefficient for hybrid powerplant SFC as a function of thrust
$SFChybridC_i$	Third order polynomial curve fitting coefficient for hybrid powerplant SFC as a function of thrust
$SFChybridD_i$	Third order polynomial curve fitting coefficient for hybrid powerplant SFC as a function of thrust
S_H	Horizontal tail area (see Eq. 5-3)
SOC	State-of-charge (see Eq. 4-50)
S_{ref}	Reference area
S_V	Vertical tail area (see Eq. 5-4)
S_{wet}	Wetted area
t	Time
T	Temperature
tc	Thickness-to-chord ratio
T_C	Cold temperature/cold side temperature
T_H	Hot temperature/ambient temperature/hot side temperature
T_S	Static temperature
TS	Thrust split (see Eq. 3-1)

TSA	Third order polynomial curve fitting coefficient for thrust split as a function of thrust
TSB	Third order polynomial curve fitting coefficient for thrust split as a function of thrust
TSC	Third order polynomial curve fitting coefficient for thrust split as a function of thrust
TSD	Third order polynomial curve fitting coefficient for thrust split as a function of thrust
T_T	Total temperature
u	Tank ullage
U	Speed/Velocity
U_D	Dive speed
U_S	Stall speed
V	Voltage
V_{OC}	Open-circuit voltage
Vol	Volume
W	Mass flow rate
W_F	Fuel flow rate
W_{fus}	Fuselage width
Z	Surge Margin

Subscripts

0 or ∞	Freestream
ACDC	AC/DC converter
avg	Average
batt	Battery or Battery Pack
Blade(s)	Blades
cell	Battery cell
clean	Clean wing/clean tank
climb	Climb
components	Components
cr	Critical
cruise	Cruise
DCAC	DC/AC converter

DCDC	DC/DC converter
DD	Drag divergence
descent	Descent
discharge	Discharging/Discharge
DP	Design Point
duct	Duct casing
E	Energy
ES	Electric energy storage
exit	Exit
fan	Fan
fuel	Fuel
fus	Fuselage
GB	Gearbox
gen	Generator
GT	Gas turbine
hold	Hold
HT	Horizontal tail
hub	Hub
i	Section/Station/element number
ideal	Ideal
in	Inlet or Input
ins	Installed
insulation	Insulation
j	Blown section of the wing/jet
main	Main mission
max	Maximum
mean	Mean
min	Minimum
motor	Motor
net	Net
no cryo	Without cryocooler
nom	Nominal
nose	Nose

OD	Off-design
out	Output
P	Power
payload	Payload
peak	Peak
profile	Profile
propeller	Propeller
propulsive	Propulsive
propulsor	Propulsors
rear	Rear
recharge	Recharging/Recharge
req	Required
reserves	Reserve mission
rotor	Rotor
SL	Sea-level
SOC	State-of-charge
Stage	stages
tail	Tail or empennage
tank	Tank
tip	Tip
tot	Total
turbofan	Turbofan
upper	Upper
VT	Vertical tail
Whkg ⁻¹	Watt-hours per kilogram
wing	Wing
with cryo	With cryocooler

1 INTRODUCTION

1.1 Motivation

With rising fuel costs and increased concern about the impacts of greenhouse gas emissions on the environment, research has been extensive in aerospace to develop solutions to mitigate these issues. Distributed Propulsion (DP) is a novel concept that has been proposed as having the potential to offer significant improvements in aircraft performance to achieve ambitious future environmental targets set out by organisations such as the European Union [1] and NASA [2]. The European Union through its Flightpath 2050 vision for the future of European aviation has set out that aviation technologies and procedures available should achieve a 75% reduction in CO₂ emissions, 90% reduction in NO_x emissions, 65% reduction in perceived noise emissions for flying aircraft in 2050 relative to typical state of the art aircraft in 2000 and aircraft movements are to be emission-free when taxiing.

Turboelectric Distributed Propulsion (TeDP) is a special form of DP whereby electrical power is transmitted from one or more gas turbines to a series of distributed propulsors [3], and this architecture allows for unique benefits and technology integration options. Optimal TeDP system architecture is likely to vary for different transport mission roles as a result of differing design drivers and this affects which technologies are feasible to employ with TeDP. For shorter missions flexibility of operation, reduced weight and turnaround times are more important design drivers but for longer missions propulsion system efficiency and aerodynamic efficiency become more important. Investigating the behaviour of the TeDP system, sub-systems and associated synergies will provide insight into how DP system architectures for highly-integrated propulsion airframe design configurations evolve for different aircraft mission roles.

Also, what needs to be considered is that electrification of the propulsion system architecture through the employment of TeDP likely to bring about new modes of operation. For a standard TeDP system architecture such as the one

defined in [4], the presence of electric energy storage opens up the potential for propulsion system architecture hybridisation. This feature allows for even further benefits at the aircraft level via the increase of overall propulsion system efficiency and the potential for reduced CO₂, NO_x and noise emissions. To meet future environmental targets, a TeDP system in synergy with hybrid electric propulsion (HeP) may be needed. Studies detailing requirements and system behaviour for HeP systems would be of importance in characterising practical hybrid electric distributed propulsion (HeDP) system architectures for future aircraft.

1.2 Research Aim and Objectives

A shift from predominantly mechanical architectures used today with gas turbine propulsion systems to more electric architectures such as TeDP opens up a number of synergistic technology options. This architectural shift is likely to result in greater propulsion system complexity and an opening up of the integrated airframe-propulsion design space. To understand better the system interrelationships in TeDP & HeP architectures, the main aim of the research is to assess the performance and feasibility of conceptual integrated aircraft-TeDP-HeP configurations and to understand how the overall system changes for different aircraft mission roles. To achieve this aim the following objectives were set:

1. Development of a multidisciplinary and integrated method to assess the impact of TeDP & HeP sub-system integration on aircraft and propulsion system performance as a function of configuration and mission requirements.
2. Assess suitable propulsion system architectures to identify which architecture potentially performs best for a given civil transport mission.
3. Assess the impact of novel technology integration on overall system performance and architecture
4. Consider the effect of the propulsion system on aircraft design, performance, aerodynamics and structures

1.3 Research Contribution

The following are a selection of issues that arise in the research of DP systems:

- An effective way to account for the performance benefits of a number of synergistic technologies from disparate disciplines opened up by DP on integrated aircraft-propulsion system performance for conceptual design purposes
- Understanding of sub-system operation and how this affects performance at propulsion system and aircraft levels
- Understanding of how the integrated propulsion-airframe design space evolves and changes as a result of an aircraft utilising a TeDP system
- What are the developmental needs for the critical enabling technologies of TeDP to achieve highly ambitious performance targets for the 2035-2050 timeframe?

Taking into account these issues the following research question was proposed:

Research Question

How does the airframe-propulsion system design space open up beyond conventional architectures with the implementation of a TeDP/HeDP system taking into account TeDP/HeDP sub-system interactions and potential new technology synergies?

Scientific contribution

To answer the research question, the following was proposed as scientific contribution:

The development of a method that assesses how aircraft mission operation affects sub-system performance while simultaneously being able to identify how key individual parameters of identified sub-subsystems affect mission operation of potential future electrified propulsion system architectures for civil aerospace. To be taken into account in the method are key characteristics of electric architectures such as advanced power management, advanced energy management, novel off-design control schemes and novel aerodynamic

synergies such as propeller-wing-slipstream lift augmentation. This effectively allows one to examine the physical behaviour of the system from the component level all the way to the aircraft-level and then to see the effects aircraft mission operation has on individual component performance and sub-systems.

Publications

- Modelling of Distributed-Propulsion Low-Speed HALE UAVs Burning Liquid Hydrogen
 - o Main Author: Luca Gallo
 - o Co-Authors: **Bernard Tashie-Lewis**, Panagiotis Laskaridis, Paul Miller, Mark Husband
 - o Conference: SAE 2015 AeroTech Congress & Exhibition
- Simulation and Performance Investigation of Turbo-electric Distributed Propulsion System
 - o Main Author: Danda R. Jaganmohana
 - o Co-Authors: **Bernard Tashie-Lewis**, Sampath Suresh, Panagiotis Laskaridis
 - o Conference: ISABE 2017

1.4 Thesis Structure

The thesis comprises of eight chapters and four appendices. Among the eight chapters there are two modelling chapters (chapter 4 & chapter 5) and two results chapters (chapter 6 and chapter 7). The two results chapters are written as case studies for a specific aircraft role that show how the methodology (chapter 3) has been applied.

- **Chapter 1 – Introduction:** This chapter highlights motivation behind research, gives overview of research aim & objectives, and discusses research contribution
- **Chapter 2 – Literature Review:** A background of distributed propulsion technology and key historical developments of the technology are

discussed together with a review of key aspects of turboelectric distributed technology

- **Chapter 3 – Methodology:** The multidisciplinary and integrated method employed to make a scientific contribution is explained in this chapter along with a brief review of previous approaches
- **Chapter 4 – Propulsion System Modelling:** Details for modelling sub-systems of the propulsion system are presented along with methods for assessing the propulsion system at design point and off-design
- **Chapter 5 – Aircraft Modelling:** Details for modelling at the aircraft level are discussed. Key aspects covered in the chapter are airframe sizing, flight performance assessment, cryogenic fuel use and assessment of hybrid electric flight.
- **Chapter 6 – Regional Transport:** Application of methodology for regional civil transport mission is carried out with turboelectric and hybrid electric operation assessed
- **Chapter 7 – Medium Haul Transport:** Application of methodology for medium haul civil transport mission is carried out. Performance assessment of using hydrogen as a fuel and employment of a geared hybrid electric architecture is undertaken
- **Chapter 8 – Conclusions and Future Work:** Conclusions of research effort are presented and recommendations for future work are made
- **Appendix A:** Thermodynamic gas properties definitions and propeller properties definitions
- **Appendix B:** Assumptions, models and equations used for sizing and mass estimation of epicyclic differential gearbox are contained in this appendix
- **Appendix C:** Assumptions, method and equations used for sizing of a centrifugal compressor are contained within this appendix
- **Appendix D:** Maximum landing weight and maximum zero fuel weight definition
- **Appendix E:** Hybrid electric propulsion definitions at electric energy storage level and propulsion system level contained in this appendix

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- [4] ARMSTRONG, M. J. et al (2012) Propulsion System Component Considerations for NASA N3-X Turboelectric Distributed Propulsion System. *SAE International Journal of Aerospace*, 5 (2), pp. 344-353

2 LITERATURE REVIEW

2.1 Distributed Propulsion

2.1.1 Background

From literature read there are quite a few definitions for DP. A recent one by Gohardani [1] defines DP as an aircraft featuring 3 prime-mover/propulsor units or more, where the propulsor can either be coupled or decoupled to the prime-mover. However, one limitation to this definition is that typical 4-engined aircraft already in service today such as the Boeing 747 (turbofan-powered) and the Airbus A400M (turboprop-powered) would fall under this definition. Whether these aircraft can be considered aircraft with DP is debatable and this is one weakness with the definition proposed in [1]. In fairly recent studies [2], [3], [4] & [5], an emphasis in these studies is the placing of engine units to bring about synergistic performance benefits and new capabilities in integrated airframe-propulsion system performance. As a result novel airframe-propulsion configurations are created with architectures different than what is found in the Boeing 747 and Airbus A400M. An alternative definition to distinguish DP architectures from conventional architectures states that DP is the distribution of airflows and forces generated by the propulsion system about an aircraft in ways to improve the vehicle's aerodynamics, propulsive efficiency, structural efficiency and aeroelasticity [3].

Considering the definitions for DP presented, from literature one of the first concepts of a propulsion system that could be called a DP system is the system designed by Griffith in [6] (see Figure 2-1). In 1954 Griffith offered up a novel design different to the predominantly propeller-powered aircraft in service in those times, in which a plurality of distributed gas turbine units were spaced spanwise along an aircraft with at one master combustion engine unit designed to drive GT units and auxiliary equipment. Additionally, equipment was on board to enable the master unit to be controlled independently of distributed units. Some of the claimed benefits of this design were:

1. Vertical thrust capability through control of master unit independent of slave units
2. The adoption of the smaller distributed units for reduced installation penalties on the wing
3. Differential thrust from distributed engines and main engine units
4. Reduced fuel consumption in cruising conditions when a bypass engine is used as the master unit
5. Possibility for thrust vectoring by the presence of a common mounting structure that is angularly movable with respect to the main portion of the wing structure
6. Lateral control of aeroplane possible by differential adjustment of the fuel supply to the distributed turbine units
7. A blown flap effect of the wing when the flap is lowered because the thrust line of the distributed units is directed downwards with respect to the longitudinal axis of the aircraft to provide a direct lifting force on the aircraft

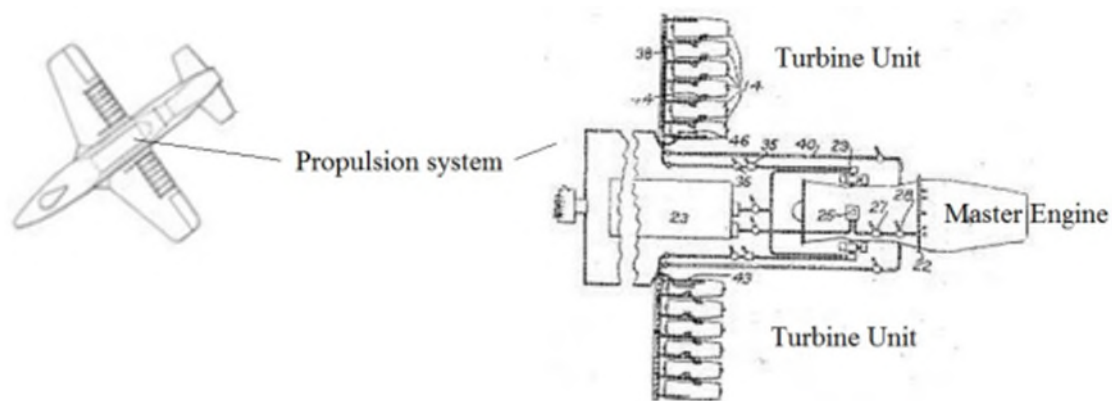


Figure 2-1 Early DP concept by Griffith and Rolls-Royce [6]

Most of these claims have been supported by recent research efforts except for claim 4 which is debatable. From a similar configuration investigated in [2] a series of small GTs distributed spanwise across the leading edge of the wing of a tube-and-wing airframe was considered. In the configuration it was found that for a given thrust requirement, as the number of gas turbines increases so too does the specific fuel consumption (SFC). The SFC during cruise increases by

40% for 10 engines to about 60% for 30 engines. Reasons given for this were mainly due to reduced turbomachinery size leading to early separation and aerodynamic losses, Increased difficulty of combustor scaling, tip clearance increases relative to blade height for a given level of manufacturing technology, and geometric requirements for turbine cooling flow passages not being able to be met leading to lower permissible turbine entry temperatures (TETs) and thermal efficiency. With these limitations in mind, most of the recent work on DP has focused on the feasibility of distributing a number of propulsors driven by a few engine cores rather than distributing small GTs.

2.1.2 Distributed Propulsors Driven by a Few Engine Cores

This form of DP entails a series of propulsors being driven by a few fuel-efficient engine cores and has been investigated for feasibility as part of NASA's Subsonic Fixed Wing (SFW) N+3 project [7]. For this form of DP three main types have been identified from literature and these are:

1. Gas-Driven Propulsors
2. Gear-Driven Propulsors
3. Electrically Driven Propulsors

1. Gas-Driven Propulsors

One of the first concepts of significance that can be classified under this configuration was the Vertical/Short Take-Off and Landing (V/STOL) Air-Deflection and Modulation (ADAM III) fighter concept in the late 1960's (see Figure 2-2). In the concept, gas generators and their inlets were installed near the fuselage to provide hot gas to the wing mounted turbines that drove high bypass ratio turbofans. The hot gases from the gas generators were routed through long ducts across the wing span to the locations where the turbines and fans were installed. The inlets and nozzles for the turbofans and turbines were also all within the wing structure away from the gas generators and provided distributed thrust to the vehicles. The concept was studied for various missions but the design never went into production with the problem of ducting hot gas through the wing structure identified as a potential bottleneck [8].

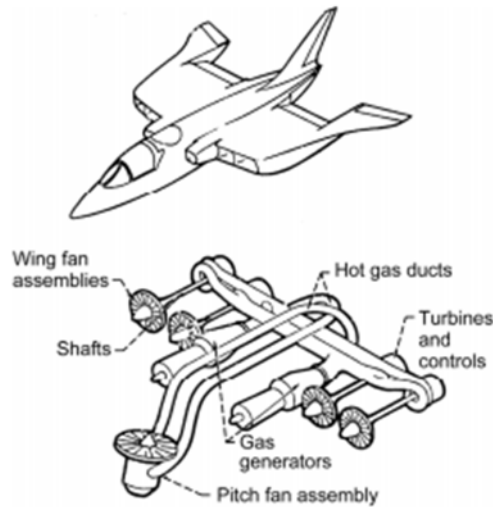


Figure 2-2 ADAM III fighter configuration using gas driven multi-fans propulsion system [8]

Another gas-driven multi-fan configuration was conceived in the 1970's and a working model was tested for STOL transport operation (see Figure 2-3). In this configuration 16 tip-driven fans were powered by high-pressure discharge air from the main gas generators [9]. High-pressure discharge air from the low-pressure compressor (LPC) stages powered 1.25 fan pressure ratio (FPR) tip driven fans mounted on a hinged flap to achieve high lift via supercirculation. The supercirculation effect coupled with a massive suction effect in front of inlets allows for increased lift and the delay of flow separation on the upper surface of the wing.



Figure 2-3 STOL transport using LPC discharge tip-driven multi-fans [9]

2. Gear Driven Propulsors

In more recent times to meet new technology targets set out by organisations such as NASA (see Table 2-1) and the European Union a gear driven DP arrangement has been considered [4] & [10]. The geared transmission enables the use of low pressure bypass diversion air for thrust generation and eliminates the hot ducting problem of gas-driven configurations. From the literature read there was not one gear driven propulsor configuration that utilises distributed propellers, with the few identified all utilising distributed fans. Possible reasons for this are that it would be too complicated mechanically and most of the studies for DP in recent times have been assessed at mission ranges where flight is carried out at transonic Mach numbers for which conventional propeller flight is inefficient.

Table 2-1 NASA's Technology Goals for Future SFW Vehicles [11]

<i>Corners of the trade space</i>	N ^a +1 (2015 EIS) Generation Conventional Tube and Wing (relative to B737/CFM56)	N ^a +2 (2020 IOC) Generation Conventional Blended Wing Body (relative to B777/GE90)	N ^a +3 (2030-2035 EIS) Advanced Aircraft Concepts
Noise (cumulative below Stage 4)	-32 dB	-42 dB	55 LDN at average airport boundary
LTO NO _x emissions below CAEP 6)	-60%	-75%	Better than -75%
Performance: Aircraft Fuel Burn	-33% ^b	-40% ^b	Better than -70%
Performance: Field Length	-33%	-50%	Exploit metroplex ^c concepts

^a“N” represents current state-of-the-art aircraft as stated in parentheses

^bAn additional reduction of 10 percent may be possible through improved operational capability

^cConcepts that enable optimal use of the airports (with shorter runways within the metropolitan areas)

In [4], NASA considered distributed propulsion with 12 small engines for a notional Cruise Efficient Short Take-Off and Landing (CESTOL) transport vehicle. It was mentioned that the small engines could be replaced with a number of high bypass ratio (BPR) multi-fan engines. For these multi-fan engines, one engine core drives three fans with power transmission through a gearing system. The combination of this DP system coupled with a blended-wing-body (BWB) airframe configuration enabled powered lift with an internally blown flap (IBF) for STOL operation; integration of substantial engine noise shielding for noise reduction; and high cruise efficiency through the use of a BWB airframe.

Studies in [10], [12], [13] & [14] employed a similar DP configuration of gear-driven multi-fans for the “Silent Aircraft Initiative” by Cambridge and MIT. Similar to [4] DP was employed with a BWB airframe in the SAX-40 which had three partially embedded multi-fans (see Figure 2-4). In addition to the technologies employed for the reduction of noise in the SAX-40, the use of partially embedded engines combined with the BWB allows for the concept to take advantage of boundary layer ingestion (BLI) for aerodynamic & propulsive efficiency improvement. BLI performance was not extensively studied in the project but from the study in [15], the ingesting of BLI air is likely to impose a distortion penalty on the engine if turbomachinery is not designed to eliminate or minimise it. This penalty can result in up to a 15% drop in engine SFC depending on the engine-airframe configuration. Other limitations of the propulsion system arrangement highlighted in the study included additional duct losses; increased mechanical design complexity; general reliability compared to current commercial aero-engines; increased Low Pressure (LP) system weight; increased weight arising from an oil system required for both cooling and lubrication (critical to mission operation); and developing of a dependable

cooling system for mechanical transmission with acceptably sized heat exchangers.

Another geared configuration was considered in [16] and this time DP was applied to a conventional Tube-and-Wing (T&W) airframe. In the concept propulsive power is supplied to the aircraft via two geared turbofans and one fuselage fan. The fuselage fan was mounted at the rear of the fuselage to take advantage of BLI. For this configuration the fuselage was shaped for minimised distortion at the fan inlet however the potential for tail scrape is increased by reduced ground clearance at the rear-section of the fuselage. Additionally from [16] the design parameter thrust split (TS) was introduced to represent the ratio of thrust produced by the fuselage fan to the total thrust produced by the propulsion system. At an optimal TS of 27%, a 9.9% increase in block specific air range (SAR) over an advanced conventional configuration was found with this being driven by increases in propulsive and aerodynamic efficiency.

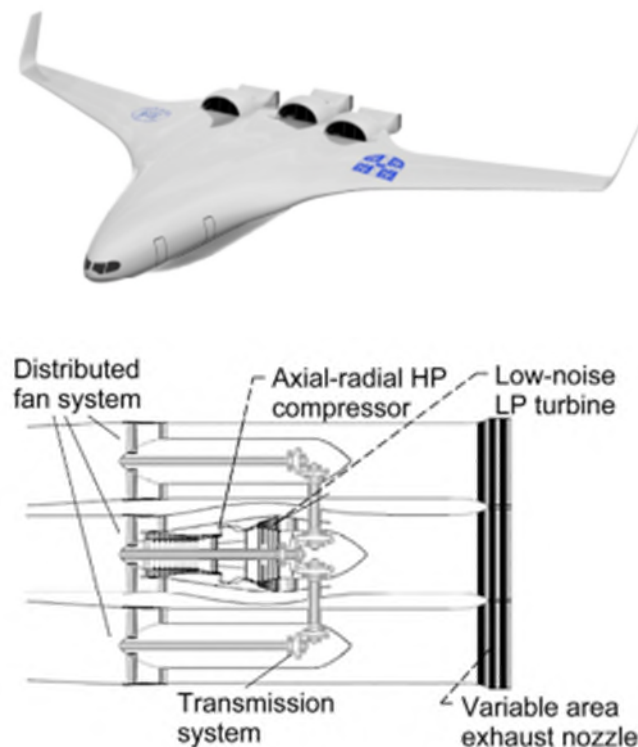


Figure 2-4 SAX-40 Silent Aircraft Initiative Concept using gear driven three fans with single-engine core propulsion system [12], [13] & [14]

3. Electrically Driven Propulsors

With electrically driven propulsors as the name implies involves the transmission of power to a series of distributed propulsors via electric transmission. Concepts and design studies of this configuration have been carried out for the use for both distributed fans [11], [17], [18] & [19] and distributed propellers [19] & [20]. The transmission of electric power from a common core allows for a distribution of power from one core to multiple small propulsive devices. This arrangement allows many small partially embedded fans while retaining the superior efficiency of large core engines, which are physically separated but connected through electric power lines to the fans [11]. The arrangement allows for greater flexibility in the placement of propulsors to better take advantage of integrated DP-airframe synergies. A special form of this type of DP is TeDP where electrical power is transmitted from one or more GTs to a series of distributed propulsors.

2.2 Turboelectric Distributed Propulsion Technology

2.2.1 Superconducting Turboelectric Distributed Propulsion

With the CESTOL airframe as a baseline [4], NASA considered a turboelectric-powered variant in [11]. The major change in the concept was the employment of a number of high temperature superconducting (HTS) motors to drive distributed fans rather than using many small conventional engines. For the configuration sixteen 35 inch fans were distributed along a large portion of the upper aft wing span to maximise BLI benefit. The combination of fans and generators in the configuration achieves a higher effective bypass ratio (eBPR) than the turbofans used in the CESTOL concept (10 vs 9.4) promoting fuel efficiency. The parameter eBPR in this context is defined as the ratio of all bypassed mass flow in the propulsion system to the mass flow through the engine cores. The power driving these electric fans is generated by two wing tip located turbogenerators through HTS transmission lines, with the wing tip location of the turbogenerator chosen as a design solution because of the potential for wing-tip wake vortices reduction. However just because in the study in [11] situated the core engines at the wing-tips, does not mean other

advantageous locations may exist when vehicle aerodynamics, structural performance and payload carrying capacity are considered. For the superconducting machinery it was envisioned that both the stator and rotor windings are fully superconducting, however for most state-of-the-art machines that are in service today only the rotor windings are fully superconducting. This is a major limitation because the main benefits for the use of superconducting machinery over conventional electronic machinery is significant increases in power density and transmission efficiency (exceeding 99% vs 95%) and for HTS machines it is the stator windings which are the high-power windings where most of the power losses occur.

Another major challenge however from the incorporation of HTS machines is that they need to be cooled to cryogenic temperatures for efficient operation and this imposes a weight penalty on the aircraft and power penalty on the propulsion system. For the study in [11] the use of cryocoolers was proposed as a cooling option, and from preliminary weight and performance estimation the combined weight of the TeDP system was 6,800 kg compared to 4,500 kg for conventional small distributed turbofans and 4,000 kg for conventional large non distributed turbofans; for which the Thrust SFC (TSFC) for the TeDP configuration and conventional large non distributed turbofans was the same but the TSFC for the conventional small distributed turbofan configuration being approximately eleven percent worse. It was proposed that liquid hydrogen (LH_2) could be used as a coolant for the HTS machinery to eliminate cryocooler power and mass penalties with the mass of the liquid cooled TeDP system estimated to be eighteen percent lighter than the cryocooled TeDP system. With the use of LH_2 for cooling, a synergy with the propulsion system opens up in the form of LH_2 also being able to be used as fuel. The gravimetric energy density of LH_2 is about three times greater than kerosene based jet fuel [21], [22] & [23] which allows for reduced SFC and combustion with no CO_2 emissions. However despite these benefits liquid hydrogen has a volumetric energy density approximately 3.6 times lower than kerosene and its use imposes a greater volume penalty on the aircraft.

The TeDP-BWB concept in [11] was studied and analysed further in [24] with the N3-X aircraft (see Figure 2-5) and compared to a conventional configuration in the Boeing 777-200LR and a similar sized BWB aircraft with two geared large ultra-high BPR (UHBPR) turbofans in the N3A. One of the key differences in the N3-X different from the previous study in [11] was the use of LH₂ to be used simultaneously for cooling and as a fuel. In addition to increased gravimetric energy density of LH₂ compared to kerosene, LH₂ can be pre-heated before entering into a gas turbine chamber for increased thermal efficiency. In studies carried out in [25] it was found that in an aero gas turbine that for the same TET, in a novel gas turbine hydrogen cycle with fuel pre-heated by exhaust gases that cruise SFC can be reduced by up to 2.6%. This is due to the additional energy imparted into the GT for a fixed fuel flow rate since fuel temperature is increased. Another significant change to the TeDP system architecture from previous studies was the incorporation of power electronics between the generators and the motors. Through the use of cryogenic power converters with improved transmission efficiency (from 95% to 99.93% [26]) and power density (from 16 kW/kg to 33 kW/kg [26]) compared to ambient temperature solid-state switching power converters; operation of the propulsors at any rotational speed independent of the speed of the GT output shaft is possible. This in effect physically decouples the GT from the propulsors by converting alternate current to direct current and vice versa with a combination of power converters acting in tandem as a continuously-variable-ratio gearbox.



Figure 2-5 N3-X aircraft TeDP concept

The N3-X aircraft with LH₂ cooling is compared to the cryocooled N3-X, the N3A and the Boeing 777-200LR in Table 2-2. The N3-X concepts in Table 2-2 have a higher eBPR than the conventional Boeing 777-200LR due to more ambitious propulsor design targets. Compared to the UHBPR N3A aircraft, the distribution of a number of small distributed fans for the N3-X concepts allows lesser installation penalty than the N3A concept and a higher eBPR. Also, the N3-X with cryocooling concept has a lower eBPR than the N3-X with liquid hydrogen cooling due to the need for greater core mass flow required of the turbogenerator to supply power to the cryocooler. Additionally the mail slot inlet nacelle allows for BLI in the N3-X which allows for increased propulsive efficiency however it was mentioned that the use of a mail-slot inlet could be limited by ice, snow, rain, insects and bird ingestion. For the N3-X propulsion system the distortion penalty from the fan was nominally set at 1% efficiency loss, although more detailed studies such as that conducted in [27] indicate a 1-2% efficiency drop and 2-2.5% drop in FPR is more atypical. Increased propulsive efficiency, increased eBPR, the use of LH₂ and higher transmission efficiency with the use of superconducting machinery result in a propulsion system for the N3-X that has a lower TSFC than the UHBPR turbofan of the N3-A and the conventional GE90-115B. The additional cooling power requirement from using a cryocooler caused a higher TSFC for the N3-X configuration with kerosene as a fuel when compared to the N3-X configuration with liquid hydrogen as a fuel. The reduced TSFC of the TeDP configurations leads to reduced aircraft fuel burn which allows for reduced aircraft size which also reduces aircraft thrust requirements.

From Table 2-2 the reduction in thrust requirements causes reduced turbomachinery size of N3-X's propulsion system compared to the N3-A's UHBPR turbofans and GE90-115B. Additionally the reduction in total propulsor mass (15 smaller fans and associated installation is lighter than two large fans for the same thrust requirement and FPR) further serves to reduce the turbomachinery weight of the N3-X's propulsion system. Reduction in turbomachinery weight causes a reduced overall propulsion system mass however this reduction is offset to some extent by the additional weight of

electric components. The N3-X configuration with LH₂ as a fuel has a propulsion system that is lighter than the cryocooled N3-X due to the fact that the need for a cryocooler is eliminated. However the elimination of a cryocooler is counteracted to a certain extent at the aircraft level by the need to store hydrogen in cryogenic tanks that which imposes a weight and size penalty on the aircraft.

Table 2-2 Integrated Aircraft-Engine comparison [24]

	Boeing 777-200 LR/GE90-115B	N3A	N3-X (cryo)	N3-X (LH ₂)
<i>Propulsion System at Aerodynamic Design Point (ADP)</i>				
eBPR	8.5	27.2	30.5	31.7
Average Propulsive efficiency	87.0%	90.2%	96.4%	96.4%
TSFC (installed) (lbm/hr/lbf)	0.578	0.4667	0.3451	0.294
Installed net thrust	55697	24173	19293	19293
<i>Propulsion System Weight</i>				
Turbomachinery weight (core and fan) (lb)	33622	28887	15335	15335
Gearbox/Electrical weight – includes cryocooler (lb)	0	592	13938	11841
Installation weight (Inlet, Nacelle, Nozzle, Pylon) (lb)	8829	13377	3029	3029

Propulsion System Weight total (lb)	42451	42856	32302	30205
<i>Aircraft Level</i>				
Empty weight (lb)	340800	285800	267800	267400
Payload weight (lb)	118100	118100	118100	118100
Block Fuel (lb)	279800	133700	83500	78500
Reduction in block fuel (lb)	N/A	52.2%	70.2%	71.9%
Take-off Gross Weight (TOGW) (lb)	768700	551000	479300	473500

2.2.2 Gas Turbine Performance

Efficiency

One key difference between proposed TeDP concepts and conventional GT architectures is the decoupling of the thrust producing device from the power producing GT. Typically for civil transport role aero GTs, power is transmitted mechanically from the output shaft of a GT to either a propeller (turboprop) or a fan (turbofan). The performance achievable of aero GTs has increased significantly over the past century relative to the original Whittle turbojet engine (see Figure 2-6). Considering present day state-of-the-art high BPR turbofans, core thermal efficiency has improved by about 32 to 38 percentage points and propulsive efficiency has improved by about 10 to 20 percentage points when compared to the original Whittle turbojet. Efficiency improvements in GTs over the past century have been driven by increases in TET, overall pressure ratio (OPR) and BPR. With current technology the maximum achievable OPR for GTs is about 50+ and maximum achievable TET is 1,800 K with blade cooling. However continual improvement of GT performance by means of increasing

TET and OPR would necessitate the use of advanced cooling and material technology for an ever-diminishing thermal efficiency gain and increased technical investment [28].

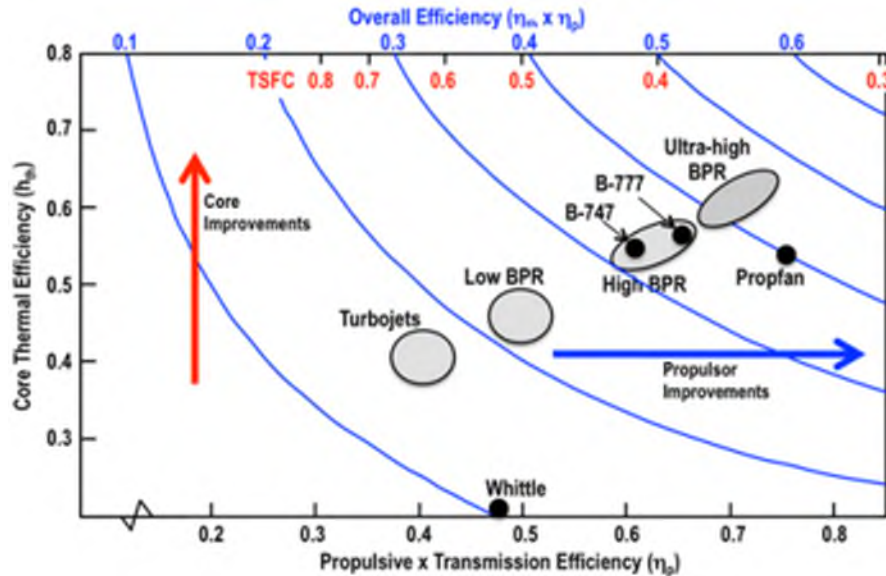


Figure 2-6 Overall engine efficiency as a function of core thermal efficiency, propulsive efficiency and transmission efficiencies [28]

Novel Cycles

With a decoupled core engine, the architecture of the core GT can be modified and optimised without limiting the performance of the propulsor. This opens up the potential for the use of novel cycle layouts such as the cycles studied in [19] & [29]. In [29] a recuperated Brayton cycle was considered for a small GT (mass flow rate = 5.67 kg/s) being used for a High-Altitude-Long-Endurance (HALE) Surveillance unmanned mission role. A 15% increase in vehicle endurance over a baseline configuration was estimated to be possible and this increase was mentioned to be possible by the incorporation of a heat exchanger (HE). Design TET for this GT was 1,523 K and OPR 15.8. This benefit in novel cycle use however is very mission specific as Unmanned Aerial Vehicles (UAVs) typically operate at lower thrust levels than what is required for a civil transport role. Thus for civil transport role with a larger engine, TET and OPR would be higher than what is achieved with a UAV and this allows for increased thermal efficiency without the need for adopting a novel cycle layout. This to

some extent negates the need for a novel cycle layout for a GT being used for a civil transport role. Also, for a shorter distance being flown on a civil transport mission, incorporation of heat exchangers is likely to impose a greater weight penalty on the aircraft.

Use of Hydrogen

A change of aircraft fuel from kerosene to a fuel with a higher lower heating value (LHV) will reduce engine fuel flow rate given GT cycle design parameters and work output remain constant. This because the amount of heat needed to maintain the same thermal efficiency remains constant but since heat addition for a gas turbine is a function of fuel flow rate and LHV, for constant heat addition an increase in LHV means fuel flow rate has to be reduced if the same amount of heat into the engine is to be maintained. Being able to reduce the fuel flow rate of an engine means that for a given GT thrust or power requirement at the propulsion system level, less fuel will be burnt at the aircraft level. Two such fuels that have been considered in recent TeDP studies which have a higher LHV than kerosene that can be utilised with GTs are hydrogen [18], [19], [24] & [30] and methane [30]. With an emphasis on hydrogen its main benefits as an aviation fuel are reduced fuel burn for a given GT work output and cycle, and zero CO₂ emissions. Limitations for the use of hydrogen as a fuel are that for a given kilo of fuel more water vapour will be emitted compared to kerosene. Also, its volumetric energy density at ambient temperatures and pressures is significantly less than kerosene. This is one of the reasons why for the studies in [18], [19] & [24] the use of hydrogen as a fuel was assessed in its liquid form for which hydrogen in its liquid form has a volumetric energy density about two times greater than gaseous hydrogen stored at 800 bar [31]. However hydrogen only exists as a liquid at cryogenic temperatures which brings into play tank design challenges such as size constraints, embrittlement, permeation, tank shape, manufacturing process and weight minimisation. These constraints are what drive tank design [32] and limit size, shape and materials that can be used for the construction of safe and reliable LH₂ cryotanks

whereas kerosene can be stored in integral tanks in a wing which aid in providing wing bending relief and a lighter aircraft structure.

2.2.3 Distributed Propulsor Architecture

In TeDP the propulsors are decoupled from the predominantly power generating GT. This opens up a number of possible design synergies depending on the architecture of the propulsive device considered. Two possible arrangements identified from the literature read are:

- Distributed ducted fans studied in [18],[19], [24], [30] & [33], and
- Distributed propellers studied in [19] & [20]

Typically with more conventional aero GT architectures than TeDP, propellers are used for aircraft flying at subsonic speeds from Mach 0 to Mach 0.6 due to the fact that turboprops achieve higher propulsive efficiencies than similarly sized turbofans and turbojets (see Figure 2-7). This is because for turboprop configurations a significantly greater amount of air is bypassed through a propeller than for a high bypass ratio turbofan which means lower jet velocities for propellers compared to jet velocities for a high bypass turbofan. At Mach numbers above 0.6 however, propulsive efficiencies achievable for propellers using conventional design methods rapidly drops due to the onset of wave drag at the blade tips (Propfans are exceptions that use specially designed propellers with swept back blade tips that delay the onset of wave drag that enable efficient flight up to higher Mach numbers of 0.7+) [34]

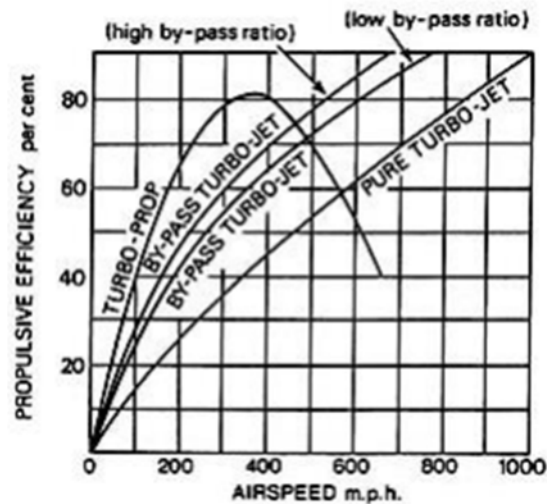


Figure 2-7 Propulsive efficiency for a selection of GT configurations [35]

For TeDP architectures a series of ducted fans can be installed on an airframe for higher effective BPRs with reduced installation penalty than a similarly sized turbofan with the same BPR [19] & [33]. This feature combined with the option of partially embedding fans with a given airframe for synergistic aerodynamic benefit, means that the propulsive efficiency of a ducted fan configuration can approach or even be larger than the propulsive efficiency achievable with turboprop configurations at subsonic Mach numbers between 0 and 0.6 [36]. In the transonic speed region, propulsive efficiencies greater than 90% are possible for ducted-fan-TeDP configurations with BLI and a BWB airframe [33]. With the BWB the relatively large span of the inner-wing section allows for a large number of distributed propulsors to be arranged on its upper surface and this upper surface can be aerodynamically shaped to optimise boundary layer growth and improve overall vehicle aerodynamics [37].

A gap in understanding emerges in the understanding of the multi-disciplinary trade-offs of tightly-integrated-airframe-TeDP system configurations as a result of the incorporation of partially-to-fully embedded distributed propulsors. For instance in [24], distributed ducted fans are situated at the rear of the inner wing with propulsors partially embedded in the airframe to take advantage of BLI. This arrangement allowed for a 71.9% reduction in fuel over present day state-of-the-art architecture. However propulsors can be situated at the rear of the

fuselage to take advantage of BLI as is the case in [16] & [18] in which the use of a conventional airframe is considered and this to a certain extent reduces design complexity, manufacturing difficulties and regulatory requirements of using TeDP with a novel airframe at the aircraft level. Additionally for aircraft flying at subsonic speeds less than Mach 0.6, a series of distributed propellers can be situated along the wing for a lift-augmentation effect that allows for wing resizing to meet field length constraints and reduce drag [20]. A summary of possible synergies and design disciplines affected by a series of distributed propulsors is shown in Table 2-3.

Table 2-3 Summary of synergies of decoupled distributed propulsors and disciplines affected

Discipline	Synergies, benefits and associated physical phenomena
Aerodynamics	<ul style="list-style-type: none"> • BLI associated performance benefits and penalties [18], [24] & [33] • Propeller slipstream-wing lift augmentation [19] & [20] • Supercirculation/blown flap effects [38] • Elimination of wing tip vortices for reduced lift induced drag [39] • Reduced nacelle drag associated with reduction in propulsor size^a
Weight and Structures	<ul style="list-style-type: none"> • Bending relief effect on wing to reduce overall wing mass due to distribution of propulsors over wing [40] • Reduction in total propulsor mass associated with reduction in propulsor size as propulsor mass is roughly proportional to a fan diameter that is raised to a power greater than one (with fan weight estimation method in [41] this power is 2.7) • Reduction in nacelle weight associated with reduction in propulsor size • Empennage size reduction [18]

	<ul style="list-style-type: none"> • Reduction in LP system weight and elimination of a gearbox between core engine and propulsor for overall propulsion system weight reduction
Dynamics and Stability	<ul style="list-style-type: none"> • Vectored thrust/vehicle control [42] • Increased pitch effector performance [11] • Aircraft centre of gravity and neutral point changes
Noise	<ul style="list-style-type: none"> • Airframe noise shielding of forward radiating engine noise to reduce acoustic emissions, dependent on airframe chosen (e.g. BWB airframe [14]) • Reduced core jet velocity, reduced propulsor jet velocity and embedded propulsors resulting in reduced noise emissions [43]
Safety	<ul style="list-style-type: none"> • Inherent redundancy built in with distributed propulsors if one or more propulsors fail • Increased safety against possible engine rotor blade detachment depending on where propulsors and core engine(s) are located

^aWith an increased number of propulsors, for the same mass flow rate and thrust requirement, overall propulsor size reduces as number of propulsors increase

2.2.4 Electrical Systems & Architecture

In [11] a generic TeDP system's electric architecture was defined to have two turbogenerators supplying power to superconducting motors through HTS transmission lines with power converters present and is shown in Figure 2-8. It was stated that this architectural layout would add considerable flexibility to the propulsion and vehicle architecture by functioning as a continuously-variable-ratio gearbox allowing for propulsive fans and the output shaft of GTs to operate at optimal speeds throughout a flight envelope. Despite this function in [11] a fixed speed ratio was considered between GT output shaft and the propulsive fans with a solid-state converter present for power management. The type of

converters to be used, whether Alternate Current/Alternate Current (AC/AC), Direct Current/Direct Current (DC/DC), AC/DC or DC/AC was not explicitly stated.

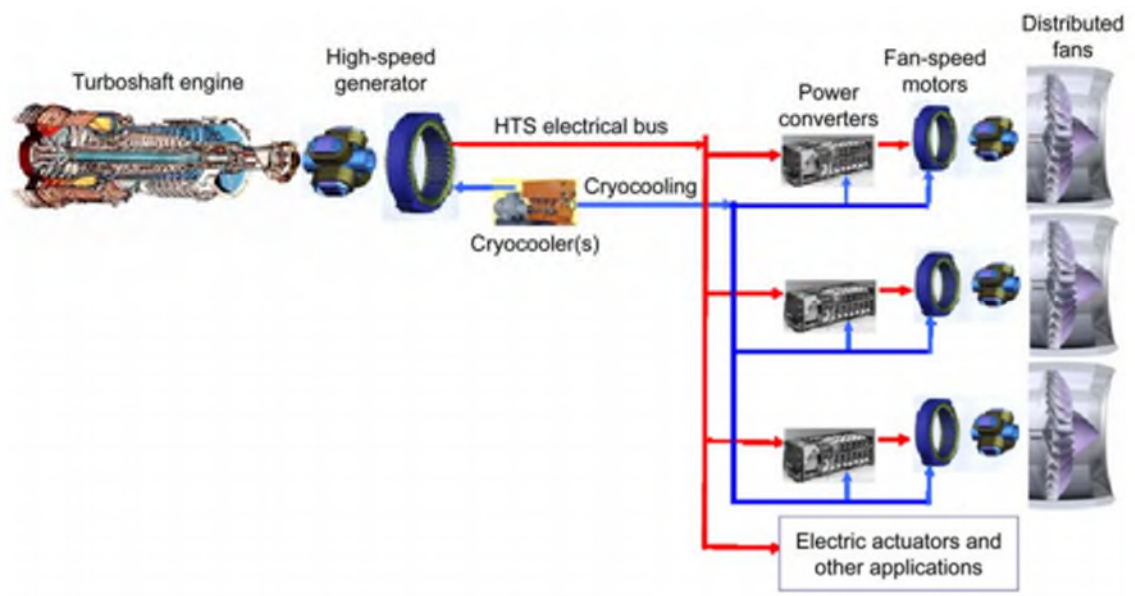


Figure 2-8 Generic TeDP system architecture [11]

The system architecture in [24] & [26] (see Figure 2-9) defines the power converters to be employed for N3-X’s TeDP system to be inverters which convert DC current to AC current. In the configuration stator windings in the generator delivers AC power which is rectified and transmitted as DC power by superconducting transmission lines, however it is not mentioned whether this rectification is part of the generator design or is a stand-alone component of the system. It is this DC power generation that enables propulsors to be situated anywhere on the vehicle and to run at a different speed to the main GT.



Figure 2-9 Components in a superconducting TeDP system

In [44] the power management and distribution (PMAD) of the N3-X’s propulsion system in [24] is better defined (see Figure 2-10). The rectification of the AC power from the generator considered as a stand-alone component as an AC/DC

power converter and superconducting fault current limiters (SCFCL) are considered as part of the architecture to limit the fault current rating of components downstream with circuit breakers (CBs), reactors and switches providing further equipment protection. A benefit of SCFCL is that the weight of components downstream of the SCFCL can be reduced at the cost of increased SCFCL weight. Additionally with the architecture an energy storage device is incorporated into the architecture to reduce engine transients for increased thrust response and extended GT life-time. Not stated however in [44] is that the presence of an energy storage device in the architecture opens up the option for using a turboelectric system as a hybrid TeDP system as studied in [45]

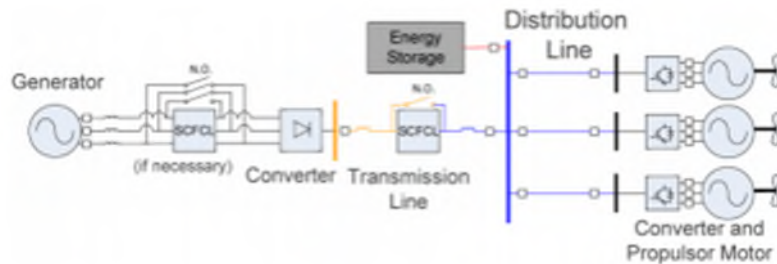
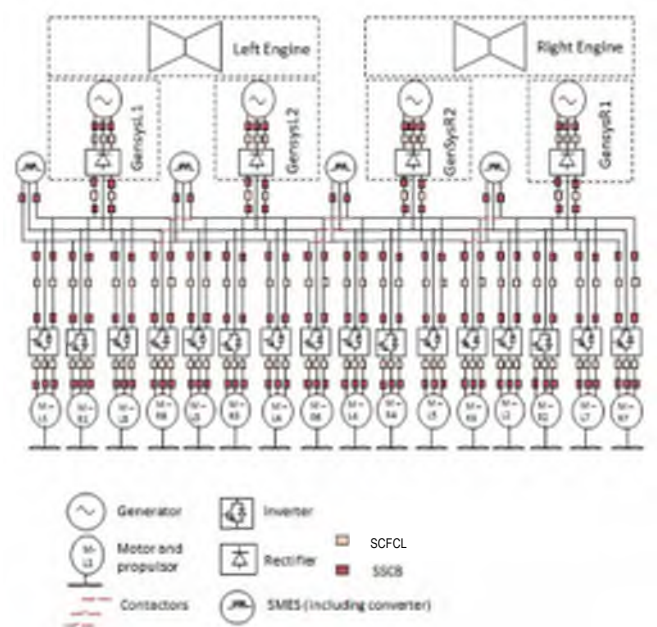


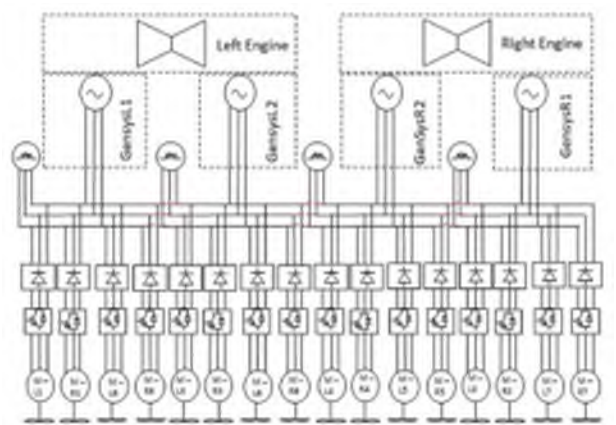
Figure 2-10 General description of electric power generation, distribution and consumption in a TeDP system [44]

The transmission of current in the electrical network of the TeDP system defined in [44] is predominantly DC. This architecture was assessed further in [46] but different from [44] a power converter is stated to be needed to interface energy storage devices to the electrical network to ensure adequate controllability in flight. This presence of another power converter poses another weight and size penalty for the propulsion system when integrated with an aircraft. Considering this plus the weight of other power electronics in electric architecture of a TeDP system, studies in [46] compared the baseline DC architecture to a hybrid AC/DC architecture and AC architecture (see Figure 2-11). At present only the DC architecture has a proposed protection system from the literature read, and super magnetic energy storage (SMES) was considered as an energy storage device for DC and hybrid AC-DC architectures for its ability to absorb and deliver large amounts of power [47]. In all three

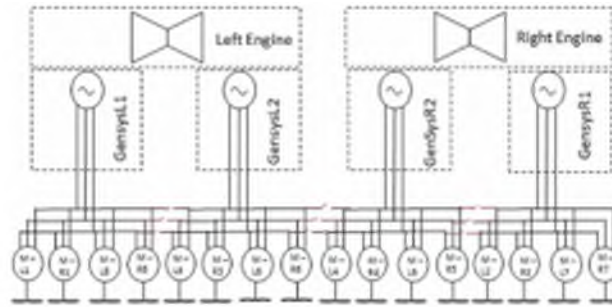
architectures two generators are present per GT for redundancy in the event of a one engine failure. Interconnection between four main sub-systems powered by each generator is present to reduce motor power rating which reduces electrical system mass and size penalty.



(a)



(b)



(c)

Figure 2-11 Candidate Electric Architectures for TeDP (a) DC architecture with protection system comprising of solid-state circuit breakers (SSCB) and SCFCLs (b) Hybrid AC-DC architecture (c) AC-AC architecture [46]

The hybrid AC-DC architecture is predominantly AC with back-to-back converters for each motor to allow for independent speed optimisation. With this configuration the number of power converters in the electrical system can be reduced. For the AC architecture power electronics can be completely eliminated which results in the generators and motors being run synchronously however the ability to independently run GT and propulsors at different speeds is lost. Additionally with the AC architecture the energy storage device is eliminated as it was mentioned that the energy storage device would not be able to interface with the network. Considering the differences in architecture, a comparison of the performance of the three architectures at the electrical network level presented in [46] is shown in Table 2-4. The elimination of the power electronics results in the AC architecture being much lighter than the hybrid AC-DC architecture which is lighter than the DC architecture and the efficiency of the AC architecture is also the highest. At the electrical network level the AC architecture also performs best. A gap in knowledge exists in the evaluation of these architectures at the integrated aircraft-propulsion system level. This is because the trade-off between off-design operation and weight savings possible of the candidate architectures would need to be assessed.

Table 2-4 Comparison of mass and efficiencies of candidate architectures at optimal operating points without protection [46]

	Candidate Architecture		
Parameter	DC	AC-DC	AC
Efficiency ^a (%)	70.82	72.44	97.47
Mass (kg)	34860	32690	5065

^aEfficiency in this context defined as the ratio of total motor power demand to the summation of total motor power demand, cryocooler power and electrical power losses

For DC and hybrid AC-DC architectures considered in [46], hybridisation of the propulsion system is possible if the architecture is operated to have propulsive fans receiving a proportion of power from an electrical energy storage device such as a battery or a flywheel [48]. From studies carried out in [19] for a HALE unmanned mission and [49], [50] & [51] for a transport role, increasing hybridisation at the aircraft level is characterised by increased aircraft weight and enhancement to overall propulsion system efficiency (discharge efficiencies of current state-of-the art batteries, is circa 99% [52] & [53]). Increase in aircraft weight and increased propulsion system efficiency act as two counteracting effects on aircraft performance with the former effect serving to worsen aircraft fuel burn and overall energy consumption and the latter serving to improve it. For the HALE unmanned mission studied in [19], for a tube-and-wing aircraft weighing 15,000 kg an endurance increase of close to 10% over a baseline configuration was determined for an energy degree-of-hybridisation H_E of 17.7%. The increase in vehicle performance however was eroded as H_E increased as the battery starts to have a deadweight effect on the airframe. The effect of changing the degree-of-hybridisation for power H_P however was not assessed in the studies in [19] but it was considered in the studies in [51]. Here the trends are similar in the fact that with block fuel as a mission performance parameter, for a given aircraft design range there is an optimal H_P for minimum block fuel. To adequately assess and characterise a hybrid electric propulsion (HeP) system architecture, H_E and H_P need to be accounted for [54], along with

an assessment of the off-design performance of core GT(s) to determine to what extent SFC is mitigated during off-design operation [19].

Characterising the propulsion system as a HeP system this further opens up the design space and system configurations possible for electrified aircraft. From an architectural point of view it should be noted that although the DC and hybrid AC-DC architectures can function as complex series hybrid electric distributed propulsion (HeDP) systems, a parallel HeP architecture such as the one defined Figure 2-12 is not a TeDP architecture as there is synchronous mechanical transmission between the GT and the propulsor. Another HeP aircraft concept in [51] assessed a quad-fan concept with two geared turbofans and two electric fans. This distinction is important because it is the first step towards the understanding, characterising and quantifying the benefits of the electrified-GT propulsion architectures and suitably defining the integrated electrified-GT-airframe propulsion design space.

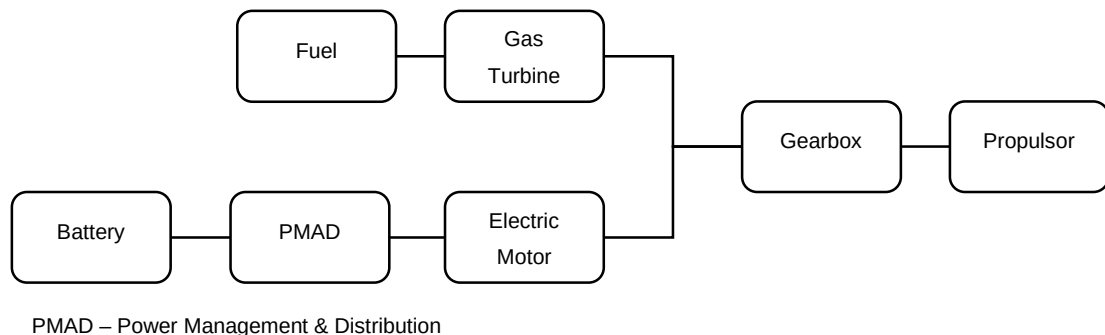


Figure 2-12 Block diagram representation of parallel HeP configuration/architecture considered in [51]

The full realisation of the benefits of electrifying the propulsion system with TeDP and similar architectures is still far away as a number of the technologies needed for concept feasibility have only been developed to low Technology-Readiness-Levels (TRLs) [44]. A summary of the weights and efficiencies of present day state-of-the-art electrical machinery compared to present future estimates for electrical machinery is shown in Table 2-5. From Table 2-5 future estimates for energy and power densities of the electrical components in a TeDP system obtained from literature are highly ambitious with the exception

being the SCFCL. For this component the technology associated with its development has already reached a TRL of 5-6.

Table 2-5 Electrical machinery comparison

Power/Energy Density			
Component	Present day state-of-the-art	Future target	Change (Future vs Present)
Energy Storage	1.30 MJ/kg ^a [55]	6.12 MJ/kg ^b [55]	+370.8%
Generator	8.20 kW/kg ^c [56]	41.59 kW/kg ^d [24]	+407.2%
Motor	8.20 kW/kg ^c [56]	22.03 kW/kg ^d [24]	+168.7%
Power Converter	16.4 kW/kg [26]	32.9 kW/kg ^e [26]	+100.0%
SCFCL	4.20 kW/kg [57]	N/A	N/A
Transmission Lines	18.27 kW/kg ^f [21] & [58]	227.1 kW/kg ^g [26]	+1143%
Efficiency			
Component	Present day state-of-the-art	Future target	Change (Future vs Present)
Energy Storage	0.99 [52]	From literature read no data present	N/A
Generator	0.95 ^c [56]	0.9998 ^d [24]	+5.242%
Motor	0.95 ^c [56]	0.9997 ^d [24]	+5.232%
Power Converter	0.95 [48]	0.9993 ^e [26]	+5.189%
SCFCL	1.00 [57]	N/A	N/A
Transmission Lines	0.999997 ^h	0.999998 ^g [26]	+0.0001%

^aLithium-Sulphur Battery

^bLithium-Air Battery

^cHalbach array architecture

^dFully superconducting rotors and stators made with MgB₂

^eCryogenic operation

^fCopper

^gSuperconducting

^hCopper cable sized to the same dimensions and operating requirements as the HTS transmission line. Electrical resistivity of copper at 298 K is $1.712 \times 10^{-8} \Omega\text{m}$ [21]

2.2.5 Cooling Systems

The use of superconducting technology is a key technology for the feasible implementation of TeDP with aircraft. However the use of superconducting machines requires operation at cryogenic temperatures and hence an on-board cooling system is required. Identified ways of cooling electric machinery are with a cryocooler or a cryogen. Present day cryocoolers are too heavy for TeDP application with the lightest existing coolers having power densities of 55 W/kg [26]. Additionally the upper limit of efficiency for existing cryocoolers is about 30% of Carnot efficiency. For feasibility of a cryocooled TeDP system an aggressive power density target of 329 W/kg has been set for cryocoolers to achieve. With cryogen cooling the cryocooler is eliminated and the cryogen can be used synergistically with the propulsion system as fuel for the aircraft. However this comes at a cost of having to install cryogenic tanks in the airframe that imposes a volume and mass penalty on the aircraft [18], [19] & [59].

2.3 Summary

DP is the distribution of airflows and forces generated by the propulsion system about an aircraft in ways to improve vehicle aerodynamics, propulsive efficiency, structural efficiency and aeroelasticity. TeDP is a special DP configuration which has been proposed as a technology solution to meet future environmental and performance targets. In TeDP power is transmitted electrically from a few core GTs to a series of distributed propulsors. For this arrangement the employment of fully superconducting machines has been identified as a required technology for feasible employment of TeDP on a given aircraft. For superconducting TeDP a summary of the benefits realisable for the technology are listed below:

1. V/STOL capability of TeDP airframes

2. Taking advantage of BLI for propulsive and aerodynamic efficiency improvement
3. Increased eBPR for cruise efficient operation
4. Strategic location of cores and propulsors for airframe structural relief
5. Use of hydrogen as both a fuel and coolant for reduced CO₂ emissions and reduced SFC.
6. Reduced propulsion system installation penalties for a given eBPR
7. Reduced noise by better and tighter integration between airframe and propulsion system
8. Employment of hybrid electric architectures for reduced fuel burn
9. Higher transmission efficiencies
10. Reduction of LP system weight
11. Running of propulsors and core at optimal speeds during off-design for minimum off-design SFC
12. Elimination of oil for cooling and lubrication of a geared transmission system
13. Use of novel cycle layouts for core engine architecture becomes attractive for reduced fuel burn
14. Reduced turbomachinery mass
15. Minimisation of GT transients for increased GT lifetime and increased thrust response
16. Improved propulsion system efficiency
17. Reduction of mechanical design complexity
18. Potential reduction of wing-tip vortices by locating either propulsors or engine cores at the wing-tip
19. Taking advantage of synergistic propeller-slipstream-wing interactions for drag reduction and/or field length reduction
20. Novel airframes such as the BWB to become potentially feasible by taking advantage of integrated synergies

However the use of TeDP is not without its shortcomings and potential limitations of the technology include the following:

1. Distortion penalty on fan turbomachinery for BLI configurations
2. Increased system complexity
3. Cryogen storage installation penalties if a cryogen is used for cooling of machinery and/or used as a fuel
4. Present unavailability of required technologies
5. Feasibility of using technology is dependent on achieving very ambitious performance goals
6. Additional turbomachinery weight in novel cycle layouts
7. Deadweight penalty of electrical energy storage device in a hybrid electric propulsion system if oversized
8. Increased cooling power requirements
9. Increase in transmission weight

From the literature reviewed it has been seen that benefits realised with TeDP are configuration and mission specific and benefits and synergies from one configuration may not be applicable to another configuration and vice-versa. The main gaps in understanding emerge in the understanding of the multi-disciplinary trade-offs of tightly-integrated airframe-TeDP configurations as a result of the incorporation of partially-to-fully embedded distributed propulsors. Being specific, the following knowledge gaps arise:

- A procedure for adequately quantifying optimal GT/turbogenerator location on the airframe for best aerodynamic and structural efficiency rather than just a statement of implicit performance benefits
- Most of the technology specified for concept feasibility has only been developed to low TRLs and thus there is the challenge of applying existing models, methods and theories to technologies that haven't even been realised yet. Technoeconomic factors can change a great deal between now and the N+3 timeframe and thus it is not certain whether existing models, methods and theories prove concept feasibility without full-scale development. A research gap lies in the formulation of procedures that can adequately assess the integrated electrified-

propulsion-airframe design space as critical enabling technologies develop from low to high TRLs.

- By elimination of the gearbox the adoption of electric architecture presents the opportunity for novel off-design control schemes. Understanding the control and sizing differences at off-design between TeDP, HeP and HeDP architectures, conventional architectures at the aircraft level has not been formalised from literature reviewed

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3 METHODOLOGY

3.1 Introduction

The stated aim of the research is to assess the performance and feasibility of conceptual integrated aircraft-TeDP configurations and to understand how the overall system changes for different aircraft mission roles. To achieve this, the objective of developing a multidisciplinary and integrated method was set. By the achievement of this objective all the other research objectives can be met. Considering a TeDP system as a holistic entity its performance is a function of propulsor configuration, airframe type, GT performance, electric architecture, energy storage and mission profile (see Figure 3-1). From the integrated nature of TeDP a systems engineering approach is employed in formulating the methodology, as it focuses on the system as a whole and tackles the complexity that arises as a result of a number of interrelated components and subsystems working together to achieve a common objective [1].

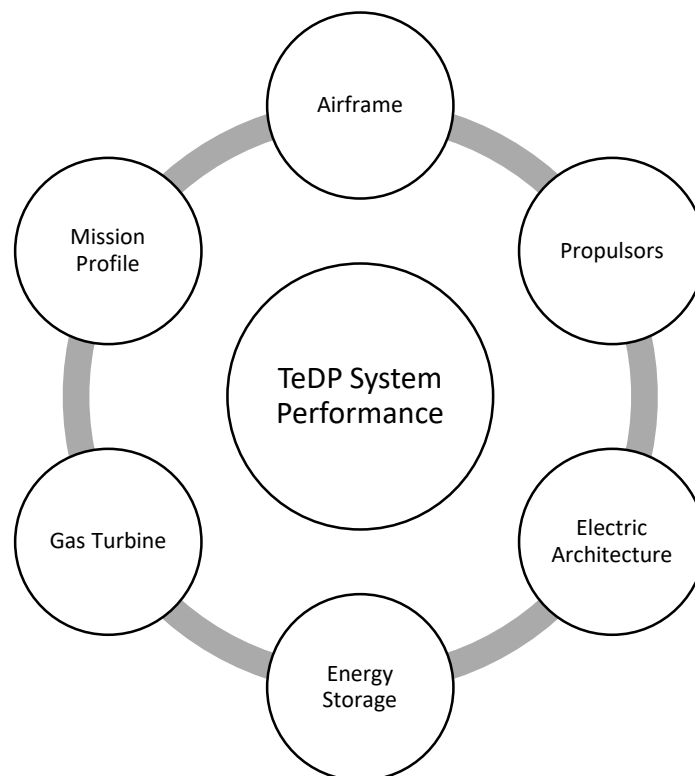


Figure 3-1 TeDP System Relationships

Previous studies which have used systems-like methods in the assessment of TeDP system architectures with the studies in [2] & [3] utilising methods that assess the TeDP system performance at the propulsion system level and the studies in [4] & [5] utilising methods that assess TeDP system performance at the integrated airframe-propulsion system level. For HeP architectures a methodology for assessing hybrid electric aircraft performance at the aircraft level has been formulated in [6]. In [3] the TeDP engine cycle with BLI at design point and off-design was characterised. The method developed was then employed in the assessment of TeDP system performance for the N3-X aircraft. Research carried out in [2] covered similar ground but there was more emphasis on the modelling of the distortion effect of BLI on the distributed fans through the use of the discretized miller approach. For studies carried out at the aircraft level previous approaches to assessing TeDP system performance have been mission specific. Studies in [5] looked at the application of TeDP technology for a HALE UAV technology demonstrator. The purpose of this demonstrator was for the HALE UAV to act as a test-bed for TeDP technology to prove concept feasibility before full scale development for a civil transport role. The multi-disciplinary and multi-fidelity method employed in [5] allowed for the capturing of the effects of the use of cryogenic fuel, novel cycles, HeP, natural laminar flow shaping, novel airframes and propeller slipstream-wing lift augmentation. Studies in [4] employed a similar integrated method for a long range transport which captured synergistic vehicle aerodynamics in BLI and laminar flow control in the assessment of TeDP system performance for both a T&W airframe and a BWB. In [6] instead of characterising the propulsion system by thrust and fuel flow tables in the assessment of propulsion system performance at the aircraft level, an approach was utilised that characterises the propulsion system by thrust and energy tables. The characterising of the propulsion system this way allowed for the accounting of propulsive power being supplied by two energy sources.

For the method employed here different from previous research; is the development of an approach that allows for the assessment and understanding of TeDP system architectures for the entire design space as a function of

mission profile, GT performance, airframe type, propulsor architecture, electric architecture and energy storage (see Figure 3-1) for civil transport aircraft. An emphasis is placed on the understanding of how the propulsion system affects aircraft sizing and performance. The top-level method employed is shown in Figure 3-2. The method is to be used to assess TeDP and similar architectures at a conceptual level as a pre-cursor to the development of higher fidelity methods for preliminary and detailed design.

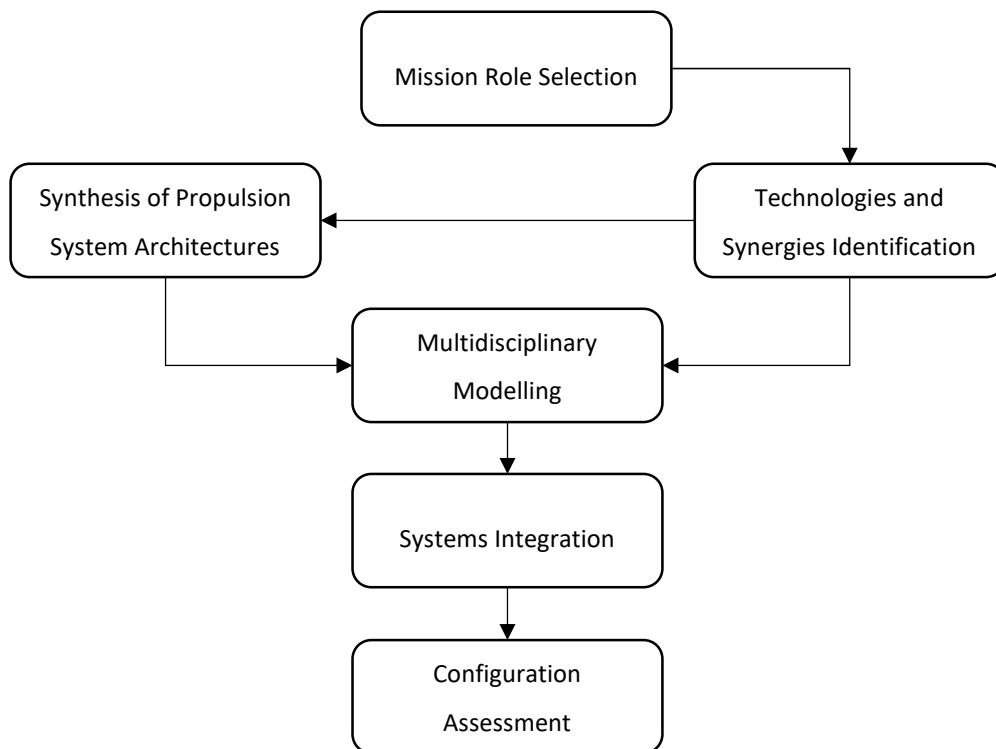


Figure 3-2 Top-Level Method

3.2 Mission Role Selection

To adequately assess the performance of TeDP system architectures at the aircraft level, baseline missions need to be defined. The choice of mission has consequences on integrated aircraft-propulsion system performance. For a given aircraft mission using a TeDP system the following factors are going to affect the mission performance:

- Heat losses from the electrical system
- Specific energy and power density of the electrical system

- Flight speed and altitude
- Propulsor selection – whether using fans or propellers
- Mission objective – civil transport role, military combat, surveillance, etc.
- Duration of flight spent at propulsion system off-design conditions
- Importance of terminal area operations
- Aircraft aerodynamics – compressibility effects become a concern at higher Mach numbers
- Safety and technological risk

These factors present a challenge in realising the practical benefits of TeDP for civil transport roles. To reduce the risk in the employment of TeDP for a civil transport aircraft a HALE unmanned mission was considered in [5]. The HALE UAVs considered in [5] were to act as a technology demonstrator with identified technologies for the HALE mission to be scaled up for civil transport roles. For civil transport missions in the current research, regional and medium-haul missions are considered. For the regional transport the baseline aircraft is to be an aircraft similar to the Bombardier Q400 EHGW and for the medium haul mission the baseline aircraft is to be an aircraft similar to the Airbus A320-200.

3.3 Technologies & Synergies Identification

To adequately assess TeDP system performance the components and sub-systems that comprise the integrated TeDP system has to be defined. This allows for the consideration of the effect different technology and synergy options have on system performance. To understand the effect different technology and synergy options have on system performance the following procedure is followed:

1. Identification of sub-systems that comprise the TeDP system
2. Identification of components for each sub-system
3. Identification of possible technology options for each component and/or sub-system
4. Identification of direct sub-system interrelationships for sub-systems defined

5. Identification of possible synergies and physical effects that occur for each sub-system interrelationship
6. Consideration of physical effects that affect performance for a given aircraft mission
7. Down-selection of technologies and synergies to be considered for a given mission

The identified sub-systems that comprise the TeDP system for the current research are (1) propulsors, (2) GTs, (3) airframe, (4) transmission system, (5) cooling system and (6) energy storage. For GT and the propulsor, the components that comprise each is dependent on the architecture defined for each. For the transmission system the main components it consists of are the generators, motors, power converters, transmission lines and gearboxes (if a geared architecture is defined) [7]. For the airframe identified key components are the wing, fuselage and empennage. The energy storage for the TeDP system can either be in the form of stored fuel, stored electrical energy or a hybrid of both.

At the sub-system level the following technology options were identified:

- Airframe – BWB [8], T&W [4] & [9] and Triple Fuselage [5] & [10]
- Cooling System – Cryocooled or Cryogen-Cooled [8]
- Energy Storage – All-Fuel [8], All-electric [11] or Hybrid [6]
- GT – Turbojet [3], Turboshift [3] & [8] or Turbofan [3] & [12]
- Propulsor – Ducted Fan [8] or Propeller [13]
- Transmission – AC, DC [7] or Geared [9] & [14]

At the component level the following technology options were identified:

- Electric Energy Storage – Batteries or Flywheels [15]
- Empennage – T-tail [9], conventional tail, inverted Y-Tail [10] or Tailless [8]
- Fuel – Hydrogen, Kerosene [8] or Methane [16]
- Generator – Halbach array [17] or AC Fully Superconducting [8]
- Motor – Halbach array [17] or AC Fully Superconducting [8]

- Power Converters – Ambient or Cryogenic [18]
- Transmission Lines – Copper or HTS [18]

For TeDP an overview of how the identified sub-systems directly interact with each other is shown in Figure 3-3. For the considered TeDP system the propulsor only directly interacts with the airframe and transmission system. The interaction with the transmission system transfers power from either a GT and/or electric energy storage device to the distributed propulsors. With the airframe the propulsor imposes a weight and sizing penalty on the airframe while the sizing of the airframe dictates the thrust requirements for the propulsor. Identified synergies that open up through the interaction of the propulsors with the airframe are wing bending relief; increased aileron effectiveness; BLI; noise reduction; supercirculation; slipstream lift augmentation; empennage size reduction; thrust vectoring and rudder control. In addition to the propulsors the energy storage, GT and transmission all impose weight and size penalties on the airframe. For the case where superconducting machines and cryogenic power converters are employed, a cooling system is needed to keep machinery operating at cryogenic temperatures. The cooling system used whether a cryogen or a cryocooler imposes a weight and volume penalty on the airframe and a power requirement from the GT. If a cryogen is used for cooling and is also used as a fuel then the performance of the cooling system is dependent on the properties of the cryogen used. Considering a decoupled GT in addition to providing power it can also be situated on the airframe in as such a way to provide wing bending relief and to reduce wing tip vortices. Additionally the heat used to run this GT comes from internal energy stored in fuel. This fuel depending on its properties whether cryogenic or not will require its own systems.

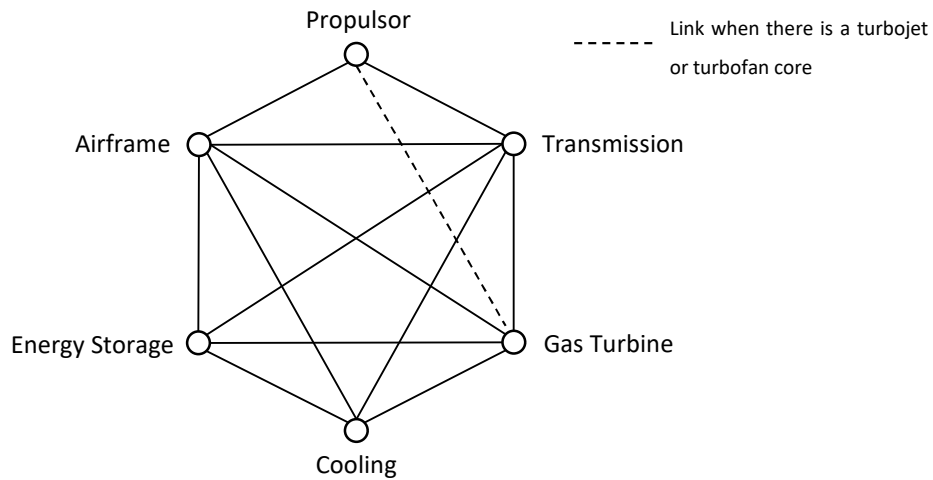


Figure 3-3 Web diagram showing direct interrelationships between TeDP sub-systems

In considering the technologies and synergies applicable for a given mission, ideally a pre-design quantitative method should be used to assess the different technology options as a function of mission role. However, for TeDP where a standardised architecture and system has not yet been defined at a high TRL, this would be a time-consuming task beyond the scope of this PhD. Instead rationales are discussed on why specific synergies and technologies are considered for a given mission.

For the HALE mission in [5], cruise efficient operation and payload carrying capability were key design drivers. As a consequence efficient engine operation and airframe aerodynamics were key factors affecting overall mission performance. For a regional mission, terminal area operations and route flexibility become increasingly important when compared to the HALE mission. As a result total fraction of mission time spent at cruise is reduced and off-design operation of the engine is of increasing importance. Additionally, increased powerplant weight is likely to have a greater impact on overall mission performance. Also, turboelectric operation through the use of DC architecture can improve the off-design operation of propulsors being operated for the regional mission. Considering these factors the following options were considered for the regional mission:

- Propulsors – Propellers
- Airframe – T&W
- Transmission – DC and Geared
 - o Generators – Halbach array and fully superconducting
 - o Motors – Halbach array and fully superconducting
 - o Power Converters – Ambient and cryogenic
 - o Transmission Lines – Copper and HTS
- Energy Storage – All-fuel and hybrid
 - o Fuel – Kerosene
 - o Electric Energy Storage – Batteries
- GT – Turboshaft
- Cooling – Cryocooled
- Physical Synergies – Wing bending relief and slipstream lift augmentation

Batteries are considered ahead of flywheels because at present they achieve higher specific energies than present day state-of-the-art flywheels (200 W.h/kg versus 100 W.h/kg) [15] and have much lower self-discharge rates [19]. However with current future estimates, flywheels are expected to have much higher specific energies than the best known battery technology today in Lithium-Air batteries. 2700 W.h/kg is the estimated specific energy possible for a carbon nanotube flywheel with superconducting bearings [19] against 1700 W.h/kg estimated for Lithium-Air batteries [20]. This novel flywheel design is expected to achieve comparable self-discharge rates as typical batteries of today.

Compared to an aircraft flying a regional mission, the longer distances a typical aircraft flies for a medium haul mission means flight at transonic speed is essential for economic operation. Also for longer flight distances there are increases in overall flight time and time spent at cruise. As a result cruise efficient operation increasingly becomes a concern over increased powerplant weight and flying at transonic speeds rules out the use of distributed propellers.

Considering these factors the following options were considered for the medium-haul mission:

- Propulsors – Ducted Fans
- Airframe – T&W
- Transmission – DC and Geared
 - o Generators – Fully Superconducting
 - o Motors – Fully Superconducting
 - o Power Controllers – Cryogenic
 - o Transmission Lines – HTS
- Energy Storage – All-Fuel and Hybrid
 - o Fuel – Kerosene and Hydrogen
 - o Electric Energy Storage – Batteries
- GT – Turbojet (power extraction for propulsors from LP shaft) and Turbofan
- Cooling – Cryocooled and Cryogen Cooled
- Physical synergies – wing bending relief and empennage size reduction

Using the Airbus A320-200 as a reference (length of airframe = 37.6 m) [21], at 11280 m and Mach 0.78 an appreciable boundary layer (BL) height of around 30 cm at the rear of the fuselage is expected using Schlichting's models for boundary layer formation over a flat plate [22]. Although this makes the employment of BLI propulsors an attractive proposition for a medium haul mission, with BLI assessment already carried out in a number of previous studies, focus was made on assessing hydrogen-use and hybrid electric operation.

3.4 Synthesis of Propulsion System Architectures

From technology options identified suitable propulsion system architectures need to be synthesized to assess propulsion system performance and to understand how TeDP and HeP system architectures affect aircraft design and performance. Depending on the way the propulsion system architecture is characterised, its operation and performance can greatly differ if characterised

differently. To systematically understand the effect different system architectures have on overall performance the following procedure for system architecture synthesis is followed:

1. Define which identified sub-systems characterise the propulsion system
2. Synthesize propulsion system architectures from previously defined technology options at the sub-system level. With the emphasis on TeDP and HeP, these architectures are to be characterised by energy storage and transmission system technology options

3.5 Multidisciplinary Modelling

The employment of TeDP will result in the propulsion system having a greater impact on aircraft (1) noise, (2) aerodynamics, (3) structures, (4) design, (5) weights and (6) flight dynamics and stability. In addition to this the tighter coupling of the airframe and the propulsion system means the design and sizing of the aircraft will have a greater impact on propulsion system performance than for a conventional architecture. Henceforth at the sub-system level numerical modelling needs to adequately take into account the physics of the disciplines previously listed so that the impact of novel technology integration can be better assessed and understood. The multidisciplinary modelling here aims to focus on the potential of TeDP on improving overall vehicle performance and to understand system interrelationships. Unfortunately due to time constraints the effects of TeDP on vehicle noise, flight dynamics and stability is not to be considered due to project time constraints and is beyond the scope of the research.

For the analysis of TeDP system performance, the TeDP system has to be assessed at different system levels. For investigation in the present study, the levels of interest are the propulsion system level and the aircraft level. Due to the number of sub-systems in play when assessing TeDP system performance, the sub-systems, disciplines and technologies that characterise the propulsion system level and aircraft level need to be identified. For assessment at the propulsion system level the disciplines/sub-systems that affect modelling are listed below:

- Propulsor sizing & performance
- Transmission system
- Gas Turbine sizing & performance
- Energy Storage
- Cooling System
- Synergistic Technologies

For assessment at the aircraft level the disciplines that affect modelling are listed below as:

- Airframe Sizing
- Technology Integration
- Flight Performance

For modelling at the aircraft level, assessment of flight stability and control is beyond the scope of current work.

3.6 Systems Integration

The investigation of Turboelectric and Hybrid Electric distributed propulsion architectures brings into play a high degree of complexity due to the large number of possible interactions between sub-systems. To adequately integrate the number of technologies associated with TeDP and HeDP a suitable approach to systems integration is needed to simplify system assessment. For the approach in the present study a top-down approach is adopted to reduce the complexity involved in evaluating TeDP and HeDP systems. The steps used in this approach are listed below

1. Characterising the propulsion system with suitable parametric descriptors in a way that allows for comparison between alternative system architectures
2. From sub-systems defined, establish modes of operation of the integrated system
3. Determining relationships and interactions between the propulsion system and the aircraft to determine propulsion system performance

parameters that directly affect aircraft performance i.e. propulsion system-aircraft integration

4. With propulsion system outputs assess integrated system performance at the aircraft level

3.6.1 Propulsion System Characterisation

In assessing the performance of a TeDP, consideration has to be made to the fact that thrust can be generated from both the core gas turbine and the propulsors. A useful way of characterising this feature of the turboelectric system is by the definition of thrust split, TS , between the propulsors and core gas turbines (See Eq. 3-1 where $F_{propulsor}$ is the thrust produced by the propulsor and F_{net} is the overall net thrust produced by the propulsion system). The definition of a TS is a good option in representing the size of the propulsion system as the size of the core GT and propulsors are roughly proportional to the thrust they produce [3]. Additionally by varying the TS an optimum distribution of energy/power between the core GTs and distributed propulsors can be achieved [23].

$$TS = \frac{F_{propulsor}}{F_{net}} \quad 3-1$$

In accounting for the performance of a hybridised TeDP system (HeDP), an assessment needs to account for the propulsor being driven by power from two energy sources. In [24] it was argued that irrespective of vehicle type a suitable elaboration of any dual-energy-storage-propulsion-system requires two parametric descriptors involving ratios between the electric energy source to that of the entire dual-energy-storage-propulsion-system. These are H_E (Eq. 3-2 where E_{ES} is the amount of energy stored in the electric energy source and E_{tot} the total energy stored on board of the aircraft) and H_P (Eq. 3-3 where $PW_{ins,max,tot}$ is the total maximum installed power of the HeP system and $PW_{ins,max,ES}$ is the maximum installed power of the electric energy source). H_E compares the energy stored in the electric energy source to the total energy stored on board, and H_P compares the maximum installed power of the electric energy source to the total maximum installed power of the HeP system.

$$H_E = \frac{E_{ES}}{E_{tot}} \quad 3-2$$

$$H_P = \frac{PW_{ins,max,ES}}{PW_{ins,max,tot}} \quad 3-3$$

3.6.2 Modes of Operation

From sub-systems identified, TeDP systems can operate in a different number of modes. By classifying these different modes of operation, performance assessment and design space exploration can be simplified and undertaken in a systematic manner. From sub-systems defined with relevant technology options, the list of all the possible modes of operation of the turboelectric system are listed below:

1. Turboelectric operation with thrust generation from propulsor only
2. Turboelectric operation with thrust generation from propulsor only and with cryocooling
3. Turboelectric operation with thrust generation from propulsor only and with cryogen cooling
4. Turboelectric operation with thrust generation from propulsor and GT
5. Turboelectric operation with thrust generation from propulsor and GT, and with cryocooling
6. Turboelectric operation with thrust generation from propulsor and GT, and with cryogen cooling
7. Hybrid operation with thrust generation from propulsor only
8. Hybrid operation with thrust generation from propulsor only and with cryocooling
9. Hybrid operation with thrust generation from propulsor only and with cryogen cooling
10. Hybrid operation with thrust generation from propulsor and GT
11. Hybrid operation with thrust generation from propulsor and GT, and with cryocooling
12. Hybrid operation with thrust generation from propulsor and GT, and with cryogen cooling

13. All-electric operation with thrust generation from the propulsor
14. All-electric operation with thrust generation from the propulsor and with cryocooling
15. All-electric operation with thrust generation from the propulsor and with cryogen cooling

All-electric operation is not considered further because with batteries as the electric energy storage option of choice, considering lithium-ion batteries, best case estimate for specific energy from literature read is 1500 W.h/kg [25]. With this specific energy battery weights are still expected to be too high with maximum range an all-electric aircraft can fly limited. Additionally thrust generation from the propulsor only is not considered as system performance at the aircraft level is expected to be better when thrust is generated from both the propulsor and core GT(s) when aerodynamic integration issues and presence of electric equipment are considered [2], [3], [4] & [23]. This is because for a given thrust requirement, TET and OPR; increase in thrust split is associated with an increase in propulsor size and a reduction in core GT size. Although propulsive efficiency is expected to increase with an increase in propulsor size, associated electric machinery such as motors and power controllers will increase and this will impose a weight penalty on the aircraft which counteracts propulsive efficiency increase. Also, increase in propulsor size will increase propulsion system drag penalty on an airframe. Additionally, if GT size is overly reduced, it is expected that turbomachinery component efficiencies will fall due to worse aerodynamic performance [26] and this will eventually lower GT thermal efficiency.

3.6.3 Propulsion System – Aircraft Integration

Typically a new aircraft design is launched when it is perceived that there is a requirement to fulfil a need beyond the capability of existing aircraft [27]. For the present research effort this need is the aspiration to design aircraft that meet ambitious future environmental such as CO₂ emissions, NO_x emissions, noise targets and fuel burn targets. For a given mission the typical performance requirements that need to be specified are listed below:

- Range
- Payload
- Maximum speed
- Take-off and landing field length limitations
- Climb performance

Mission requirements determine speeds for the aircraft at different flight phases, and aircraft sizing and flight performance determine aircraft thrust requirements at different flight phases. Typically for GT engines for civil transport, speed requirements and thrust requirement of the aircraft at different mission flight phases are used for the design of propulsion system [28]. For a given GT, key performance parameters needed for the assessment of aircraft performance are SFC, weight, size, location on the airframe, noise and emissions [28] & [29]. In considering the performance of TeDP systems however, potential integration synergies subtly change the relationship between propulsion system and the aircraft. For the case where the propulsors are propellers, accounting for propeller-wing interaction effects means the characteristics of the slipstream emanating from needs to be accounted for in integrated performance assessment. Taking into account propeller-wing interaction, propulsion system characterisation and the different modes the propulsion system can operate, Figure 3-4 shows a summary of the systems integration procedure created and used for assessment of propulsion system performance at the aircraft level.

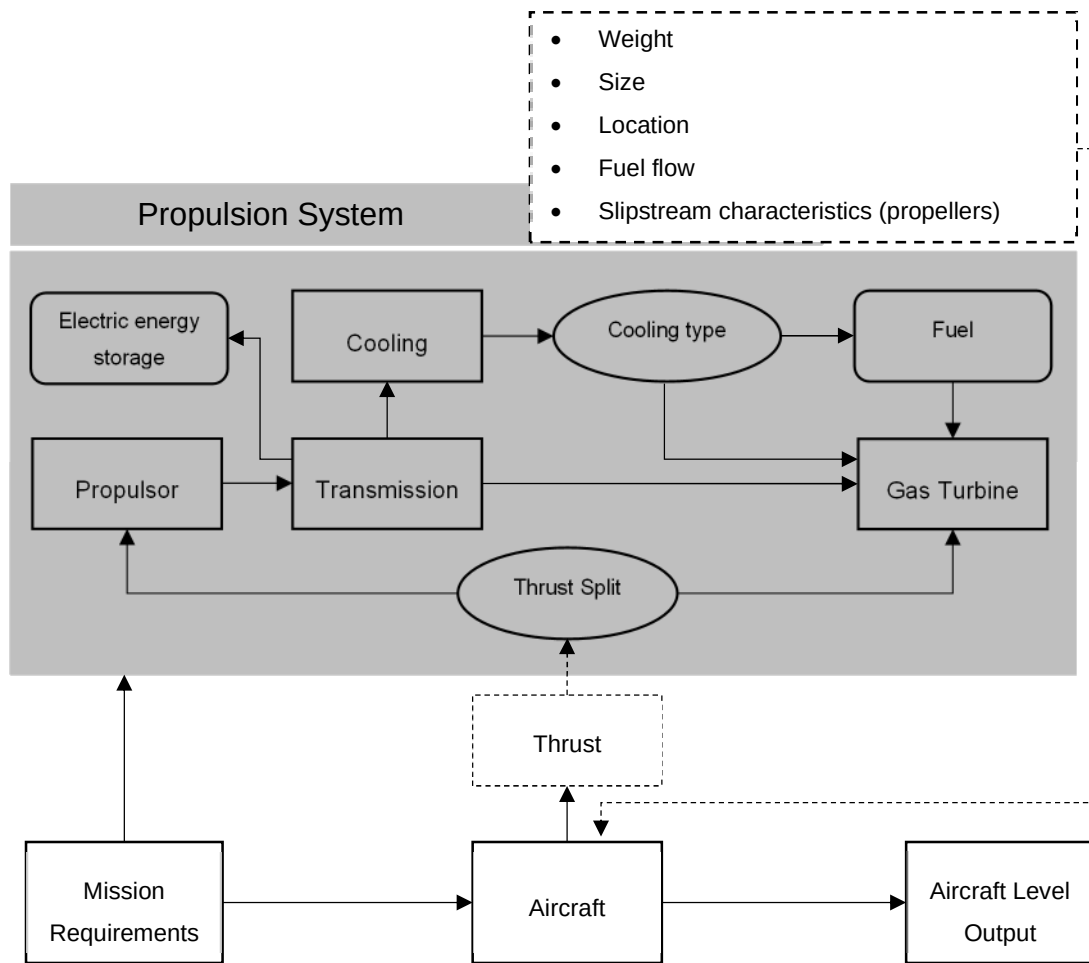


Figure 3-4 Systems integration overview for assessment of TeDP & HeP performance at the aircraft level

3.7 Configuration Assessment

With TeDP highlighted as a concept that can bring forth performance improvements for aircraft in the 2035-2050 timeframe over present day state-of-the-art technology; a suitable baseline configuration needs to be defined as a benchmark to adequately quantify TeDP system benefits and limitations over the baseline. The baseline can either be a present day state-of-the-art configuration or an advanced version of a conventional architecture depending on what type of technology scenario is to be assessed. In assessing the system it would be possible to model each sub-system and couple them with an optimiser so that an optimum system can be determined for a given set of constraints. However the automated nature of optimisers leads to a black-box

approach that would not highlight the trade-off between opposing effects e.g. the trade-off between reduced SFC and an increase in engine weight when the bypass ratio is increased [5]. As such a parametric approach is employed for the conceptual assessment of TeDP and HeP architectures. The parametric approach is used to assess performance from a component level to the aircraft level and enables the assessment of alternate configurations compared to a defined baseline for a given set of performance metrics (See Figure 3-5).

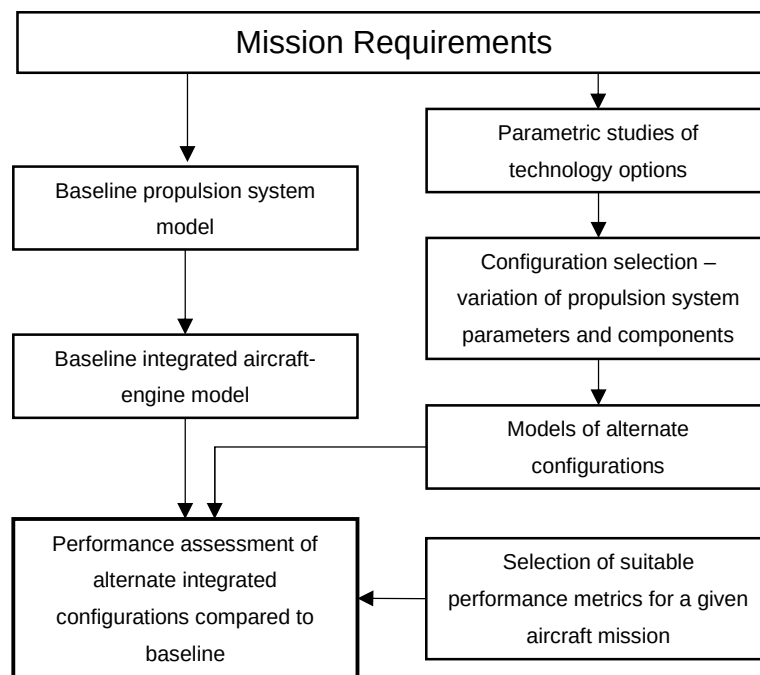


Figure 3-5 Configuration Assessment

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4 PROPULSION SYSTEM MODELLING

4.1 Propulsion System Architecture Synthesis

As established in the methodology, TeDP system performance is a function of propulsor configuration, airframe type, GT performance, electric architecture, energy storage and mission profile. Since the propulsion system is defined to be comprised of the GT, transmission system, propulsors, energy storage and cooling system; propulsion system architectures can be synthesized from identified technology options. The technology options for energy storage, transmission, GT and propulsors are shown in Table 4-1.

Table 4-1 Propulsion system sub-systems technology options

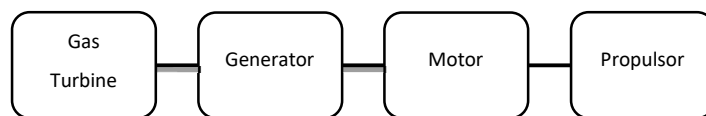
Energy Storage	Transmission	Gas Turbine	Propulsors	Cooling
All-Fuel	AC	Turboshaft	Ducted Fans	Cryocooled
Hybrid	DC	Turbofan	Propellers	Cryogen cooled
	Geared	Turbojet		

For propulsion system architecture synthesis, propulsion system architecture is to be characterised by energy storage technology option and transmission technology option. With this characterisation the following propulsion system architectures are synthesised:

- Propulsion System Architecture #1 All-Fuel/AC (AC Turboelectric)
- Propulsion System Architecture #2 All-Fuel/DC (DC Turboelectric)
- Propulsion System Architecture #3 All-Fuel/Geared
- Propulsion System Architecture #4 Hybrid/AC*
- Propulsion System Architecture #5 Hybrid/DC (DC Hybrid Turboelectric)

- Propulsion System Architecture #6 Hybrid/Geared (Geared Hybrid Electric)

The schematics of the synthesized architectures under consideration are shown in Figure 4-1. From these architectures it should be noted that Propulsion System Architectures (PSA) #3 and PSA#6 are not a TeDP configurations as the capability of distributing multiple propulsors through electric power transmission for synergistic performance improvement is lost. PSA#4 is not shown in Figure 4-1 because this architecture as defined from previous studies in [1], is not electronically possible because the ability of the electric energy storage device to interface with the main electric network is lost. For the geared architectures an additional classification needs to be made for propulsor type. PSA#3 with a ducted fan is essentially a geared turbofan and PSA#3 with propellers is a geared turboprop, and PSA#6 is the parallel hybrid version of both.



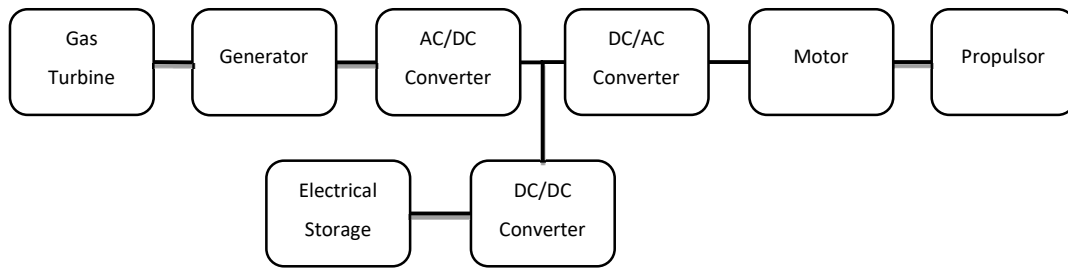
(a)



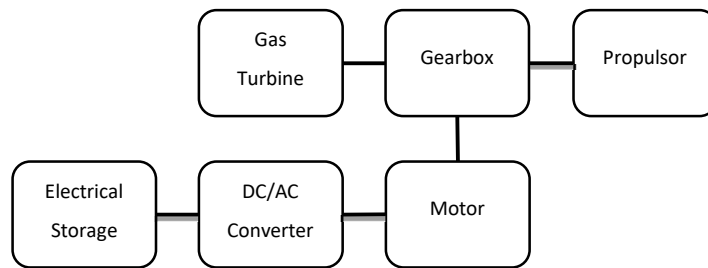
(b)



(c)



(d)



(e)

Figure 4-1 Synthesized PSAs (a) PSA#1 – AC Turboelectric (b) PSA#2 – DC Turboelectric (c) PSA#3 – Geared (d) PSA#5 – DC Hybrid Turboelectric (e) PSA#6 – Geared Hybrid Electric

For the hybrid electric architectures (PSA#5 and PSA#6) the electric energy storage device from technology options is identified to be a DC device. Thus for PSA#5, a DC/DC converter is defined to interface the electric energy storage device to the main electric network for the architecture. For the parallel hybrid architecture (PSA#6), since the motor is considered to be an AC device, a DC/AC converter is considered instead of a DC/DC converter.

For a given PSA the following rules were applied:

- Number of generators is equal to number of GTs defined
- Number of AC/DC converters is equal to number of GTs defined
- Number of DC/AC converters is equal to the number of propulsors defined
- Number of motors is equal to the number of propulsors defined
- Number of gearboxes is equal to number of GTs defined

- Number of DC/DC converters equal to the number of electric energy storage devices

From technologies and synergies identified the following PSAs are considered for each mission:

- Regional PSA#2, PSA#3, PSA#5
- Medium PSA#3, PSA#5, PSA#6

4.2 Propulsors

4.2.1 Ducted Fans

In order to simulate the performance of a decoupled ducted fan, a numerical tool was created. For the modelling of the fan, it is assumed that the ducted fan operates in an International Standard Atmosphere (ISA) which is modelled using equations defined in ESDU 77022 [2] and its structure consists of an inlet duct, fan, motor duct and a convergent nozzle [3] (see Figure 4-2). For the modelled ducted fan it is assumed that the mass flow ratio, *MFR*, of the intake is equal to 1. The definition of *MFR* is shown in Eq. 4-1 where A_{∞} is free-stream cross-sectional area at station ∞ and A_{10} is intake highlight area. For higher accuracy, calculations should take into account how *MFR* is likely to change throughout a flight envelope. With a design *MFR* of one at top-of-climb or ceiling, *MFR* is likely to have a value greater than one at take-off and a value less than one during cruise.

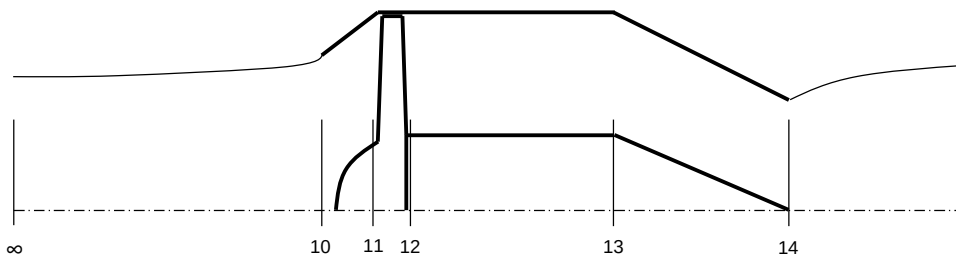


Figure 4-2 Ducted fan schematic

$$MFR = \frac{A_{\infty}}{A_{10}}$$

4-1

For the accurate calculations of thermodynamics properties at each station to within 0.25%, enthalpy and entropy polynomials are used [4]. Using enthalpy and entropy polynomials; the ratio of specific heats, γ , isobaric specific heat capacity, C_P , specific enthalpy, H and entropy polynomial temperature function F_T , for a given thermodynamic station are defined in appendix A. The parameters are defined for specific gas constant, R , of dry air having a value of $287 \text{ Jkg}^{-1}\text{K}^{-1}$.

For a number of podded distributed propulsors, $n_{propulsor}$, the mass flow rate for an individual fan unit, W_{fan} , is determined in Eq. 4-2 where $W_{propulsor,tot}$ is the total mass flow through a distributed propulsor array.

$$W_{fan} = \frac{W_{propulsor,tot}}{n_{propulsor}} \quad 4-2$$

Total inlet area for a distributed propulsor array, A_{inlet} , is determined in Eq. 4-3 where ρ is the density and U is air speed.

$$A_{inlet} = \frac{W_{propulsor,tot}}{\rho_{10}U_{10}} \quad 4-3$$

At station 10 using isentropic flow equations, the total temperature, T_T , is determined in Eq. 4-4 where M_{10} is the inlet Mach number and T_S is static temperature.

$$T_{T,10} = T_{S,10} \frac{2 + (\gamma - 1)M_{10}^2}{2} \quad 4-4$$

Total pressure, P_T , at station 10 is determined in Eq. 4-5, where P_S is static pressure

$$P_{T,10} = P_{S,10} \left(\frac{T_{T,10}}{T_{S,10}} \right)^{\frac{\gamma}{\gamma-1}} \quad 4-5$$

At station 11 assuming an adiabatic diffuser, T_T and total specific enthalpy, H_T , at station 11 are equivalent to the values at station 10 (see Eq. 4-6 and Eq. 4-7)

$$T_{T,11} = T_{T,10} \quad 4-6$$

$$H_{T,11} = H_{T,10} \quad 4-7$$

P_T at station 11 is determined as a function of P_T at station 10 and the inlet pressure recovery, IPR , in Eq. 4-8

$$P_{T,11} = P_{T,10} IPR \quad 4-8$$

For flow of air through the fan, P_T at station 12 is estimated as the product of P_T at station 11 and the propulsor FPR. To estimate the T_T at station 12 the change in entropy across the fan has to be determined. However the change in entropy is a function of the total temperature at the fan exit station. Henceforth an iterative procedure is used to determine T_T and H at station 12 using entropy and enthalpy polynomials as described in [4].

For the motor duct between the fan and the nozzle, Fanno flow is assumed which means that the total temperature in the duct remains constant and that there are frictional losses in the duct. To account for losses due to friction, a duct pressure loss, DPL , is defined and P_T at station 13 is determined as a function of DPL and station 12 P_T in Eq. 4-9.

$$P_{T,13} = P_{T,12}(1 - DPL) \quad 4-9$$

To check for whether the nozzle is choked the critical static pressure, $P_{S,cr}$ first needs to be determined (see Eq. 4-10). If the $P_{S,cr}$ is greater than the ambient P_S the nozzle is choked, otherwise the nozzle is unchoked. If the flow is choked $P_{S,14}$ is equal to $P_{S,cr}$ otherwise $P_{S,14}$ is equal to the ambient pressure.

$$P_{S,cr} = P_{T,13} \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma}{\gamma - 1}} \quad 4-10$$

When the nozzle is unchoked M at the nozzle exit is determined in Eq. 4-11 as a function of the nozzle pressure ratio, NPR . For the choked nozzle case M_{14} is equal to one.

$$M_{14} = \sqrt{\frac{2NPR^{\frac{\gamma-1}{\gamma}} - 2}{\gamma - 1}} \quad 4-11$$

With M at the nozzle exit determined, T_S at the nozzle exit is determined in Eq. 4-12

$$T_{S,14} = \frac{2T_{T,13}}{2 + (\gamma - 1)M_{14}^2} \quad 4-12$$

The ρ at the nozzle exit is estimated from the ideal gas equation in Eq. 4-13

$$\rho_{14} = \frac{P_{S,14}}{RT_{S,14}} \quad 4-13$$

U at the nozzle exit is determined as a function of exit M and T_S in Eq. 4-14

$$U_{14} = M_{14}\sqrt{\gamma RT_S} \quad 4-14$$

For the determination of propulsor thrust using the “overall net thrust” definition the expansion of the jet leaving the nozzle needs to be accounted for. Using the nozzle performance polynomial in [5] supplied by G Doulgeris a nozzle thrust coefficient, C_X , is calculated as a function of NPR in Eq. 4-15.

$$\begin{aligned} C_X = & -0.00000005NPR^6 + 0.00000412NPR^5 - 0.00012692NPR^4 \\ & + 0.00192109NPR^3 - 0.01434825NPR^2 + 0.03787371NPR \\ & + 0.94732668 \end{aligned} \quad 4-15$$

For a given thrust requirement, F , if the nozzle is unchoked W_{fan} is determined in Eq. 4-16 and for the choked case in Eq. 4-17 [4].

$$W_{fan} = \frac{F}{n_{propulsor}(C_X U_{14} - U_{\infty})} \quad 4-16$$

$$W_{fan} = \frac{F}{n_{propulsor} \left[(C_X U_{14} - U_{\infty}) + \frac{P_{S,14} - P_{S,10}}{\rho_{14} U_{14}} \right]} \quad 4-17$$

With W_{fan} determined the nozzle exit area A_{exit} (Eq. 4-18) and total propulsor mechanical power $PW_{propulsor,tot}$ (Eq. 4-19) can be determined.

$$A_{exit} = \frac{W_{fan}}{\rho_{14} U_{14}} \quad 4-18$$

$$PW_{propulsor,tot} = n_{propulsor} W_{fan} (H_{T,12} - H_{T,11}) \quad 4-19$$

At the design point by the specification of a flight condition (altitude and M), IPR , thrust requirement, DPL , FPR and η_{fan} ; the created tool via Eqs. 4-2 to 4-19 determines $W_{propulsor,tot}$, $PW_{propulsor,tot}$, A_{inlet} , A_{exit} and thermodynamic data at each fan station.

The tool was validated by comparing the design point performance of a ducted fan modelled with the tool coupled with a turboshaft modelled in Turbomatch 2.0 to a thermodynamically equivalent separate-streams-turbofan modelled in Turbomatch 2.0 (see Figure 4-3). Turbomatch is a validated engine performance code that is used for simulation of the steady-state and transient performance of a gas turbine [6] with multi-fuel capability. For the turboshaft coupled to the ducted fan, it is configured to produce a residual thrust via a convergent nozzle. Design flight condition for both propulsion systems under consideration is specified for an altitude of 15,240 m and Mach number of 0.3. The cycle is defined is for a TET of 1,200K, core OPR of 20 and BPR of 10. A summary of the engine performance parameters at the design point of both engines including the cycle parameters and flight condition is shown in Table 4-2.

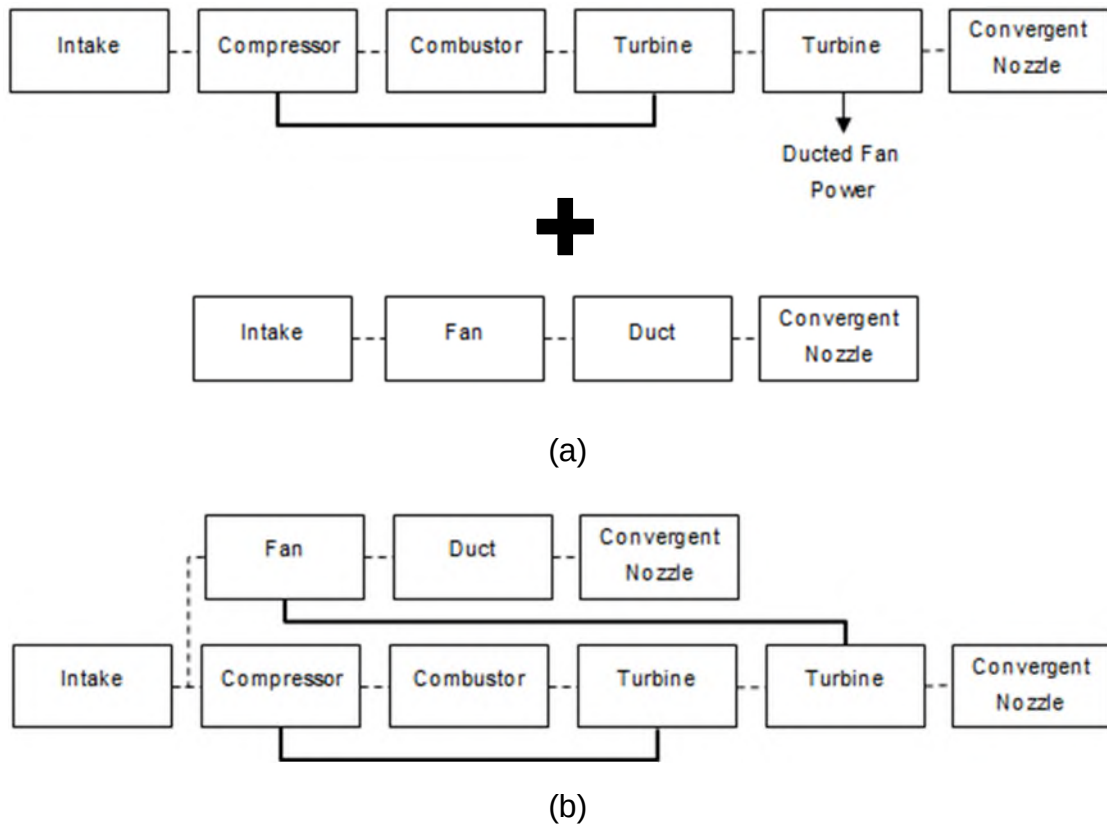


Figure 4-3 Architectures used for ducted fan validation (a) Turboshaft with ducted fan (b) Separate stream turbofan

Table 4-2 Ducted fan tool validation engine performance parameters

Parameter	Value
Altitude	15240 m
Mach	0.3
TET	1200 K
OPR	20
BPR	10
Mass flow rate	22 kg/s
IPR	1.0

Fan isentropic efficiency	0.87
Bypass duct pressure loss	0.002
HPC pressure ratio (PR)	20
Burner pressure loss	0.04
Combustion efficiency	0.99
High Pressure Turbine (HPT) isentropic efficiency	0.88
Low Pressure Turbine (LPT) isentropic efficiency	0.88
Transmission efficiency	1.0

A summary of the results of the simulation of the Turbomatch turbofan and ducted fan coupled with a Turbomatch turboshaft is shown in Figure 4-4. In Figure 4-4 the increase in FPR increases the “cold thrust”. As the cold thrust is increased more power is extracted from the core flow of the turbofan and turboshaft. This has the effect of reducing “hot” thrust from the core flows and since TET is fixed with the fuel flow rate remaining constant for all FPRs the SFC falls. However as more power is extracted from the core flow the hot thrust reduces towards zero and the effect of hot thrust reduction dominates which causes overall net thrust to reduce and as a consequence SFC. For all FPR’s considered, the propulsion system that comprises of a turboshaft and a ducted fan produces similar amounts of thrust as the turbofan. Mass flow between bypass and core flows being fixed by BPR and equivalent fan design parameters, results in nearly equal power extraction from core flow. This nearly equal power extraction from core flow for both configurations results in both propulsion system configurations producing similar amounts of thrust. For the simulations carried out the mean absolute error between the simulated points is 0.021% with the sources of this error being the difference in the calculated

ambient atmospheric pressure and differences in the values of the nozzle thrust coefficient when the nozzle is choked.

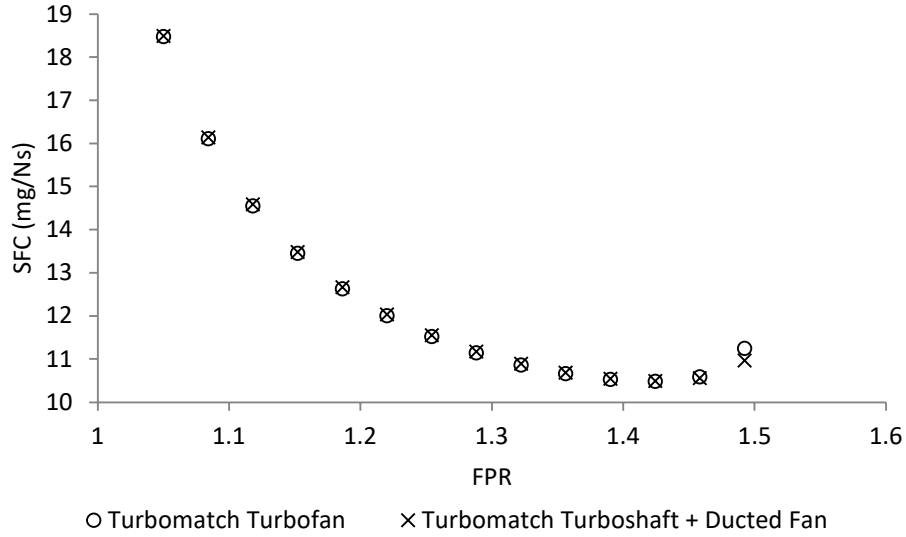


Figure 4-4 Ducted fan tool validation

For the off-design performance assessment of the ducted fan an approach is needed that accounts for the performance of the fan at all flight conditions for integrated aircraft-propulsion system performance. For a fixed nozzle area fan the following method was employed:

1. A design point flight condition is fixed
2. Ducted fan design point calculations are carried out to determine $A_{exit,DP}$ and $W_{fan,DP}$
3. A fan map is created using Turbomatch by specifying ducted fan design point flight condition, FPR_{DP} , $\eta_{fan,DP}$, DPL and $W_{fan,DP}$
4. A range of flight M is defined
5. Selection of a flight M
6. Selection of a relative rotational speed (PCN) line on the created fan map
7. From defined values of corrected non-dimensional mass flow (WAC), FPR and η_{fan} ; through thermodynamic calculations for each ducted fan station, A_{exit} is calculated for each operating point on the PCN line

8. The two operating points where calculated A_{exit} is closest to $A_{exit,DP}$ are determined and interpolation between these points is carried out to determine operating WAC, FPR and η_{fan} for the given PCN line
9. Steps 6 to 9 are repeated for all fan map PCN lines to create the operating line for a given M
10. Steps 5 to 9 are repeated until all flight M s are considered to create fan operating lines for each M . Fan off-design performance is then defined by the created operating lines

It should be noted that for the procedure listed above, steps 6 to 8 work for the modelling of a fan from near above-idle speeds and during unchoked operation. At low speeds (particularly during sub-idle operation) the relationship between FPR and WAC becomes collinear (horizontal on the fan map) which means there would more than one value of WAC for a given FPR. This makes direct interpolation of parameters on a speed line problematic. Similarly at sufficiently high enough speeds where the flow through a fan chokes, the relationship between FPR and WAC becomes collinear as well, with a vertical line on the fan map for a given speed created. This means there would be more than one value of FPR for a given WAC. For a more extensive computation of the fan map, beta lines should be used on the fan map for assessment of fan performance at low and high speeds, with interpolation carried out on created beta lines.

For sizing of the fan the procedure shown in Figure 4-5 is employed. A code in MATLAB was created that carries out the procedure. In the procedure through fan design inputs and thermodynamic station data; fan size and speed are estimated by guessing a mean stage loading (see Eq. 4-20 for definition where n_{stage} is the number of fan stages, $U_{mean,fan}$ is fan mean blade speed and MSL is mean stage loading [4]) within a suitable range to meet a tip Mach number target. In the procedure although a multi-stage fan could be assessed, for studies carried out a single stage fan was considered since fans with relatively high FPRs were not assessed. In the procedure in Figure 4-5; inlet hub-to-tip

ratio, inlet Mach number and exit Mach number are specified based on typical values from [4].

$$MSL = \frac{H_{T,fan,exit} - H_{T,fan,in}}{U_{mean,fan}^2 n_{stage}} \quad 4-20$$

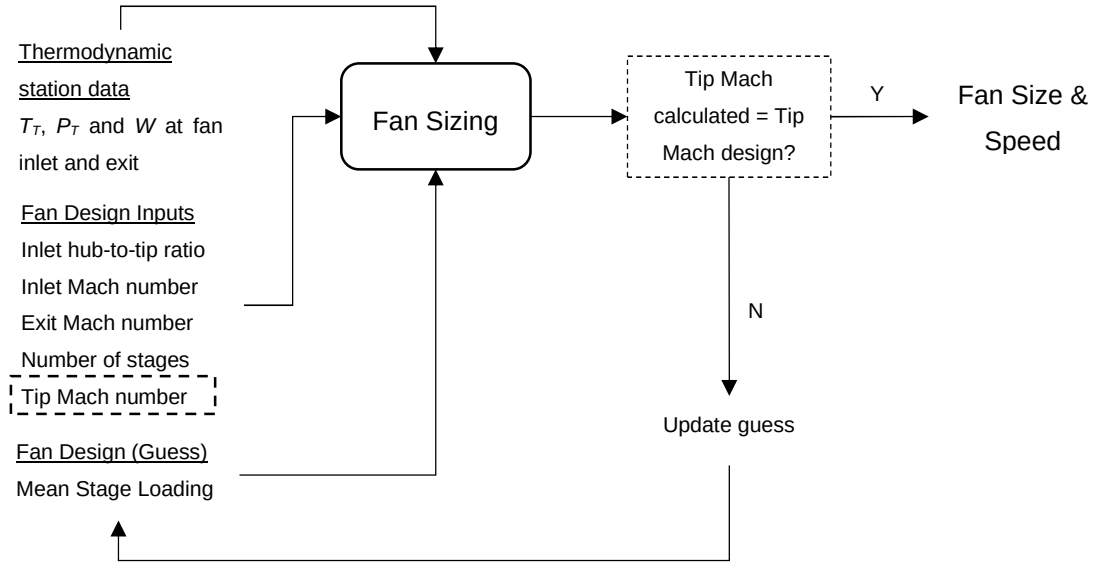


Figure 4-5 Fan sizing procedure

To size the fan T_S and P_S at the fan face first needs to be determined. Thus static temperature at the fan face, $T_{S,fan,in}$, is determined in Eq. 4-21.

$$T_{S,fan,in} = \frac{2T_{T,fan,in}}{(2 + (\gamma - 1)M_{fan,in}^2)} \quad 4-21$$

$P_{S,fan,in}$ is determined as a function of T and γ in Eq. 4-22

$$P_{S,fan,in} = P_{T,fan,in} \left(\frac{T_{S,fan,in}}{T_{T,fan,in}} \right)^{\frac{\gamma}{\gamma-1}} \quad 4-22$$

To determine the area of the fan, A_{fan} , the compressible flow, Q , curve and fan blockage factor, K_B , need to be determined. Eq. 4-23 is used to determine the value of the Q curve and the blockage factor is determined as a function of the number of fan stages from typical K_B values in Table 4-3 [7].

$$Q = 1000 \sqrt{\left(\frac{2\gamma}{(\gamma-1)R}\right) \left(\frac{P_T}{P_S}\right)^{\frac{-2}{\gamma}} \left(1 - \left(\frac{P_T}{P_S}\right)^{\frac{1-\gamma}{\gamma}}\right)} \quad 4-23$$

Table 4-3 Typical values for K_B

At inlet to stage number	1	2	3	4	5 and subsequent
K_B	0.99	0.95	0.92	0.9	0.88

At the fan face, $A_{fan,in}$ is determined in Eq. 4-24

$$A_{fan,in} = \frac{W_{fan} \sqrt{T_{T,fan,in}}}{K_B K_{fan,in} \cdot Q_{fan,in} \cdot \frac{P_{T,fan,in}}{1000}} \quad 4-24$$

From the determination of $A_{fan,in}$ and the specification of an fan inlet hub-to-tip ratio, $htr_{fan,in}$, the tip diameter of the fan $d_{fan,tip,in}$ is determined in Eq. 4-25.

$$d_{fan,tip,in} = \frac{4A_{fan,in}}{\pi(1 - htr_{fan,in}^2)} \quad 4-25$$

With $d_{fan,tip,in}$ determined, fan hub diameter, $d_{fan,hub,in}$, (Eq. 4-26) and mean fan diameter, $d_{fan,mean,in}$, (Eq. 4-27) are determined

$$d_{fan,hub,in} = htr_{fan,in} d_{fan,tip,in} \quad 4-26$$

$$d_{fan,mean,in} = \frac{d_{fan,hub,in} + d_{fan,tip,in}}{2} \quad 4-27$$

The same procedure is used to determine $A_{fan,in}$ is used to determine the exit area of the fan, $A_{fan,exit}$, with T_T and P_T at the fan exit used to determine $T_{S,fan,exit}$ (Eq. 4-28) and $P_{S,fan,exit}$ (Eq. 4-29).

$$T_{S,fan,exit} = \frac{2T_{T,fan,exit}}{(2 + (\gamma - 1)M_{fan,exit}^2)} \quad 4-28$$

$$P_{S,fan,exit} = P_{T,fan,exit} \left(\frac{T_{S,fan,exit}}{T_{T,fan,exit}} \right)^{\frac{\gamma}{\gamma-1}} \quad 4-29$$

With P_S and T_S at the fan exit determined with the use of the Q curve, $A_{fan,exit}$ is determined in Eq. 4-30.

$$A_{fan,exit} = \frac{W_{fan} \sqrt{T_{T,fan,exit}}}{KB_{fan,exit} \cdot Q_{fan,exit} \cdot \frac{P_{T,fan,exit}}{1000}} \quad 4-30$$

Assuming constant $d_{fan,tip}$; $d_{fan,hub,exit}$ is determined in Eq. 4-31 and with $d_{fan,hub,exit}$ determined, $d_{fan,mean,exit}$, is determined in Eq. 4-32.

$$d_{fan,hub,exit} = \sqrt{d_{fan,tip,exit}^2 - \frac{4A_{fan,exit}}{\pi}} \quad 4-31$$

$$d_{fan,mean,exit} = \frac{d_{fan,hub,exit} + d_{fan,tip,exit}}{2} \quad 4-32$$

From the initial guess of a mean stage loading and determination of total fan enthalpy at fan inlet and exit; fan mean blade speed is determined from Eq. 4-33.

$$U_{mean,fan} = \sqrt{\frac{H_{T,fan,exit} - H_{T,fan,in}}{MSL_{fan} \cdot n_{stage}}} \quad 4-33$$

With $U_{mean,fan}$ determined, fan angular speed, ω_{fan} , is determined in Eq. 4-34.

$$\omega_{fan} = \frac{2U_{mean,fan}}{d_{fan,mean,in}} \quad 4-34$$

With ω_{fan} determined, fan tip speed, $U_{fan,tip}$, (Eq. 4-35) and $M_{fan,tip}$ (Eq. 4-36) are determined.

$$U_{fan,tip} = \omega_{fan} \frac{d_{fan,tip}}{2} \quad 4-35$$

$$M_{fan,tip} = \frac{U_{fan,tip}}{\sqrt{\gamma R T_{s,fan,in}}} \quad 4-36$$

For the case where the speed the fan is to rotate at is already fixed before sizing of the fan e.g. for an AC turboelectric architecture (See Sec. 4.3.1) or a geared architecture, the procedure for fan sizing in Figure 4-5 changes to the procedure in Figure 4-6 which is also coded in MATLAB. With an input angular speed, tip Mach number and mean stage loading are calculated as outputs by re-arranging of Eqs. 4-33 to 4-36.

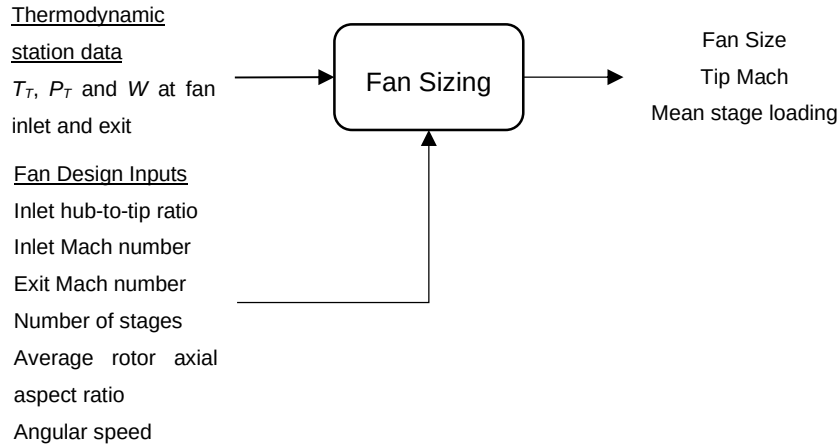


Figure 4-6 Fan sizing procedure for an input angular speed

4.2.2 Propellers

The numerical tool QMIL was used for sizing of the propeller. QMIL generates propeller geometries for the minimum induced loss condition and it uses an extension of the method in [8] to account for the radially-varying self-induction velocity which gives consistency with the heavily-loaded actuator disc limit [9]. This makes QMIL a suitable tool for the sizing of propellers for a regional transport mission since propellers for the mission role are expected to be heavily loaded. After the propeller is sized the numerical tool QPROP was used for performance assessment of a variable pitch and variable speed propeller. In assessing the designed propeller, the tool solves the overall propeller system

that is characterised by theories of Betz [10], Goldstein [11], Theodorsen [12] and Larrabee [8] with a global Newton method.

To assess propeller performance throughout the flight envelope the procedure in Figure 4-7 is employed. In the procedure before design of the propeller, an aerofoil for blade cross-section is selected with the main criterion being a desired aerodynamic profile. Then drag polar information of this aerofoil at a reference Reynold's number is obtained using the numerical tool XFOIL. The reference Reynold's number can be computed at any point on a given flight envelope for an estimated propeller chord (taken at the 75% radial point on a propeller blade). The general definition for Reynold's number utilised is shown in appendix A. Meanwhile, XFOIL [13] uses a viscous formulation in the determination of 2D lift coefficient, C_l and 2D drag coefficient, C_d at a given angle of attack, α for an aerofoil. For computation of propeller blade geometry generation for the minimum induced loss condition, 2D lift coefficient and 2D drag coefficient are calculated for a number of angles of attacks within a suitable range. This creates a drag polar for selected aerofoil at the reference Reynold's number.

From parabola fitting of the drag polar (see Figure 4-8); the zero lift drag coefficient $C_{d0,propeller}$, lift coefficient for zero lift drag, $C_{l,Cd0,propeller}$, 2D maximum lift coefficient, $C_{l,max,propeller}$, and minimum lift coefficient within range of $\alpha_{propeller}$'s considered, $C_{l,min,propeller}$, are obtained. From line fitting of the lift curve, lift coefficient at zero angle of attack, $C_{l0,propeller}$, and the lift curve slope, $C_{la,propeller}$ are obtained. A drag coefficient function C_{d2} is defined for the solution of $C_{d,propeller}$ in QMIL where QMIL uses a 2-piece quadratic function to determine $C_{d,propeller}$ as a function of reference Reynold's number, C_l and a Reynold's number scaling factor (see Eq. 4-37 where Re is Reynold's number, Re_{ref} is reference Reynold's number and Re_{exp} is the Reynold's number scaling factor). The Reynold's number scaling factor is used for the crude estimation of $C_{d,propeller}$ at off-design Reynold's numbers different to the reference Reynold's number. Since propellers are being considered for the regional transport mission, a value of -0.2 is chosen for the Reynold's number scaling factor to

represent a large propeller with significant turbulent flow. If calculated $C_{l,propeller}$ for a blade element is greater than $C_{l,Cd0}$ then C_{d2} is equal to $C_{d2,up}$ (see Figure 4-8 for definition) from the parabola fit, however if calculated $C_{l,propeller}$ for a blade element is less than $C_{l,Cd0}$ then C_{d2} is set equal to $C_{d,low}$ (see Figure 4-8 for definition).

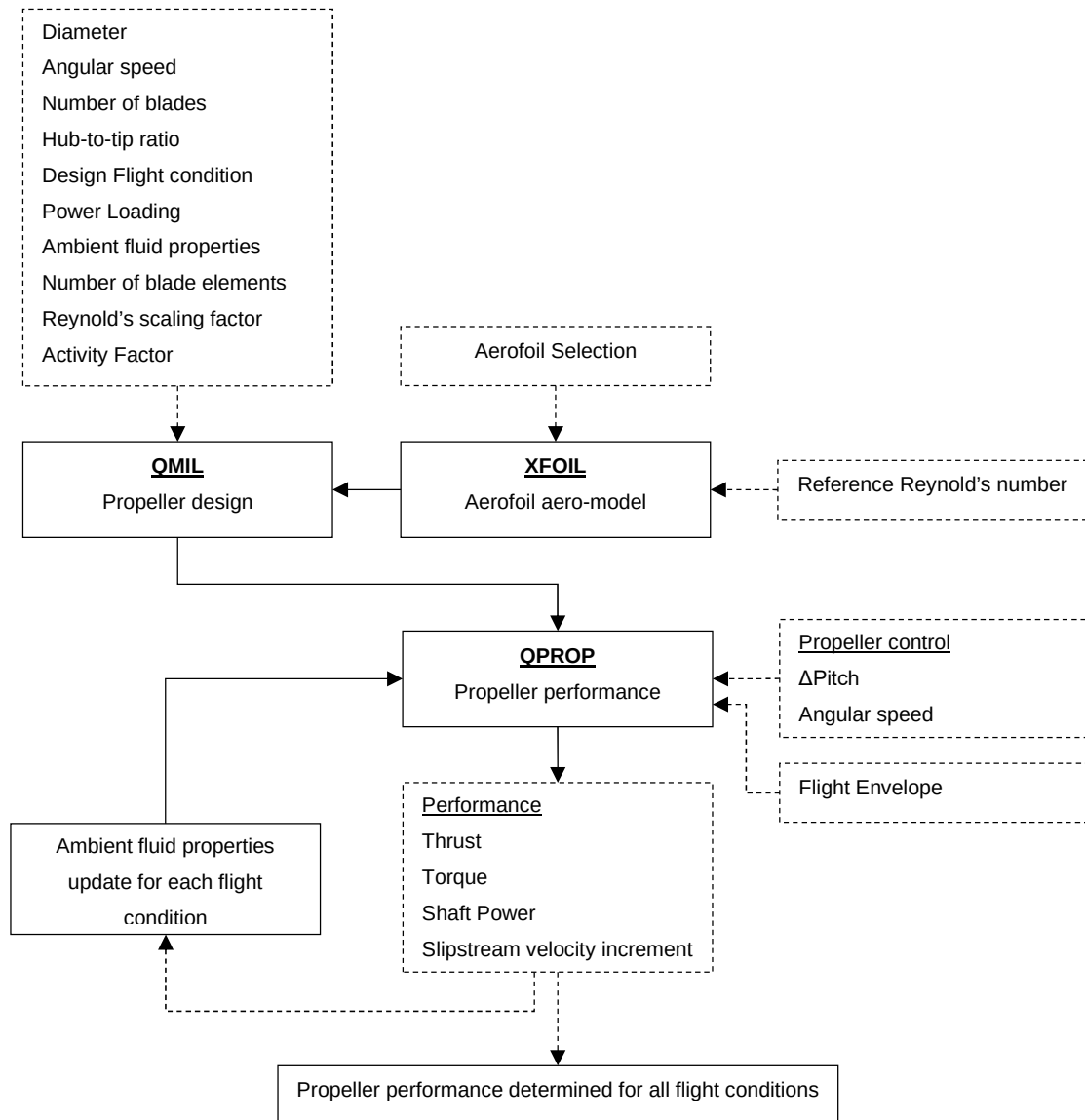


Figure 4-7 Procedure for propeller performance assessment throughout the flight envelope using QMIL and QPROP

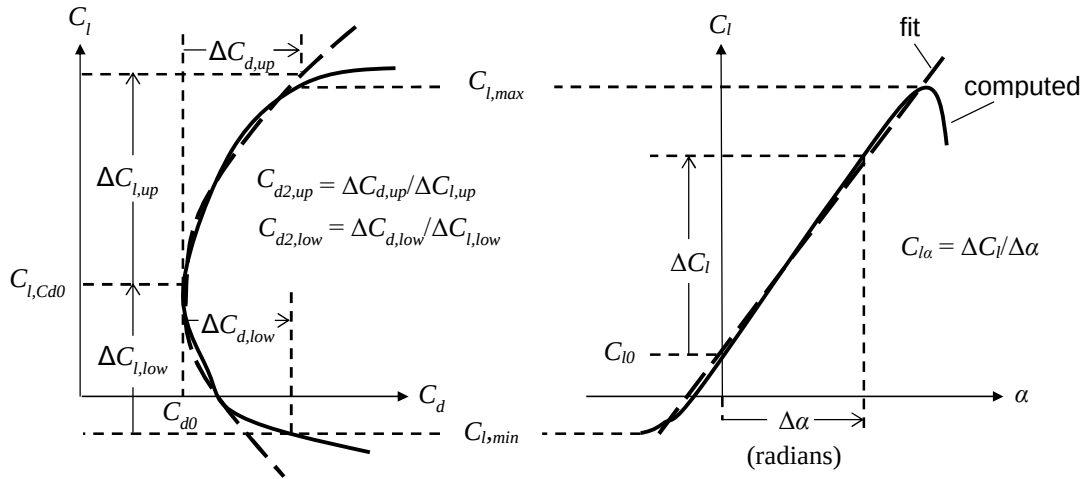


Figure 4-8 Aerofoil drag polar characterisation for QMIL

$$\begin{aligned}
 C_{d,propeller}(C_{l,propeller}, Re) &= \left[C_{d0,propeller} + C_{d2}(C_{l,propeller} - C_{l,Cd0,propeller})^2 \right] \\
 &\times \left(\frac{Re}{Re_{ref}} \right)^{Re_{exp}}
 \end{aligned}
 \tag{4-37}$$

For a specified activity factor (AF), propeller diameter, $d_{propeller}$ and propeller hub-to-tip ratio; the blade chord profile along the blade radius is characterised. The relationship between these variables is shown in the activity factor definition in appendix A. However, for sizing of the propeller in QMIL, activity factor is not defined as an input but instead blade lift coefficient profile at selected radial stations is defined as an input. With a desired lift coefficient profile initially unknown, the lift coefficient profile is guessed and then within QMIL the propeller drag profile is determined. Within QMIL once blade lift coefficient and drag profile are known; calculations are carried out to determine blade chords and pitch angles for blade elements along the blade radius to meet a shaft power requirement for specified design angular speed ($\omega_{propeller,DP}$), number of blades (n_{blades}), flight condition and ambient fluid properties (density, ρ , speed of sound, a , and dynamic viscosity, μ). With blade chords determined, an activity factor for the sized propeller blade is calculated outside of QMIL. If this calculated activity factor matches the specified activity factor to within a set

convergence margin, the propeller geometry is fixed; otherwise if it does not then the guess for the lift coefficient profile is updated and QMIL is ran again. This procedure is repeated until calculated activity factor matches specified activity factor and is coded with MATLAB (see Figure 4-9) . Once propeller geometry is fixed, by the setting of the radially-constant pitch angle change and $\omega_{propeller}$, propeller thrust $F_{propeller}$, efficiency, propeller torque, propeller shaft power and slipstream velocity increment ($\Delta U_{propeller}$) are computed with QPROP for all flight conditions on a defined flight envelope (see Figure 4-7). For the assessment of propeller performance; the definitions utilised for advance ratio, efficiency and power coefficient are shown in appendix A.

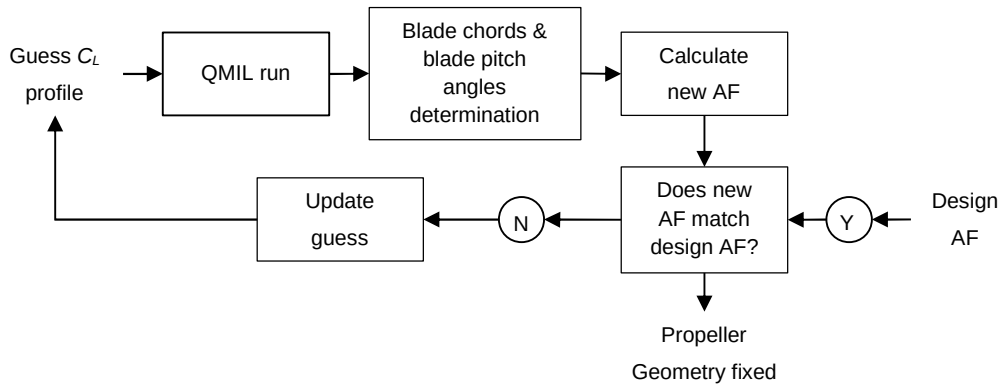


Figure 4-9 Procedure to size propeller utilising QMIL for a specified activity factor

Propeller mass is estimated using the semi-empirical correlation in [14]. In [14] the mass of an eight-bladed propeller was estimated as a function of diameter, tip speed and disc loading in Eq. 4-38 where K_1 is a curve fitting coefficient.

$$m_{propeller} = K_1 d_{propeller}^{2.48808} U_{tip}^{0.3} \left(\frac{PW_{propeller}}{d_{propeller}^2} \right)^{0.2925} \quad 4-38$$

The correlation was originally built to estimate the mass of what were state-of-the-art propellers in 1979 and at this technology level a value of 0.6235 for K_1 in SI units. Using modern technology composite propeller blades in the Dowty R408 propellers for verification purposes; the value of K_1 was amended to have a value of 0.4177. This value is determined by substituting known propeller mass, diameter, tip speed and diameter for the Dowty R408 propellers in Eq.

4-38 and solving for a new K_1 . Additionally since Eq. 4-38 is the correlation for estimating the mass of an eight-bladed propeller, the equation was modified in Eq. 4-39 so that the mass of an arbitrary propeller with n_{blades} can be determined.

$$m_{propeller} = n_{blades} \frac{K_1}{8} d_{propeller}^{2.48808} U_{tip}^{0.3} \left(\frac{PW_{propeller}}{d_{propeller}^2} \right)^{0.2925} \quad 4-39$$

4.2.3 Propulsive Device Efficiency

Typically with conventional GT architectures for aero application the propulsor is coupled with the power generating device. However with turboelectric and hybrid electric architectures this does not have to be case. This has a consequence in assessing performance at the aircraft level of hybrid electric architectures. Typically the results of flight performance calculations will yield a thrust requirement that the propulsion system has to produce and for a given thrust requirement associated SFC is determined. Usually this SFC is determined from a thrust-SFC look-up table. With thrust and SFC known, segment fuel burn for a given mission is determined. For a conventional architecture, given thrust requirement and mission point; the power produced by core of a GT does not need to be explicitly known for flight performance calculations to be undertaken if the thrust-SFC relationship is known. However, for hybrid electric architectures the presence of electric energy storage adds another variable to the assessment of mission fuel burn and thus the determination of segment fuel burn from just the two parameters of thrust and SFC is no longer sufficient. By taking into account the propulsive device efficiency, η_{PD} , as defined in [15] and shown in Eq. 4-40; the determination of propulsor power requirement for a given mission point within a given aircraft performance tool can be determined. It should be noted that for a propeller its propulsive device efficiency is equal to its propulsive efficiency which is also equal to propeller efficiency. With propulsor power requirements determined within a given aircraft performance assessment tool, for a given electric energy

storage power rating and transmission losses, GT shaft power output is more readily determined and with this the fuel flow rate of the GT can be determined as a function of GT shaft power output. With fuel flow rate determined segment fuel burn can be estimated.

$$\eta_{PD} = \frac{F_{propulsor} U_0}{PW_{propulsor}} \quad 4-40$$

4.3 Transmission

For assessment of the effect of the transmission system on system performance the following assumptions were made:

- For power transfer through the transmission system components are treated as black boxes with each having fixed efficiency. For higher fidelity, electric components should be modelled. From the modelling of the electric components performance maps should be obtained which can then be appropriately matched to GT and propulsor performance requirements. However with electric machinery transmission efficiency for a given component is not expected to vary much at off-design, a pseudo black box approach is sufficient for getting a good estimate of overall system performance at the aircraft level.
- For AC turboelectric, DC turboelectric and DC hybrid turboelectric architectures interconnectivity from power sources (GT and electric energy storage) to propulsors is assumed. With this interconnectivity each propulsor receives power from all power sources and this allows for increased transmission system redundancy and fault tolerance [16]
- Electric components in the transmission system and gearbox are rated for maximum power transferred through the component for a design mission profile. For AC turboelectric, DC turboelectric and DC hybrid turboelectric architectures; for the case when GT One-Engine-Inoperative (OEI) power output from GTs is different GT End-of-Runway (EoR) power output, with electrical system interconnectivity assumed

generators and associated AC/DC converters are rated for OEI power requirements. Meanwhile motors and associated DC/AC converters are rated for EoR power requirements.

- Power converters coupled to electric energy storage in hybrid architectures rated for maximum power discharged from electric energy storage device
- Mass of electric components estimated as the ratio of rated power to component power density or the ratio of rated torque to torque density, similar to the approach in [17]. Component power and torque densities are determined from values obtained from public literature
- Size of electric components is expected to impose a volume penalty on a given aircraft, but assessment of this is not considered here and is beyond the scope of the work. This point is mentioned as a factor that should be considered for future work.
- Centre-of-gravity of electric components on an airframe is assumed to coincide with centre-of-gravity of GT. In addition to flight stability, this consideration means that for assessment of wing bending relief, electric component mass is aggregated with GTs for the assessment.
- For the conventional architecture with a gearbox (PSA#3), gearbox mass is estimated using the empirical relation in [18] where gearbox mass is estimated as a function of output torque and gear ratio. For parallel hybrid architecture (PSA#6) however, the empirical relation in [18] is not sufficient because the relation assumes that the gearbox only has one input shaft and one output shaft. For parallel hybrid operation this is not applicable because a gearbox would need to be able to mechanically integrate inputs from two shafts (one shaft from the electric motor output and the second shaft from the gas turbine output) with the output shaft being that of the propulsor. Two options for gearbox architecture that can achieve this are the use of a summation gearbox or a differential gearbox. A differential gearbox is considered here as the technology has already been implemented in hybrid vehicles, albeit for automotive

application [19] & [20]. Assumptions, sizing approach and mass estimation for the differential gearbox are elaborated upon in appendix B.

4.3.1 AC Turboelectric

For the AC turboelectric architecture (see Figure 4-1(a)), propulsor power is determined in Eq. 4-41 where $PW_{GT,tot}$ is total GT power, η_{gen} is generator efficiency and η_{motor} is motor efficiency. For this architecture the generators and motors run synchronously where the generator rotates at the same speed as the output shaft of the GT and the propulsors rotate at the same speed as the motors. With the frequency of the motor equal to frequency of the generator, the angular speed of the motor is fixed by the angular speed of the generator with the relation in Eq. 4-42, where n_{poles} is the number of machine poles.

$$PW_{propulsor,tot} = PW_{GT,tot} \eta_{gen} \eta_{motor} \quad 4-41$$

$$\omega_{motor} = \frac{\omega_{gen} n_{poles,motor}}{n_{poles,gen}} \quad 4-42$$

4.3.2 DC Turboelectric

For the DC turboelectric architecture (see Figure 4-1(b)) propulsor power is determined in Eq. 4-43, where η_{ACDC} is AC/DC converter efficiency and η_{DCAC} is DC/AC converter efficiency. For this architecture angular speed of the propulsor is independent of the angular speed of the output shaft of the GT.

$$PW_{propulsor,tot} = PW_{GT,tot} \eta_{gen} \eta_{ACDC} \eta_{DCAC} \eta_{motor} \quad 4-43$$

4.3.3 Geared

For the geared architecture (see Figure 4-1(c)) propulsor power is determined in Eq. 4-44, where η_{GB} is gearbox efficiency. The angular speed of the propulsor is fixed by the speed of the output shaft of the GT for a given gearbox gear ratio, GR (see Eq. 4-45).

$$PW_{propulsor,tot} = PW_{GT,tot}\eta_{GB} \quad 4-44$$

$$\omega_{propulsor} = \omega_{GT,out}GR \quad 4-45$$

4.3.4 DC Hybrid Turboelectric

This architecture (see Figure 4-1(d)) is the hybridised version of the DC turboelectric architecture and it functions as a series hybrid. Power transfer to the propulsor from two energy sources is determined in Eq. 4-46, where η_{DCDC} is DC/DC converter efficiency, $PW_{ES,out}$ is the power discharged from the electric energy storage device and $\eta_{discharge}$ is the discharge efficiency of the electric energy storage device. This discharge efficiency is determined from simulation of the electric energy storage device. Similar to the DC Turboelectric architecture angular speed of the propulsor is independent of the angular speed of the output shaft speed of the GT.

$$PW_{propulsor,tot} = \eta_{motor}\eta_{DCAC}(PW_{GT,tot}\eta_{ACDC}\eta_{gen} + PW_{ES,out}\eta_{discharge}\eta_{DCDC}) \quad 4-46$$

For the case where the electric energy storage device is fully discharged during a mission; recharging of the electric storage device is possible with this architecture and undertaking this during a mission may be desirable. Aircraft range could be increased for instance. For the architecture, recharge power into the electric storage device is determined in Eq. 4-47, where $\eta_{recharge}$ is the recharge efficiency of the electric energy storage device.

$$PW_{ES,in} = \eta_{recharge}\eta_{DCDC}\left(PW_{GT,tot}\eta_{gen}\eta_{ACDC} - \frac{PW_{propulsor,tot}}{\eta_{motor}\eta_{DCAC}}\right) \quad 4-47$$

4.3.5 Geared Hybrid Electric

This architecture (see Figure 4-1(e)) is the hybridised version of the geared architecture and it functions as a parallel hybrid. Power transfer to the propulsor

from two energy sources is determined in Eq. 4-48. Angular speed of the propulsor is fixed by the angular speed of the output shaft of the GT (see appendix B).

$$PW_{propulsor,tot} = (PW_{GT,tot} + PW_{ES,out}\eta_{discharge}\eta_{DCAC}\eta_{motor})\eta_{GB} \quad 4-48$$

For the case where the propulsor is a ducted fan, the core GT, gearbox and ducted fan do not need to be considered as separate entities and instead the HeP system can be modelled as a geared turbofan receiving power intake from a battery. For this case the power intake into the turbofan is defined in Eq. 4-49. With power intake determined, a turbofan with gearbox transmission losses and a power intake is modelled with Turbomatch and HeP performance at the propulsion system level is obtained.

$$PW_{turbofan,intake} = PW_{ES,out}\eta_{discharge}\eta_{DCAC}\eta_{motor} \quad 4-49$$

4.4 Gas Turbine

With TeDP system performance being assessed at the aircraft level, GT performance throughout the flight envelope for a given mission profile needs to be determined. Turbomatch was used for the simulation of GT performance at design point and off-design for specified engine configuration, TET, OPR, flight conditions, GT component efficiencies, pressure losses, component performance maps and shaft power requirement (or thrust requirement). Considering hybrid electric performance at off-design, variable nozzles are considered to lower LP shaft surge TET for configurations where the power rating of an electric energy storage device is high enough to cause the thrust produced at LP shaft surge TET with fixed area nozzles to be higher than the minimum thrust required for a given mission point. This happens with fixed area nozzles because for a given thrust requirement, increasing electric energy storage power transfer to propulsors reduces power required from GTs and for a non-resized GT this will cause operation to move closer to surge as shaft speed reduces.

For component sizing and shaft speeds determination; components considered for the GT are intake, fans, axial compressors, centrifugal compressors, annular burners and convergent nozzles; with speeds of the fans and the compressors fixing speeds of the turbines. Component sizes and speeds were determined through a created code in MATLAB using suitable relations. For the fans, size and speed are determined in a similar manner as for a ducted fan utilising Eqs. 4-21 to 4-26. For axial compressors size and speed are determined using relations in the compressor design manual in [7] for a preliminary design compressor. For preliminary design of the compressors in [7]; compressor size and speed is determined for input thermodynamic station data, inlet hub-to-tip ratio and mean stage loading. Burner size is estimated using the relations in [4] where size is determined from thermodynamic station data and burner design parameters such as equivalence ratios, burner loading, temperature distribution and local Mach numbers. Turbines are sized using the relations in the axial turbine design manual in [21] where size is determined from thermodynamic station data, number of stages, inlet hub-to-tip ratio, mean stage loading and reaction. Assumptions and sizing procedure for the centrifugal compressor are elaborated upon in appendix C.

For the simulation of idle performance, simulation is limited to baseline mechanical configurations integrated with a propulsor (i.e. a turbofan or a turboprop with the architecture definition in Figure 4-1(c)). This is due to limitations with Turbomatch at present in simulating idle performance. Turbomatch does not solve for GT idle characteristics directly but an estimation of idle performance can be made by determining the minimum throttle that yields a valid GT off-design solution for a given GT configuration and selected component maps. For turboprops, idle is simulated by first setting GT output shaft PCN to the PCN that matches propeller idle or minimum RPM, where output shaft PCN at idle is equal to the ratio of propeller idle RPM to design point RPM. Then GT TET is reduced until GT residual thrust approaches 0. Idle TET is then fixed by the TET that causes this and performance at this operating point is read-off. For turbofan configurations, it is assumed idle LP shaft speed is approximately 60% of design LP shaft speed for 1st segment descent from

cruise altitude to 10,000 ft. For second segment descent from 10,000 ft to sea-level it is assumed LP shaft speed is approximately 40% of design LP shaft speed. It is assumed to be lower than what is the case for 1st segment descent because of lower calibrated airspeed. Thus for idle simulation, depending on the flight condition, TET is reduced until LP shaft speed is approximately 60% of LP design shaft speed for 1st segment descent and or 40% of design LP shaft speed for second segment descent. Then GT performance is read-off at the TET that results in the shaft speed being met.

4.5 Energy Storage

4.5.1 Fuels

For stored energy in the form of fuel, the rate of energy consumption is directly proportion to the rate at which fuel is burned in the GT. For a given aircraft mission this causes a fall in the fuel carried on-board and this results in aircraft mass falling as fuel is burned. Combustion characteristics of the fuel as it is burned are assessed through GT simulation in Turbomatch.

4.5.2 Electric Energy Storage

Considering an electric storage device for hybrid electric propulsion, for a given aircraft mission the flight physics of energy consumption from typical electric energy storage is different to fuel burn. With fuel burn aircraft mass reduces, whereas with electric energy consumption the mass of the electric energy storage typically remains constant. To characterise the amount of stored electric energy on-board the aircraft the parameter State-of-Charge, *SOC*, is used (see Eq. 4-50 for definition, where E_{ES} is the energy in the electric energy storage device and $E_{ES,max}$ is the maximum amount of energy that can be stored in the electric energy storage device).

$$SOC = \frac{E_{ES}}{E_{ES,max}} \quad 4-50$$

The identified electric energy storage technology option from Sec. 3.3 was to employ a battery. The battery type chosen for mission application are lithium-ion

batteries due to their higher energy density [22] compared to other battery options. One major limitation to their use is performance deterioration with cycling due to crack propagation in the electrodes of lithium-ion cells [23].

In capturing the performance of batteries a number of models exist for the capturing of battery performance and these models fall into three main categories. These categories are electrochemical models, mathematical models and electrical models. Electrochemical models are generally used for detailed battery design as they are mainly used to optimise the physical design aspects of batteries, characterise fundamental mechanisms of power generation and relate battery design parameters with macroscopic and microscopic information [24]. This option for modelling is not selected as electrochemical models require days of simulation time and difficult to obtain proprietary battery-specific information. Additionally the fidelity of electrochemical models would be higher than what is required with assessment at the conceptual stage. A mathematical model is not used as mathematical models are mostly too abstract to embody any practical meaning and their results can be inaccurate in the order of 5% to 20% error [24].

Electrical models use a circuit-equivalent approach and represent the battery as a network of voltage sources, resistors and capacitors (see Figure 4-10). Electrical models have accuracies that lie between electrochemical and mathematical models with errors between 1% and 5%. The electrical battery model in [24] & [25] is employed here as it is comprehensive enough to capture all the dynamic and electrical characteristics of a battery. These characteristics are time-dependent usable capacity, non-linear open-circuit voltage, V_{OC} , and transient response. This model was validated against experimental results for polymer Lithium-ion batteries to within 0.4%. Limitations of the model is that it neglects the effects of self-discharge, cycle number and temperature, which are expected to be factors affecting performance for turboelectric application on a civil transport aircraft where power ratings are expected to be relatively high with aircraft operating multiple flights for a given time period.

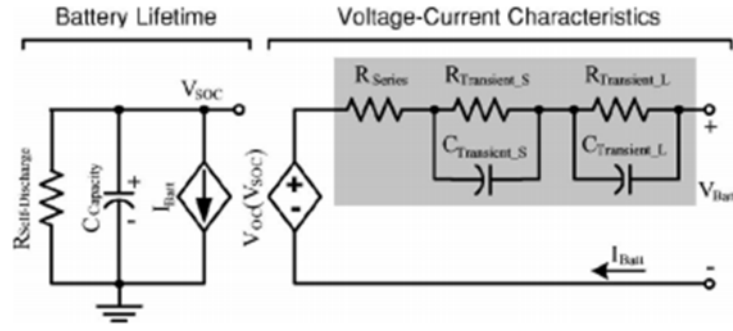


Figure 4-10 Electrical battery cell model proposed in [24]

Total Internal resistance of the cell of the battery, $R_{elec,i,tot}$ is determined as the summation of a step response series resistor, $R_{elec,series}$, a short time step transient resistor, $R_{elec,TS}$ and a long time step transient resistor, $R_{elec,TL}$ (see Eq. 4-51). The resistances represent different electrochemical processes such as ohmic losses, activation losses and concentration losses, and are determined as a function of SOC.

$$R_{elec,i,tot} = R_{elec,series} + R_{elec,TS} + R_{elec,TL} \quad 4-51$$

The resistances and open circuit voltage used for modelling are taken from [24] and shown in Table 4-4.

Table 4-4 Overview of battery cell parameters taken from [24]

$V_{oc} (V)$	$-1.031 \cdot e^{-35 \cdot SOC} + 3.685 + 0.2156 \cdot SOC - 0.1178 \cdot SOC^2 + 0.321 \cdot SOC^3$
$R_{elec,series} (\Omega)$	$0.1562 \cdot e^{-24.37 \cdot SOC} + 0.07446$
$R_{elec,TS} (\Omega)$	$0.3208 \cdot e^{-24.37 \cdot SOC} + 0.04669$
$R_{elec,TL} (\Omega)$	$6.6030 \cdot e^{-155.2 \cdot SOC} + 0.04984$

For a known V_{oc} profile as a function of SOC and $R_{elec,i}$ profile, by the setting of a cell current, I_{cell} , cell voltage, V_{cell} , is determined in Eq. 4-52. By setting an initial SOC (typically 100%) of a battery before discharge on an aircraft mission and setting a nominal cell current, $I_{cell,nom}$; a nominal cell voltage, $V_{cell,nom}$ is determined. $I_{cell,nom}$ and $V_{cell,nom}$ are then used to set the design point current and voltage of the battery cell. The nominal cell current is fixed by a suitable

discharge current for a known battery chemistry. In [24] discharge currents of 80, 320 and 640 mA were tested for a 850 mAh TCL PL-383562 polymer Li-ion battery.

$$V_{cell} = V_{OC} - R_{elec,i,tot} I_{cell} \quad 4-52$$

Battery efficiency is calculated in Eq. 4-53, where the voltage drop at the internal resistance is treated as a loss. The parameter η_{batt} is equal to $\eta_{discharge}$ when the battery is discharging and equal to $\eta_{recharge}$ when battery is recharging.

$$\eta_{batt} = 1 - \frac{I_{cell}^2 R_{elec,i,tot}}{V_{OC} I_{cell}} \quad 4-53$$

For a battery pack composed of n_{cells} in series and $m_{modules}$ in parallel (see Figure 4-11), by applying Kirchhoff's laws n_{cells} is set by the requirement of the battery pack to meet a maximum operating voltage requirement (see Eq. 4-54) and $m_{modules}$ is set by the requirement of the battery pack to meet a peak power requirement (see Eq. 4-55).

$$n_{cells} = \frac{V_{max}}{V_{cell,nom}} \quad 4-54$$

$$m_{modules} = \frac{PW_{peak}}{V_{max}I_{cell,nom}} \quad 4-55$$

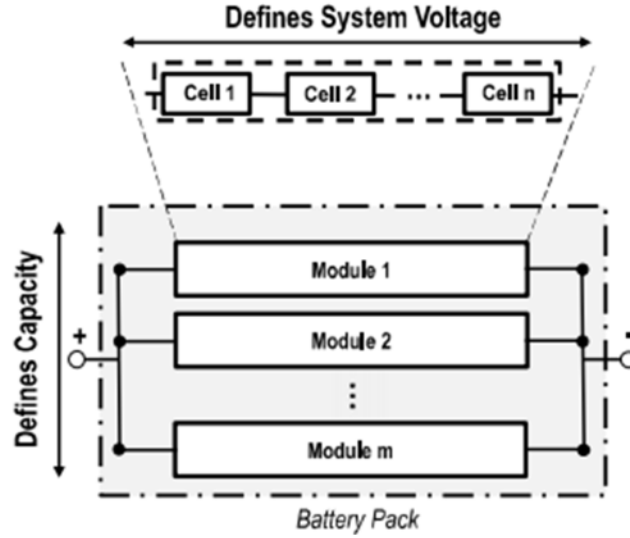


Figure 4-11 Design of a battery pack consisting of n_{cells} in series and $m_{modules}$ in parallel from [25]

For n_{cells} in series, battery pack voltage is calculated in Eq. 4-56 and for $m_{modules}$ battery pack current is calculated in Eq. 4-57.

$$V_{batt} = V_{cell}n_{cells} \quad 4-56$$

$$I_{batt} = I_{cell}m_{modules} \quad 4-57$$

In addition to runtime and I - V characteristics, the battery pack capacity, C_{batt} , also needs to be included in the integrated system representation [26]. For a defined battery mass, m_{batt} , and battery specific energy in W.h/kg, $\rho_{energy,batt}(W.h/kg)$, C_{batt} is determined in Eq. 4-58.

$$C_{batt} = \frac{m_{batt} \rho_{energy,batt} (\text{W.h.kg}^{-1})}{V_{max}} \quad 4-58$$

For a given mission power requirement and battery SOC, cell current is iterated until calculated power ($PW = I_{batt} V_{batt}$) matches the mission power requirement. With the power matched; battery pack current, voltage and C-rate are determined. To track changes in SOC for a given mission m_{batt} has to be defined as an input. With battery mass known $E_{ES,max}$ is determined in Eq. 4-59. Also, for a given time step dt , the change in stored energy of the electric energy storage device, dE_{ES} , is determined in Eq. 4-60. For a given point in time on a given aircraft mission, the instantaneous amount of stored electric energy is determined in Eq. 4-61 where $E_{ES,i}$ is the amount of stored electric energy corresponding to a SOC at a specific point in time. If initial SOC at the start of the mission is set to 100% then $E_{ES,0}$ is equal to $E_{ES,max}$.

$$E_{ES,max} = m_{batt} \rho_{energy,batt} \quad 4-59$$

$$dE_{ES} = PW dt \quad 4-60$$

$$E_{ES,i} = E_{ES,i-1} - dE_{ES} \quad 4-61$$

It should be noted that battery size is expected to impose a small volume penalty on the aircraft but assessment of this is beyond the scope of this work. Additionally, due to weight, the location of the battery on the airframe is expected to affect aircraft flight stability and is also beyond the scope of current work. For future work volume penalty from the installation of a battery pack on an aircraft and location of the battery pack on the airframe should be considered.

4.6 Cooling

4.6.1 Cryocooling

The first technology option identified for cooling is through the use of a cryocooler. In assessing the impact of the cryocooler on integrated system performance, cooling power requirements and cryocooler mass need to be

determined. In assessing cryocooler power requirements the analytical model by Berg et al in [27] is utilised which determines cooling power requirement, PW_{CC} , as a function of machinery cold operating temperature, T_C , ambient “hot” temperature, T_H , cooler efficiency as a fraction of Carnot efficiency, η_{fc} , and thermal load, $Q_{H,cold}$, at T_C (see Eq. 4-62). $Q_{H,cold}$ is assessed for the “maximum power transmitted” condition from a set of GTs for a given mission.

$$PW_{CC} = \frac{Q_{H,cold}}{\eta_{fc}} \left(\frac{T_H - T_C}{T_C} \right) \quad 4-62$$

Cryocooler power requirement is assumed to be constant for a given mission as cryocoolers typically performance typically experience “on-off” performance when operation shifts from design to off-design. Additionally cryocooler power demand is taken as a power offtake from core GTs.

Cryocooler architectures come in a number of types from Stirling cryocoolers to Reverse Brayton cryocoolers. Previous work indicates that reciprocating cryocoolers are likely to be unfavourable for high power, weight-sensitive and high reliability applications compared to Brayton cycle cryocoolers [28] & [29]. Thus for the current work a Reverse Brayton architecture for the cryocooler is assumed. In [30] a survey of existing Reverse Brayton cryocoolers was carried out and based on this an empirical relation for determining cryocooler mass, m_{cryo} , as a function of PW_{CC} is obtained (see 4-63).

$$m_{cryo} = \frac{27.5PW_{CC}}{1000} \exp[-1.225(\log_{10} PW_{CC})] \quad 4-63$$

In the consideration of cryocooler size, it is expected that the cryocooler would impose a small volume penalty on an aircraft. However due to project time constraints this penalty has not been assessed in detail.

4.6.2 Cryogen Cooling

It is assumed that heat from electrical machinery operational inefficiencies is transferred into cryogenic fuel and the fuel is pre-heated to bring fuel temperature up to ambient temperature to mitigate any performance penalties

of burning “cold” fuel. Assessment of the weight and size of pre-heating machinery (e.g. heat exchangers) is beyond the scope of the current studies.

4.7 Propeller-Wing Interaction

Capturing the effects of synergistic propeller-wing interactions has the potential to reduce drag and resize a wing when propellers are distributed along a wing [26] & [31]. As a first step towards assessing performance for distributed propeller configurations, propeller-wing interaction effects are captured for twin-propeller configurations using the engineering method in [32]. The method in [32] is utilised here for conceptual study since it estimates the aerodynamic characteristics of practical V/STOL aircraft configurations at a suitable level of fidelity for engineering calculations. The method has been previously used for assessment of propeller-wing combinations in [26]. Assumptions of the approach in [32] include:

- Swirl effects downstream of the propeller are not captured
- The interference of the wing extending beyond the slipstream is assumed to be relatively small
- Slipstream contraction behind the propeller disk at the wing leading edge is not accounted for
- The slipstream emanating from multiple propellers on the same wing semi-span can be assumed to merge into a wide rectangular slipstream provided the propellers are separated by no more than 10% of the propeller diameter
- For validity wing chord should not be more than twice the diameter of the propeller

With these assumptions, according to [32] results from utilising the engineering method are approximated to within about 3% of higher fidelity theoretical calculations. Accounted for with the method includes (1) change in effective angle of attack in blown section of the wing, (2) change in lift curve slope in blown section of the wing, (3) change in induced drag factor in the blown

section of the wing and (4) upwash on the unblown section of the wing induced by propeller slipstream.

To account for changes in profile drag due to the increase in dynamic pressure over the wing in the blown section (not captured in [32]), the blown wing zero lift drag coefficient, C_{D0j} , is calculated for an increased Reynold's number due to propeller jet velocity, U_j . Accounting for increased U in the blown section of the wing due to the propeller jet, the total change in profile drag, $F_{D,profile}$, in the blown section of the wing is estimated using a similar method of superposition to that which is used to determine change in induced drag in [32]. $\Delta F_{D,profile}$ in the blown section of the wing is estimated in Eq. 4-64 where U_0 is freestream velocity, C_{D0} is wing zero lift drag coefficient and S_J is blown section wing area. For installed propellers, change in induced drag (determined through use of engineering method in [32]) and $\Delta F_{D,profile}$ in blown section of the wing are computed for each point on a defined aircraft mission profile.

$$\Delta F_{D,profile} = \frac{1}{2} \rho \sum_{jets} S_j (U_j^2 C_{D0j} - U_0^2 C_{D0}) \quad 4-64$$

At the aircraft level propeller-wing interaction effects are only captured during take-off, climb and cruise mission phases of the main mission. Propeller-wing interaction effects at other mission phases is not expected to significantly affect overall mission performance.

4.8 Integrated Performance Assessment

To assess the performance of the propulsion system, an adequate procedure needs to be defined to simplify the complexity that arises from integrating a number of different sub-systems. At a top-level for given mission requirements the steps followed for assessing the propulsion system are listed below:

1. Design point assessment
2. Off-design assessment for a range of GT throttle or GT LP shaft speed settings for each flight condition defined
3. Sub-systems sizing
4. Weight estimation

4.8.1 Design Point Assessment

4.8.1.1 General

The procedure for propulsion system design point assessment starts with the assessing of the propulsor power requirement. Propulsor power requirement is determined from the outputs of the propulsor design point simulation. If propulsors are ducted fans the propulsors are simulated using the relations and tools described in Sec. 4.2.1, and if propulsors are propellers the propulsors are simulated using the relations and tools described in Sec. 4.2.2. For the case where the propulsors are ducted fans, either $W_{propulsor}$ or $F_{propulsor}$ can be a design input. If $F_{propulsor}$ is the design input then $W_{propulsor}$ is the output and if $W_{propulsor}$ is the design input then $F_{propulsor}$ is the design output. Now considering the integrated propulsion system after propulsor power requirement is determined; the total power offtake from core GTs at the design point is determined after taking into account transmission losses, cooling power requirements and power hybridisation (see Sec. 4.3 and Sec. 4.6). With total power offtake from core GTs determined; the performance of the GT is simulated and engine fuel flow rate, mass flow rate and overall net thrust are determined. Overall engine TSFC is determined by dividing GT fuel flow rate by overall net thrust produced by the propulsion system ($F_{propulsor} + F_{GT}$). An overview of this created design point assessment procedure is shown in Figure 4-12. It should be noted that the procedure shown in Figure 4-12, power hybridisation due to an electric energy storage device does not have to be accounted for at design point. For example in a retrofit case, propulsion system can be configured to have electric energy storage only discharge power at propulsion system off-design with its power output set to 0 W at propulsion system design point so as to not resize core gas turbines and propulsors.

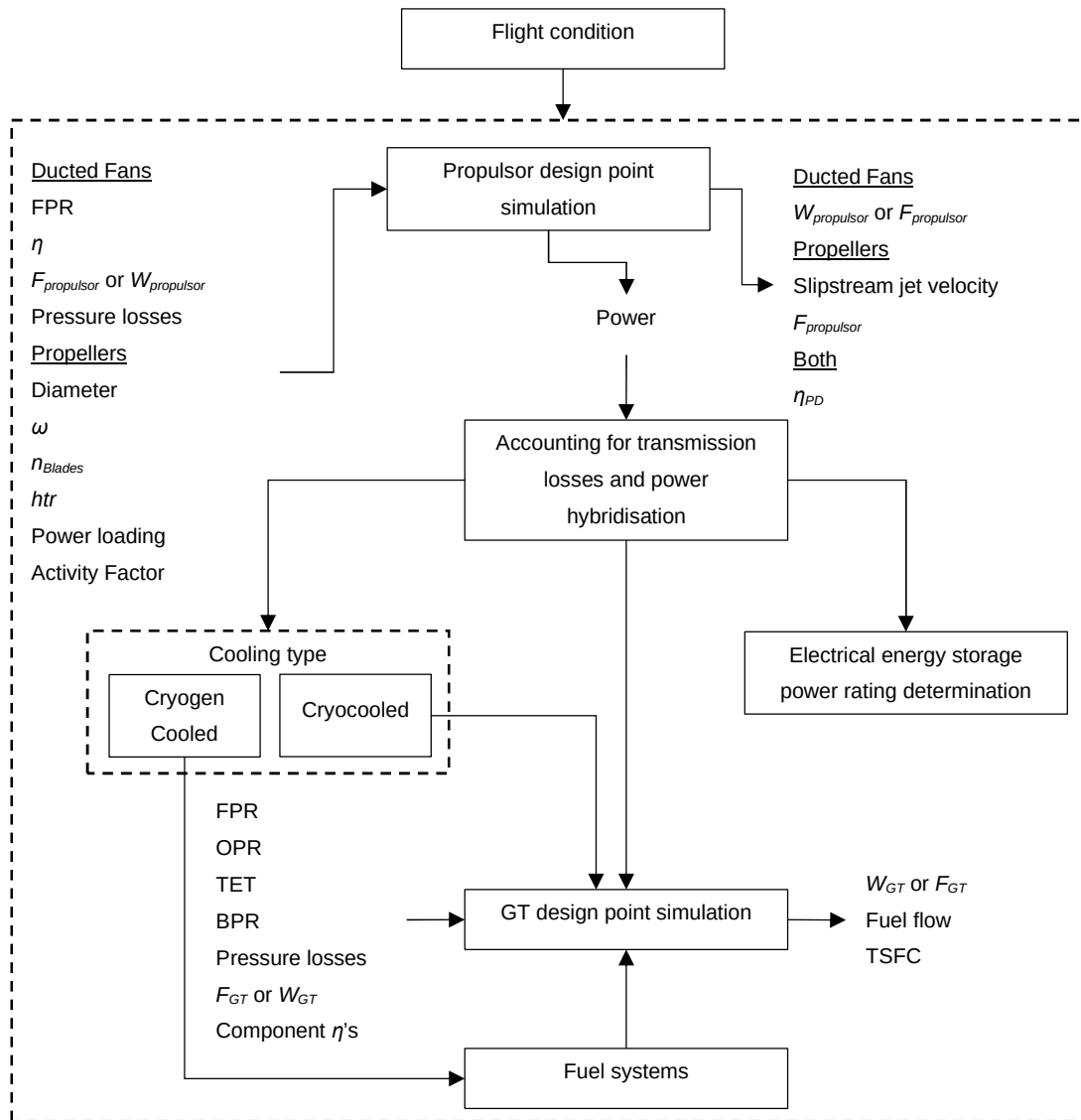


Figure 4-12 Design Point Assessment Procedure

4.8.1.2 Retrofitted Turboelectric Ducted Fan

For a retrofitted turboelectric propulsor case, where an already sized GT is run at off-design and propulsor is operated at design point (e.g. a rear-mounted BLI propulsor being retrofitted to an already existing aircraft with designed core turbofans) the procedure in Figure 4-12 changes to that in Figure 4-13. The difference with this procedure is that GT is simulated first at the design point before a desired thrust split sets $F_{propulsor}$ at the design point and off-design GT thrust requirement, $F_{GT,OD}$. Integrated TSFC is then assessed for the combination of the design point propulsor and off-design GT. For this procedure

it is assumed core GT is either a turbofan or a turbojet. From the procedure in Figure 4-13 it should be noted that power hybridisation due to an electric energy storage device does not have to be accounted for at design point. The propulsion system can be configured to have the electric energy storage device to only discharge power at propulsion system off-design with electric energy storage device power output fixed at 0 W at the propulsion system's design point.

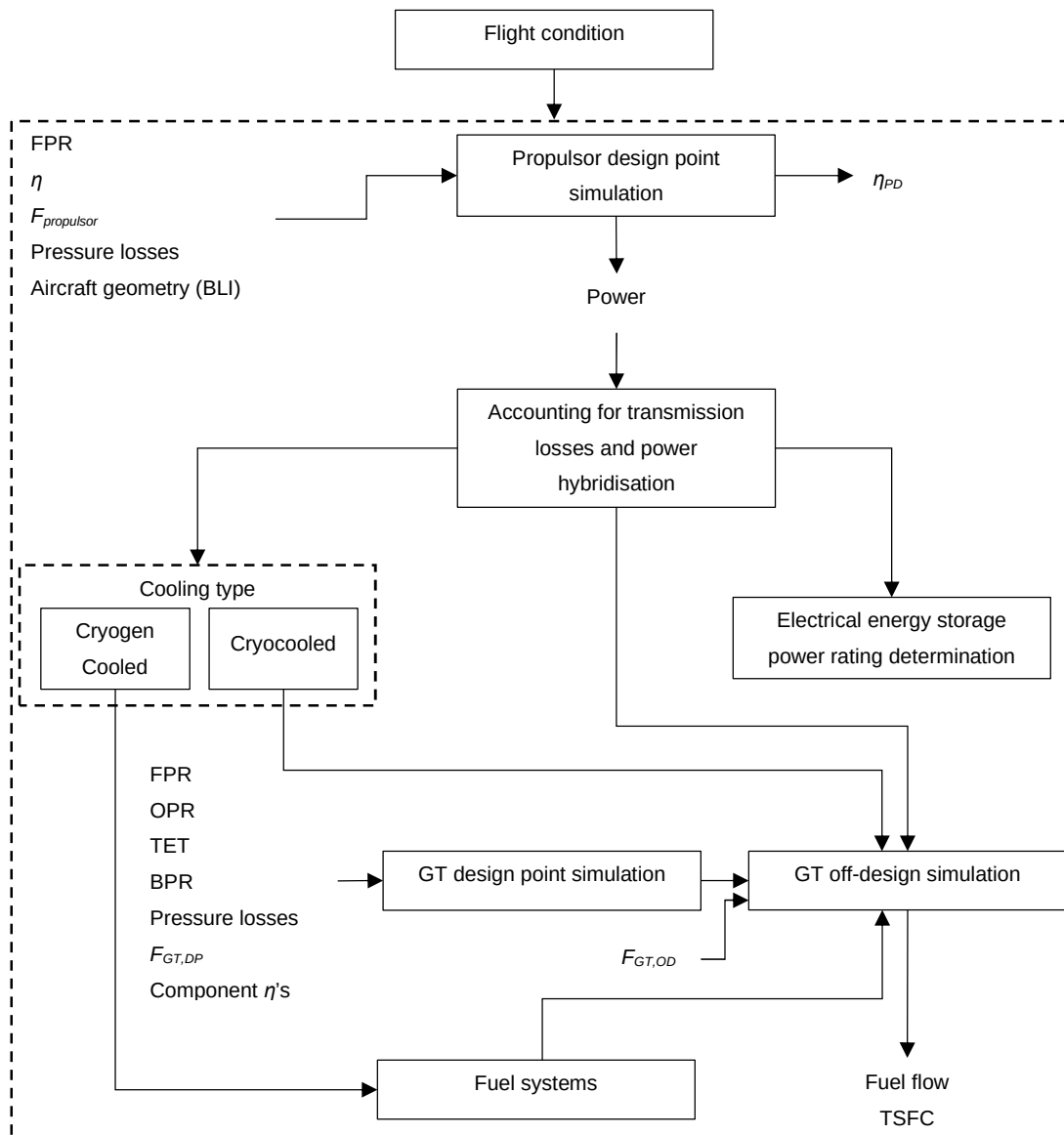


Figure 4-13 Design point assessment procedure for a retrofitted ducted fan

4.8.2 Off-Design Assessment

4.8.2.1 AC Turboelectric

Although in current studies AC turboelectric architecture is not being assessed for mission roles under consideration, an off-design approach is presented here for future assessment purposes. Assuming the nozzle area of the GT does not vary at off-design, the off-design simulation starts with the simulating of propulsor off-design performance for a given shaft speed and off-design flight condition. With the propulsor simulated; propulsive device efficiency, thrust and power requirement are determined. From Sec. 4.3.1 the speed of the propulsor fixes the speed of the output shaft of the GT and power offtake required of the GT is determined by taking into account transmission losses and any cooling power requirements if a cryocooler is employed. GT off-design simulation is then carried out to determine TET, GT thrust and fuel flow rate. Overall propulsion system TSFC is then determined by dividing the fuel flow rate by the summation of GT thrust and propulsor thrust. An overview of the created procedure employed is shown in Figure 4-14.

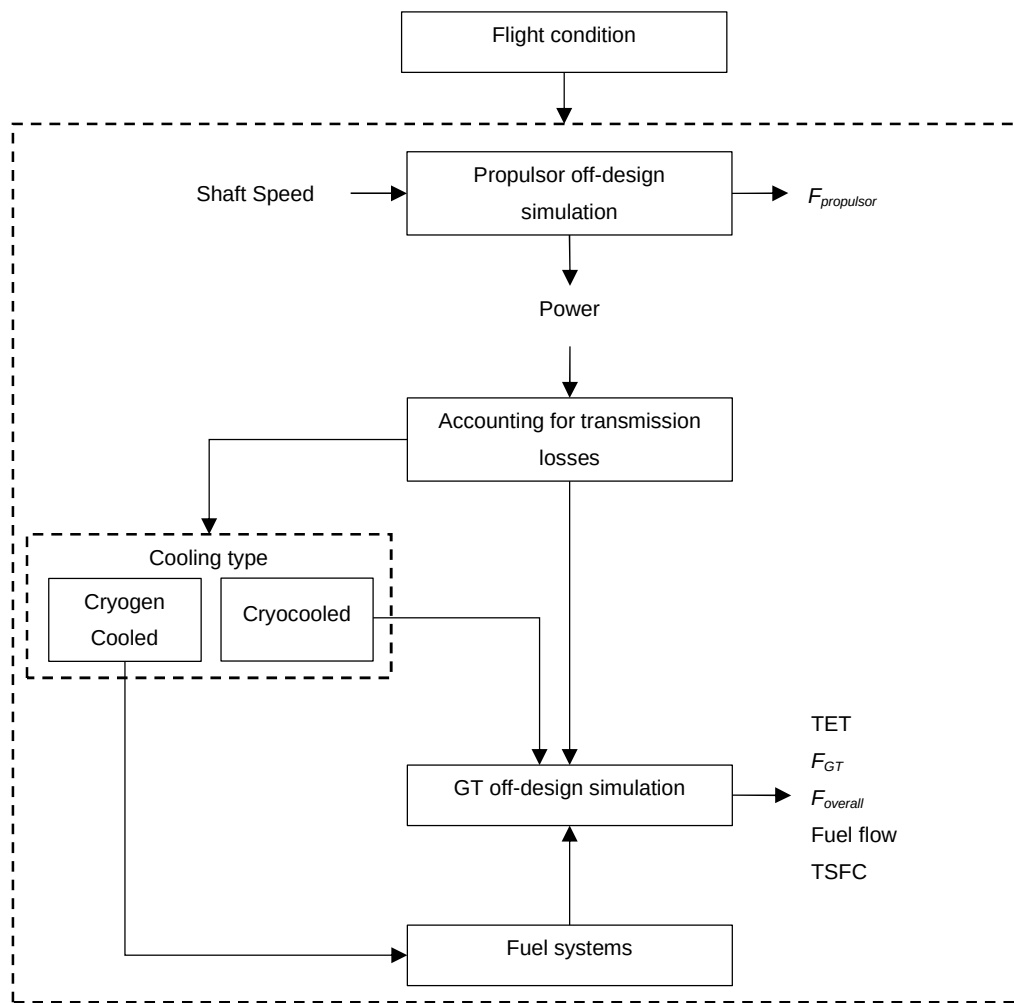


Figure 4-14 Off-design assessment procedure for AC Turboelectric architecture

4.8.2.2 DC Turboelectric

One key characteristic of the DC turboelectric architecture is that the shaft speed of the propulsors is independent of the output shaft speed of the GT, in which the power controllers enable this by acting as an infinitely variable ratio transmission [33]. An infinitely variable gear ratio transmission means that the ratio in speeds between the output shaft of the GT and the propulsor shaft speed can have any value; plus the ratio can vary from design point value throughout a given flight envelope. At off-design this allows for both GT and propulsors to operate at optimal speeds for maximum efficiency throughout a given flight envelope. To assess this feature, suitable matching between the propulsor and the GT needs to be carried out to determine what speed propulsors and GTs operate at for minimum TSFC during off-design.

Ducted Fans

For ducted fans as the propulsor choice and assuming a turbojet or turbofan core, off-design performance assessment starts with the setting of a fan PCN. With fan PCN set, by simulating the off-design performance of the propulsor; propulsive device efficiency, thrust and power requirement are determined. From the determination of propulsor power requirement by taking into account transmission losses and any cryocooler power requirements (if a cryocooler is part of the architecture), the power offtake from the GT is determined. For a calculated power offtake, GT off-design performance is simulated for a suitable range of LP PCN's (LP PCN is the GT off-design simulation handle). From the simulation of GT off-design performance for a range of LP PCN's, the LP PCN that yields minimum integrated propulsion system TSFC is determined. This PCN is subsequently set as LP PCN for the given fan PCN and GT performance for the optimum LPC PCN is read-off. An overview of the created off-design assessment procedure for the DC turboelectric architecture with ducted fans is shown in Figure 4-15.

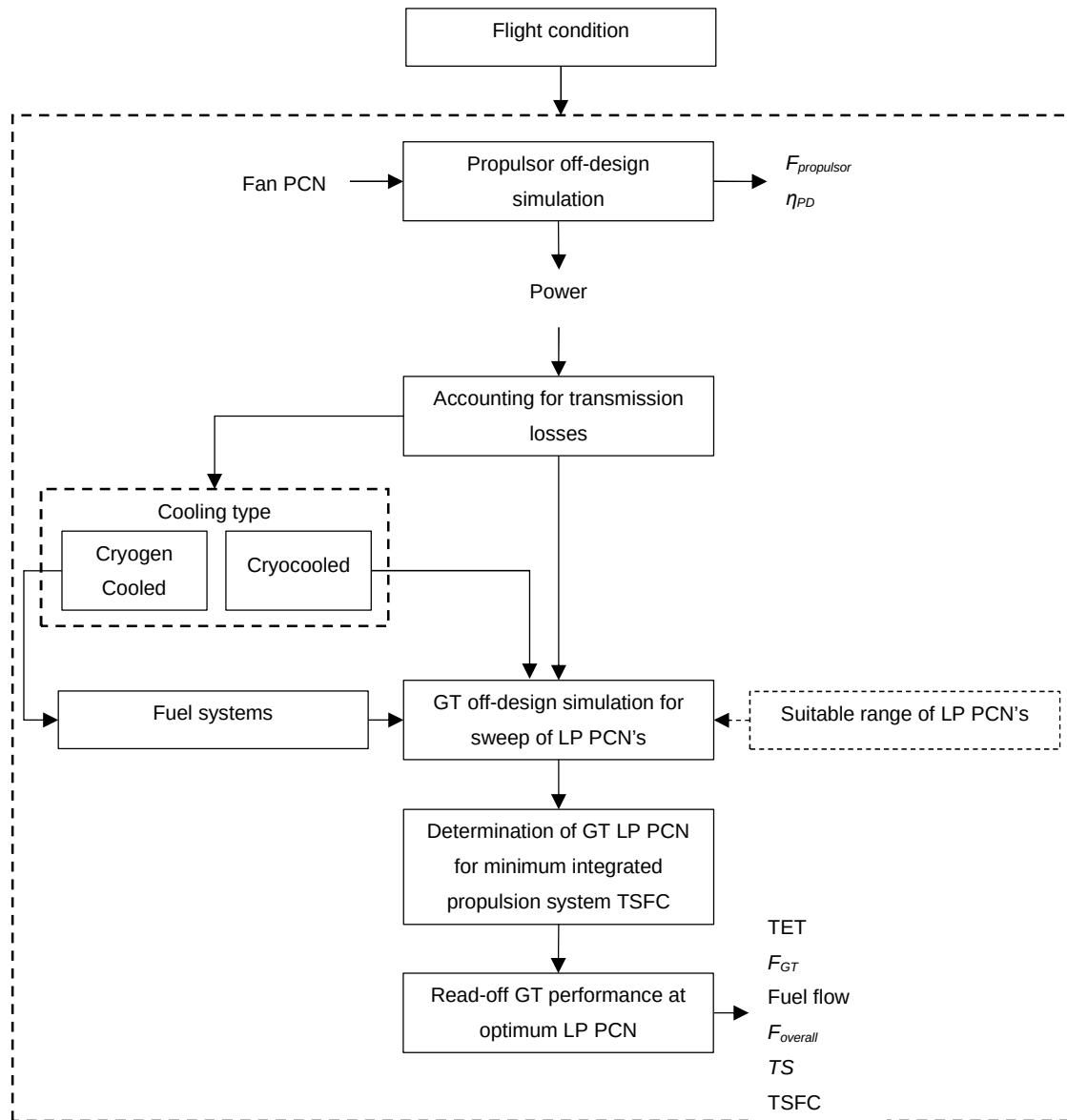


Figure 4-15 Off-design assessment procedure for DC turboelectric architecture with ducted fans

Propellers

In considering the performance for the DC architecture with propellers and assuming a turboshaft core, the load on the output turbine of the GT can be controlled electrically by the generator with output turbine speed effectively kept constant for all off-design operating points [34]. Additionally, with this configuration the rotational speed of the propellers can be controlled independently of the GT while pitch angle variation of the propeller controls the

amount of power 'swallowed' by the propellers. Taking into account these features, off-design performance assessment starts with the setting of GT TET (this fixes the engine throttle) as opposed to the setting of propeller speed. This is done because since GT output turbine rotational speed is constant at off-design, by the setting of TET GT power offtake can be determined without prior knowledge of the characteristics of the propulsor. With this approach GT off-design simulation is carried out to determine core thrust, fuel flow rate and power offtake. Power transmitted to the propellers for a given engine throttle is then determined after subtracting any transmission losses and cooling power requirements from the power offtake. For off-design simulation of the propeller, a suitable propeller rotational speed is selected and pitch angle is varied until propeller shaft power matches the power transmitted from the GT and with this done propeller performance is read-off for the matched pitch angle. The off-design simulation is repeated for a suitable range of propeller rotational speeds. From the multiple off-design simulations of the propeller, the propeller rotational speed that yields maximum propeller efficiency is determined and propeller performance is read-off for this optimum RPM. With propeller performance at off-design determined for an optimum RPM; propeller thrust, slipstream jet velocity, overall propulsion system thrust, TS and integrated propulsion system TSFC are determined. An overview of this created procedure for assessing off-design performance of the DC turboelectric architecture with propellers is shown in Figure 4-16.

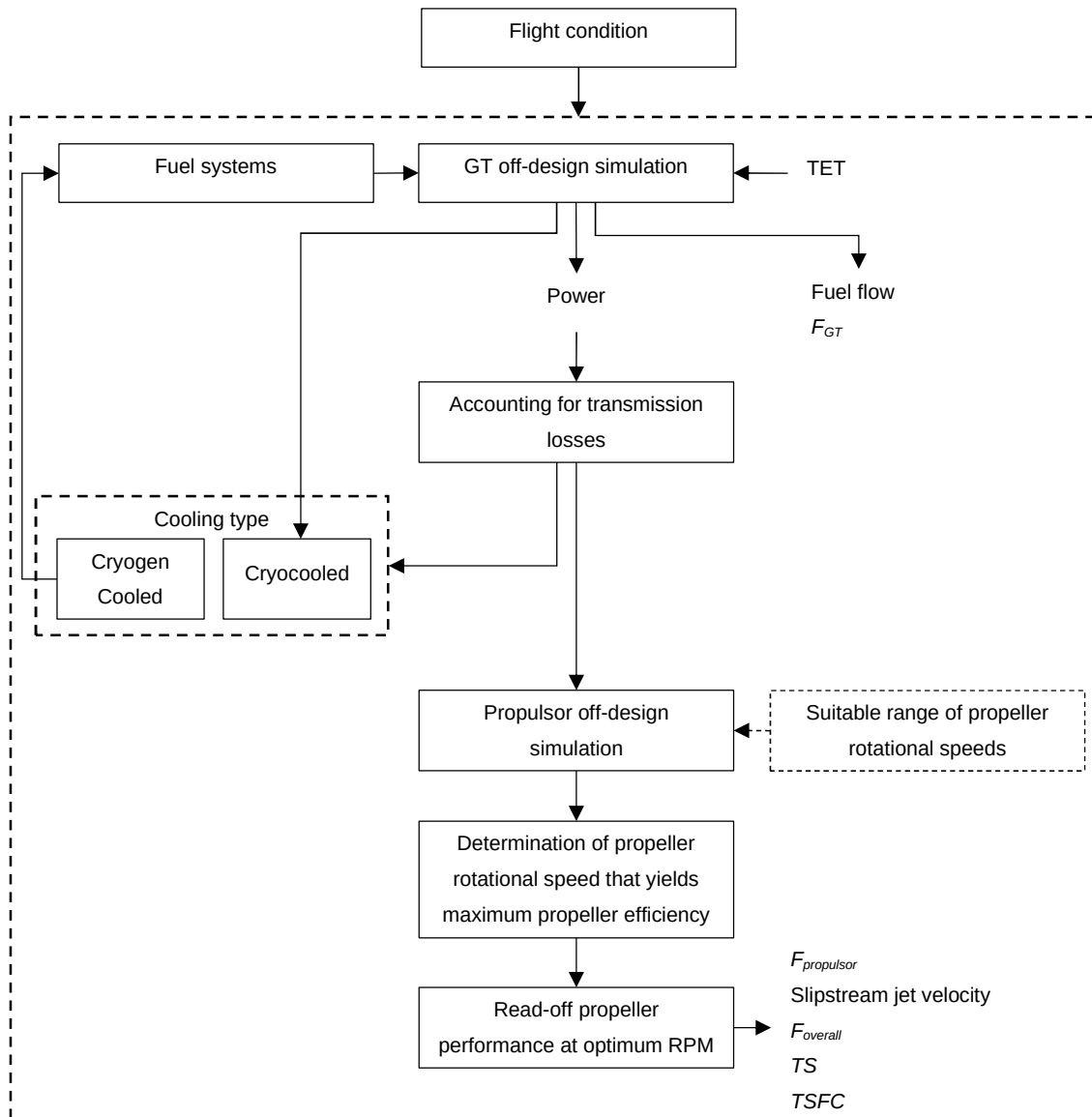


Figure 4-16 Off-design assessment procedure for DC turboelectric architecture with propellers

4.8.2.3 Geared

For the geared architecture with ducted fans off-design performance is assessed using Turbomatch with TET as calculation handle. Propulsion system thrust and TSFC are then determined for a set TET.

For the geared architecture with propellers, off-design performance assessment starts with the setting of a GT output shaft rotational speed and engine TET. With these parameters set; off-design simulation of the GT is carried out to determine core thrust, fuel flow rate and shaft power output. Propeller rotational

speed is then determined using the relation in Eq. 4-45. Power transmitted to the propellers is then determined by subtracting transmission losses from GT shaft power output. With transmitted power to the propeller determined; propeller off-design simulation is carried out to determine propeller efficiency, propeller thrust, slipstream jet velocity, overall thrust, TS and integrated TSFC. For the off-design simulation, propeller pitch angle is varied until propeller shaft power matches power transmitted from the GT. An overview of the created off-design performance assessment procedure for the geared architecture with propellers is shown in Figure 4-17.

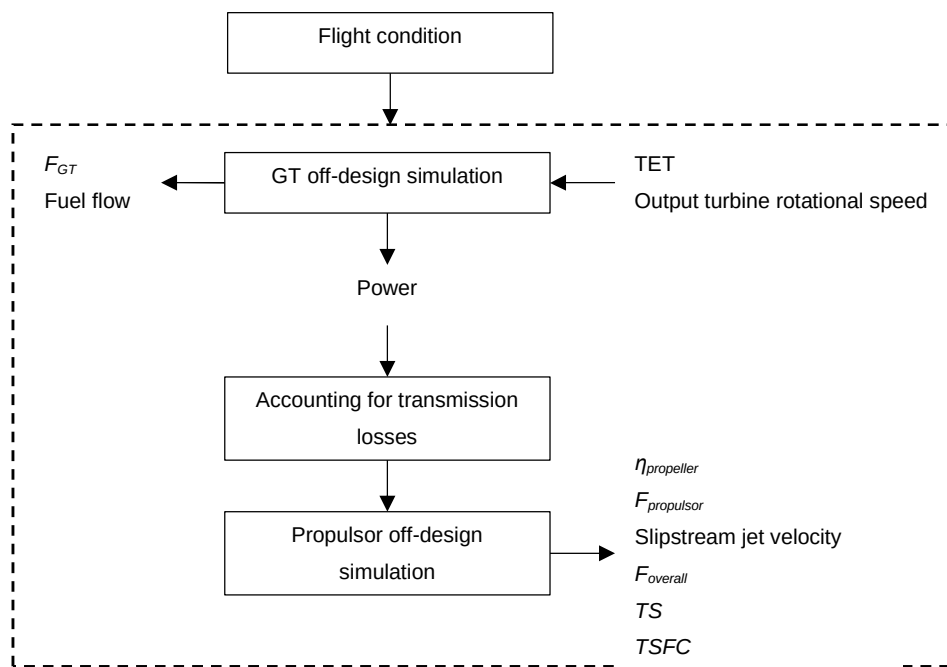


Figure 4-17 Off-design assessment procedure for geared architecture with propellers

For hybridised version of the geared architecture, whether with propellers or ducted fans, if superconducting machines and cryogenic power converters are considered, cooling losses are determined in addition to transmission losses.

4.8.2.4 Mission Assessment

To assess aircraft flight performance, propulsion system performance has to be assessed at off-design for multiple flight conditions and engine ratings. Normally at the aircraft level thrust-SFC lookup tables are used to determine engine SFC

for a given mission flight condition and calculated thrust requirement. The data for the lookup tables is obtained from performance simulation of the propulsion system for a range of engine ratings (fixed by TET) for each flight condition of a given mission envelope. Due to lack of project time, third order polynomial curve fits of simulated data are used instead of data lookup tables. With polynomial curve fits propulsion system SFC at the aircraft level is determined as a function of calculated thrust requirement in Eq. 4-65 for a given flight condition. The polynomial curve fitting coefficients in Eq. 4-65, $SFCA$ to $SFCD$, are determined from the curve fitting of a third order polynomial to calculated propulsion system SFC and thrust at different engine ratings for a given flight condition. For a mission that has ' n ' defined flight conditions, one polynomial is created for each defined flight condition. For studies here, MATLAB was utilised for creating the polynomial curve fits and determining the polynomial coefficients with the procedure in Figure 4-18. Using third order polynomial curve fits of simulated data captures the general performance trend of the propulsion system at the aircraft level albeit at a limitation of increased local errors for simulated points off the curve fitted trendline.

$$SFC = SFCA.F^3 + SFCE.F^2 + SFCEC.F + SFCD$$

4-65

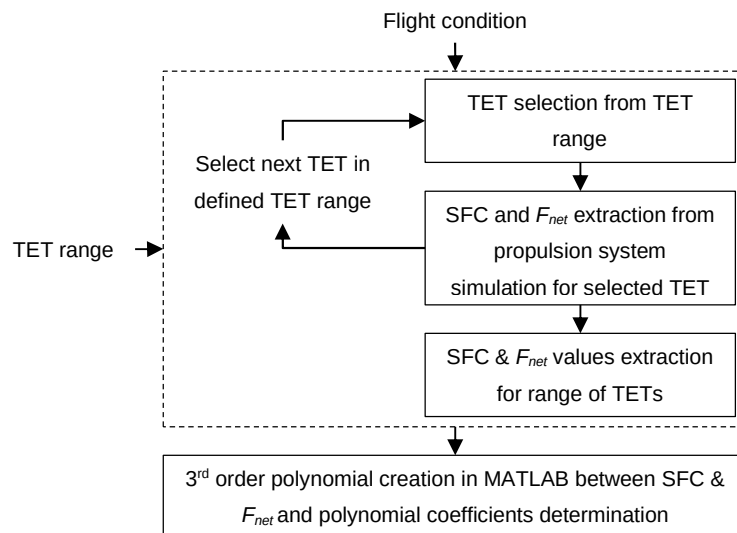


Figure 4-18 Third order polynomial curve fit between SFC and net thrust creation procedure for a given flight condition

4.8.2.5 Hybrid Electric Mission Assessment

For the assessment of aircraft performance with a hybrid electric powerplant, simply assessing propulsion system SFC as a function of aircraft thrust requirement is no longer sufficient. Consideration has to be made to the fact that power being transmitted to the propulsor is coming from an electric energy storage device in addition to the GT. For the electric energy storage device both its SOC and discharged power will alter propulsion system TSFC and have to be taken into account in the assessment of aircraft performance. Utilising a third order polynomial curve fit at the aircraft level, $\eta_{discharge}$ can be determined as a function of SOC in Eq. 4-66. The polynomial curve fitting coefficients in Eq. 4-66, $etadisA$ to $etadisD$, are determined from the curve fitting of a third order polynomial to calculated $\eta_{discharge}$ values for a range of SOC's (from a minimum defined SOC to SOC at start-of-discharge). Discharge efficiency for a given SOC is calculated using Eq. 4-53 for a defined cell current, cell resistance profile and open circuit voltage profile.

$$\eta_{discharge} = etadisA.SOC^3 + etadisB.SOC^2 + etadisC.SOC + etadisD \quad 4-66$$

For the geared hybrid electric configuration with ducted fans the propulsion system as explained in Sec. 4.3.5 is considered as a geared turbofan with power intake from an electric energy storage device. In assessing the power intake (see Eq. 4-49) $\eta_{discharge}$ varies as a function of SOC; thus engine SFC has to be determined as a function of flight condition, GT rating, electric energy storage power rating and electric energy storage SOC. At the aircraft level utilising third order polynomial curve fits for a given flight condition and electric energy storage power rating, SFC's ($SFC_{SOC,i}$) for a range of SOC's from SOC_1 (SOC_{min}) to SOC_n (SOC_{max}) are determined in Eq. 4-67 as a function of calculated thrust requirements. The polynomial curve fitting coefficients, $SFChybridA_i$ to $SFChybridD_i$, are determined from curve fitting of third order polynomials to calculated propulsion system SFC and thrust at different engine ratings for a given SOC, battery power rating and flight condition. This is done for all flight conditions on a defined flight envelope at SOC's ranging from SOC_{min} to SOC_{max} . For studies here, MATLAB is utilised to create the

polynomial curve fits and determine the polynomial curve fitting coefficients with the procedure in Figure 4-19. To find SFC at a given mission point, propulsion system SFC is determined by calculating $SFC_{SOC,i}$ for a suitable range of SOC's (from the minimum SOC specified for a given mission to SOC at the start-of-discharge) in Eq. 4-67 and then interpolating for SFC between the calculated $SFC_{SOC,i}$'s that correspond to the SOC_i 's nearest in value to calculated SOC for the electric energy storage.

$$\begin{aligned}
 SFC_{SOC,1} &= SFChybridA_1.F^3 + SFChybridB_1.F^2 & ; SOC_i = SOC_1 \\
 &+ SFChybridC_1.F + SFChybridD_1 \\
 \\
 SFC_{SOC,2} &= SFChybridA_2.F^3 + SFChybridB_2.F^2 & ; SOC_i = SOC_2 \\
 &+ SFChybridC_2.F + SFChybridD_2 \\
 \\
 &\vdots & \vdots \\
 \\
 SFC_{SOC,n} &= SFChybridA_n.F^3 + SFChybridB_n.F^2 & ; SOC_i = SOC_n \\
 &+ SFChybridC_n.F + SFChybridD_n
 \end{aligned}
 \tag{4-67}$$

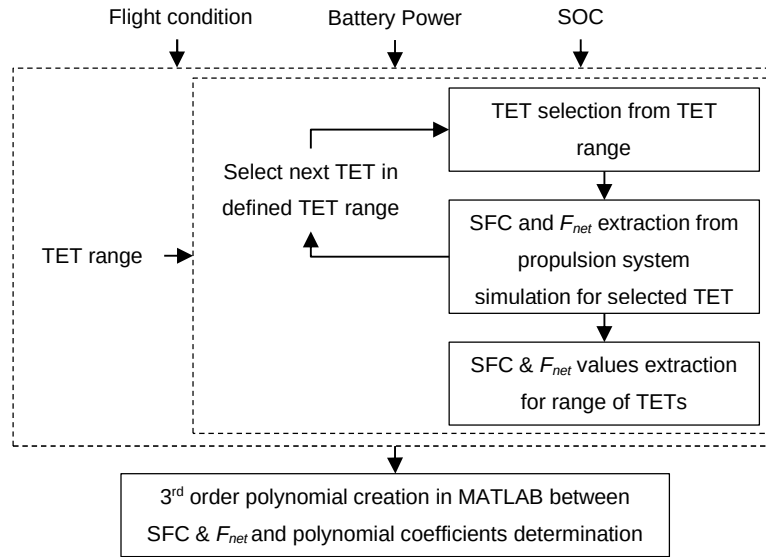


Figure 4-19 Third order polynomial curve fit between SFC and net thrust creation procedure for a given flight condition, battery power rating and SOC

For the other hybrid electric powerplant configurations being considered, consideration has to be made to the characteristics of the propulsor, core GT and electric energy storage. To assess flight performance in the approach here GT fuel flow rate is first calculated before calculating integrated TSFC. Integrated TSFC is determined by dividing GT fuel flow rate by calculated thrust requirement. For the determination of GT fuel flow rate at the aircraft level the propulsor and GT are decoupled to adequately assess the effect a hybrid power source has on GT performance. From the decoupled system the parameters of interest that characterise the system are TS , propulsive device efficiency, core thrust, propulsor thrust and fuel flow rate. By determining the values for these parameters in an aircraft performance tool, flight performance of the hybrid-electric aircraft can be determined.

Utilising third order polynomial curve fits at the aircraft level TS is determined as a function of calculated thrust requirement (see Eq. 4-68), propulsive device efficiency is determined as function of calculated propulsor thrust (see Eq. 4-69), core thrust is determined as a function of GT shaft power (see Eq. 4-70), propulsor thrust is determined as a function of propulsor shaft power (see Eq. 4-71) and fuel flow rate (W_F) is determined as a function of GT shaft power (see Eq. 4-72). The polynomial curve fitting coefficients in Eq. 4-68, TSA to TSD , are determined from the curve fitting of a third order polynomial to calculated TS and thrust at different engine ratings for a given flight condition. The polynomial curve fitting coefficients in Eq. 4-69, η_{PDA} to η_{PDD} , are determined from the curve fitting of a third order polynomial to calculated propulsive device efficiency and propulsor thrust at different engine ratings for a given flight condition. The polynomial curve fitting coefficients in Eq. 4-70, $FGTA$ to $FGTD$, are determined from the curve fitting of a third order polynomial to calculated GT thrust and GT shaft power output at different engine ratings for a given flight condition. The polynomial curve fitting coefficients in Eq. 4-71, F_{propA} to F_{propD} , are determined from the curve fitting of a third order polynomial to calculated propulsor thrust and propulsor shaft power at different engine ratings for a given flight condition. The polynomial curve fitting coefficients in Eq. 4-72, f_{flowA} to f_{flowD} , are determined from the curve fitting of a third order polynomial

to calculated fuel flow rate and GT shaft power output at different engine ratings for a given flight condition. The polynomials for $TS-F$, $\eta_{PD}-F_{propulsor}$, $F_{GT}-PW_{GT}$ and W_F-PW_{GT} are computed for every flight condition on a defined flight envelope. For studies here the procedure in Figure 4-20 using MATLAB is employed to create the hybrid-electric polynomial curve fits and to determine polynomial coefficients.

$$TS = TSA.F^3 + TSB.F^2 + TSC.F + TSD \quad 4-68$$

$$\eta_{PD} = etaPDA.F_{propulsor}^3 + etaPDB.F_{propulsor}^2 + etaPDC.F_{propulsor} + etaPDD \quad 4-69$$

$$F_{GT} = FGTA.PW_{GT}^3 + FGTB.PW_{GT}^2 + FGTC.PW_{GT} + FGTD \quad 4-70$$

$$F_{propulsor} = FpropA.PW_{prop}^3 + FpropB.PW_{prop}^2 + FpropC.PW_{prop} + FpropD \quad 4-71$$

$$W_F = fflowA.PW_{GT}^3 + fflowB.PW_{GT}^2 + fflowC.PW_{GT} + fflowD \quad 4-72$$

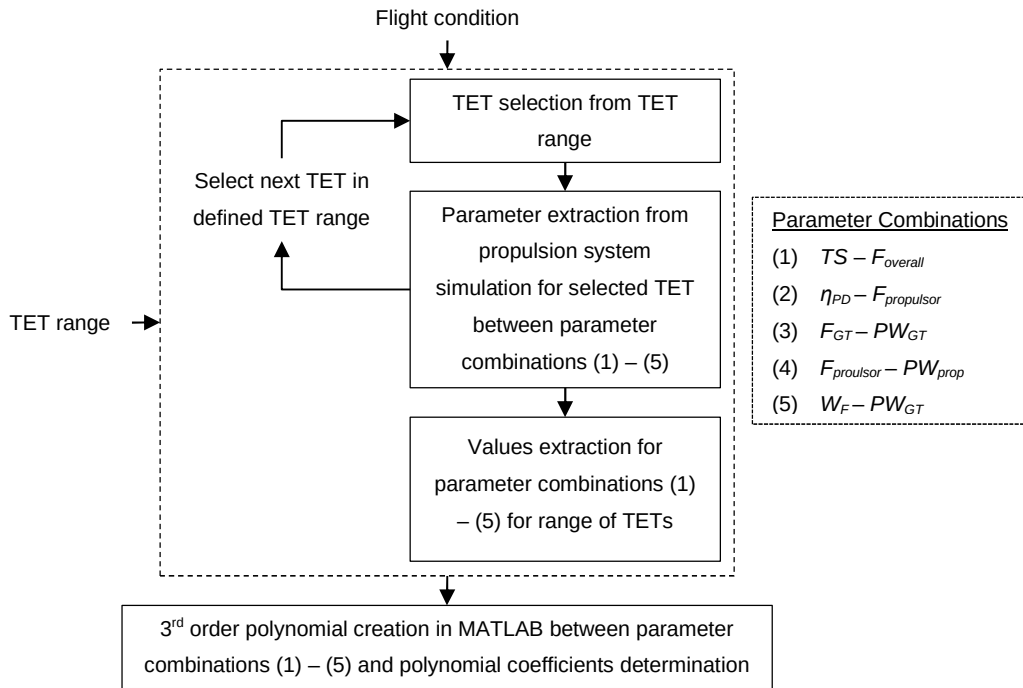


Figure 4-20 Third order polynomial curve fit creation procedure for hybrid-electric flight parameters for a given flight condition

To adequately assess the effect an increase in battery power output has on performance at the propulsion system level; overall system efficiency is defined in Eq. 4-73 to ascertain how efficiently thrust is produced by the hybrid powerplant in comparison to a baseline non-hybrid propulsion system. In Eq. 4-73, PW_{batt} is battery power and U_{∞} is freestream velocity.

$$\begin{aligned} \text{System Efficiency} &= \frac{\text{Thrust Power}}{\text{Supplied Fuel Power} + \text{Supplied Battery Power}} \\ &= \frac{F_{net} U_{\infty}}{W_F \cdot \text{LHV}_{\text{fuel}} + PW_{batt}} \end{aligned} \quad 4-73$$

4.8.3 Sizing

Sub-systems of the propulsion system are sized according to models and assumptions described in Secs. 4.2-4.6.

4.8.4 Weight Estimation

Total propulsion system weight is set as the sum of the weight of all the sub-systems and sub-system weights are determined according to models and assumptions described in Secs. 4.2-4.6.

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5 AIRCRAFT MODELLING

5.1 Introduction

With a stated objective of the research effort being the consideration of the effect the effect of the propulsion system on aircraft design, performance, aerodynamics and structures; a suitable platform is needed that captures the unique characteristics of TeDP and HeP at the aircraft level. An existing platform that assesses the impact of conventional and novel propulsion systems on aircraft configurations is the numerical tool CADI (Conceptual Aircraft Design Integration) originally developed by Kirner, R. in Fortran [1]. For performance assessment in current studies this tool has been used here with modifications made to it to achieve current project aim & objectives. Modifications made to the tool include the following:

- Assessment of flight performance with profile drag coefficient changing at each point on the flight envelope rather than assuming constant profile drag coefficient throughout the flight envelope
- Computation of wave drag through the Korn equation method
- Computation of cruise performance with the higher accuracy integrated range method rather than using the Breguet Range equation
- Computation of cruise performance for multiple input flight conditions for a stepped cruise profile and modifying the code of the tool so that descent starts at final cruise flight condition
- Computation of take-off performance with the balanced field length approach
- Ability to assess climb performance for input thrusts and rate-of-climbs at defined climb flight conditions
- Flight performance assessment taking into account a defined reserve mission profile
- Flight performance assessment taking into account the variation of TSFC with thrust requirement for a given flight condition using propulsion system performance polynomials (see Sec. 4.8.2.4 & Sec. 4.8.2.5) rather than assuming constant TSFC for a given flight condition

- Aircraft mass estimation taking into account the possibility of a fuselage and empennage that is fabricated utilising composite material
- Installed powerplant mass estimation taking into account an electrical installation factor
- Capability to assess the performance of a retrofitted aircraft with known systems, equipment and wing mass
- Assessment of the performance of propeller-powered flight taking into account the physics of propeller-wing interaction
- Assessment of aircraft performance at off-design
- Assessment of aircraft performance for different payload-range combinations
- Capability of assessing hybrid electric flight performance
- Sizing and flight performance assessment of tube-and-wing aircraft utilising cryogenic fuel
- Changes to coding of the tool to make the code more robust

Design point assessment for a given aircraft configuration starts with the specification of propulsion system, design mission requirements and general aircraft design characteristics. Design mission requirements are range, payload, flight altitudes, flight speeds and reserve mission profile. General aircraft design characteristics includes geometry type, component locations, aircraft material, airframe geometry sizing parameters and high-lift device characteristics. With propulsion system, design mission requirements and general aircraft design characteristics specified; performance assessment starts with the sizing and mass estimation of the components of the aircraft that do not vary with wing size. For civil transport T&W considered here these components are set to be the fuselage and operational items. With these assessed an initial maximum take-off weight (MTOW) or maximum take-off mass (MTOM) is guessed so that wing size and mass can be determined. For studies here the initial MTOW is guessed by multiplying the summation of the masses of the fuselage, operational items, payload and powerplant by a factor greater than one (1.75 was arbitrarily used in CADI). The masses of the fuselage mass, operational items, payload and powerplant are already known before MTOW is known; what

is unknown are the masses of the wing, systems, equipment and empennage. With the actual mass of the aircraft expected to be greater than the summation of the masses of the fuselage, operational items, payload and powerplant when the other components of the aircraft such as the wing, systems, equipment and empennage are considered; multiplication of the fixed masses by a factor greater than one results in a guess for MTOW that is expected to be close in value to the actual MTOW of an aircraft provided the multiplication factor is not too large or too small.

With MTOW guessed so that wing size and mass can be determined, empennage size and mass is determined as a function of wing size. With wing size and empennage size determined, aircraft drag can be estimated and this enables the computation of aircraft performance. With aircraft performance computed, aircraft fuel burn is estimated and this coupled with an estimation of aircraft equipment mass leads to a new estimate for aircraft MTOW. From [3] aircraft equipment is considered to be the aggregate of equipment carried on board the aircraft, aircraft systems (except the fuel system which is considered as part of the powerplant) and the landing gear. In [3] it is estimated as a function of MTOW. If the new estimate for aircraft MTOW does not match the initial guess to within a given convergence margin; initial guess MTOW is updated with calculated MTOW and aircraft sizing, mass estimation and performance calculations are repeated until MTOW converges. Once MTOW converges the procedure for assessing the aircraft at design point comes to an end. An overview of this procedure for the design point assessment of a given aircraft configuration is shown in Figure 5-1. For non-hybrid configurations, MTOW convergence typically happens in fewer than 20 iterations. For hybrid configurations, MTOW convergence typically happens in fewer than 30 iterations.

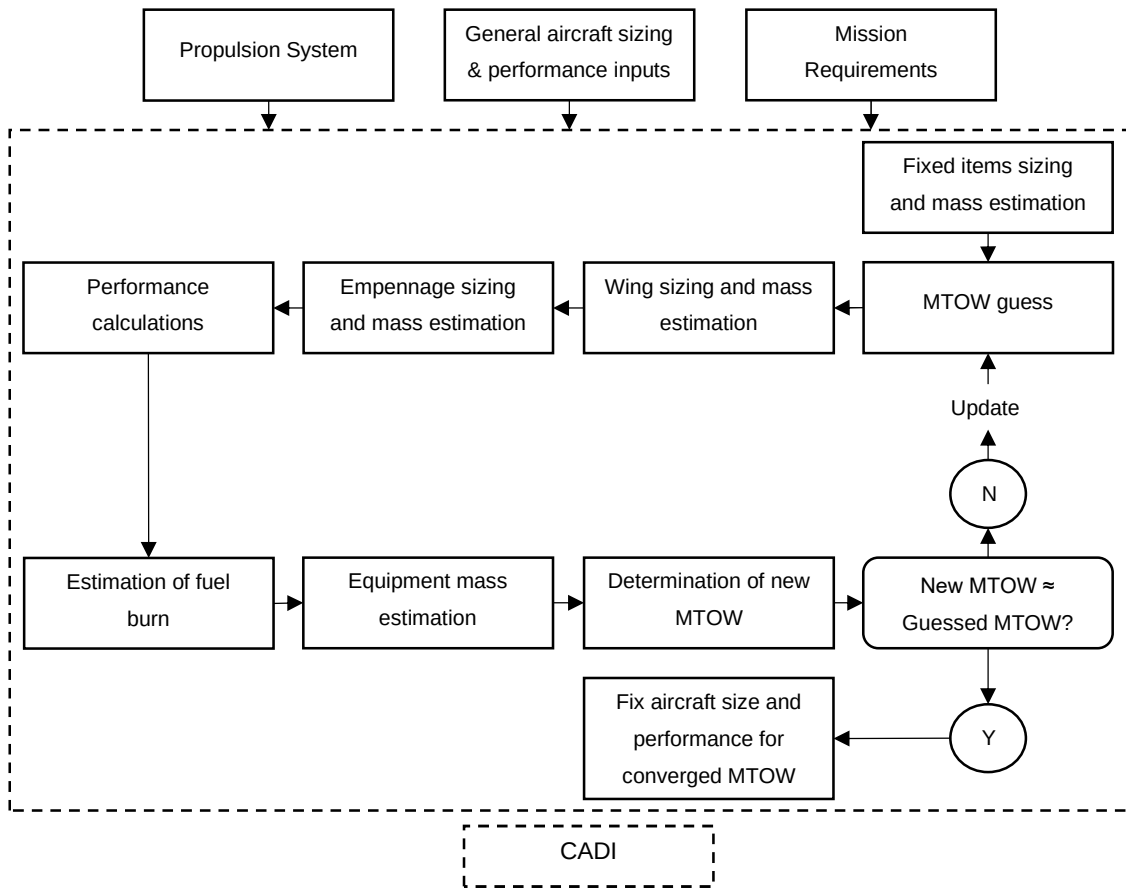


Figure 5-1 Aircraft design point assessment procedure in CADI

With aircraft components and MTOW fixed from design point simulation, aircraft performance calculations are carried for different payloads to determine aircraft range and thus payload-range points on the payload-range diagram. Additionally aircraft performance for off-design missions is determined by specifying new mission requirements and engine settings. With the new mission requirements and engine settings, aircraft off-design performance is determined by guessing an aircraft take-off weight (TOW) or take-off mass (TOM) and carrying out aircraft performance calculations until TOW converges.

5.2 Airframe Sizing

5.2.1 Wing

The wing is sized for the clean stall speed requirement, $U_{S, \text{clean}}$, at sea-level in Eq. 5-1, where S is the wing area, m_{TO} is aircraft take-off mass, $C_{L, \text{max, clean}}$ is

clean wing maximum 3D lift coefficient and g is gravitational acceleration. In Eq. 5-1, the value for $C_{L,max,clean}$ is set based on typical design values for a civil transport aircraft (usually between 1.5 and 1.6). With S determined, for a specified wing aspect ratio (AR_{wing}), wing span (b) is determined from Eq. 5-2.

$$S = \frac{2m_{TO}g}{\rho C_{L,max,clean} U_{S,clean}^2} \quad 5-1$$

$$b = \sqrt{AR_{wing} \cdot S} \quad 5-2$$

The wing geometry is parametrised in Figure 5-2 and accounts for wing kink, storage of fuel in the wings, sweepback and taper; where $\Lambda_{0.25}$ is quarter chord sweep angle, $\Lambda_{0.5}$ is half chord sweep angle, b_{cs} is wing centre section span, b_{fuel} is fuel tank span and b_{kink} is the centre section to kink distance. Wing section chords and $\Lambda_{0.5}$ are calculated as a function of b , specified taper ratios and specified $\Lambda_{0.25}$.

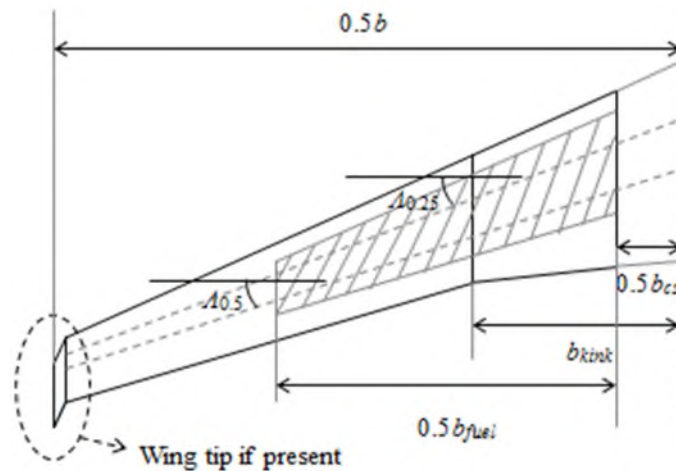


Figure 5-2 Transport tube-and-wing half-span geometry (shaded area represents stored fuel in the wing)

5.2.2 Fuselage

The fuselage is sized to hold the payload and is defined by the specification of five design parameters (see Figure 5-3).

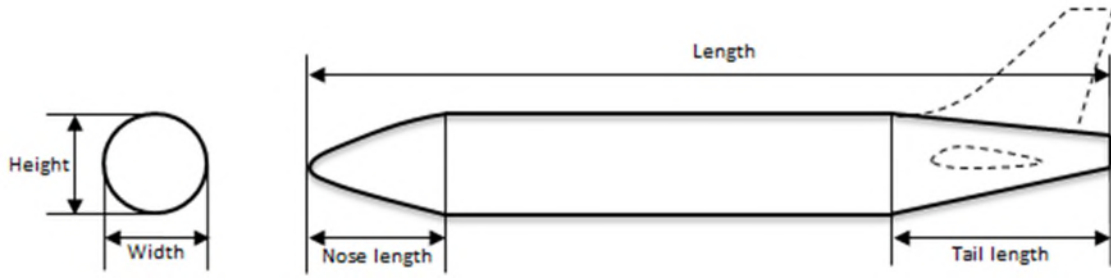


Figure 5-3 Fuselage sizing

5.2.3 Empennage

The horizontal and vertical tail of the empennage can be sized with empirical and geometry parameters using the volume coefficient method [2] & [3]. This approach is suitable for sizing a number of different tail types such as conventional tails and t-tails at the conceptual level. Using tail volume coefficients the area of the horizontal tail, S_H , is determined in Eq. 5-3 and the area of the vertical tail, S_V , is determined in Eq. 5-4, where c_{HT} is the horizontal tail volume coefficient, c_{VT} the vertical tail volume coefficient, MAC the mean aerodynamic chord of the wing and I_{arm} is the tail moment arm. Values for c_{HT} and c_{VT} are fixed by typical values for tail volume coefficients of historical aircraft designs.

$$S_H = \frac{c_{HT} \cdot S \cdot MAC}{I_{arm}} \quad 5-3$$

$$S_V = \frac{c_{VT} S b}{I_{arm}} \quad 5-4$$

By specifying an aspect ratio for the horizontal tail and vertical tail, tail span is determined in Eq. 5-5. Then for a defined tail taper ratio and quarter chord sweep angle, the tail root chord and half chord sweep angle are determined.

$$b_{tail} = \sqrt{AR_{tail} S_{tail}} \quad 5-5$$

5.3 Aircraft Component Mass Estimation

For the T&W airframe considered for TeDP in a HALE UAV application, the main structural components of the airframe were defined to be the wing, fuselage, empennage and tail boom [4]. For civil transport application in addition to these components (excluding the tail boom); operational items (crew & associated personal items, safety equipment such as emergency oxygen and life rafts, freight equipment, water and food), systems, aircraft equipment and the landing gear need to be accounted for. The mass of all these items are estimated using a combination of empirical and semi-empirical relations obtained from public literature. The relations are used for aircraft mass estimation in CADI. For the design of potential future aircraft utilising turboelectric propulsion, hybrid electric propulsion and structures utilising composite materials; modifications to were made to the equations utilised to estimate fuselage mass, empennage mass and installed powerplant mass.

The mass of the operational items, aircraft systems, aircraft equipment and landing gear are estimated using the empirical relations in [3]. The mass of the operational items are estimated in Eq. 5-6 where n_c is the number of crew, $n_{p,max}$ the maximum number of passengers an airframe can hold and F_{OP} is operating items factor whose value is dependent on the transport mission role. The mass of the systems, aircraft equipment and landing gear are aggregated as one item, m_{sys} , for mass estimation purposes. The value of m_{sys} is determined from Eq. 5-7 where C_4 is the systems factor whose value is dependent on the aircraft type.

$$m_{op} = 85n_c + F_{OP}n_{p,max} \quad 5-6$$

$$m_{sys} = C_4m_{TO} \quad 5-7$$

The quasi-analytical procedure by Torenbeek in [5] is used for the estimation of wing mass for the transport aircraft where in [5] wing mass is defined as the sum of the mass of the primary wing structure and secondary wing structure in Eq. 5-8. The primary structure consists of the wing panels, ribs, spars, webs and “non-ideal weights”; whereas the secondary structure consists of tip

structures, leading edge support & devices and trailing edge support & devices. Mass of the ideal primary structure is computed through integrating the span-wise material to resist bending and shear loads from the air-load distribution which fixes the ideal weight of the primary structure. Accounting for bending relief and mass penalties on the primary structure of the wing to allow for “non-optimum” effects such as torsional stiffness requirements, manholes, access hatches, fail safety, reinforcements, attachments and support structure; the primary structure mass is determined in Eq. 5-9 where R_{relief} is the bending relief factor. The bending relief factor accounts for bending relief from the weight of the wing, stored fuel in the wings, discrete engines attached to the wing and a negative relief from wing tips. For the secondary structure, statistical relations in [5] are used to determine mass. The method in [5] is applicable to cantilevered wings with a primary structure of predominantly Al-alloy.

$$m_{wing} = m_{prim} + m_{sec} \quad 5-8$$

$$m_{prim} = m_{prim,ideal} R_{relief} + \Delta m_{non\ ideal} \quad 5-9$$

For the structural assessment of the fuselage, the carrying of human passengers on board means that the fuselage has to be pressurised at altitude. Accounting for this and taking into account the possibility of a fuselage designed with advanced composites, the empirical relation for estimating the mass of a pressurised fuselage in [3] is used and modified with a fuselage material factor, $F_{M,fus}$, in Eq. 5-10, where L_{fus} is fuselage length, W_{fus} is fuselage width, h_{fus} is fuselage height, p is the cabin maximum working differential pressure in bar and C_2 is the fuselage mass coefficient which depends on the type of pressurised fuselage. The parameter $F_{M,fus}$ gives a rough estimate of the effect of designing the fuselage with different materials [2]. For a predominantly Al-alloy fuselage the value of $F_{M,fus}$ is unity and for an advanced composite fuselage $F_{M,fus}$ is set to have a value of 0.9.

$$m_{fus} = F_{M,fus} C_2 p (9.75 + 5.48 W_{fus}) \left(\frac{2L_{fus}}{W_{fus} + h_{fus}} \right) (W_{fus} + h_{fus})^2 \quad 5-10$$

The mass of the empennage for the transport aircraft is estimated using a modified form of the semi-empirical method in [3]. The modification accounts for design of the empennage with different materials through the definition of tail material factor, $F_{M,tail}$. The parameter $F_{M,tail}$ gives a rough estimate of the effect of designing the empennage with different materials [2]. With the modification horizontal tail mass is estimated in Eq. 5-11 and vertical tail mass is estimated in Eq. 5-12, where U_D is aircraft dive speed and k_{12} is the vertical tail mass factor. For a predominantly Al-alloy empennage the value of $F_{M,tail}$ is unity and for an advanced composite empennage $F_{M,tail}$ is set to have a value of 0.83. The value of k_{12} is dependent on the vertical location of the horizontal tail relative to the vertical tail. For a conventional tail k_{12} is equal to unity and for a T-Tail k_{12} is 1.5 [3].

$$m_H = F_{M,tail} 0.047 U_D S_H^{1.24} \quad 5-11$$

$$m_V = F_{M,tail} 0.065 k_{12} U_D S_V^{1.5} \quad 5-12$$

For estimating the mass of the powerplant at the aircraft level, the additional mass arising from the engine mounting, engine exhaust system, nacelle and fuel system needs to be considered alongside the basic powerplant mass. By the definition of a powerplant installation factor C_3 to account for these additional items, total installed powerplant mass is determined in Eq. 5-13 where m_{ENG} is the mass of the basic engine. The value for C_3 in Eq. 5-13 is dependent upon the type of aircraft and engine [3]. For turboelectric and HeP architectures installation of the electric machinery needs to be considered. To do this an electrical installation factor, C_{3E} , is considered to account for any mountings required for electric machinery. With this factor Eq. 5-13 is modified and becomes Eq. 5-14 where m_{elec} is the mass of the electric transmission (including cryocooler mass) and m_{ENG} is defined to be mass of mechanical sub-systems of the basic propulsion system.

$$m_{PP} = C_3 m_{ENG} \quad 5-13$$

$$m_{PP} = C_3 m_{ENG} + C_{3E} m_{elec} \quad 5-14$$

With the mass of the operational items, fuselage, empennage, wing mass and equipment mass and installed propulsion system mass determined; aircraft operating empty weight (OEW) or operating empty mass (OEM) is determined in Eq. 5-15 where m_{PP} is powerplant mass. For a determined OEW, TOW is determined as the sum of OEW, payload mass, a main mission fuel burn and a reserve fuel. Definitions for maximum zero fuel weight (MZFW) or maximum zero fuel mass (MZFM) and maximum landing weight (MLW) or maximum landing mass (MLM) are presented in appendix D.

$$OEW = m_{op} + m_{sys} + m_{fus} + m_{wing} + m_H + m_V + m_{PP} \quad 5-15$$

5.4 Drag Assessment

For assessment of aircraft performance, aircraft drag needs to be appropriately determined at each flight condition defined by mission requirements. For a given flight condition; total aircraft drag is determined as the sum of the zero-lift drag, lift-induced drag, wave drag and trim drag.

5.4.1 Zero-Lift Drag

Zero-lift drag is determined with the component build-up method. The method calculates the parasitic drag of airframe components through the accounting of skin friction drag, pressure drag due to viscous separation and interference drag [2] & [6]. The effect of skin friction drag is estimated through the skin friction coefficient, C_f , the effect of pressure drag is estimated through a component form factor, FF and interference effects are captured with an interference factor, Q_{int} . With these parameters for a given airframe component, zero-lift drag is estimated in Eq. 5-16 where S_{wet} is component wetted area and S_{ref} is the reference area used for the calculation of drag coefficient. For the assessment of aircraft drag S_{ref} is usually set to be equal to S [6] and thus for studies here

S_{ref} is fixed by S . For the estimation of total aircraft C_{D0} , C_{D0} is determined for each component of the airframe and then by summing C_{D0} for all airframe components, total aircraft zero-lift drag coefficient is determined (see Eq. 5-17).

$$C_{D0,component} = C_f \cdot FF \cdot Q_{int} \frac{S_{wet}}{S_{ref}} \quad 5-16$$

$$C_{D0,tot} = \sum C_{D0,component} \quad 5-17$$

For a turbulent boundary layer C_f is determined for each airframe component using the Prandtl-Schlichting formula (see Eq. 5-18).

$$C_f = \frac{0.455}{(\log Re)^{2.58} (1 + 0.144 M^2)^{0.65}} \quad 5-18$$

For the fuselage FF is determined with Eq. 5-19 where A_{fus} is fuselage cross-sectional area. Additionally for the fuselage Q_{int} is equal to 1.0 [6].

$$FF_{fus} = 1 + \frac{2.2}{\left[\frac{L_{fus}}{\left(\frac{4A_{fus}}{\pi} \right)^{0.5}} \right]^{1.5}} - \frac{0.9}{\left[\frac{L_{fus}}{\left(\frac{4A_{fus}}{\pi} \right)^{0.5}} \right]^3} \quad 5-19$$

For the wing FF is determined in Eq. 5-20 where FF^* is a parameter fixed by the thickness to chord ratio, tc . For a wing FF^* is defined in Eq. 5-21. For the wing Q_{int} is set to 1.0 [6].

$$FF = (FF^* - 1) \cos^2 \Lambda_{0.5} + 1 \quad 5-20$$

$$FF_{wing}^* = 1 + 3.3tc_{wing} - 0.008tc_{wing}^2 + 27tc_{wing}^3 \quad 5-21$$

For tail surfaces FF is determined in the same manner as the wing in Eq. 5-20 but with FF^* defined differently. FF^* for tail surfaces is defined in Eq. 5-22 and is fixed by empennage tc . For tail surfaces Q_{int} is set to 1.2 [6].

$$FF_{tail}^* = 1 + 3.52tc_{tail} \quad 5-22$$

In the consideration of nacelle drag, drag estimation is complicated by the intricate geometry of many nacelles and the inter-relationship with the definition of engine thrust. For initial estimates here and assuming wing-mounted engines, a value of 1.25 for the product of FF and Q_{int} is used [6].

5.4.2 Lift-Induced Drag

Lift-induced drag is assumed to be proportional to the square of the lift with a proportionality factor called the drag-due-to-lift factor (see Eq. 5-23 where C_{DI} is 3D lift induced drag coefficient and K is the drag-due-to-lift factor).

$$C_{DI} = KC_L^2 \quad 5-23$$

The value of K is determined using the Oswald Span Efficiency method where K is determined as a function of wing effective aspect ratio, AR_{eff} , and an Oswald span efficiency factor, e , which accounts for the extra drag generated by a wing due to non-elliptic lift distribution and wing separation drag (see Eq. 5-24) [2]. AR_{eff} is equal to wing AR except for the case where the wing has winglets. When the aircraft has winglets AR_{eff} is approximately equal to $1.2AR$.

$$K = \frac{1}{\pi \cdot AR_{eff} \cdot e} \quad 5-24$$

The Oswald span efficiency factor can be estimated based upon empirical data from actual aircraft. For leading edge sweep angles less, Λ_{LE} , than 30° e is determined in Eq. 5-25 and for Λ_{LE} 's greater than 30° e is determined in Eq. 5-26.

$$\Lambda_{LE} < 30^\circ \quad e = 1.78(1 - 0.045AR^{0.68}) - 0.64 \quad 5-25$$

$$\Lambda_{LE} > 30^\circ \quad e = 4.61(1 - 0.045AR^{0.68})(\cos \Lambda_{LE})^{0.15} - 3.1 \quad 5-26$$

5.4.3 Wave Drag

For aircraft configurations flying at transonic speeds, compressibility effects due to shockwave formation cause an increase in drag that is non negligible. For the assessment of this wave drag the wing is assumed to consist of a single strip and using Lock's fourth power law the wave drag coefficient is estimated in Eq. 5-27 where M_{cr} is the critical Mach number [7].

$$C_{DW} = \begin{cases} 0 & M \leq M_{cr} \\ 20(M - M_{cr})^4 & M > M_{cr} \end{cases} \quad 5-27$$

According to [7] M_{cr} is estimated as function of the drag divergence Mach number, M_{DD} , in Eq. 5-28.

$$M_{cr} = M_{DD} - \sqrt[3]{\frac{0.1}{80}} \quad 5-28$$

M_{DD} is estimated as a function of wing C_L , $\Lambda_{0.5}$, wing tc and a Korn factor κ_A in Eq. 5-29. For supercritical aerofoils $\kappa_A = 0.95$ and for conventional aerofoils $\kappa_A = 0.87$ [7].

$$M_{DD} = \frac{\kappa_A}{\cos \Lambda_{0.5}} - \frac{tc}{\cos^2 \Lambda_{0.5}} - \frac{C_L}{10 \cos^3 \Lambda_{0.5}} \quad 5-29$$

A limitation of wave drag modelling here is that the wing has been modelled as a single strip. For more rigorous calculations the wing should be modelled as multiple strips with the wave drag coefficient determined for each wing strip before overall wing wave drag coefficient is determined.

5.4.4 Trim Drag

It is possible for trim drag to be a significant contributor to total aircraft drag. At the initial project design stage, however, insufficient data is available to accurately estimate it. A typical figure for a well design aircraft is about 5 drag counts [6] and this value is used here for aircraft trim drag.

5.5 Flight Performance

The flight profile for a civil transport aircraft is split into several phases. These mission phases are take-off, climb, cruise, descent, approach and hold. Typically these phases can be split into two distinct mission profiles. These are the main mission profile and the reserve mission profile (see Figure 5-4). Under normal operating conditions the reserve mission is not flown but fuel that is used for the reserve mission must be carried on board the aircraft at all times and therefore has an impact on aircraft weight, aerodynamics and performance [8]. Thus flight performance is assessed for both the main mission and reserve mission. For flight analysis here start-up, taxi-out and taxi-in mission phases are not considered as the computation of these phases is not expected to significantly impact overall mission performance. Flight performance calculations for main mission and the reserve mission are coded in CADI.

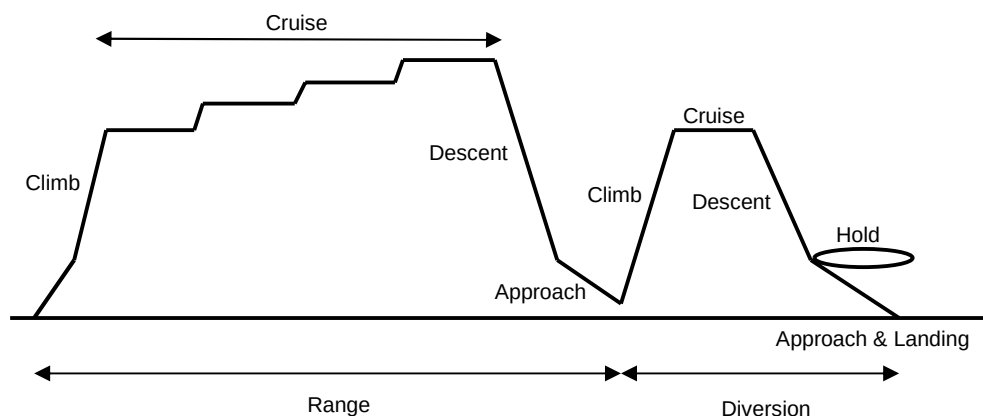


Figure 5-4 Flight mission with stepped cruise

5.5.1 Main Mission

For the main mission computation occurs sequentially to determine block fuel burn and total mission time for input propulsion system data and aircraft data (see Figure 5-5). With inputs defined, performance assessment starts with the estimation of Take-Off Field Length (TOFL) and take-off fuel burn through take-off calculations. For the determination of TOFL; input TOW, airframe size, flight condition, take-off BPR (ducted fans as propulsors), take-off net thrust (ducted fans as propulsors), shaft power (propellers as propulsors), propeller efficiency

(propellers as propulsors) and propeller diameter (propellers as propulsors) are required. For the estimation of take-off fuel burn, in addition to inputs needed for TOFL determination, by determining take-off SFC through input SFC-thrust polynomials take-off fuel burn is determined. Aircraft mass at the end-of-take-off is then determined by subtracting take-off fuel burn from aircraft TOW. Known aircraft mass at the end-of-take-off coupled with input airframe size, climb flight conditions, climb thrust at specified flight conditions and climb SFC-thrust polynomials; are used for climb calculations to determine climb fuel burn, time-to-height, top-of-climb conditions and horizontal distance covered during climb.

For the assessment of cruise performance, computation requires the top-of-climb conditions and the horizontal distance flown during cruise to be known. However for an input aircraft range horizontal distance flown during cruise is unknown because descent has not been computed. To tackle this; for input aircraft range and landing field length (LFL); the horizontal distance covered during descent is initially guessed and horizontal distance covered during cruise is determined by subtracting TOFL, LFL, horizontal distance covered during climb and horizontal distance covered during descent from the overall aircraft range requirement. Cruise calculations are then carried for input airframe size and cruise SFC-thrust polynomials to determine cruise fuel burn, time elapsed during cruise and top-of-descent conditions. Known top-of-descent conditions coupled with input descent flight conditions and engine idle performance are used for descent calculations to determine descent fuel burn, time elapsed during descent, end-of-mission aircraft mass and a new horizontal distance covered during descent. After descent calculations are carried out a check is made to see if calculated horizontal distance covered during descent matches the guessed value to within a set convergence margin. If it does not match then the guess for horizontal distance covered during descent is updated with calculated value and horizontal distance travelled during cruise is re-calculated. With horizontal distance travelled during cruise recalculated, cruise and descent calculations are repeated until guessed horizontal distance covered during descent matches calculated horizontal distance covered during descent. Subsequently total main mission time is determined by summing the time

elapsed during each mission phase and main mission block fuel is determined by summing the fuel consumed during each mission phase.

Within CADI for the simulation of main mission performance, eleven climb flight conditions and eleven descent flight conditions are required for performance computation. For cruise performance up to 10 flight conditions for a stepped cruise profile can be specified.

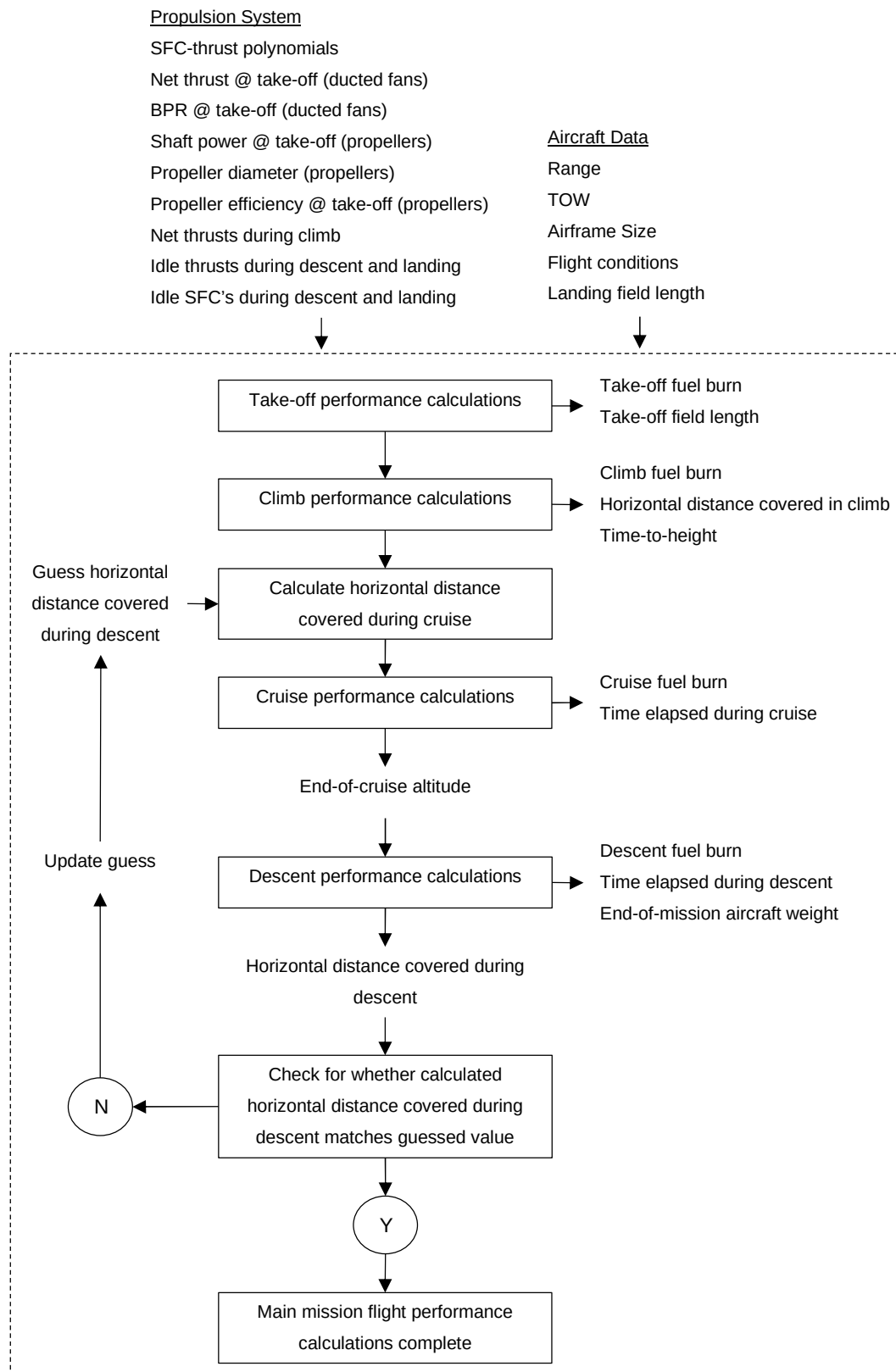


Figure 5-5 Overview of main mission performance assessment procedure

5.5.2 Reserve Mission

For the reserve mission similar to the main mission computation occurs sequentially to determine reserve mission time and fuel burn for input propulsion system and aircraft data (see Figure 5-6). For these inputs performance assessment starts with climb performance calculations to estimate climb fuel burn, horizontal distance covered during climb, time-to-height and top-of-climb conditions. For cruise performance calculations similar to the main mission, for input aircraft diversion range the horizontal distance covered during descent is initially guessed and horizontal distance covered during cruise is determined by subtracting horizontal distance covered during climb and horizontal distance covered during descent from the diversion range requirement. Cruise calculations are then carried out to determine cruise fuel burn and time elapsed during cruise. After cruise calculations are carried out, descent calculations from top-of-descent to specified hold flight condition are carried out. At hold, loiter calculations estimate hold fuel burn. After hold descent resumes and performance calculations are completed to determine descent fuel burn, time elapsed during descent and a new horizontal distance covered during descent. After descent, calculations are completed a check is made to see if calculated horizontal distance covered during descent matches the guessed value to within a set convergence margin. If it does not match then the guess for horizontal distance covered during descent is updated with the calculated value and horizontal distance covered during cruise is re-calculated. With horizontal distance covered during cruise re-calculated; cruise, hold and descent calculations are repeated until guessed horizontal distance covered during descent matches calculated horizontal distance covered during descent. Subsequently total reserve mission time is determined by summing the time elapsed during each reserve mission phase.

Within CADI for the simulation of reserve mission performance, up to ten flight conditions for climb to diversion cruise altitude can be specified. For descent from diversion cruise altitude to approach up to ten flight conditions can be specified. For diversion cruise and loiter at hold flight condition only one flight condition is specified for each in CADI.

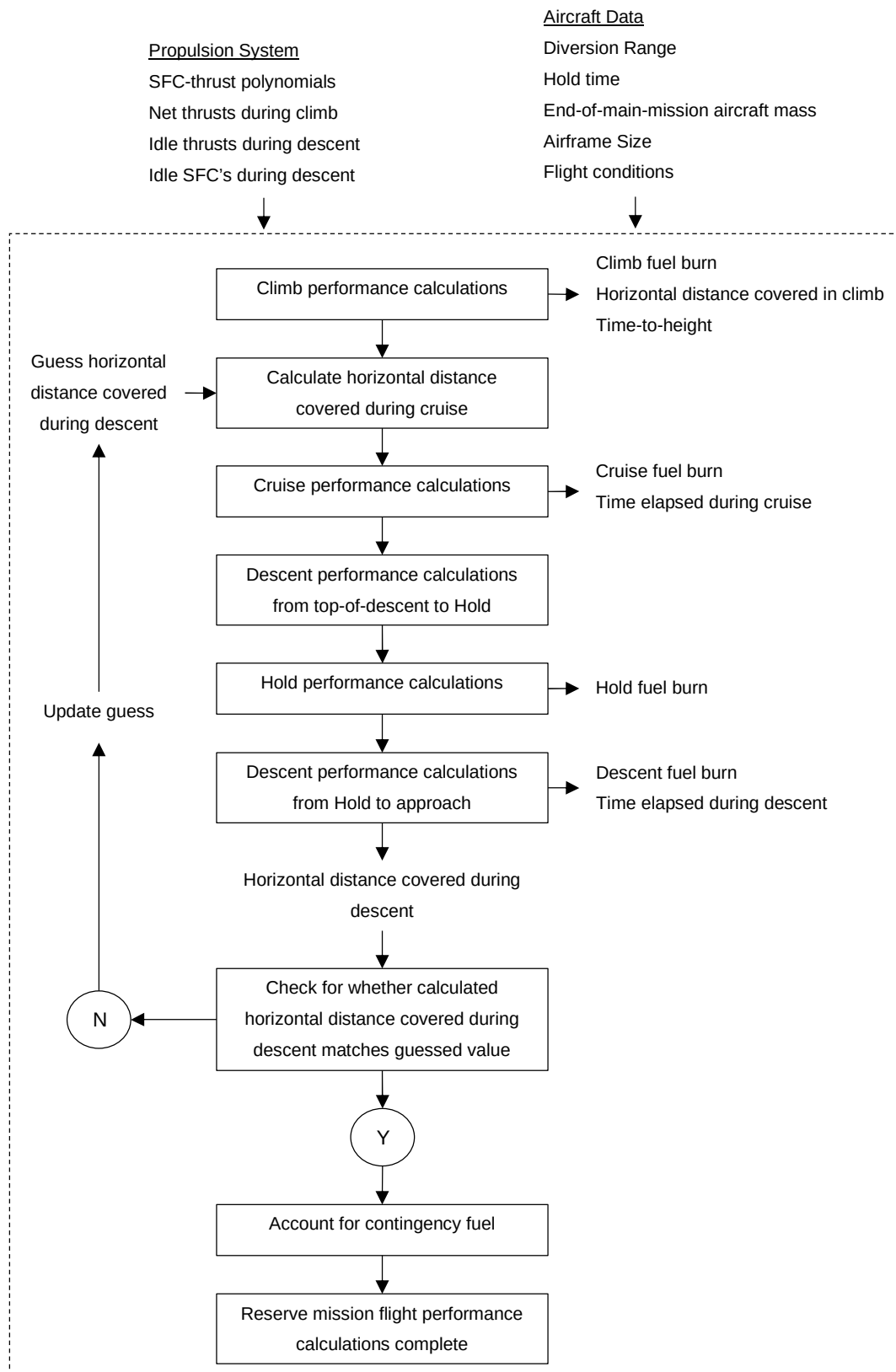


Figure 5-6 Overview of reserve mission performance assessment procedure

According to ICAO Annex 6, Part 1, Section 4.3.6; to determine total amount of reserve fuel; contingency fuel needs to be considered alongside the diversion and hold [9]. The contingency fuel is estimated as a percentage of main mission block fuel [6] & [8] and the typical value for this percentage is 5% according to EU-OPS 1.255 [10]. With contingency fuel estimated as 5% of main mission block fuel; total reserve mission fuel is estimated as the sum of diversion, hold and contingency fuel.

5.5.3 Take-Off

Take-off performance is modelled using the “balanced field length” (BFL) approach. With the approach BFL is the total take-off distance including obstacle clearance when an engine fails at a specified decision speed, the speed at which, upon an engine failure, the aircraft can either brake to a halt or continue the take-off in the same total distance. This BFL is estimated using the empirical relation in Eq. 5-30 [2] and TOFL is set to be equal to BFL.

$$\frac{BFL}{0.3048} = \frac{0.863}{1 + 2.3B_1} \left(3.28084 \frac{\frac{F_W}{S}}{\rho \cdot g \cdot C_{L,climb}} + h_{obstacle} \right) \left(\frac{1}{\frac{F_{avg}}{F_W} - B_2} + 2.7 \right) + \left(\frac{655}{\sqrt{\frac{\rho}{\rho_{SL}}}} \right) \quad 5-30$$

For jet powerplants average thrust during take-off, F_{avg} , is estimated in Eq. 5-31 where $F_{takeoff,static}$ is take-off static thrust. The relation gives fairly accurate results for jet powerplants up to a BPR of 10.

$$F_{avg} = 0.75 F_{takeoff,static} \left(\frac{5 + BPR}{4 + BPR} \right) \quad 5-31$$

For propeller powerplants F_{avg} is estimated in Eq. 5-32.

F_{avg}

$$= 5.75(0.00134102PW_{propeller,tot}) \left[\frac{\left(\frac{\rho}{\rho_{SL}}\right) n_{propeller} (3.28084 d_{propeller})^2}{0.00134102PW_{propeller,tot}} \right]^{\frac{1}{3}} \quad 5-32$$

In Eqs. 5-30 to 5-32 the following parameters are defined:

- ρ_{SL} is sea-level air density
- F_W is aircraft weight
- $B_1 = \gamma_{flight,climb} - \gamma_{flight,min}$
- $\gamma_{flight,climb} = \arcsine[(F - F_D)/F_W]$, 1-engine-out, climb speed, end-of-runway
- $\gamma_{min} = 0.024$ 2-engine; 0.027 3-engine; 0.030 4-engine
- $C_{L,climb}$ is lift coefficient at $1.2U_S$
- $h_{obstacle}$ is obstacle height in feet; 35 ft for commercial aircraft, 50 ft for military aircraft
- $B_2 = 0.01C_{L,max} + 0.02$ for flaps in take-off position

Take-off fuel burn is estimated by assuming take-off lasts for two minutes [6] and multiplying this duration by engine end-of-runway thrust and end-of-runway SFC. End-of-runway SFC is determined as a function of end-of-runway thrust from input thrust-SFC polynomial for this flight condition.

5.5.4 Climb

The accurate prediction of climb performance is complex due to the fact that many flight parameters change during the climb. Thrust, drag and weight all vary with altitude, flight speed and fuel burn. While analytical derivation of these variations is possible, analytical integration over the height of climb is fraught with difficulty [3]. For the purposes of aircraft synthesis this problem is overcome to some extent by limiting the climb to a constant calibrated airspeed (CAS) climb until cruise Mach number is reached; and then continuing climb at constant Mach number. Additionally for civil aircraft operation, Aircraft Traffic

Control requirements usually limit the initial climb to 250 knots CAS up to a height of 10,000 ft.

To allow for the computation of time elapsed, horizontal distance covered and fuel consumed during climb, the climb is divided into multiple altitude steps. With this done climb calculations start with the determination of the rate-of-climb, RoC [8]. At conditions of zero gust and crosswinds RoC is equal to the change in height with respect to time (see Eq. 5-33 where h is flight altitude)

$$RoC = \frac{dh}{dt} \quad 5-33$$

Taking into account the forces acting on the aircraft, assuming aircraft lift (F_L) is equal to aircraft weight and specifying propulsion system thrusts at climb flight conditions, aircraft rate-of-climb with no acceleration is given by Eq. 5-34 where U is aircraft true air speed (TAS) and m is instantaneous aircraft mass. For the calculation in Eq. 5-34, U is determined from specified aircraft climb flight condition and drag is determined after computing the drag coefficient for the specified flight condition.

$$RoC_0 = \frac{F - F_D}{m \cdot g} U \quad 5-34$$

Since the aircraft is not climbing at constant TAS, an acceleration factor needs to be taken into account to capture the physical effect of the aircraft accelerating as it climbs [6] & [11]. The acceleration factor is a correction for the fact that the aircraft is not climbing at constant TAS and is defined in Eq. 5-35. With the acceleration factor defined, the rate-of-climb with acceleration is given in Eq. 5-36. At top-of-climb and end-of-initial climb to 10,000 ft, the acceleration factor has a value of 1.

$$K_A = 1 + \frac{U}{g} \frac{dU}{dh} \quad 5-35$$

$$RoC_a = \frac{RoC_0}{K_A} \quad 5-36$$

The time, horizontal distance covered and weight (thus fuel consumed) of the aircraft at the end of each climb step depends on the RoC . The RoC in turn depends of the drag, thrust and mass of the aircraft during the climb step. The first step towards solving these parameters comes from the determination of the time to climb from a lower altitude h_1 to a higher altitude h_2 . The time to climb from h_1 to h_2 is calculated in Eq. 5-37.

$$t = \int_{h_1}^{h_2} \frac{1}{RoC_a} dh \quad 5-37$$

The integration of Eq. 5-37 is not straightforward because of the interrelationships of the variables involved. For this reason an iterative procedure is adopted for each climb step. For the procedure an initial step fuel burn guessed. From the initial guess for step fuel burn, aircraft weight at the end of the climb step is determined and subsequently RoC_a is determined from Eq. 5-36. With RoC_a determined, the time elapsed for a climb step is found from Eq. 5-37. With engine SFC determined as a function of thrust from input thrust-SFC polynomial for a given climb flight condition, a new climb step fuel burn is estimated with Eq. 5-38 and 5-39. If the calculated new climb step fuel burn does not match the guessed step fuel burn to within a set convergence margin, the guessed step fuel burn is updated with the calculated value and climb calculations for the climb step are repeated until calculated fuel burn matches guessed fuel burn.

$$W_F = SFC \frac{F}{n_{GT}} \quad 5-38$$

$$m_{fuel} = n_{GT} W_F t \quad 5-39$$

The total time spent during climb is calculated by summing the time spent at each climb step in the climb profile (see Eq. 5-40).

$$t_{climb} = \sum_{i=1}^n t_i \quad 5-40$$

Horizontal distance covered for each climb step was estimated as the product of CAS and time elapsed for a given climb step. For more accurate calculations the horizontal distance covered for a climb step should be estimated as the product of time elapsed, TAS and the cosine of the flight path angle. The total horizontal distance covered during climb is calculated by summing the horizontal distance covered during each climb step in the climb profile (see Eq. 5-41 where s is horizontal distance travelled and s_{climb} is total distance travelled during climb).

$$s_{climb} = \sum_{i=1}^n s_i \quad 5-41$$

In a similar manner total fuel consumed during climb is computed by summing the fuel consumed at each climb step.

5.5.5 Cruise

The total distance travelled during cruise for an aircraft in straight line un-accelerated flight for a given change in aircraft mass is given in Eq. 5-42.

$$s = - \int_{m_1}^{m_2} \frac{U \frac{F_L}{F_D}}{SFC \cdot m \cdot g} dm \quad 5-42$$

In the classical solution of Eq. 5-42 and assuming straight and level flight; aerodynamic efficiency, SFC and flight speed are kept constant for a given

change in aircraft mass from m_1 to m_2 to form the Breguet Range Equation (see Eq. 5-43).

$$s = \frac{U}{SFC} \frac{\frac{F_L}{F_D}}{g} \ln\left(\frac{m_1}{m_2}\right) \quad 5-43$$

The above equation implies that for constant TAS the aircraft is constantly climbing to maintain the same lift coefficient and aerodynamic efficiency. This is because for constant TAS and lift coefficient, the density reduction has to be proportional to reduction in aircraft weight as fuel is burned. This is only possible if the aircraft gradually climbs as cruise progresses. This density reduction effect is underestimated in Eq. 5-43, as a reduction in density will change parasitic drag appreciably. As a result aerodynamic efficiency will change as aircraft weight and altitude changes. In addition, SFC will change with changes in altitude. Also in practice, air traffic control does not allow cruise-climb profiles and instead insist that airlines adopt a stepped cruise-climb profile [6] & [8].

According to [12] a better approach for estimating the distance travelled during cruise for an aircraft is to perform a numerical integration of the basic range equation in Eq. 5-42. The technique is known as the integrated range method and the range is given by dividing aircraft mass change in 'n' steps so that distance travelled during cruise is given by Eq. 5-44.

$$s_{cruise} = \sum_{i=1}^n \frac{U_i}{W_{F,i}} (m_i - m_{i-1}) \quad 5-44$$

Provided mass steps are small enough the approach in Eq. 5-44 accounts for constant TAS and altitude flight as well as variations in SFC and drag. For performance assessment here instead of dividing the mass of the aircraft into discretized mass steps and calculating range, time and fuel consumed; the cruise is divided into a number of steps and for a specified distance travelled and time interval, fuel burn during cruise is determined with Eq. 5-45 for a given flight condition. Additionally for a given time interval, the range covered during each cruise step is given in Eq. 5-46 and total calculated distance travelled

during cruise is equal to the number of cruise steps multiplied by range covered for a time step. In Eq. 5-45 SFC is determined as a function of calculated drag from input SFC-thrust polynomial for specified cruise flight condition. Also, the time interval in Eq. 5-45 is set so that aircraft cruise ends near the point where calculated distance flown during cruise matches specified distance travelled during cruise and thus n in Eq. 5-45 is known once a total cruise time is estimated. Once total cruise time is estimated n is determined by dividing total cruise time by set time step.

$$m_{fuel,cruise} = \sum_{i=1}^n \frac{SFC_i F_{W,i} \Delta t}{\left(\frac{F_L}{F_D}\right)_i} \quad 5-45$$

$$ds = U dt \quad 5-46$$

To improve overall efficiency airliners are allowed to perform a step climb cruise. By specifying required altitudes the aircraft can fly at during cruise, aircraft cruise altitude is set to increase when the specific air range of flying at a higher altitude is greater than specific air range of continuing flight at a lower cruise altitude.

5.5.6 Descent

For descent computation is carried out in a similar way to climb calculations with the exception being that drag is greater than thrust and therefore the aircraft glides during descent. The rate-of-descent during the glide with no acceleration, RoD_0 , is given in Eq. 5-47. Thrust in Eq. 5-47 is fixed by engine idle thrust for a specified flight condition.

$$RoD_0 = \left(\frac{F_D - F}{F_W} \right) U \quad 5-47$$

The rate-of-descent with acceleration, RoD_a , is determined in Eq. 5-48.

$$RoD_a = \frac{RoD_0}{K_A} \quad 5-48$$

The time to descend for a given descent step from a higher altitude h_1 to a lower altitude h_2 is given in Eq. 5-49.

$$t = \int_{h_2}^{h_1} \frac{1}{RoD_a} dh \quad 5-49$$

To solve Eq. 5-49 an initial step fuel burn is guessed. From the initial guess for step fuel burn, aircraft weight at the end of the descent step is determined and subsequently RoD_a is determined from Eq. 5-48. With RoD_a determined the time to complete a descent step is solved with Eq. 5-49. With engine SFC fixed by input engine idle SFC for a specified descent flight condition, a new descent step fuel burn is determined through Eq. 5-38 and Eq. 5-39. If the calculated new descent step fuel burn does not match the guessed step fuel burn to within a set convergence margin the guessed step fuel burn is updated with the calculated value and descent calculations for the descent step are repeated until calculated fuel burn matches guessed fuel burn. Total time spent during descent is then calculated by summing the time spent at each descent step in the descent profile (see Eq. 5-50).

$$t_{descent} = \sum_{i=1}^n t_i \quad 5-50$$

Horizontal distance covered for each descent step was estimated as the product of CAS and time elapsed for a given descent step. For more accurate calculations the horizontal distance covered for a descent step should be estimated as the product of time elapsed, TAS and the cosine of the flight path angle. The total horizontal distance covered during descent is calculated by summing the horizontal distance covered during each descent step in the descent profile (see Eq. 5-51).

$$S_{descent} = \sum_{i=1}^n S_i \quad 5-51$$

In considering the speed the aircraft flies at during descent, the descent starts from end-of-cruise altitude with a speed corresponding to cruise Mach number and this continues to 10,000 ft. From 10,000 ft descent continues at 250 kts CAS till approach. If cruise CAS is less than 250 kts, descent is carried out at cruise CAS from end-of-cruise altitude to approach.

5.5.7 Hold

Hold calculations are computed in a similar way to cruise calculations with the exception being that time spent during the mission phase is known before calculations are carried out. This is because time spent at hold is usually set by airworthiness regulations. For a specified flight condition and hold time, hold fuel burn is determined in Eq. 5-52.

$$m_{fuel,hold} = \frac{SFC \cdot F_W \cdot t_{hold}}{\left(\frac{F_L}{F_D}\right)} \quad 5-52$$

5.5.8 Landing & Approach

Landing fuel burn is estimated by assuming landing lasts for six minutes [6] and multiplying this duration by engine idle thrust at approach and engine idle SFC at approach. LFL is specified as a main mission performance input for a given aircraft configuration. Approach speed before landing is specified to be 1.3 multiplied by U_S .

5.6 Cryogenic Fuel Use

An identified technology option for TeDP application in the methodology was the use of hydrogen as a fuel. As previously mentioned benefits of using hydrogen as a fuel are its high gravimetric energy density and zero CO₂ emissions produced from its combustion when compared to kerosene. However

hydrogen's volumetric energy density at standard conditions is significantly lower than kerosene's. This is why for aircraft application hydrogen has been assessed in its liquid form in previous studies [4], [13] [14] & [15] to minimise volume penalty on the aircraft. Since hydrogen only exists as a liquid at cryogenic temperatures, cryotanks are required for the storage of LH₂. For performance assessment at the aircraft level, size and mass of LH₂ cryotanks needs to be determined to adequately assess the impact of using LH₂ as a fuel. To do this hydrogen use is modelled at two levels, with these levels being the tank level and the higher aircraft level.

5.6.1 Modelling at the Tank Level

Hydrogen cryotanks are assumed to be made of composite materials because of recent developments in composite material application for cryogenic hydrogen storage in space application [4] coupled with the fact that at present composite cryotanks achieve higher gravimetric efficiencies than state-of-the-art metallic tanks [16]. For the present study tanks are assumed to be cylindrical in shape although ideally tanks should either be spherical in shape or cylindrical with hemispheric heads, however, this discrepancy is not expected to make a significant impact on overall performance trends. Clean tank (without insulation) gravimetric efficiency is set to be 93.7%. This value is based on public domain information about the Boeing composite cryotank design in [16] and setting LH₂ density to be 71 kg/m³. The Boeing composite cryotank design had a 'fluted-core' tank wall structure to reduce overall tank weight where and thus it is assumed cryotanks to be assessed have this design. Meanwhile, clean tank gravimetric efficiency is defined in Eq. 5-53 where $\eta_{grav, clean}$ is clean tank gravimetric efficiency and $m_{tank, clean}$ is clean tank mass; and insulated tank gravimetric efficiency is determined in Eq. 5-56 where $\eta_{grav, clean, insulation}$ is insulated tank gravimetric efficiency and $m_{insulation}$ is the mass of tank insulation. For tank ullage consideration, tank volume is assumed to be 2% greater than fuel volume to account for contingencies and to permit unforeseen expansion of LH₂ [17].

$$\eta_{grav, clean} = \frac{m_{fuel}}{m_{fuel} + m_{tank, clean}} \quad 5-53$$

$$\eta_{grav, ins} = \frac{m_{fuel}}{m_{fuel} + m_{tank, clean} + m_{insulation}} \quad 5-54$$

In the consideration of boil-off from the cryotank there are two types of boil-off that need to be accounted for. These are a boil-off due to vented gas and a boil-off due to pressurant gas. Typically the two boil-offs are of the same order of magnitude [18] and for studies carried out here the boil-off due to pressurant gas is assumed to be equal to the boil-off due to vented gas. To minimise boil-off due to vented gas, Load-Responsive Multilayer Insulation (LRMLI) is selected as insulation choice. LRMLI is chosen over foams and aerogels because for a given heat leak LRMLI insulation is lighter [19].

Accurate thermal performance modelling and weight prediction for the cryogenic hydrogen storage system are beyond the scope of this thesis. However for assessment at the aircraft level the estimation of the storage system's vented gas boil-off and installed/insulated gravimetric efficiency is carried out.

For the estimation of vented gas boil-off, according to [4] vented gas boil-off is estimated as a function of heat flux across the insulation, q , flight time, tank surface area, $A_{surf, tank}$ and fuel latent heat (see Eq. 5-55). For hydrogen fuel latent heat is 446,000 J/kg.

$$m_{fuel, boil, vent} = \frac{q \cdot t \cdot A_{surf, tank}}{446000} \quad 5-55$$

The heat flux across the insulation is estimated using the semi-empirical relation in [15] which estimates insulation thermal performance in Eq. 5-56 by accounting for the effects of ambient temperature and different insulation layer materials. In Eq. 5-56 C_{t1} , C_{t2} and C_{t3} are empirical coefficients that describe insulation layer materials, T_H and T_C are hot and cold side temperature respectively, N_t is the layer density of the insulation in layers/cm, n_{layer} is the total number of layers, C_g is a coefficient that is fixed by the interstitial gas, n_g is

an exponent that is fixed by the interstitial gas, P_{int} is interstitial gas pressure in torr and ϵ is the emittance of the layers.

$$q = C_{t1} \frac{N_t^{C_{t2}} (T_H + T_C) (T_H - T_C)}{2(n_{layer} + 1)} + C_{t3} \epsilon \frac{T_H^{4.67} - T_C^{4.67}}{n_{layer}} + \frac{C_g P_{int} (T_H^{n_g} - T_C^{n_g})}{n_{layer}} \quad 5-56$$

For estimation of $m_{fuel,boil,vent}$ in Eq. 5-55, q is estimated at a specified cruise flight condition because this is the condition an aircraft carrying cryogenic fuel on-board will spend the largest proportion of its time in flight. Then for a known reference amount of fuel and fuel density, the fuel volume is determined. After fuel volume is determined, by specifying a reference tank radius, overall tank dimensions and $A_{surf,tank}$ are determined. With $A_{surf,tank}$ and q found, $m_{fuel,boil,vent}$ in Eq. 5-55 is then found for a reference flight time. To then find fraction of fuel boiled off due to gas vent off, $m_{fuel,boil,vent}$ is divided by the reference amount of fuel. An overview of this procedure to determine vented gas boil-off is shown in Figure 5-7. The procedure was implemented into a tool coded with MATLAB. With boil-off due to pressurant gas assumed to be equal to boil-off due to vented gas, overall boil-off fraction is estimated by multiplying vented gas boil-off fraction by two.

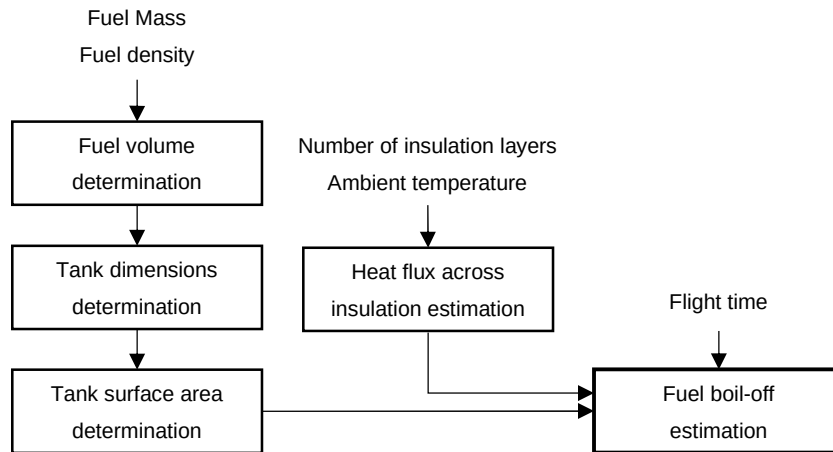


Figure 5-7 Vented gas boil-off estimation procedure

In order to estimate mass of the hydrogen storage system at the aircraft level, mass of the clean tank and insulation needs to be estimated. For a reference amount of fuel, clean tank mass is fixed by specified clean tank gravimetric

efficiency. From [19] experimental data correlating insulation mass per unit area for one to four insulation layers was obtained. This data is used for the estimation of insulation mass here. For layers greater than four, insulation mass per unit area is obtained by linear extrapolation of the data in [19]. Thus for a specified number of insulation layers, insulation mass per unit area is obtained from the experimental data. With tank surface area already determined, insulation mass is now obtained by multiplying tank surface area by estimated insulation mass per unit area. The overall storage mass is then estimated by summing insulation mass and clean tank mass. Then installed tank gravimetric efficiency is determined by dividing the reference fuel amount by the summation of reference fuel, tank mass and insulation mass. An overview of the procedure for tank with insulation gravimetric efficiency estimation procedure is shown in Figure 5-8. This procedure was implemented into a tool coded with MATLAB.

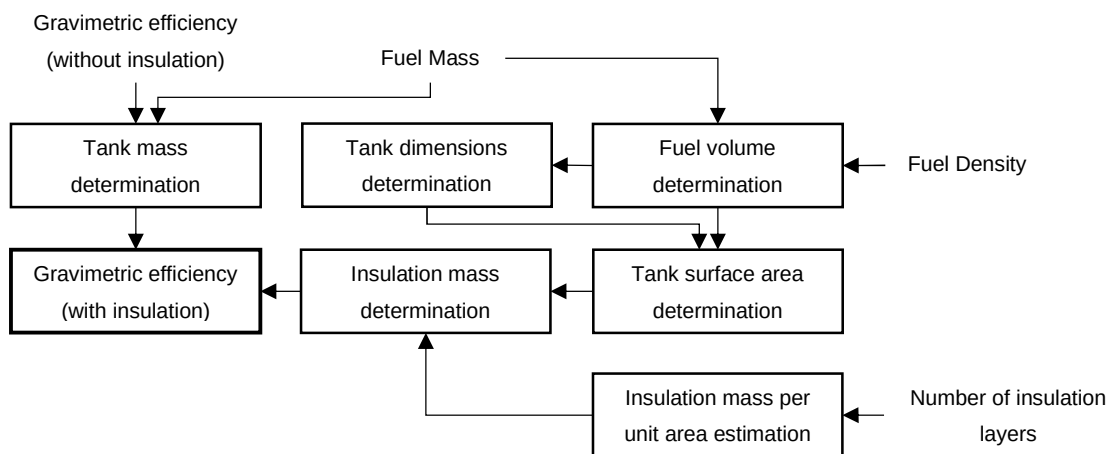


Figure 5-8 Tank with insulation gravimetric efficiency estimation procedure

5.6.2 Modelling at the Aircraft Level

5.6.2.1 Cryotank Integration

To assess impact of using cryogenic fuel at the aircraft level, tank design parameters that directly affect the aircraft design need to be identified. For studies here these design parameters are the boil-off fraction, tank ullage, cryogenic fuel density and tank gravimetric efficiency. These parameters

characterise the effect of the hydrogen storage system on total fuel stored in airframe, fuel volume requirements, tank size and tank mass.

For the integration of cryotanks with the aircraft it is assumed that cryotanks are installed in the fuselage and that cryotanks are sized for the maximum fuel mission at MTOW (see Figure 5-9). Cryotanks are installed in the fuselage rather than the wing because their installation in a wing section would result in thick wing section which would be aerodynamically inefficient. Since cryotanks are installed in the fuselage, kerosene fuel tanks in the wing are eliminated and thus bending relief in wing due to presence of fuel is eliminated. Also, cryotanks are sized for the maximum fuel mission at MTOW instead of the design mission because the maximum fuel mission represents the condition that imposes the largest fuel requirement on the cryogenic fuel storage system.

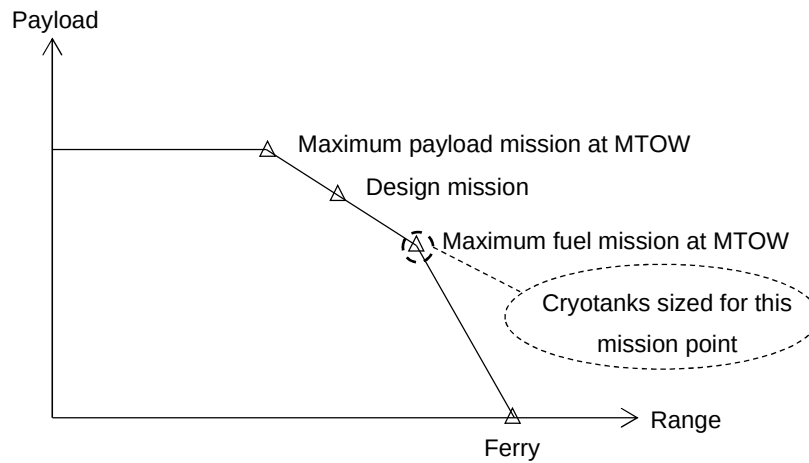


Figure 5-9 Typical payload-range diagram showing cryotanks sizing point

With tanks integrated into the aircraft, the OEW changes from the definition in Eq. 5-15 to the form in Eq. 5-57 where m_{tank} is total tank mass with insulation.

$$OEW = m_{op} + m_{sys} + m_{fus} + m_{wing} + m_H + m_V + m_{PP} + m_{tank} \quad 5-57$$

5.6.2.2 Fuselage Sizing and Mass Estimation

From [14] for medium haul application installed cryotanks in the fuselage can be located both the aft section of the fuselage and above the passenger compartment for the least increase in construction weight and wetted area. For modelling of the cryotanks here, the ratio in volume between aft tanks and tanks above the passenger compartment is set to 1:1 and fuselage width is kept constant as fuselage height and length change as a result of tank installation (see Figure 5-10 where nose length and tail length have been kept constant from Figure 5-3). For aircraft sizing & performance assessment purposes it is assumed that 'rear' tank diameter is fixed by the nominally specified fuselage height when the fuselage does not hold cryotanks (see Eq. 5-58). Thus as stored fuel in the rear tank increases, tank diameter remains constant but tank length increases. For the 'upper' tank its length is fixed by both the nominally specified fuselage length when the fuselage does not hold cryotanks and rear tank length (see Eq. 5-59 where L_{tail} is tail length and L_{nose} is nose length). Thus size changes of the upper tank are dependent on two variables. The first being its length is fixed by changes in fuel stored in the rear tank. The second is due to stored fuel increase in the upper tank, as fuel stored in the upper tank increases, tank diameter increases and this has the effect of increasing overall fuselage height.

$$d_{tank, rear} \approx h_{fus, nom} \quad 5-58$$

$$L_{tank, upper} = L_{fus, nom} - (L_{tail} + L_{nose}) + L_{tank, rear} \quad 5-59$$

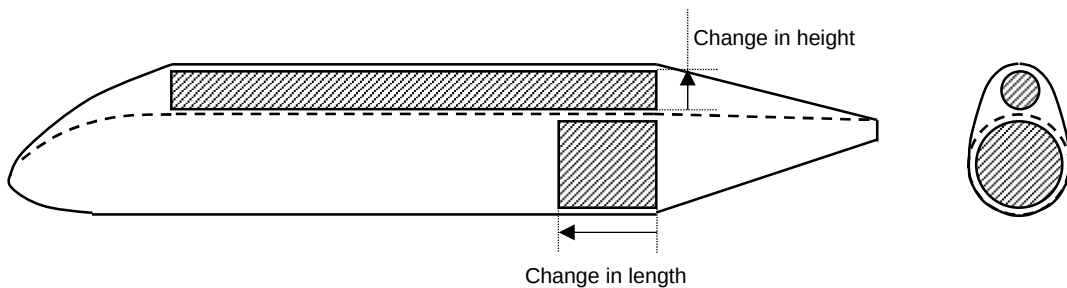


Figure 5-10 Cryogenic tube-and-wing fuselage

For the non-cryogenic tube-and-wing fuselage mass was estimated using the semi-empirical relation in Eq. 5-10. To account for possible fuselage weight increase due to the need for additional reinforcements to house tanks in the fuselage, an arbitrary cryogenic fuselage structural factor, F_{cryo} , is applied to Eq. 5-10 to yield Eq. 5-60 which is then used for the determination of fuselage mass. Values greater than one for the cryogenic fuselage structural factor means fuselage mass has been penalised by need for structural reinforcement. In Eq. 5-60 L_{fus} is set as the sum of $L_{fus,nom}$ and the change in fuselage length due to the need to hold cryotanks, and h_{fus} is the sum of $h_{fus,nom}$ and the change in fuselage height due to the need to hold cryotanks.

$$m_{fus} = F_{cryo} F_{M,fus} C_2 p (9.75 + 5.48 W_{fus}) \left(\frac{2L_{fus}}{W_{fus} + h_{fus}} \right) (W_{fus} + h_{fus})^2 \quad 5-60$$

5.6.2.3 Aircraft Sizing and Performance Assessment

For the sizing and performance assessment of the aircraft with cryogenic fuel, different from kerosene powered aircraft consideration needs to be made to the fact that fuselage size and mass is no longer fixed by payload alone and that it is now a variable item that changes as the amount of fuel stored on board changes. Thus the fixed items for the calculation independent of the fuel burn are the specified design payload and operational items. Taking this into account, for input propulsion system, mission requirements and general aircraft sizing & performance inputs; performance assessment starts with an initial sizing and mass estimate of the fuselage. Initial fuselage size is fixed by fuselage length, height and width when the aircraft is not carrying any cryogenic fuel. With this carried out initial fuselage mass is estimated with Eq. 5-60. After this is carried out an initial MTOW is guessed so that wing size and mass can be determined. With wing size determined, empennage size is determined as a function of wing size. The sizing of the wing and empennage allows for flight performance calculations to be made for a design payload and range. With aircraft performance computed, aircraft fuel burn is estimated and this coupled with an estimation of aircraft equipment mass leads to a new estimate for

aircraft MTOW. If the new estimate for aircraft MTOW does not match the initial guess to within a given convergence margin; initial guess MTOW is updated with calculated MTOW and aircraft sizing, mass estimation and performance calculations are repeated until MTOW converges.

Once the MTOW converges for the first time, hydrogen cryotanks need to be sized for the maximum fuel mission at MTOW and subsequently the new fuselage size and mass can be determined as a function of stored fuel. This is done by setting range equal to the range for maximum fuel at MTOW (payload for this range is calculated as an output). With this done flight performance calculations are carried out to determine the maximum fuel required to be stored in cryogenic tanks. This fuel requirement, $m_{fuel,req}$, is determined by taking into account boil-off and summing main mission block fuel, $m_{fuel,main}$, and reserve mission fuel, $m_{fuel,reserves}$, calculated from flight performance calculations for maximum fuel mission at MTOW (see Eq. 5-61 where bf is boil-off fraction). For the determined fuel requirement and previously calculated insulated tank gravimetric efficiency, tank mass with insulation is estimated in Eq. 5-62. Tank volume is then estimated so that fuselage size with installed cryotanks can be determined. This is carried out by first estimating total volume occupied by the cryogenic fuel (see Eq. 5-63). Fuel volume is determined by dividing the tank fuel requirement by a specified fuel density. Total tank volume is then determined by taking into account the volume penalty imposed by tank ullage requirement in Eq. 5-64 where u is tank ullage. Since a 1:1 ratio in volume split between rear tank and upper tank was assumed, for a determined total tank volume requirement the volume in each of these tanks is half of the total tank volume requirement. With volume in each of the tanks known, tank dimensions for the rear and upper tanks are determined and subsequently a new fuselage size is determined (see Sec, 5.6.2.2). For the new fuselage size, fuselage mass is then re-estimated using Eq. 5-60. The new fuselage mass coupled with estimated tank mass leads to a new estimate for MTOW. If the new MTOW does not match guessed MTOW that converged the design mission to within a given convergence margin; aircraft sizing, mass estimation and performance calculations are repeated until the two MTOW's converge. Once the MTOW

converges for a second time, payload-range calculations are carried out for the maximum payload mission at MTOW to determine maximum payload and subsequently relevant performance outputs for the design mission are read-off. An overview of this procedure for sizing and performance assessment of an aircraft utilising cryogenic fuel is shown in Figure 5-11 and was integrated into CADI.

$$m_{fuel,req} = (1 + bf)(m_{fuel,main} + m_{fuel,reserves}) \quad 5-61$$

$$m_{tank} = \left(\frac{1 - \eta_{grav,ins}}{\eta_{grav,ins}} \right) m_{fuel,req} \quad 5-62$$

$$Vol_{fuel} = \frac{m_{fuel,req}}{\rho_{fuel}} \quad 5-63$$

$$Vol_{tank} = Vol_{fuel}(1 + u) \quad 5-64$$

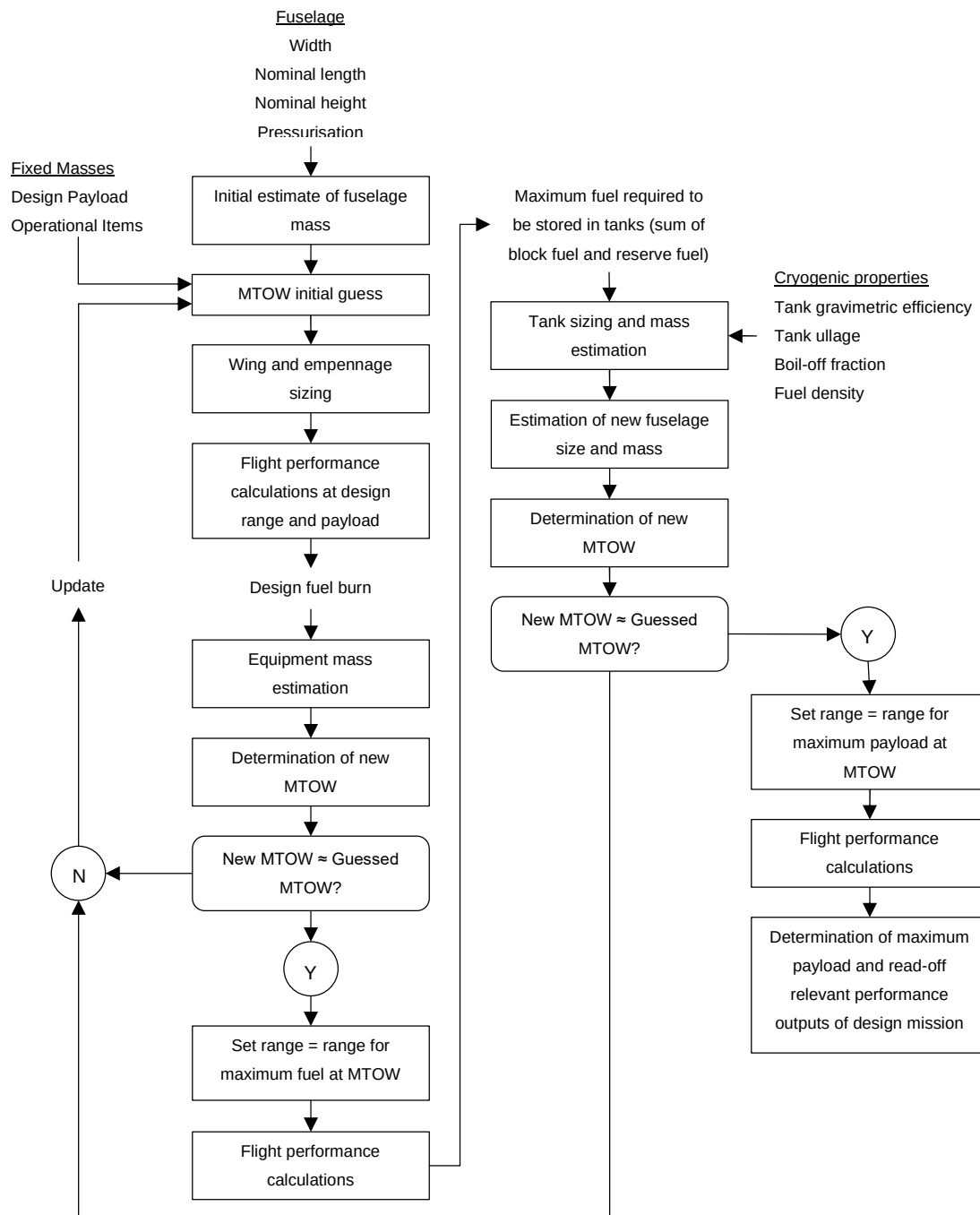


Figure 5-11 Overview of procedure for sizing and performance assessment of aircraft with cryogenic fuel

5.7 Hybrid Electric Flight Performance

For hybrid electric aircraft flight performance, power driven to the propulsor from the two distinct energy/power sources needs to be accounted for. For HeP with GTs, the reduction in GT power output for a given aircraft thrust requirement as

power discharged from a battery increases is an additional degree-of freedom that has to be taken into account in flight performance calculations.

In the methodology in [20] to tackle this extra degree-of-freedom for a given electrical power output from the battery, battery mass is increased iteratively so that the battery is depleted to a specified minimum SOC at the end of the mission. A limitation of this approach is it does not allow for a fuller understanding of the design space because what happens when the battery does not discharge to the specified minimum SOC at the end of the mission is not computed. In certain cases this might be a practical issue if for instance an aircraft diverts from its originally stated mission due to an unforeseen occurrence or bad weather conditions.

Another approach to the assessment of hybrid electric flight performance was the “battery cycle build-up” approach defined in [4]. In the approach by defining a battery cycle to be the sum of the recharge and discharge time, a battery pack continuously recharges and discharges until no more fuel is left in the aircraft. After each cycle completed aircraft endurance increases by the time it takes to complete one battery cycle until no fuel is left in the aircraft. The first limitation in applying this approach here is that this approach was utilised for a HALE UAV where vehicle endurance is the main performance parameter of interest, for a civil transport role where the main aim is to maximise profit from payload over a given aircraft range, use of this approach might lead to a counteractive effect of designing an unnecessarily “bulky” aircraft. The second limitation of the approach is that for operational reasons a continuous recharge-discharge hybrid scheme might not be desired.

For flight performance calculations here a flexible approach is employed by assessing flight performance at each mission segment for a specified battery mass and power rating and then for each mission segment assessing fuel burn, distance travelled and time elapsed. Battery use ends when battery SOC is depleted to a specified minimum value. For overall mission assessment; hybridisation is only considered here for take-off, climb and cruise phases of the main mission. Also it is assumed that the mission starts with the battery fully

charged, thus initial SOC at the start of the mission is equal to an upper SOC limit specified for a given mission. Additionally, for a given mission the propulsion system will operate in different modes, for assessment here relevant modes and their definitions are given in appendix E. It should be noted that this approach is not only applicable for a battery employed as an electric energy storage device, but to any electric energy storage device that is characterised with a fall in SOC as energy is depleted from it while its mass remains constant. The flight performance assessment procedures created in current studies are implemented into CADI for the computation of hybrid-electric flight.

5.7.1 DC Hybrid Turboelectric & Geared Hybrid Electric with Propellers Flight Performance

For the assessment of hybrid electric performance for DC hybrid turboelectric and geared hybrid electric with propellers architectures, characteristics of the battery and transmission need to be taken into account in addition to the GT-propulsor combination. For the battery its mass, power rating, upper SOC limit, lower SOC limit and discharge efficiency profile are needed for performance assessment. For the transmission, efficiencies of each component that comprise it are required for the determination of overall transmission efficiency between the propulsor and the GT. These parameters in addition to hybrid propulsion system performance polynomials (defined in Sec. 4.8.2.5) and general propulsion system performance parameters in the take-off net thrust (ducted fans as propulsors), take-off BPR (ducted fans as propulsors), take-off shaft power (propellers as propulsors), propeller diameter (propellers as propulsors), propeller efficiency (propellers as propulsors) and thrust rating during climb, are used to characterise the propulsion system inputs required for flight performance assessment. With propulsion system performance inputs defined; aircraft data required for flight performance assessment are aircraft range, TOW, airframe size and specified flight conditions where the propulsion system operates in hybrid mode.

With inputs defined, performance assessment for hybrid flight segments starts with setting the battery pack to discharge mode for the first hybridised mission

segment. With the battery pack in discharge mode airframe drag is determined and subsequently the thrust requirement to overcome the drag is determined. For climb and take-off mission segments, the thrust of the propulsion system is determined from defined propulsion system inputs. For cruise since straight line un-accelerated flight is assumed, the thrust requirement is equal to calculated drag. After the thrust requirement is determined, the thrust split between the propulsor and GT is determined from defined thrust split-thrust polynomial for the flight condition (see Eq. 4-68). The thrust split determined in this manner then fixes GT and propulsor thrust for zero battery power output.

To determine GT thrust and propulsor thrust for defined battery power output (fixed by the battery power rating), consideration has to be made to the fact that as battery power increases, for a given propulsor shaft power requirement the power output from the GT reduces. Taking this into account, as GT shaft power reduces, thrust produced from the GT reduces. Thus if propulsor shaft power remains constant, the increasing battery power which reduces GT thrust, will have an effect of reducing the overall thrust produced by the propulsion system. For a given thrust requirement this means that as battery power output increases, the propulsion system produces lesser and lesser thrust than what is required for a given flight segment and thus the propulsion system “under-thrusts”. To remedy this, for a given battery power output GT throttle is increased from what the throttle would be if propulsor shaft power remained constant as battery power increases. This is done to offset the under-thrust of the propulsion system. Performance calculations for this physical condition for a given battery power output starts with determining propulsive device efficiency as a function of zero-battery-power-output-propulsor-thrust from defined propulsive device efficiency-propulsor thrust polynomial for the flight condition (see Eq. 4-69). With propulsive device efficiency determined, propulsor shaft power is determined from Eq. 4-40. Then battery discharge efficiency is determined as a function of SOC from the input battery discharge efficiency profile (see Eq. 4-66). With propulsor shaft power and battery discharge efficiency determined, GT power output for a given battery power output is determined using Eq. 4-46 for DC hybrid turboelectric architecture and Eq. 4-48

for geared hybrid electric architecture. With GT power output determined, a new GT thrust is calculated as a function of GT power output from input GT thrust-GT power output polynomial for the flight condition (see Eq. 4-70). With this new GT thrust, total propulsion system thrust is determined as a sum of zero-battery-power-output-propulsor-thrust and calculated GT thrust. If this thrust does not match the thrust requirement for the mission segment, GT throttle is increased to offset under-thrusting of propulsion system in hybrid mode. As GT throttle is increased, GT shaft power output is increased and for the increased GT shaft power output, a new GT thrust is again calculated as a function of GT shaft power output from input GT thrust-GT power output polynomials. A new calculated propulsor shaft power is then determined using Eq. 4-46 for DC hybrid turboelectric architecture and Eq. 4-48 for geared hybrid electric architecture. For the newly calculated propulsor shaft power, a new propulsor thrust is determined as a function of propulsor shaft power from input propulsor thrust-propulsor shaft power polynomial for the flight condition (see Eq. 4-71) and subsequently new propulsive device efficiency is determined using Eq. 4-40. New propulsion system thrust is then determined as sum of calculated GT thrust and propulsor thrust, and GT throttle is iteratively changed until calculated propulsion system thrust matches segment thrust requirement.

After GT thrust and propulsor thrust are determined, segment fuel burn, distance travelled and time are estimated. Distance travelled and time elapsed for a given segment and mission phase are calculated as described in Sec. 5.5, but for segment fuel burn consideration has to be made to the altered power output of the GT as a result of employing a battery. In doing this estimation of fuel burn starts with estimating GT fuel flow rate. GT fuel flow rate is estimated as a function of GT shaft power output from input fuel flow rate-GT power output polynomials (see Eq. 4-72). From the estimated GT fuel flow rate, TSFC is then determined as calculated fuel flow rate divided by segment thrust requirement. With TSFC determined segment fuel burn for a specific mission phase is then calculated as described in Sec. 5.5.

After fuel burn, distance travelled and time elapsed for a mission segment are determined, the change in battery SOC has to be determined as a result of the battery discharging power for an amount of time. The change in energy level for a mission segment is determined using Eq. 4-60, where dt in this case is equal to the time travelled for a mission segment. With the change in energy level for a mission segment determined, the amount of stored energy in the battery at the end of the mission segment is determined with Eq. 4-61 and subsequently SOC at the end of the mission segment is determined with Eq. 4-50.

At the end of the first hybridised mission segment calculations move onto the next mission segment. For the next mission segment as previously carried out, propulsor thrust and core thrust are determined for the zero battery power output case and subsequently zero-battery-power-output-propulsor-shaft-power is determined. However for the next mission segment, different from the first mission segment is that after zero-battery-power-output-propulsor-shaft-power is determined a check needs to be made if the SOC is equal to or has just about fallen below input minimum battery SOC. This is because if this condition has been met the battery has to stop discharging to protect the battery from irremediable damage [20]. If this condition is met the battery is subsequently set to operate in recharge mode. In recharge mode electrical storage power output is set to 0 W and the battery is recharged with excess power produced by the GT. If the battery is not configured to be recharged the excess power from GT is fixed at 0 W. If the SOC has not fallen below the minimum SOC after previous segment calculations the battery remains in discharge mode and as calculated for the first mission segment, propulsion system calculations are carried out taking into account under-thrusting of the propulsion system to determine GT shaft power output and propulsor shaft power for the given thrust requirement. Whether the battery is in discharge mode or in recharge mode; segment fuel burn, distance travelled and flight time are determined in the same way using the approach described for the first hybrid mission segment. After this is done for end-of-segment SOC is again determined for the second segment. If the battery is in discharge mode SOC decreases further from previous segment value. If the battery is in recharge mode SOC increases if the battery has been

configured to recharge, or stays constant if the battery has not been configured to recharge.

As calculations move on to the next segment, consideration needs to be made to the fact that the battery can be either in discharge mode or recharge mode. If the battery is in discharge mode, the third segment calculations begin and follow the procedure described for the second segment. If the battery is in recharge mode a check is made to see if current battery SOC level is greater than the upper SOC limit specified for the mission before third segment calculations can begin. If SOC is not greater than the upper SOC limit specified for the mission then the battery remains in recharge mode for the next mission segment. If SOC is just about equal to the specified upper SOC limit then the battery switches back to discharge mode before the third segment begins. With battery modes fixed before the start of the third segment, segment calculations are carried out in the same way as second segment calculations up until propulsor shaft power for zero battery power output is determined. After propulsor shaft power is determined a check for whether the battery is in discharge or recharge mode is once again made. If the battery is in recharge mode, electrical storage power output remains at 0 W before recharge mode GT power output is determined. If the battery is in discharge mode, a check like the one made for the second segment calculations is made to see if SOC is less than the minimum SOC specified for the mission. If it is the battery switches to recharge mode the electrical storage power output is set at 0 W and recharge mode GT power output is determined. Otherwise if the battery SOC is not less than the minimum specified SOC for the mission the battery remains in discharge mode and propulsion system calculations are carried out taking into account under-thrusting of the propulsion system to determine GT shaft power output and propulsor shaft power for the given thrust requirement.

Once GT shaft power output is determined as a function of the battery operating mode, like for previous segments; segment fuel burn, distance travelled and flight time are determined. Additionally end-of-segment SOC is determined. For subsequent hybrid mission segments until all hybrid mission segments are

assessed, checks for what mode the battery is in and whether SOC exceeds specified upper SOC limit are made in the same way as end-of-second-segment hybrid checks before segment calculations are made. After pre-segment hybrid checks are made, segment calculations proceed in the same way as third segment calculations. This procedure of pre-segment hybrid checks and segment calculations following the same method as third segment calculations is repeated until all hybridised mission segments are assessed. Block fuel during the hybrid mission is estimated as sum of fuel burned for each hybrid mission segment. An overview of the procedure for estimating hybrid electric flight performance for stated architectures implemented into CADI is shown in Figure 5-12.

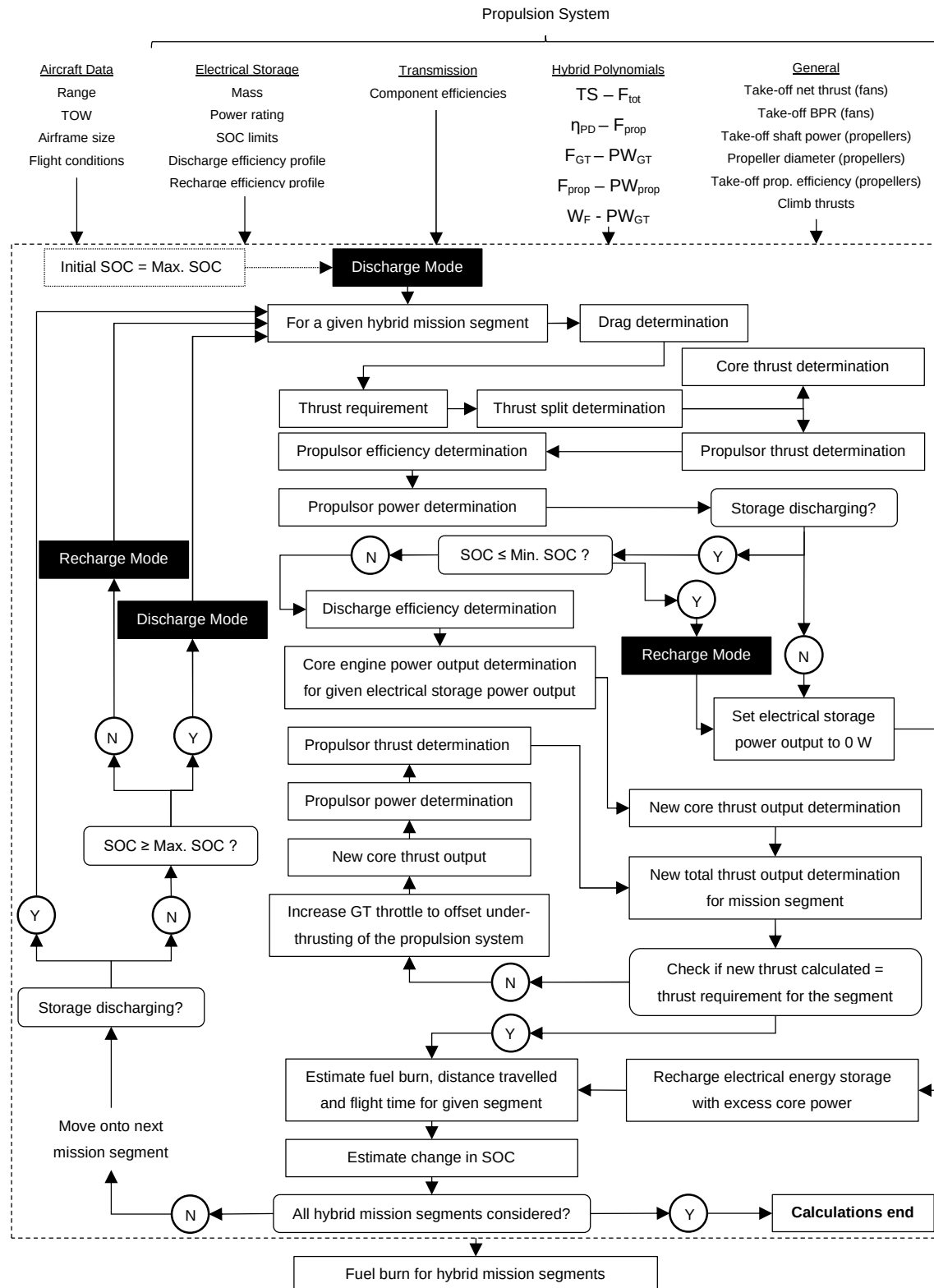


Figure 5-12 Overview of hybrid electric flight performance assessment procedure for hybrid mission segments of DC hybrid turboelectric architecture and geared hybrid electric architecture with propellers

If for operational reasons use of the hybrid capabilities of the propulsion system is delayed a mission phase or two i.e. using hybrid electric propulsion for the cruise phase as opposed to during take-off and climb, the procedure in Figure 5-12 is still employed. However the caveat is that although battery is fixed to be in discharge mode, for these mission segments battery power output is set at 0 W until cruise is reached. After cruise is reached performance calculations are then carried out as shown in Figure 5-12.

5.7.2 Geared Hybrid Electric with Ducted Fans Flight Performance

Since the geared hybrid electric architecture with ducted fans is treated as a geared turbofan with power intake from a battery, assessment of flight performance has to take into account the differences in turbofan performance when the propulsion system operates either in turbo mode or hybrid mode. With this consideration, two sets of propulsion system performance polynomials are needed. The first is SFC-thrust polynomials when the propulsion system is operating in turbo mode (see Eq. 4-65) and the second is SFC-thrust polynomials as a function of the SOC when the propulsion system is operating in hybrid mode (see Eq. 4-67). In addition to these inputs aircraft, battery and general propulsion system performance inputs are required for the computation of hybrid electric flight performance. For the aircraft data the required inputs are aircraft range, TOW, airframe size and specified flight conditions. For the battery, inputs required are battery mass, power rating, upper SOC limit and lower SOC limit. For the general propulsion system, inputs required are take-off net thrust, take-off BPR and thrusts during climb.

For the first hybrid mission segment the propulsion system is in hybrid mode and performance calculations start with determining airframe drag and subsequently aircraft thrust requirement. After the thrust requirement has been determined propulsion system SFC has to be determined for the computation of segment fuel burn. In hybrid mode propulsion system SFC is determined as a function of SOC and the thrust requirement from defined SFC-thrust-SOC polynomials for the flight condition (see Eq. 4-67). With SFC determined fuel burn is estimated in the same way as how fuel burn is computed for the non-

hybrid mission (see Sec. 5.5). Subsequently the distance travelled and time elapsed during the mission segment are also calculated in the same way as how these parameters are computed for the non-hybrid mission. After segment fuel burn, distance travelled and time elapsed are determined; battery power discharge for a unit amount of time means the change in SOC has to be determined. In a similar manner as for the DC hybrid turboelectric and geared hybrid electric with propellers architectures, the change in the energy level of the battery is first determined using Eq. 4-60. With the change in energy level determined, the amount of stored energy in the battery at the end of the mission segment is determined with Eq. 4-61 and subsequently SOC at the end of the mission segment is determined with Eq. 4-50.

At the end of the first hybridised mission segment calculations move onto the next mission segment. For the next mission segment as previously carried out, calculations proceed in the same way as the first segment up until thrust requirement is determined. However for this mission segment, different from the first mission segment is that after the thrust requirement has been established a check needs to be made if the SOC is equal to or has just about fallen below input minimum battery SOC. If this condition is met the propulsion system is set to operate in turbo mode. In turbo mode since the propulsion system is operating effectively as a 'pure' geared turbofan, SFC is determined as a function of the thrust requirement from input thrust-SFC polynomial for the flight condition, with the thrust-SFC polynomial determined from propulsion system simulation with zero battery power intake. Otherwise if the minimum SOC condition has not been met the propulsion system continues in hybrid mode and SFC is calculated in the same way as done for the first segment. After SFC is determined as a function of propulsion system mode; segment fuel burn, distance travelled and time elapsed are computed. For end-of-segment SOC calculations, if the propulsion system is in turbo mode the SOC remains constant since the battery has been turned off; otherwise SOC drops as the propulsion system remains in hybrid mode. Calculations for subsequent segments follow second segment calculation procedure until all hybridised mission segments have been assessed. Block fuel during the hybrid mission is

then estimated as the sum of the fuel burned for each hybrid mission segment. An overview of the procedure for estimating hybrid electric flight performance for the stated architecture implemented into CADI is shown in Figure 5-13.

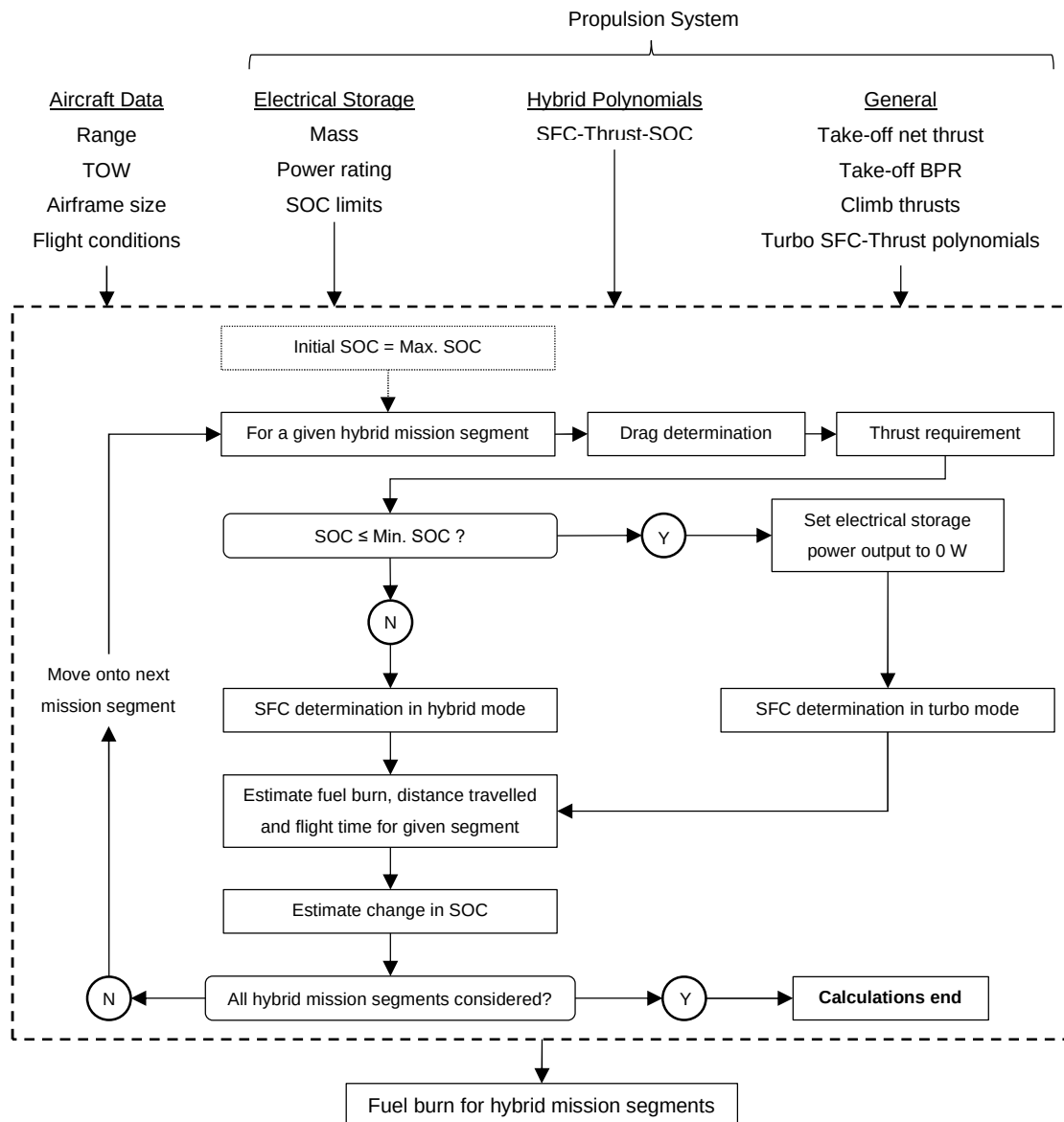


Figure 5-13 Overview of hybrid electric flight performance assessment procedure for hybrid mission segments of geared hybrid electric architecture with ducted fans

If for operational reasons use of the hybrid capabilities of the propulsion system is delayed a mission phase or two i.e. using hybrid electric propulsion for the cruise phase as opposed to during take-off and climb, the procedure in Figure

5-13 is still employed. However the caveat is that although battery is fixed to be in discharge mode, for these mission segments battery power output is set at 0 W until cruise is reached. After cruise is reached performance calculations are then carried out as shown in Figure 5-13.

5.7.3 Energy Specific Air Range

The typical figure of merit for characterizing the vehicular efficiency for fuel based aircraft is the specific air range (SAR). It describes the distance travelled for a given quantity of fuel. From [20] an extension of this figure of merit for unconventional energy sources is the energy specific air range (ESAR). ESAR determines distance an aircraft travels per unit quantity of energy (see Eq. 5-65).

$$ESAR = \frac{U \cdot \frac{F_L}{F_D}}{TSPC \cdot F_W} \quad 5-65$$

In Eq. 5-65 TSPC is the thrust specific power consumption. According to the TSPC definition given in [21], for the hybrid aircraft configurations considered in present study TSPC is given in Eq. 5-66 as the ratio of supplied power by battery and fuel to thrust.

$$TSPC = \frac{PW_{fuel} + PW_{batt}}{F} \quad 5-66$$

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6 REGIONAL TRANSPORT

6.1 Mission Requirements

For the current research effort, the baseline for the regional aircraft has been specified to be an aircraft similar to the Bombardier Q400 Enhanced High Gross Weight (EHGW) equipped with engines similar to the PW150a. With this baseline specified, mission requirements for TeDP and hybrid electric configurations are taken from the mission specifications obtained from public literature for the Bombardier Q400 EHGW. With the regional aircraft assumed to operate in ISA conditions, mission requirements for the regional aircraft are shown in Table 6-1. It should be noted that although a present day state-of-the-art configuration has been chosen as mission baseline, an advanced tube-and-wing equipped with advanced GTs that has higher component efficiencies, higher PRs and higher TET could be utilised. However with improvements in GT technology expected to only bring modest improvements in performance for ever increasing capital investment, a baseline based on present day state-of-the-art technology is sufficient to show how turboelectric and hybrid electric operation can provide performance benefits over present day state-of-the-art technology.

Table 6-1 General mission requirements for regional aircraft obtained from public literature

Design Range	1,114 n.mi [1] & [2]
Maximum payload range at MTOW	803.5 n.mi [2]
Maximum fuel range at MTOW	1514 n.mi [2]
Design mission number of passengers	74 [1] & [2]
Maximum number of passengers	86 [1], [2] & [3]
Cruise Altitude	25,000 ft [3]
Absolute Ceiling (all engines operative)	27,000 ft [1] & [3]

Maximum speed	M0.6 [1] & [2]
Landing field length	1,289 m [1]
Diversion range	100 n.mi [2]
Hold time	45 min [2]
Hold altitude	15,000 ft [2]

Also, what needs to be considered is that a regional aircraft can operate multiple types of cruise at different speeds. For current research the long range cruise and high speed cruise missions of the Bombardier Q400 EHGW are chosen in order to assess performance at different cruise speeds. With the requirements for the different mission types defined, the climb and descent profiles for each are fixed. Table 6-2 shows the mission requirements that arise as a result of taking into account different speed operation cruise types.

Table 6-2 Mission Requirements as a function of cruise type

Design mission cruise type	Long range cruise [1] & [2]
<i>Long Range Cruise (LRC)</i>	
Cruise Speed	M0.48 [1], [2] & [3]
Climb CAS	195.7 kts
Descent CAS	195.7 kts
<i>High Speed Cruise (HSC)</i>	
Cruise Speed	M0.58 [1], [2] & [3]
Climb CAS	239.5 kts
Descent CAS	239.5 kts

In absence of data available from public literature, diversion cruise speed is set to main mission long range cruise speed, diversion cruise altitude is set to main mission cruise altitude and hold speed is set to the speed the aircraft is descending at during the reserve mission at 15,000 ft.

For assessment of take-off performance, end-of-runway Mach number requirement needs to be known. With a value for end-of-runway Mach number not obtained from public literature read, it is estimated by first determining wing stall speed in clean conditions from the lift equation and then multiplying this stall speed by 1.2 to determine end-of-runway speed [4]. End-of-runway Mach number is then determined by dividing end-of-runway speed by sea-level speed of sound. For the determination of end-of-runway Mach this way, aircraft wing loading is fixed by the wing loading at take-off of the Q400 EHGW and clean wing maximum lift coefficient is chosen to be 1.5 as this value is typical for a basic aerofoil [5]. With clean stall speed estimated to be 70.7 m/s, end-of-runway Mach number at sea-level using approach described is estimated to be M0.25.

For descent performance assessment, engines are set to operate at idle conditions. With detailed assessment of GT idle performance a time-consuming task that would require the development of a dedicated numerical tool, idle thrust and SFC simulated for the baseline aircraft are utilised as idle thrust and SFC for all integrated airframe-propulsion configurations considered.

6.2 Baseline Propulsion System

The Bombardier Q400 EHGW aircraft is equipped with two PW150a turboprop engines and two Dowty R408 propellers. The Bombardier Q400 EHGW is also equipped with one Hamilton Sundstrand Power System APS 1000 T-62T-46C12 APU [6]. For performance assessment here, GT and propellers similar to the PW150a and Dowty R408 are simulated. For the simulation of the propulsion system, general architecture is represented by the architecture shown in Figure 4-1 (c). In considering the APU only its mass is considered as thermodynamic performance and size consideration of the APU is beyond the

scope of current work. Weight of the APU is fixed by the weight of the Hamilton Sundstrand Power System APS 1000 T-62T-47C12 which weighs 86 kg [7].

6.2.1 Gas Turbine and Gearbox

Taking information from [8] the GT is configured to have the following characteristics listed below

- 2 spool GT with free power turbine
- 3 stage axial LPC
- 1 stage centrifugal HPC
- Single stage LPT and HPT
- 2 stage FPT with shrouded blades
- Reverse flow annular combustor
- Bleed from LPC and HPC to cool HPT
- Offset reduction gearbox

These characteristics are used in the modelling of the GT in Turbomatch 2.0 (see Figure 6-1). For the simulation of the GT suitable inputs need to be specified so that performance can be assessed. Obtained public literature characteristics of the regional GT used for simulation are shown in Table 6-3. From Table 6-3, the characteristics of the GT set are not enough for the simulation of the GT. Thus the additional data in Table 6-4 supplied by Rolls-Royce was used for the simulation of the GT.

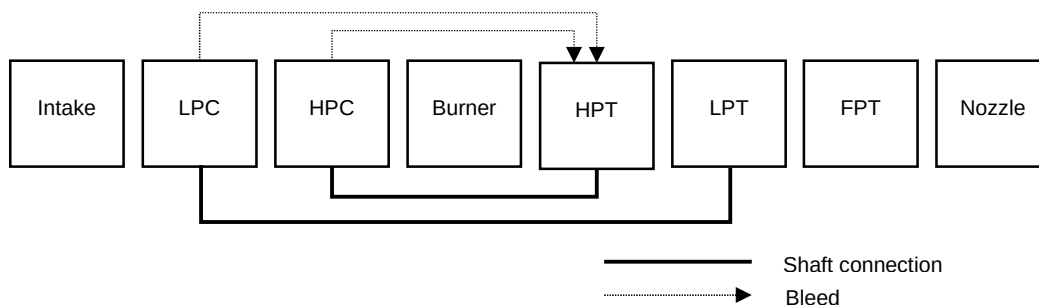


Figure 6-1 Regional GT configuration

Table 6-3 Characteristics of regional GT obtained from public literature

Dimensions and Weight	
Length	2.42 m [8]
Height	1.1 m [8]
Width	0.79 m [8]
Mass	717 kg [8]
Performance	
Design Point	Normal take-off at dry sea level static conditions [8]
Bleed from LPC	6% of inlet airflow [8]
Bleed from HPC	10% of inlet airflow [8]
Design point shaft power output	3415 kW [8]
Design Point output shaft rotational speed	1020 RPM [8]
Output shaft rotational speed for maximum take-off	1020 RPM [8]
Output shaft rotational speed for climb (LRC & HSC)	900 RPM [9]
Output shaft rotational speed for cruise (LRC & HSC)	850 RPM [10]
Fuel	Kerosene [8]

Table 6-4 Supplied data by Rolls-Royce to be used for simulation of GT performance for regional aircraft

Design point TET (K)	1487
HPT design point isentropic efficiency	86.7%
LPT design point isentropic efficiency	86.7%
FPT design point isentropic efficiency	91.2%

In addition to information from public literature for the regional GT and supplied data by Rolls-Royce for the regional aircraft; burner pressure loss, combustion efficiency, gearbox transmission efficiency and lower heating values are specified in Table 6-5 based on typical values.

Table 6-5 Regional GT performance parameters set based on typical values

Burner pressure loss	5%
Combustion efficiency	99%
Gearbox transmission efficiency	98.25% [11]
Fuel lower heating value (MJ/kg)	43.12 [11]

In simulating design point and off-design performance of the GT; the mass flow rate, compressor PRs and compressor isentropic efficiencies still need to be defined in addition to GT design values already specified. Thus mass flow rate, compressor PRs and compressor isentropic efficiencies are iterated within suitable ranges to match GT performance with obtained data from public literature on the performance of the PW150a engine. For the off-design point the throttle is unknown (thus TET is unknown) and this parameter is iterated to determine performance at the selected off-design point. Using Turbomatch, a summary of the comparison of the simulated engine with the actual engine at

the design point and given off-design point for which performance data is available is shown in Table 6-6. In Table 6-6 a decent match between simulated values and comparable values from quoted literature is obtained with average variation being less than 5%. The final values of the GT performance parameters that were iterated to determine engine performance in Table 6-6 are shown in Table 6-7. It should be noted that the maximum take-off point in Table 6-6 also corresponds to the condition the engine is set to operate in during OEI operation. The residual thrust output in Table 6-6 is the thrust produced by the nozzle that arises as a result of the GT's power turbine not extracting all of the potential available energy from the jet exiting the GT. Also, equivalent brake specific fuel consumption is the brake specific fuel consumption of the GT if all the available energy from the jet that exits the GT, with brake specific fuel consumption defined as the ratio fuel flow rate to shaft power output.

Table 6-6 Comparison of simulated engine with values obtained from public literature of engine with similar architecture

	Turbomatch Simulation	Comparable quoted values from obtained literature of engine with similar architecture	Variation between simulated and quoted values (%)
Design point – Normal take-off			
Residual Thrust (N)	3,408	3,412 [8]	-0.12%
Off-design point – Maximum Take-Off			
Shaft Power Output (kW)	3,767	3,781 [8]	-0.37%
Equivalent Shaft	4,096	4,095 [10]	0.02%

Power Output (kW)			
Residual Thrust (N)	3,871	3,750 [8]	3.23%
Brake Specific Fuel Consumption ($\mu\text{g}/\text{J}$)	73.6	73.2 [10]	0.55
Equivalent brake specific fuel consumption ($\mu\text{g}/\text{J}$)	67.7	70.8 [12]	-4.38

Table 6-7 Values of iterated GT performance parameters used to simulate regional GT performance

Design point LPC PR	2.6
Design point LPC isentropic efficiency	0.84
Design point HPC PR	6.64
Design point HPC isentropic efficiency	0.81
Maximum TET	1539 K

With the compressors of the GT sized for ceiling flight condition at design point TET (chosen since this flight condition imposes the highest PR requirement of the propulsion system), combustor sized for maximum take-off rating and turbines sized for maximum take-off rating (chosen since this flight condition imposes the highest power requirement for combustors and turbines); output speed from the power turbine at maximum take-off rating at end-of-runway is estimated to be 21,461 RPM. With output shaft RPM for the GT specified at

1020 RPM, gearbox gear ratio is determined to be 0.0475 from Eq. 4-45. With gearbox transmission efficiency set to be 98.25%, power output from the gearbox is determined at the maximum take-off rating. Then using the empirical relation in [13] to estimate gearbox mass, gearbox mass is estimated to be approximately 350 kg. For total gearbox mass (taking into account that there are two gas turbines equipped to the aircraft), total gearbox mass is estimated to be 700 kg.

6.2.2 Propellers

Propellers to be equipped are propellers similar to the Dowty R408. These propellers are variable pitch, constant speeding, feathering, reversing, use hydraulic control and have counterweights [14]. Characteristics of the propeller to be used for modelling obtained from public literature are shown in Table 6-8. In addition to the characteristics given in Table 6-8 propeller design shaft power is set to 3,415 kW which corresponds to the shaft power output of the GT at its design point. For modelling of the aerodynamic characteristics of the blade cross section in the radial direction, ARAD-A 13% aerofoil is used as the aerofoil to represent blade cross section at points along the propeller radius. This aerofoil is used because it represents propeller mid-section aerodynamic characteristics because it is not as thick as the ARAD-A 20% aerofoil cross-section (more useful for modelling blade aerodynamic properties near blade hub) and it is not as thin as the ARAD-A 6% aerofoil (more useful for modelling blade aerodynamic properties near the blade tip). In addition to data listed in Table 6-8, for the simulation of the drag polar of propeller aerofoil cross-section, the reference Reynold's number is computed at the take-off flight condition for a reference chord of 0.12m. With propeller characteristics defined, propeller blade geometry for the minimum induced loss condition is shown in Figure 6-2.

Table 6-8 Propeller characteristics for regional aircraft from obtained public literature

Diameter	4.115 m [14]
Take-off rotational speed	1020 RPM [14]

Maximum power	3,781 kW [14]
Number of blades	6 [14]
Hub-to-tip ratio	0.17 ^a [15]
Activity Factor	103.5 per blade [16]
Mass	252 kg [14]
Blade cross sectional aerofoil	Modified ARAD-A [16] & [17]
Minimum RPM ^b	660 [14]

^aEstimated from schematics in [15]

^bUsed to fix propeller idle RPM

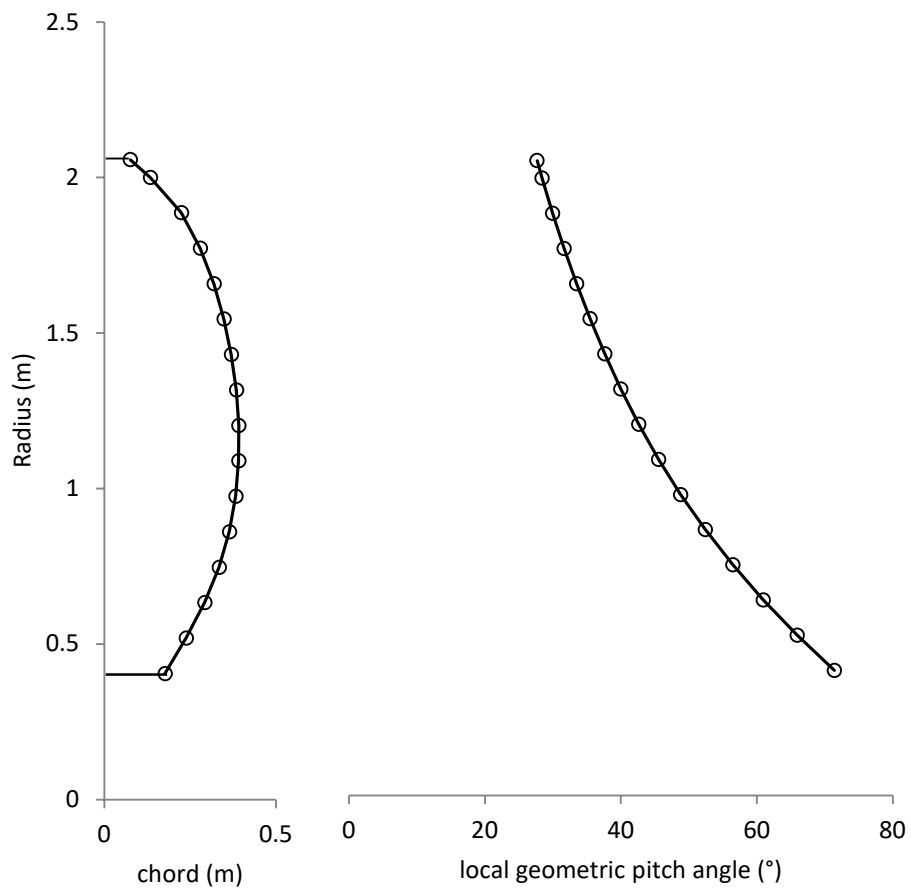


Figure 6-2 Minimum induced loss blade geometry

After propeller blade geometry was simulated using QMIL, propeller performance maps within a suitable advance ratio and pitch angle range were produced for each point on the flight envelope. Two steps were utilised for the creation of propeller performance maps for a given flight condition. (1) The first step involves the setting of a pitch angle and then varying the RPM to create the propeller characteristics. (2) Multiple propeller characteristics are created to form the propeller performance map by changing the propeller pitch angle and repeating step (1). For performance assessment for a given flight condition at a given RPM, a propeller running line is created on the propeller performance map by fixing the RPM and changing propeller pitch angle. For integration of the propeller with the GT, performance outputs for the propeller on the running line (such as thrust, shaft power, efficiency and slipstream velocity increment) are matched with the GT for a calculated shaft power output and output shaft rotational speed. A representative propeller performance map at sea-level and take-off RPM of 1020 is shown in Figure 6-3. In the propeller performance map for a given pitch angle, increasing advance ratio for constant RPM increases lift force and drag force generated by the propeller. This causes an increase in propeller shaft power. The increase in propeller shaft power corresponds to an increase in propeller power coefficient. However as flight speed increases further, viscous losses and induced losses curb the forces generated by the propeller. This eventually causes a fall in power coefficient. Additionally for a given advance ratio, increase in pitch angle increases local angles of attack at propeller blade cross-sections along the propeller radius and this increases power coefficient.

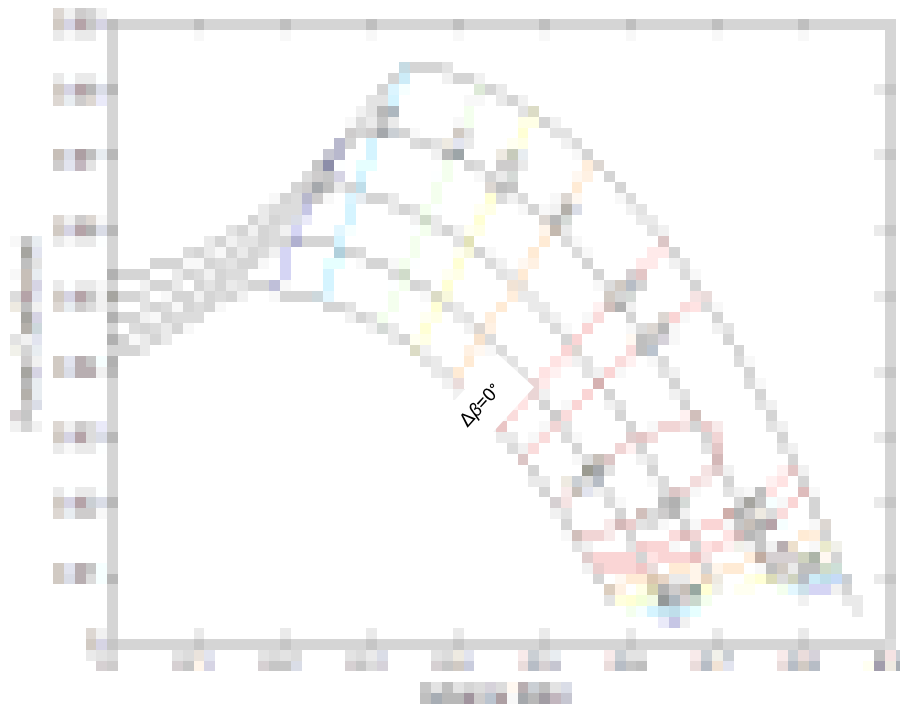


Figure 6-3 Power coefficient vs advance ratio with contours of efficiency for 2.5° pitch angle intervals at 1020 RPM and sea-level ($\Delta\beta$ is the radially constant pitch angle change)

6.2.3 Integrated Propulsion System Performance

For assessment of flight performance, the gearbox, propellers and GT need to be integrated. Matching the GT with propellers at different flight conditions and accounting for transmission losses, propulsion system thrust specific fuel consumption is obtained by dividing GT fuel flow rate by the sum of the thrust produced by the GT and propellers. For simulation carried out, an overview of integrated propulsion system performance at selected flight conditions for the propulsion system in LRC mode is shown in Figure 6-4. Thrust reduction for a given flight condition in is driven by a fall in TET. For the propeller as TET falls, propeller pitch angle falls so that the shaft power output of the GT matches the shaft power of the propeller (transmission losses are taken into account). For the baseline propulsion system in LRC mode, at flight conditions away from take-off (Sea-level, M0.25, 1020 RPM), TSFC increases primarily due to increases in GT momentum drag and a worsening of overall GT efficiency as operations moves further away from GT design point. At cruise flight condition

and RPM (7620 m, M0.48, 850 RPM) and other flight conditions near it, as TET reduces, initial TSFC reduction is driven by an increase in propeller efficiency which counteracts lower GT thermal efficiency. For the high speed cruise mission performance trends are similar except for the fact that TSFC is worsened due to increased GT momentum drag and reduced propeller efficiency (see Figure 6-5). In Figure 6-4 and Figure 6-5, thrust per engine represents thrust produced by both the GT and propeller.

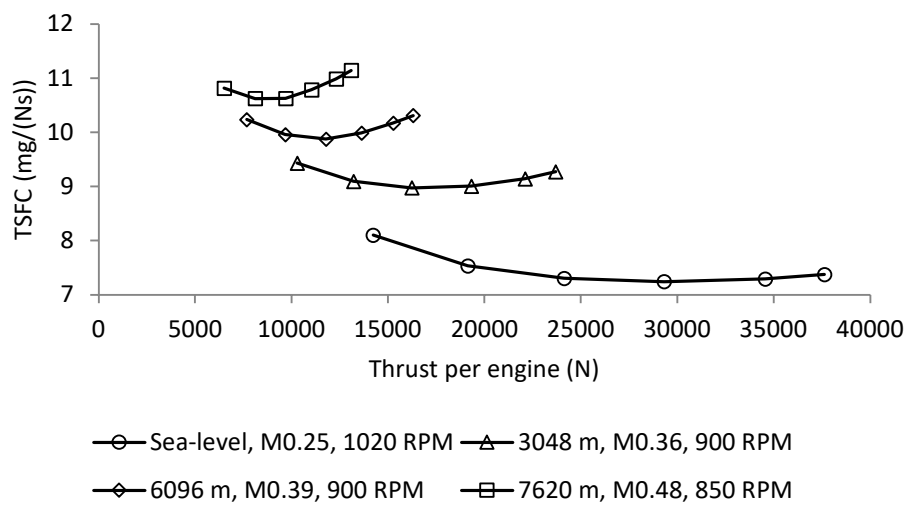


Figure 6-4 Long range cruise baseline propulsion system performance for regional aircraft at selected flight conditions

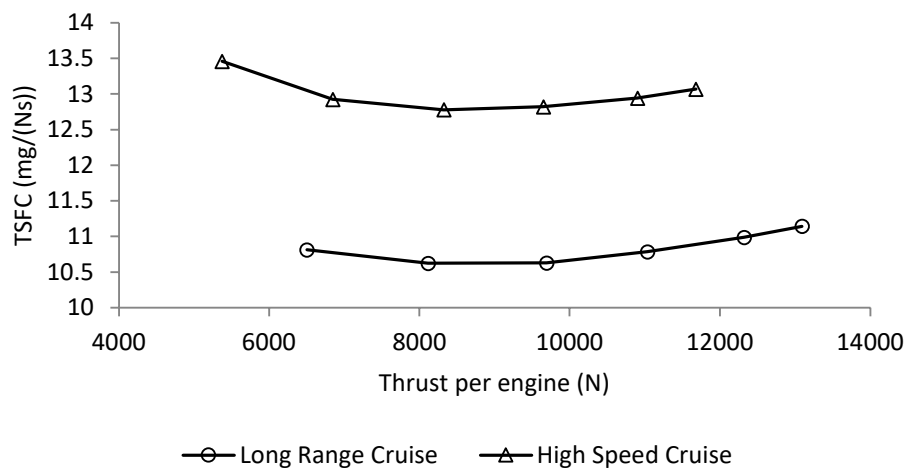


Figure 6-5 Comparison of propulsion system performance at cruise between propulsion system operating in long range cruise mode and high speed cruise mode

6.3 Baseline Aircraft

In addition to defined mission requirements and simulated propulsion system, design parameters and characteristics are required to be set for sizing and performance assessment of the baseline aircraft. Aircraft design parameters and characteristics for such a purpose are extracted from obtained data in public literature for Bombardier Q400 EHGW and are listed in Table 6-9.

Table 6-9 Regional transport aircraft design parameters and characteristics

Cabin pressure differential	0.38 bar [3] & [18]
Dihedral ^a	0°
Empennage material	Composite [3]
Empennage type	T-tail [3]
Flaps type	Two-section slotted Fowler [3]
Fuselage height	2.70 [1]
Fuselage length	32.8 [6]
Fuselage material	Aluminium alloy [19]
Fuselage width	2.70 [1]
Horizontal tail aspect ratio	5.14 [3]
Horizontal tail average thickness to chord ratio	0.1
Horizontal tail quarter chord sweep angle	8° [15]
Horizontal tail taper ratio	0.8 [15]
Horizontal tail volume coefficient	0.855 [4]
Increment in take-off lift coefficient due	0.7 [5]

to high lift devices	
Landing gear location	Wing mounted
Mass per passenger	102 kg [1] & [2]
Number of crew members	5 [1]
Relative spanwise distance from aircraft centre to engine centre	0.31 [15]
Tail moment arm	60% of fuselage length [4]
Ultimate load factor	5 ^b [4]
Vertical tail aspect ratio	1.07 [3]
Vertical tail average thickness-to-chord ratio	0.1
Vertical tail quarter chord sweep angle	34° [15]
Vertical tail taper ratio	0.9 [15]
Vertical tail volume coefficient	0.076 [4]
Wing aspect ratio	12.78 [6]
Wing average thickness-to-chord ratio	0.15 [3]
Wing material	Aluminium alloy
Wing quarter chord sweep angle	0°
Wing taper ratio	0.5

^aConsideration of dihedral is beyond the scope of current studies

^bLimit load factor multiplied by a safety factor of 1.5

From aircraft mission requirements set, aircraft design parameters set, aircraft design characteristics set and the simulated propulsion system, the baseline aircraft is simulated. A summary of top-level simulation results from the

simulation compared to the similar aircraft is shown in Table 6-10. From the simulation with the exception of take-off field length, top-level outputs of the simulated aircraft deviate from the Q400 EHGW by less than 1%. Since take-off does not constitute a significant portion of the overall mission, the higher deviation in TOFL from value for the Q400 EHGW does not overly impact overall aircraft performance. Thus the parameters such as MTOM, time-to-height, wing area and wing span only deviate by a small percentage from values for the Q400 EHGW aircraft. To better model take-off performance, the engines performance should be modelled at suitable points from static take-off to end-of-runway. Then take-off should be computed taking into account aircraft acceleration from static to lift-off and then from lift-off to obstacle clearance at end-of-runway. Also, considering payload-range (see Figure 6-6), points on the payload-range diagram deviate from values for the Q400 EHGW by 5% or less. Utilising geometry outputs from CADI, the baseline aircraft is visualised using OpenVSP in Figure 6-7.

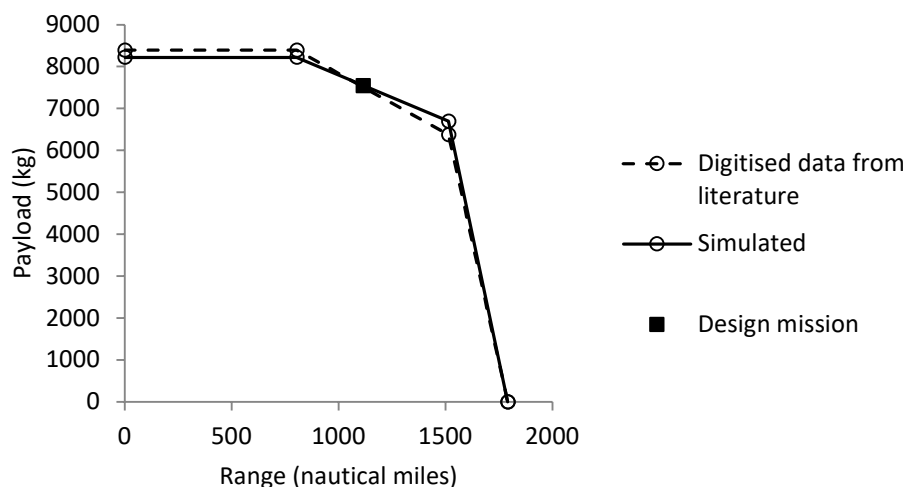


Figure 6-6 Payload-range diagram for baseline regional aircraft; digitised data obtained from [2]

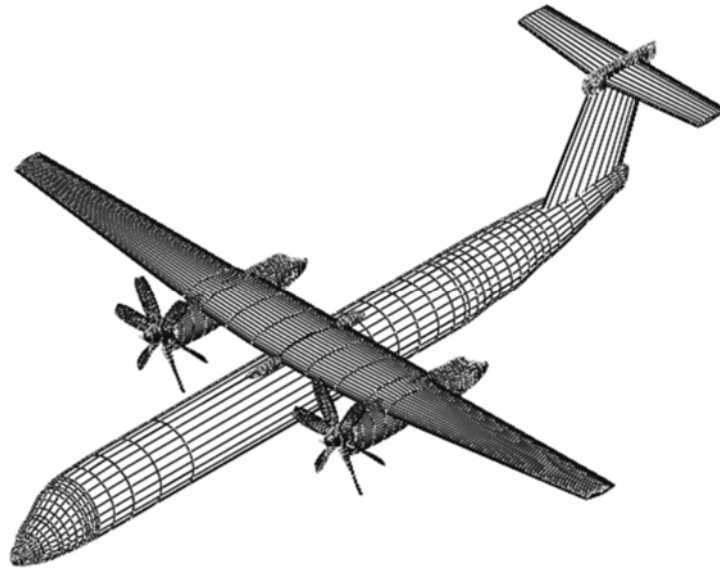


Figure 6-7 Geometry visualisation of baseline regional aircraft using OpenVSP

Table 6-10 Baseline aircraft simulation results compared with values obtained for similar aircraft from public literature at the design mission for regional transport

Parameter	CADI simulation for climb at 1457 K	Literature	Deviation
MTOM (kg)	29,401	29,574 [6]	-0.58 %
Time-to-height (min)	16.98 ^a	17.00 [3]	-0.12%
TOFL (m)	1,364	1,425 [3]	-4.28%
Wing area (m ²)	62.8	63.1 [6]	-0.48%
Wing span (m)	28.3	28.4 [6]	-0.35%

^aIterated climb TET to achieve this value

Additionally aircraft weights estimated for the simulated aircraft using CADI i.e MTOM, OEM, MZFM and MLM; deviate from values for Q400 EHGW by an average that is less than 5% (see Table 6-11).

Table 6-11 Baseline aircraft weights comparison with values obtained for similar aircraft from public literature

	CADI simulation	Literature	Deviation (%)
MTOM (kg)	29,401	29,574 [3]	-0.58
OEM (kg)	18,252	17,819 [2]	2.43
MZFM (kg)	26,476	26,308 [3]	0.64
MLM (kg)	27,446	28,123 [3]	-2.41

In considering the effect of aircraft range and with GT operating at the same engine rating as during the design mission, top-level performance parameters of interest reduce with range primarily due to lower fuel burn (see Table 6-12). Additionally considering flight at a faster speed, performance follows a similar trend as aircraft range reduces (see Table 6-13), however the aircraft does not fly at the design range in high speed cruise mode due to simulated TOM exceeding design MTOM. At 600 n.mi, fuel burn for the high speed cruise mission is greater than fuel burn at the same range in long range cruise mode primarily due to higher TSFC. At 200 n.mi however, increased flight speed for high speed mission reduces journey time to the extent that increased TSFC is offset by reduced duration of time the propulsion system is burning fuel. This results in a lower block fuel at 200 n.mi when the aircraft is operating in high speed cruise mode. Performance at reduced aircraft range and different speeds is of interest because regional aircraft typically fly missions with shorter ranges than specified design range and fly at faster cruise speeds than LRC cruise speed for economic reasons.

Table 6-12 Aircraft top-level performance summary in long-range cruise mode at different aircraft ranges for regional aircraft

Aircraft Range	Design	600 nmi	200 n.mi
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Block fuel (kg)	2654	1494	627
Reserve fuel (kg)	947	888	844
Time-to-height (min)	16.98	15.47	14.46
TOFL (m)	1364	1252	1172
TOM (kg)	29401	28181	27271

Table 6-13 High Speed Cruise Mission Top-Level Performance Summary for Regional Aircraft

Aircraft Range	600 n.mi	200 n.mi
Block fuel (kg)	1515	623
Reserve fuel (kg)	889	843
Time-to-height (min)	18.64	17.36
TOFL (m)	1253	1172
TOW (kg)	28204	27266

A breakdown of non-fuel aircraft masses for the baseline aircraft is shown in Table 6-14. These masses are the same for both the long range cruise mission and high speed cruise mission.

Table 6-14 Non-fuel aircraft masses for baseline regional aircraft

Empennage	206 kg
Equipment	6468 kg
Fuselage	2379 kg

Operational Items	1112 kg
Design Payload	7548 kg
Powerplant	4556 kg
Wing	3532 kg

6.4 Retrofit Turboelectric & Hybrid Electric Aircraft

For the purpose of assessing the effects of how a change in propulsion system configuration and architecture affects overall aircraft performance, retrofit cases are considered. For the retrofit cases at the propulsion system level, the baseline gas turbine's gearbox is removed and replaced with an electric transmission. Components of the electric transmission are components associated with the architecture in Figure 4-1(b). In accounting for the installation of electric machinery at the aircraft level, the electric installation factor from Eq. 5-14 is taken to be 1.4 [5]. Additionally, for the retrofitted aircraft it is assumed that the masses and sizes associated with the empennage, aircraft equipment, fuselage, operational items, payload and wing remain constant. Also, for the retrofit cases, unless otherwise stated, climb TET is set to be the same as baseline aircraft's TET is during its design mission. Since a time-to-height mission requirement is not a stringent requirement in current studies, climb TET does not have to be increased if overall aircraft mass increases as a result of installation of electric machinery.

6.5 Turboelectric Operation

6.5.1 Turboelectric Architecture Technology Levels

For the purposes of assessing how an evolution in technology affects overall mission performance, turboelectric operation is assessed for two technology types. The first type is a conventional turboelectric (CTE) architecture sized with present day state-of-the-art technology and the second is with a fully superconducting turboelectric (STE) architecture sized with 2035 EIS

technology assumptions. For these different technology levels, power densities of electric components in the turboelectric propulsion system are shown in Table 6-15 and transmission efficiencies for electric components are shown in Table 6-16. The large variation in performance of the superconducting turboelectric propulsion system and conventional turboelectric propulsion system is expected to make a noticeable impact on overall performance at propulsion system level and aircraft level.

Table 6-15 Power densities of electric components in propulsion system

Component	Conventional turboelectric	Superconducting turboelectric
Generator (kW/kg)	8.22 [20]	41.59 ^b or 51.62 ^c [24]
Motor (kW/kg)	8.22 [20]	22.03 [24]
Power controller (kW/kg)	16.44 [21]	32.88 [21]
Transmission lines (kW/kg)	18.27 ^a [22] & [23]	227.1 [25]

^aMaximum possible power density of copper transmission lines. Calculated as the square of current density divided by the product of electrical conductivity and density

^bStators and rotors made with MgB₂

^cStators and rotors made with BSCCO

Table 6-16 Transmission efficiencies of electric components in propulsion system

Component	Conventional turboelectric	Superconducting turboelectric
Generator	0.95 [26]	0.9998 ^b or 0.9993 ^c [24]
Motor	0.95 [26]	0.9997 ^b or 0.9994 ^c [24]
Power controller	0.95 [27]	0.9993 [21]

Transmission lines	~1.00 ^a	~1.00 [25]
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^aSizing of a copper cable for 70 MW, 2 kA, 0.112 m diameter and 33.5 m length, then taking electrical resistivity of copper at 298 K of $1.712 \times 10^{-8} \Omega\text{m}$ [22] and assuming an ohmic resistor to determine transmission efficiency

^bStators and rotors made with MgB_2

^cStators and rotors made with BSCCO

6.5.2 Cooling Requirement

For the superconducting turboelectric architecture, cooling of the electrical machines need to be accounted for. Since no cryogenic fuel is stored on board for the cooling of the machinery, it is assumed a cryocooler is installed on board to carry out the cooling. To determine the power requirement of the cryocooler; the thermal load at machine cold temperature, cryocooler efficiency, ambient temperature and machine cold temperature are needed. For the determination of thermal load at machine cold temperature, power transfer through the electric transmission is fixed by GT design point shaft power output. Also, cryocooler efficiency as a fraction of Carnot efficiency is taken to be 0.3 [24]. With these cryocooler parameters set and setting ambient conditions to be ISA sea-level conditions (ambient temperature at this condition = 288.15 K), cryocooler power requirement is calculated for a range of cold temperatures using Eq. 4-62 for two different superconducting machine materials in Figure 6-8. The two different materials are bismuth strontium calcium copper oxide (BSCCO) machines and magnesium diboride (MgB_2) machines. For both scenarios power controller efficiency is the same for both.

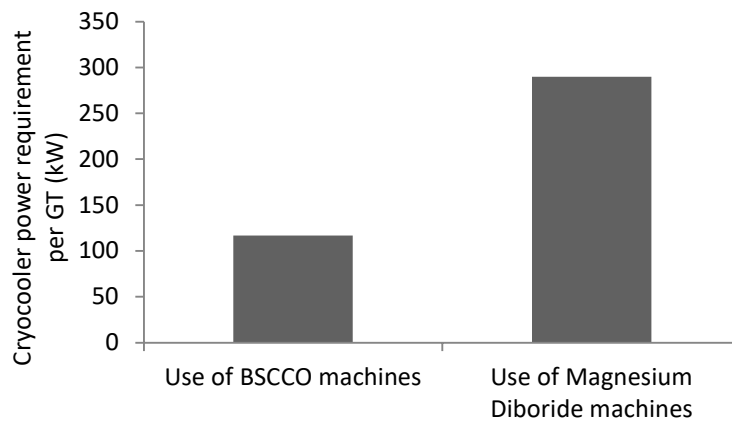


Figure 6-8 Cryocooler power requirements

From the results in Figure 6-8 it was decided that the use of MgB_2 as superconducting machine stator and rotor material would result in too high a cryocooler power offtake from the GT (at 20 K and using MgB_2 architecture transmission efficiency, cryocooler power requirement is approximately 8.46% of GT design point shaft power output). As a result BSCCO was chosen as the material of choice for superconducting machines despite slightly lower transmission efficiency. Also, BSCCO superconducting generators have higher power density than MgB_2 superconducting generators.

In integrating the cryocooler with the GT, the relationship between the two sub-systems is set to be a power offtake from the GT to run the cryocooler. In accounting for this power offtake, the effects of a change in thermodynamic cycle are not investigated and are beyond the scope of the current studies. Thus keeping thermodynamic cycle constant, the mass flow rate through the baseline GT at design point is increased to provide power to the cryocooler while also providing the power required by the propeller which is kept constant from the baseline configuration. This increase in mass flow rate through the GT at the design point effectively resizes the engine but keeps thermodynamic cycle constant. For size and mass estimation purposes of the GT, it is assumed that the increases in size and mass of the GT are directly proportional to the increase in mass flow rate through the GT (see Eq. 6-1). For a relatively small cryocooler power requirement as a fraction of GT design point power output (about 3.42% of GT design point shaft power output when using BSCCO

superconducting machines), it is expected that actual increase in GT size and mass would not differ greatly from size and mass calculated with “proportionality” assumption in Eq. 6-1. For more rigorous calculations, the proportionality assumption should take into account mass flow rate of the GT being proportional to intake area of the GT and then having GT mass being proportional to the product of GT length and intake area.

$$\frac{PW_{GT,with\ cryo}}{PW_{GT,no\ cryo}} = \frac{W_{GT,with\ cryo}}{W_{GT,no\ cryo}} = \frac{size\ of\ GT_{with\ cryo}}{size\ of\ GT_{no\ cryo}} = \frac{m_{GT,with\ cryo}}{m_{GT,no\ cryo}} \quad 6-1$$

For the conventional turboelectric machinery, although no cryocooler is considered, for practical use inefficiencies of using electric machinery will cause it to transfer heat to its surroundings and this heat will have to be managed effectively. Detailed thermal management of the conventional turboelectric machinery is beyond the scope of the current work and it is assumed aircraft thermal management system effectively manages heat transfer from conventional electric transmission machinery.

6.5.3 Turboelectric Propeller Performance

For considered electric architecture, the propeller speed can be set independently of the output speed of the GT. With output speed of the GT kept constant at off-design, load on the output turbine of the GT is controlled electrically by the generator. For optimal performance, this characteristic means the propeller can be run at its most efficient setting throughout the flight envelope. Thus on the propeller performance map for a given flight speed, both advance ratio and pitch angle will be varying as propeller shaft power changes. Also, for the turboelectric propeller system its design remains the same as the design for the baseline turboprop system.

Considering the cruise flight condition in long range cruise mode (see Figure 6-9), for the same engine throttle, power coefficient of the propeller in turboelectric mode is lower because of the higher rotational speed the propeller is allowed to rotate at. The propeller for the turboelectric case rotates at a higher rotational speed than the baseline case because the turboelectric

propeller is not constrained to be constant speed at 850 RPM during cruise. As engine throttle reduces, for the baseline case since RPM is fixed, propeller pitch angle is reduced as power swallowed by the propeller reduces. This results in power coefficient drop at constant advance ratio. For the turboelectric propeller case, simultaneous control of rotational speed and pitch angle causes propeller advance ratio to increase and pitch angle to change slightly as engine throttle reduces. The net result of simultaneous control of propeller rotational speed and pitch angle is higher propeller efficiency for a given engine throttle for the turboelectric case than for the baseline (see Figure 6-10). This is because higher propeller rotational speed operation for the turboelectric case means that for the same GT throttle, thrust is produced at a lower pitch angle due to higher circulation and thus induced drag of the propeller is lower for the turboelectric case. This allows for more efficient propeller operation. However, the difference in propeller efficiency reduces as engine throttle reduces, because the lower the transmitted power to the propeller shaft, the more advantageous to operate at higher advance ratios. This is because as shaft power reduces, efficiency increases due to reduced circulation reducing propeller blade induced drag and thus operation moves closer to pitch angle/advance ratio combination for maximum lift coefficient to drag coefficient ratio. Higher advance ratios for maximum efficiency for the turboelectric configurations at lower power coefficients means that the turboelectric propeller eventually matches the baseline turboprop at lower engine throttles during cruise. For the high speed cruise mission, performance trends are similar to that for the long range cruise mission albeit with the exception that the propellers operate at reduced efficiency, due to greater drag on the propellers operating at higher flight speed.

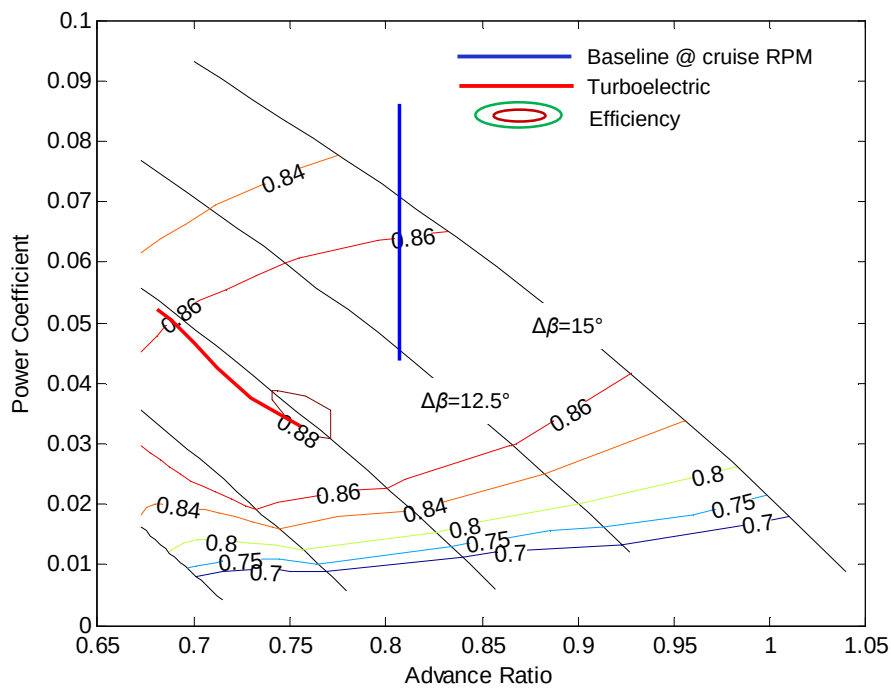


Figure 6-9 Baseline and turboelectric running lines on propeller performance map at cruise flight condition for long range cruise mission (propeller pitch angles specified at 2.5° intervals)

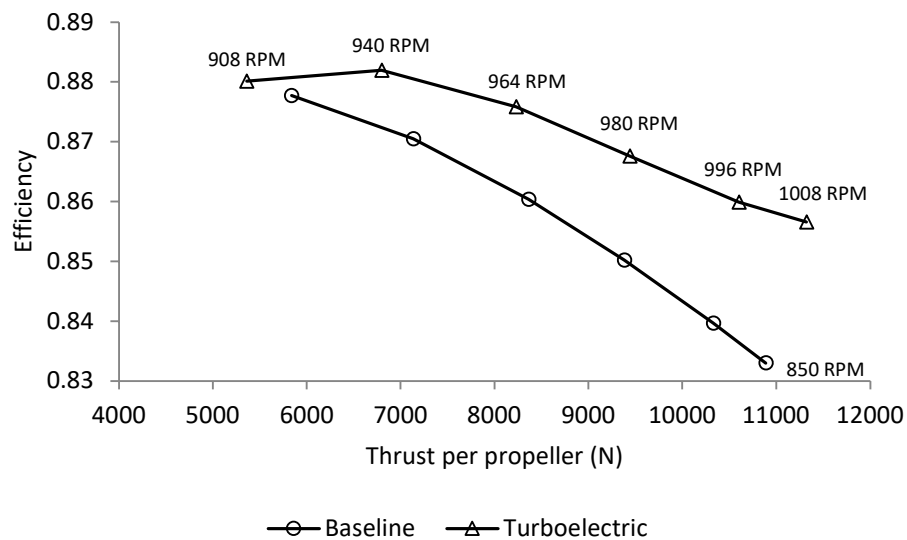


Figure 6-10 Propeller efficiency vs thrust for long range cruise mission at cruise flight condition

Although for present studies the turboelectric propeller is being controlled by both pitch angle change and rotational speed, potential limitations to this

scheme are more complex system control, increased propeller noise during climb and cruise, and increased machinery vibration for an extended period of flight. Considering increased machinery vibration further, this has the potential to impact overall lifetime of the propeller-motor combination compared to the lifetime of the propeller-gearbox combination of the baseline turboprop architecture.

With propellers optimised for a given GT throttle setting throughout the flight envelope, integrated propulsion system performance assessment is carried out taking into account architecture technology levels, electric transmission losses and gas turbine performance for both the long range cruise settings and high speed cruise settings throughout the flight envelope. Considering integrated turboelectric propulsion system performance for the long range cruise mission (see Figure 6-11), at the propulsion system level, it is found that the conventional turboelectric configuration has a higher TSFC than the baseline at all flight conditions. This is primarily due to the worse transmission efficiency between gas turbine and propellers that arises from use of the electric machinery compared to the use of a gearbox (81.45% for combined transmission efficiency for all electric components in conventional turboelectric architecture versus 98.25% gearbox efficiency). For the conventional turboelectric architecture this more than offsets the benefit of increased propeller efficiency that arises from turboelectric operation. For superconducting turboelectric application if cooling losses are not taken into account, the architecture achieves lower TSFC than the baseline configuration due to a combination of higher overall transmission efficiency and higher propeller efficiency. However, when cooling losses are taken into account, TSFC for the superconducting turboelectric architecture increases and at lower engine throttles this causes TSFC for the configuration to be higher than that for the baseline particularly because at lower engine throttle settings the fraction of cooling power offtake to total shaft power output increases. For the high speed cruise mission similar to the long range cruise mission, cooling power offtake offsets benefit of operating propeller at higher efficiency and results in propulsion system TSFC being higher than the baseline powerplant's TSFC.

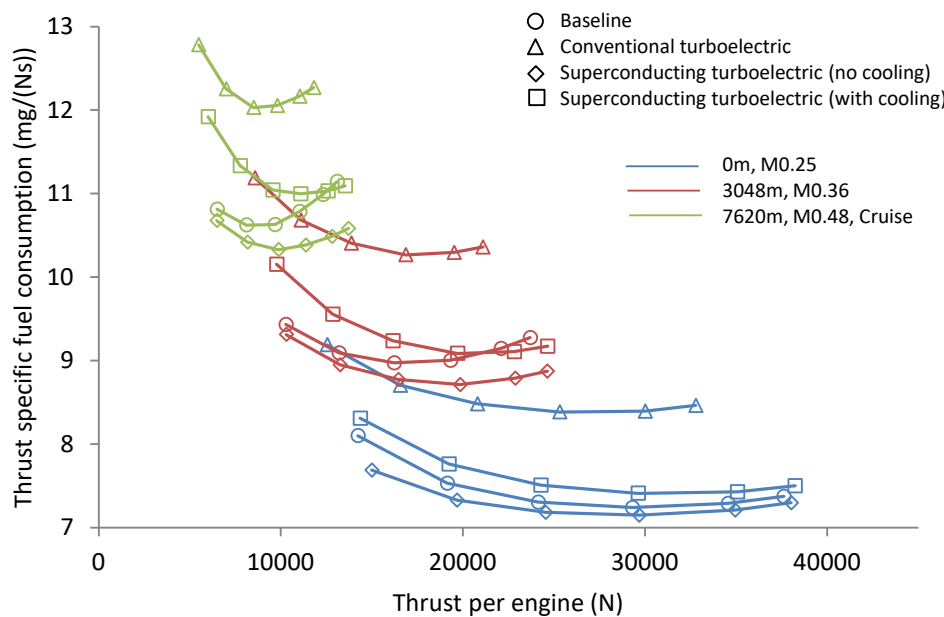


Figure 6-11 Baseline and turboelectric powerplant TSFC performance for long range cruise mission at selected flight conditions for regional transport mission

6.5.4 Propulsion System Mass Breakdown

For integration with the aircraft propulsion system mass was assessed. A summary of the masses determined for turboelectric configurations compared to the baseline configurations is shown in Table 6-17. For the baseline architecture high installation mass arises from the weight of the transmission between power turbine to gearbox and propellers, size of the exhaust system and the utilisation of the propulsion system nacelle to store the landing gear. For the architectures considered, the conventional turboelectric architecture is heaviest due to the employment of relatively low power density machines which have a high power rating despite the removal of the gearbox. For the superconducting turboelectric architecture although electric transmission is lighter than conventional turboelectric architecture's transmission, it is still slightly heavier than the baseline due to the requirement of a slightly larger gas turbine to provide the additional power needed by the cryocooler and also due to overall mass of the electric transmission being heavier than gearbox removed.

Table 6-17 Turboelectric propulsion system mass breakdown

Item	Baseline	Conventional Turboelectric	Superconducting Turboelectric
Turbomachinery (kg)	735	735	760
Gearbox (kg)	700	0	0
APU (kg)	86	86	86
Propellers (kg)	504	504	504
Cryocooler (kg)	0	0	353
Electric transmission (kg)	0	2813	922
Installation (kg)	2531	2782	2197
Total (kg)	4556	6920	4822

6.5.5 Aircraft Performance

6.5.5.1 Long Range Cruise Mission

For results for conventional turboelectric configuration, because the reserve mission did not converge for a specified climb TET of 1457 K (distance flown during climb and descent of reserve mission exceeds overall specified range for the diversion), reserve mission climb TET was increased to 1487 K. This happened because the conventional electric machinery was too heavy for the retrofitted aircraft. Although in practice this would not be done as engine life would be impaired, increasing the TET to 1487 K for the conventional turboelectric configuration allows for a performance comparison with baseline and superconducting turboelectric configuration. An alternative approach for the conventional turboelectric architecture which was not assessed here, is that

size for the core GT of the conventional turboelectric architecture could be increased to provide a sufficient amount of power to carry out reserve mission climb within a short enough distance.

In considering mission performance, the increase in propulsion system mass due to incorporation of electric machinery worsens vehicle aerodynamic efficiency and increases aircraft thrust requirements. At cruise where thrust requirement is fixed by aircraft drag, the net result of increased thrust requirements is an increase in TET for turboelectric configurations (see Figure 6-12). The TET for the conventional turboelectric configuration is highest due to the configuration having the highest powerplant mass. A likely consequence of the turboelectric aircraft configurations operating at a higher TET than the baseline during cruise is engine life and NO_x emissions will be adversely affected. Subsequently these issues will adversely impact GT economics. Additionally, if there is a TET requirement for cruise that must be met, GTs for the turboelectric architectures would need to be resized to provide the requisite mission thrust requirement at a given flight condition. For all configurations assessed in Figure 6-12, TET falls as cruise progresses due to fuel burn.

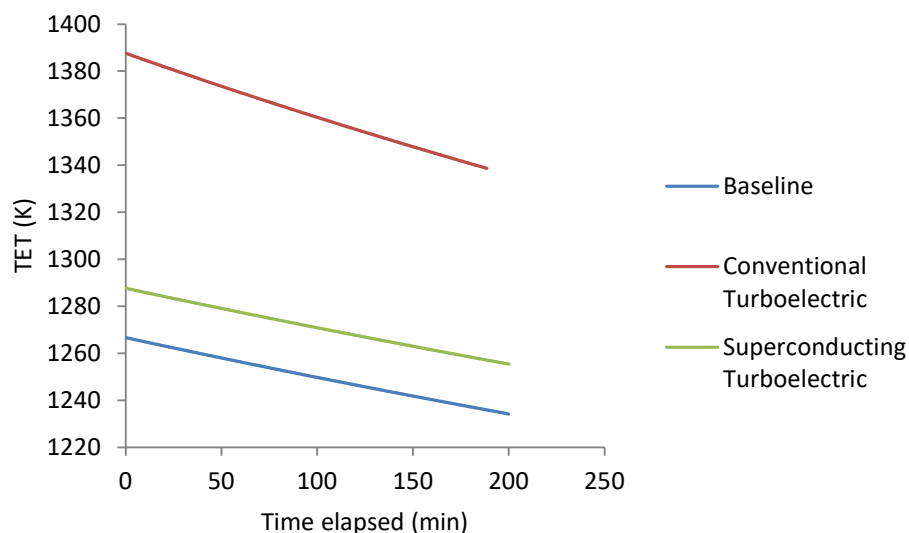


Figure 6-12 Regional transport TET variation with cruise time elapsed for turboelectric aircraft configurations at design range for long range cruise mission

Despite operating at a higher TET, the propellers for the turboelectric configurations operate at higher propeller efficiencies than the baseline case during cruise due to optimal matching between propellers and GT (see Figure 6-13). The conventional turboelectric architecture operates at slightly lower propeller efficiency than the superconducting turboelectric configuration due to the higher power demand of the propellers from the GT to meet higher flight thrust requirement. This means that the propellers of the conventional turboelectric configuration operate at higher power coefficients during cruise than the superconducting turboelectric configuration. At higher power coefficients, propeller efficiency is lower due to increased circulation having the effect of increasing blade lift and lift-induced-drag. Thus, propellers for the CTE configuration operate at lower efficiency during cruise than propellers for the STE configuration (see Figure 6-14). From Figure 6-14, for the baseline case as thrust requirement falls, power coefficient drops at constant advance ratio by a reduction in propeller pitch angle. However for the turboelectric cases, power coefficient falls due to advance ratio increase and slight fall in pitch angle.

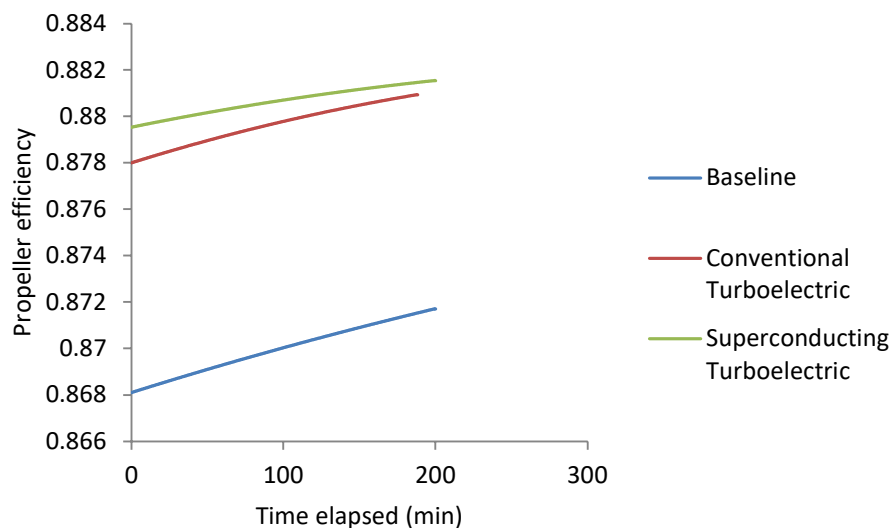


Figure 6-13 Regional transport propeller efficiency variation with time elapsed during cruise for turboelectric aircraft configurations at design range

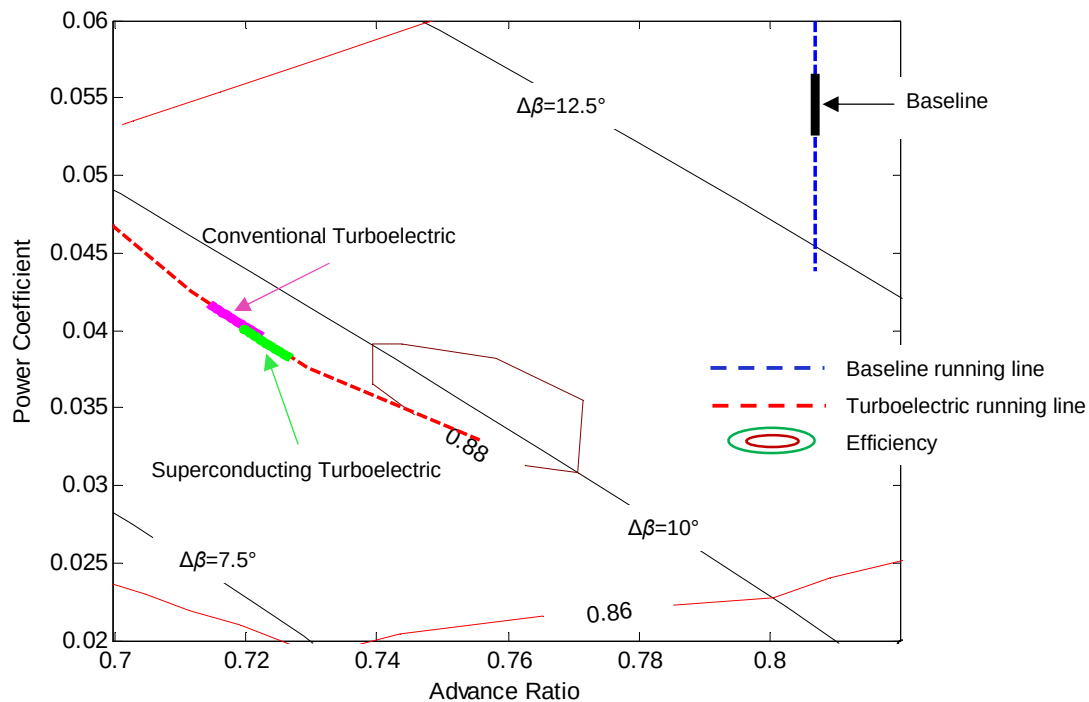


Figure 6-14 Regional transport propeller operation at cruise for turboelectric aircraft configurations at design range for long range cruise mission on propeller performance map

During cruise propulsion system TSFC for the conventional turboelectric configuration is higher than the TSFC for the baseline and superconducting turboelectric configuration primarily due to lower transmission efficiency between propellers and the GT (see Figure 6-15). Superconducting turboelectric aircraft configuration TSFC during cruise is higher than baseline TSFC primarily due to cooling power offtake from the GT offsetting higher propeller efficiency.

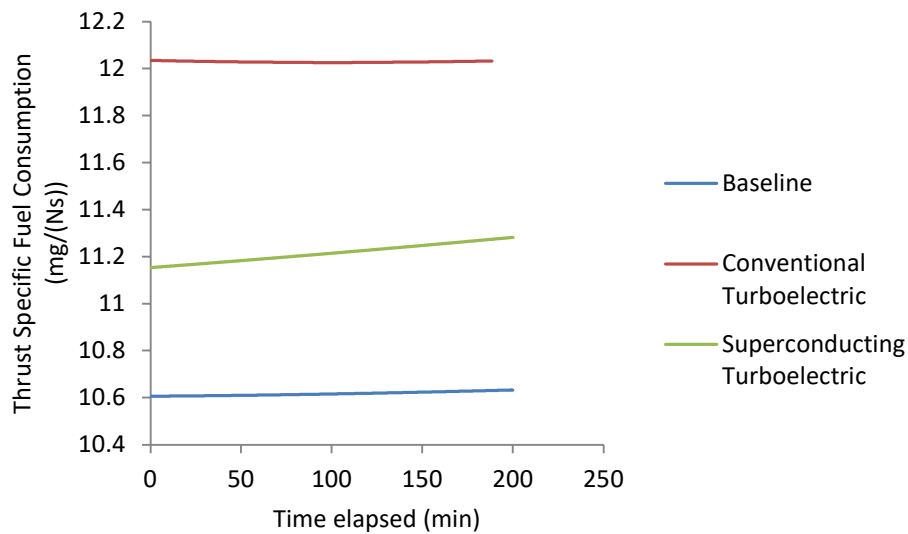


Figure 6-15 Regional transport propulsion system TSFC variation with cruise time elapsed for turboelectric aircraft configurations at the design range

Aircraft top level performance summary comparing turboelectric aircraft configurations to the baseline is shown in Table 6-18. As a result of increased TSFC and increased propulsion system weight, the turboelectric configurations burn more fuel than the baseline configuration. Also, due to increased propulsion system weight; TOFL, time-to-height and MTOM for the turboelectric aircraft configurations are higher than associated values for the baseline configuration. For the conventional turboelectric aircraft configuration, its time-to-height is considerably higher than the time-to-height of the baseline aircraft. Practically this would mean that flight with the conventional turboelectric aircraft would not be possible as it would not meet mission time-to-height requirements if this requirement was specified. It should be noted that although a retrofitted aircraft has been considered and thus wing mass has been kept constant, the situation of electric machinery on the aircraft wing could allow for a lighter wing structure due to increased bending relief. However, if electric machinery is situated in the fuselage, increased vehicle weight may necessitate structural reinforcing of the fuselage which could increase MTOM even further.

Table 6-18 Regional transport aircraft top-level performance summary for turboelectric aircraft configurations at the design range

Configuration	Baseline	Conventional Turboelectric	Superconducting Turboelectric
TOFL (m)	1364	2188	1385
Time-to-height (min)	16.98	34.67	17.06
Block fuel (kg)	2654	3350	2818
Reserve fuel (kg)	947	1161	996
MTOM (kg)	29401	32674	29881

As aircraft range reduces block fuel for all aircraft configurations reduces (see Figure 6-16). However the deviation in fuel burned between the turboelectric and baseline aircraft reduces. This is because at shorter aircraft ranges, flight time is not long enough for the detrimental effects of a heavier and more inefficient propulsion system to make a noticeable impact on block fuel. Decreasing range also has the effect of decreasing TOFL, time-to-height, reserve fuel and TOW. For the superconducting turboelectric configuration at 200 n.mi TOW decreases by 7.65%, TOFL decreases by 14.9% and time-to-height decreases by 15.9% from design range values.

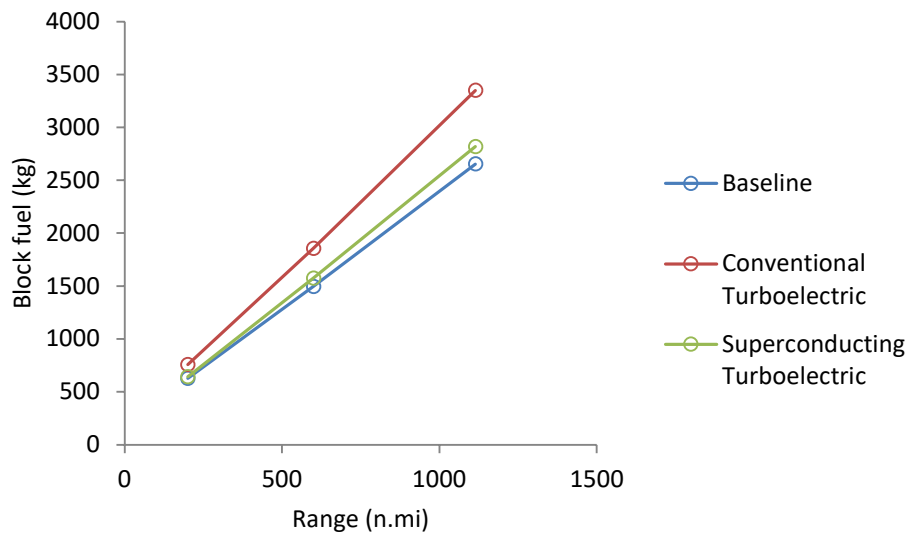


Figure 6-16 Effect of aircraft range on block fuel of turboelectric aircraft configurations for long-range cruise mission

6.5.5.2 High Speed Cruise Mission

For the high speed cruise mission the only turboelectric configuration being considered is the superconducting turboelectric configuration. This is because from the analysis for the long-range-cruise mission it is expected that there will be significant performance penalties when employing conventional turboelectric architecture for an aircraft. For the superconducting turboelectric configuration, similar to the long range cruise mission, since the GT cycle does not change, engine operating TET during cruise is greater than operating TET for the baseline due to higher thrust requirements (see Figure 6-17). Also, cruise TSFC is higher for the superconducting turboelectric configuration due to the cooling power offtake from GTs. At 600 n.mi (see Figure 6-18) and 200 n.mi (see Figure 6-19) these effects result in higher block fuel than the baseline for the superconducting turboelectric configurations. At 200 n.mi similar to the baseline, block fuel for the superconducting turboelectric configuration in high speed cruise mode is lower than block fuel in long range cruise mode due to reduced journey time offsetting higher TSFC.

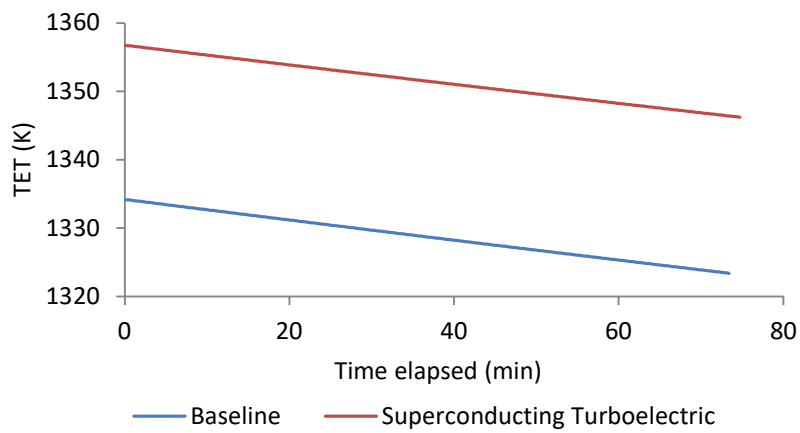


Figure 6-17 Regional transport TET variation with cruise time elapsed for turboelectric aircraft compared to baseline aircraft at 600 n.mi for high speed cruise mission

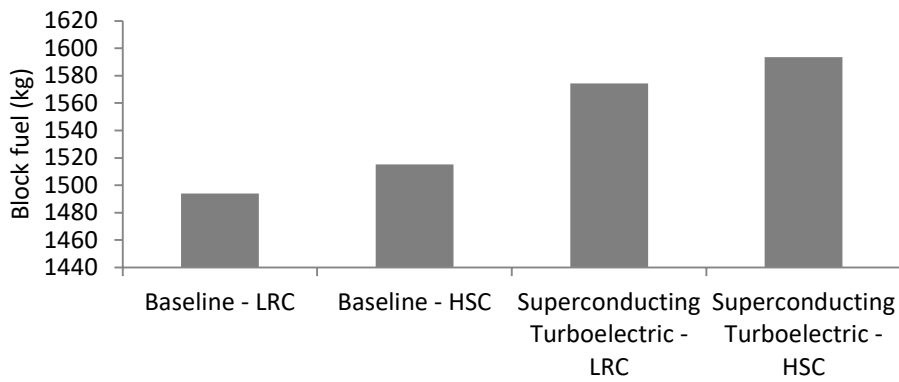


Figure 6-18 Regional transport block fuel comparison for baseline and turboelectric aircraft for mission types considered at 600 n.mi

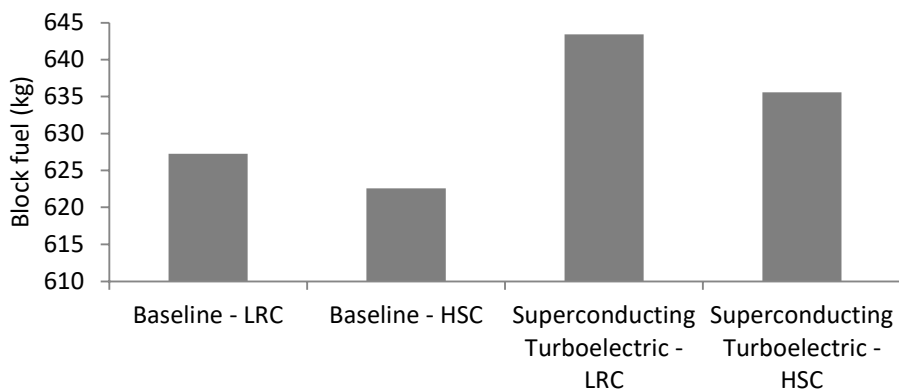


Figure 6-19 Regional transport block fuel comparison for baseline and turboelectric aircraft for mission types considered at 200 n.mi

6.6 Hybridising the Turboelectric Powerplant

6.6.1 Electric Architecture Technology Level Consideration

From analysis in the previous section, the conventional present day state-of-the-art turboelectric architecture performed significantly worse than the superconducting turboelectric architecture. Thus for hybrid electric aircraft performance assessment the conventional turboelectric architecture is not considered further. Instead for hybrid electric assessment, performance assessment here focuses on how hybridising the superconducting turboelectric architecture affects aircraft performance.

6.6.2 Battery Pack

An overview of the performance parameters used for the modelling of the battery pack is shown in Table 6-19. Values for the parameters are taken from battery pack design values from public literature. In Table 6-19 a minimum SOC is specified because according to battery technology a discharge of the batteries below a 20% SOC limit can reduce battery life or damage it severely [28]. Utilising cell parameters in Table 4-4 coupled with parameters set in Table 6-19 for battery performance simulation, the discharge efficiency profile to be used for flight performance assessment is obtained in Figure 6-20. For the profile, discharge efficiency decreases slightly as SOC decreases due to cell open circuit voltage drop.

Table 6-19 Utilised battery pack performance parameters

SOC at start-of-discharge	100% [28]
Minimum SOC	20% [28]
Nominal cell current	0.64 A [29]
Maximum voltage of battery pack	3000 V [30]

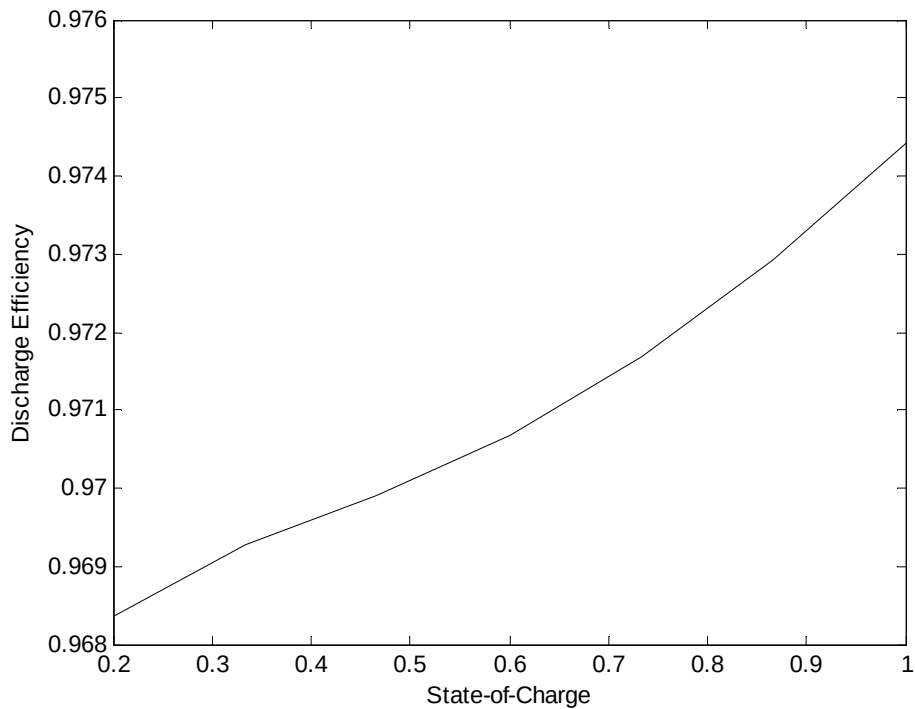


Figure 6-20 Battery discharge efficiency versus state-of-charge

For assessment of the stored energy in the battery for a given battery mass, three technology levels are considered (each technology level represents a target battery specific energy). These technology levels are listed below as:

- Present day state-of-the-art lithium ion battery technology level with battery specific energy equal to 265 Wh/kg [31]
- A future technology for year 2035+ entry-into-service with battery specific energy equal to 1500 Wh/kg [32]
- An intermediate specific energy between present day state-of-the-art and future estimates representing a stepwise-change in technology improvements in the mid-term (battery specific energy equal to 883 Wh/kg)

6.6.3 Propulsion System Mass Breakdown

Considering hybrid architecture to be the architecture defined in Figure 4-1(d), a summary of the propulsion system mass breakdown for hybrid turboelectric configurations at selected battery power ratings compared to baseline and turboelectric architecture is shown in Table 6-20. For the hybrid electric

architectures turbomachinery, APU, propeller and cryocooler masses are fixed by associated masses for the superconducting turboelectric architecture. However electric transmission mass increases as battery power rating increases due to the incorporation of a DC/DC converter to interface the battery with main electric network of the hybrid electric architecture. The DC/DC converter increases in mass as battery power rating increases. Battery mass is not included as part of the propulsion system mass and is instead considered as a stand-alone mass item at the aircraft level. This is because since it is an energy source, it is treated to similar to fuel in the aircraft mass breakdown.

Table 6-20 Regional transport hybrid turboelectric propulsion system mass breakdown

Item	Baseline	Superconducting Turboelectric	500 kW battery Hybrid Turboelectric	1000 kW battery Hybrid Turboelectric	1500 kW battery Hybrid Turboelectric	2000 kW battery Hybrid Turboelectric
Turbomachinery (kg)	735	760	760	760	760	760
Gearbox (kg)	700	0	0	0	0	0
APU (kg)	86	86	86	86	86	86
Propellers (kg)	504	504	504	504	504	504
Cryocooler (kg)	0	353	353	353	353	353
Electric transmission (kg)	0	922	937	952	966	981
Installation (kg)	2531	2197	2203	2209	2215	2221
Total (kg)	4556	4822	4843	4864	4885	4905

6.6.4 Hybrid Electric Aircraft Performance

6.6.4.1 Design Objectives and Hybrid Mission Scheme

For hybrid electric performance assessment here, emphasis is placed on the reduction of block fuel consumption and energy consumption. With these as a

primary focus and considering aircraft cruise type (whether long range cruise or high speed cruise), the cruise types considered for a given aircraft range are fixed according to which mission type yielded the minimum block fuel consumption for the baseline aircraft for a given aircraft range. Thus for each aircraft range considered the following cruise types are considered:

- Design Range – Long range cruise
- 600 n.mi – Long range cruise
- 200 n.mi – High speed cruise

Additionally when using an installed battery pack with an aircraft, a large number of operation schemes are possible. This is because there are a number of factors that affect how a battery can be used in conjunction with the GT. Depending on which factor is considered, aircraft performance will be affected. A selection of these factors is listed below:

- GT relighting and continuous battery discharge-recharge
- Initial stored energy state of battery
- Battery start-of-discharge point on the mission profile

Each of the factors listed above are effectively independent of each other and open up a large number of possible hybrid operation schemes. For the current scope of work, GT relighting and battery recharge is not considered due to the desire to understand the physics of basic hybrid electric operation before considering more complex schemes. Also since recharging is not considered, it is assumed that the initial stored energy state of the battery at the beginning of the mission is fully charged to enable hybrid operation and to assess the maximum possible benefit of utilising chosen hybrid electric architecture.

In considering battery start-of-discharge point, studies are limited to assessing the electrification of cruise (thus start-of-discharge point is initial cruise). With cruise fuel burn for the baseline aircraft at 1,114 n.mi range representing 79% of the block fuel of the baseline aircraft, it is expected that the largest reduction in block fuel and energy consumption by using batteries will happen during cruise.

6.6.4.2 Effect of Increasing Battery Power

Considering batteries for hybrid electric aircraft that has a specific energy of 1500 Wh/kg and a mass of 500 kg (see Figure 6-21) at the design range, for constant battery power output energy depletion from the battery causes SOC to reduce. For higher battery power ratings, the rate of energy depletion from the battery is increased.

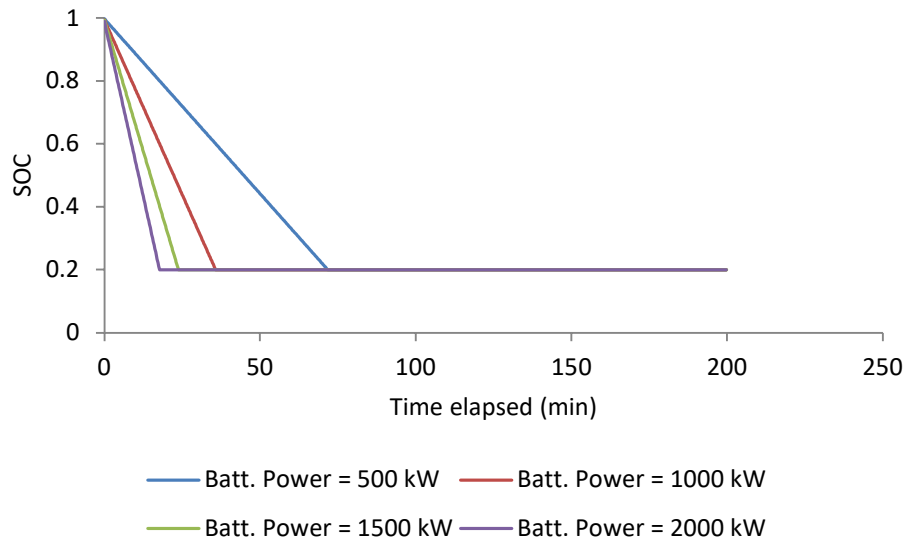


Figure 6-21 Battery SOC vs cruise time elapsed for different battery power ratings at the design range for batteries that weigh 500 kg and have specific energy of 1500 Wh/kg for regional transport

As transmitted power to propellers from battery pack increases, the power required from the GT for a required amount of thrust reduces. As a result GT TET and fuel flow rate decrease for a given amount of power discharged from the batteries. Engine TET and fuel flow rate reduction in hybrid operation is increased as battery power is increased. Even through fuel flow is reduced, operating of the GT at lower TETs further away from design point TET reduces GT thermal efficiency. However despite lower GT thermal efficiency, overall propulsion system efficiency (see Eq. 4-73) is increased as a result of discharge of power from a more efficient battery to propulsors for a given thrust requirement. This results in more efficient flight whilst the battery is being operated and this is represented by higher energy specific air range (see Figure

6-22) as battery power rating increases. However at higher battery power ratings the time at which the aircraft is operating at higher ESAR is shortened compared to the time spent at higher ESAR for lower battery power ratings. Once the battery is depleted of energy, ESAR for all hybrid electric configurations converges to ESAR for the case where power transmitted to the propellers is coming from the GT only. During both portions of the flight when the battery is operational and not operating, ESAR increases slowly due to a reduction in aircraft weight as a result of fuel burn. It should be noted that another consequence of reduced TET of the GT during hybrid operation, is that NO_x emissions are likely to be lower than what is the case for both the baseline propulsion system and the superconducting turboelectric propulsion system.

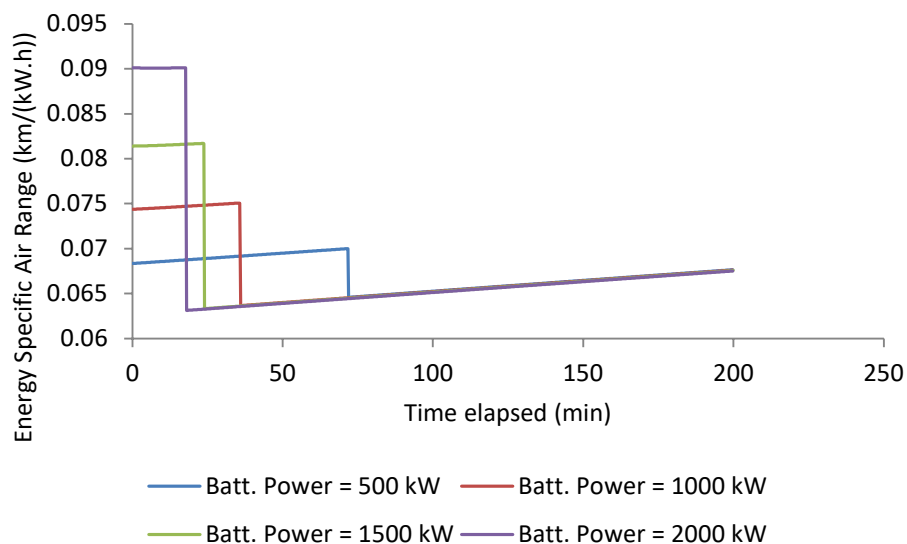


Figure 6-22 Energy Specific Air Range vs cruise time elapsed for different battery power ratings at the design range for batteries that weigh 500 kg and have specific energy of 1500 Wh/kg for regional transport

In assessing the effect of performance trends on overall block fuel, if consideration is only made to the power rating-ESAR relationship, the two effects of lower battery power rating for longer discharge time and higher battery power rating for shorter discharge time match each other and would result in equal overall block fuel for any power rating as long as a battery fully discharges before the end of a given mission. However since propulsion system

mass increases slightly as battery power rating increases as a result of electric transmission mass increase, for a fixed battery mass overall aircraft weight increases slightly with increased battery power rating. Increased aircraft weight causes an increase in time-to-height and this slightly increases climb fuel burn. Increased climb fuel burn means a slightly higher block fuel for higher battery power rating hybrid electric configurations. In comparing the fuel burn of hybrid electric aircraft configurations to superconducting turboelectric aircraft and baseline aircraft, for a 500 kg battery with specific energy of 1500 Wh/kg, the duration of time the hybrid electric aircrafts spend at increased ESAR results in reduced block fuel compared to the superconducting turboelectric configuration (see Table 6-21). However compared to the baseline, the batteries for the hybrid electric aircraft configuration do not spend a sufficient amount of time discharging to offset electrical machinery weight penalties to have a lower block fuel than the baseline aircraft. When total energy consumption is considered, benefit over superconducting electric configuration diminishes due to the additional energy expended by the battery in combination with fuel energy expenditure.

Table 6-21 Regional transport deviation in block fuel and main mission total energy consumption from baseline values for hybrid turboelectric aircraft configurations at the design range for batteries that weigh 500 kg and have specific energy of 1500 Wh/kg

	Superconducting Turboelectric	Hybrid Turboelectric – 500 kW	Hybrid Turboelectric – 1000 kW	Hybrid Turboelectric – 1500 kW	Hybrid Turboelectric – 2000 kW
Deviation in block fuel from baseline	6.18%	4.07%	4.22%	4.39%	4.51%
Deviation in total energy consumed from baseline	6.18%	5.96%	6.11%	6.28%	6.40%

It should be noted that although total energy consumption is a better measure in comparing hybrid-electric aircraft configurations to non-hybrid configurations since it takes into account consumption of energy from two energy sources; block fuel can still be an important performance parameter for evaluating hybrid-electric aircraft configurations. This is because depending on the mission objective; a minimum cost, minimum emissions and/or minimum fuel mission objective could be desired. For environmental reasons lower block fuel will allow for reduced aviation CO₂ emissions. Also, if the cost of producing electricity to provide a unit amount of energy to a battery is lower than the cost purchasing a unit amount of energy stored in fuel, economically it would be advantageous for block fuel to be minimised as much as possible even if overall total energy consumption is increased.

6.6.4.3 Effect of Increased Specific Energy

Considering batteries for hybrid electric aircraft that weigh 500 kg with a power rating of 500 kW at the design range, discharge of power from batteries depletes energy from batteries at a constant rate as battery specific energy increases (see Figure 6-23). However, due to a greater amount of stored energy in the battery as specific energy increases, it takes a longer amount of time for the battery to be depleted to minimum SOC as specific energy increases.

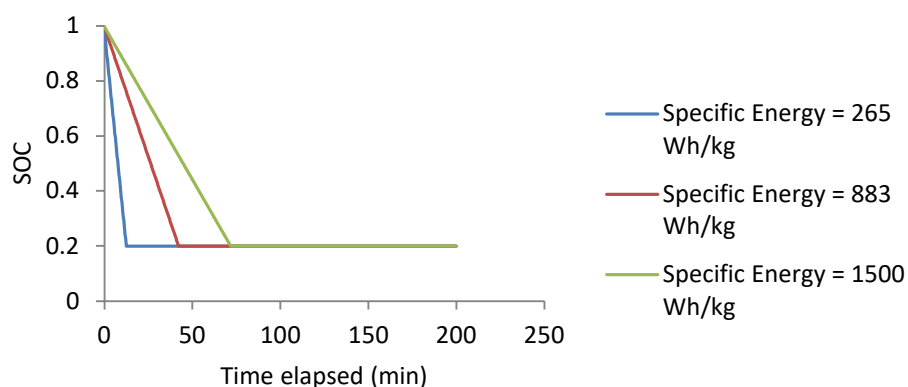


Figure 6-23 Battery SOC vs cruise time elapsed for different battery specific energies at the design range for batteries that weigh 500 kg and have power rating of 500 kW for regional transport

During the mission, GTs spend a longer time operating at a lower TET with a lower engine fuel flow rate as battery specific energy is increased. This results in reduced block fuel as battery specific energy increases. For retrofit aircraft and constant powerplant mass, reduced block fuel reduces MTOW as battery specific energy increases. Reduction of MTOW results in slightly lower TET and fuel flow rate for the same battery power output at increased battery specific energy. As explained in previous section, even though TET reduction when the propulsion system is in hybrid mode leads to less efficient GT operation, overall powerplant efficiency is higher whilst the battery is operational due to relatively high battery discharge efficiency. Higher powerplant efficiency results in higher ESAR than ESAR during phases of the flight where power transmitted to the propellers is coming from the GT only (see Figure 6-24). During period of flight while the battery is on, reduced aircraft weight results in slightly higher ESAR as battery specific energy increases. Also duration of time battery spends at increased ESAR is increased as battery specific energy increases. In Figure 6-24 during both portions of flight when battery is operational and not operating, ESAR increases due to reduction in aircraft weight as a result of fuel burn. Also, it should be noted that the rate of change of ESAR with respect to time stays constant as battery specific energy increases because battery power output remains constant. Since the rate of change of ESAR is proportional to the rate of energy consumption and the rate of energy consumption being equal to battery power output, the net result is constant rate of change of ESAR with respect to time as battery specific energy increases as long as battery power output remains constant.

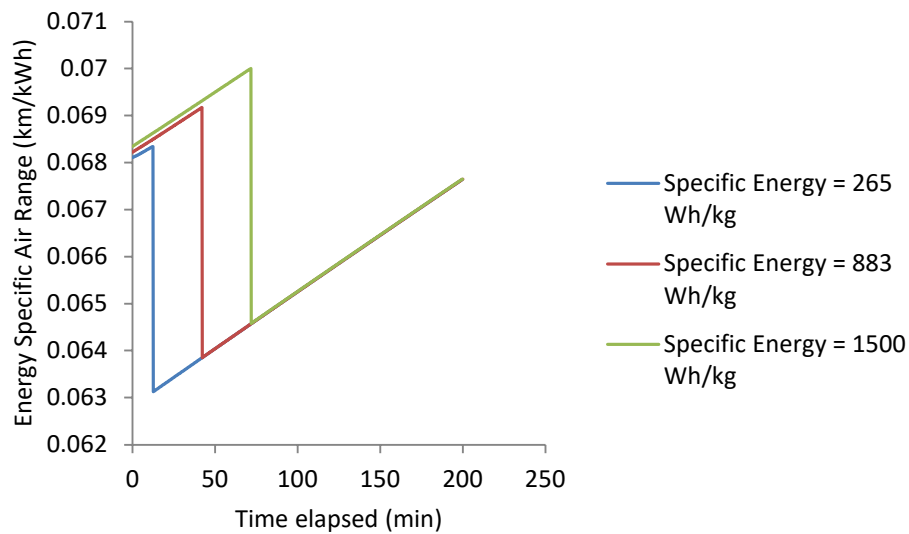


Figure 6-24 Energy Specific Air Range vs cruise time elapsed for different battery specific energies at the design range for batteries that weigh 500 kg and have power rating of 500 kW for regional transport

When the battery is turned off, the difference in ESAR at a given mission point between the hybrid electric aircraft configurations at different specific energies is reduced. This is because as the mission progresses, the difference in aerodynamic efficiency and aircraft weight at a given point in time between the hybrid electric aircraft at lower specific energy and hybrid electric aircraft at higher specific energy reduces. Since a hybrid electric aircraft configuration at a lower specific energy (for example at 265 Wh/kg) has a higher initial weight at the beginning of cruise, because it is heavier its weight actually falls at a slightly faster rate than a hybrid electric aircraft configuration with a higher specific energy (e.g. 1500 Wh/kg) during cruise. With batteries turning off at a point in time after cruise, cruise duration is long enough so that when batteries turn off, the reduced difference in aerodynamic efficiency and aircraft weight between hybrid electric aircraft configurations for a range of battery specific energies causes the ESAR between the configurations to almost match when the battery is turned off.

In comparing the fuel burn of hybrid electric aircraft configurations to baseline aircraft and superconducting turboelectric aircraft in Table 6-22, for a 500 kg

battery with 500 kW battery power rating, increasing specific energy from 265 W.h/kg to 1500 W.h/kg results in an aircraft that has lower block fuel than superconducting turboelectric configuration. However compared to the baseline, the battery for the 1500 W.h/kg hybrid electric aircraft does not spend a sufficient amount of time discharging to offset electrical machinery weight penalties to have a lower block fuel than the baseline aircraft. Considering total energy consumption, deviation in total energy consumed from baseline aircraft for hybrid electric aircraft is greater than the deviation in block fuel due to additional energy expended by the battery.

Table 6-22 Regional transport deviation in block fuel and main mission total energy consumption from baseline values for hybrid turboelectric aircraft configurations at the design range for batteries that weigh 500 kg and have power rating of 500 kW

	Superconducting Turboelectric	Hybrid Turboelectric – 265 Wh/kg	Hybrid Turboelectric – 883 Wh/kg	Hybrid Turboelectric – 1500 Wh/kg
Deviation in block fuel from baseline	6.18%	7.77%	5.98%	4.07%
Deviation in total energy consumed from baseline	6.18%	8.10%	7.09%	5.96%

6.6.4.4 Effect of Increased Battery Mass

Considering batteries for hybrid electric aircraft with power rating of 500 kW and a specific energy of 1500 Wh/kg at the design range, energy depletion from the battery due to discharge of power remains constant as battery mass increases. However since stored electrical energy on board the aircraft increases with an increase in battery mass, the time it takes to fully discharge the battery increases with an increase in battery mass. In Figure 6-25 this is represented by

the time it takes for the battery to reach minimum SOC being increased as battery mass increases. When the battery mass is too high, cruise duration is not long enough for the battery to fully discharge during cruise. This is because energy consumed from the battery during flight is less than the available stored energy in the battery. This results in minimum SOC not being reached during a mission at relatively high battery masses for 500 kW battery power output.

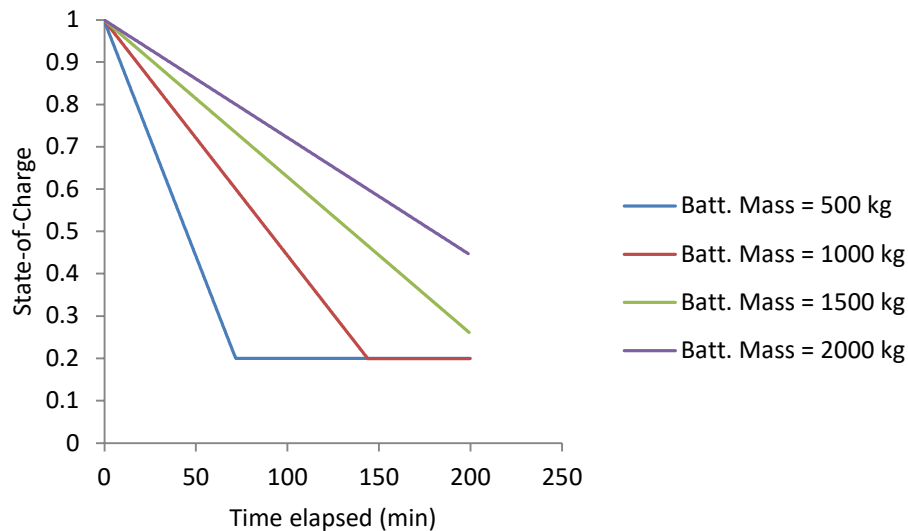


Figure 6-25 Battery SOC vs cruise time elapsed for different battery masses at the design range for batteries with power rating of 500 kW that have a specific energy of 1500 Wh/kg for regional transport

As battery mass increases, aircraft weight increases and vehicle aerodynamic efficiency is reduced. Assessing results in Figure 6-26, increased weight and lowered vehicle aerodynamic efficiency cause a decrease in ESAR for hybrid electric aircraft configurations as battery mass increases for a given operating mode of the propulsion system (whether in hybrid mode or turbo mode). However since available energy in batteries is used up at an earlier point in time during cruise (and thus powerplant transitions to turbo mode earlier), ESAR at lower battery masses falls below ESAR at increased battery masses after this point in the mission profile is reached. This is because propulsion system is more efficient when operating with power being transmitted to the propellers from GT and batteries rather than what is the case when power is transmitted to

propellers from GT only. At increased battery masses where cruise duration is not long enough for battery to fully discharge during cruise, ESAR continuously increases without reducing as transmission from both battery and GT to propellers continues throughout cruise. For all battery masses considered, during portions of the flight when battery is operational and not operating, ESAR increases due to reduction in aircraft weight as a result of fuel burn. Additionally, similar to the case as battery specific energy increases at a constant power output, increase in battery mass at constant power output means the rate of change of ESAR with respect to time remains constant.

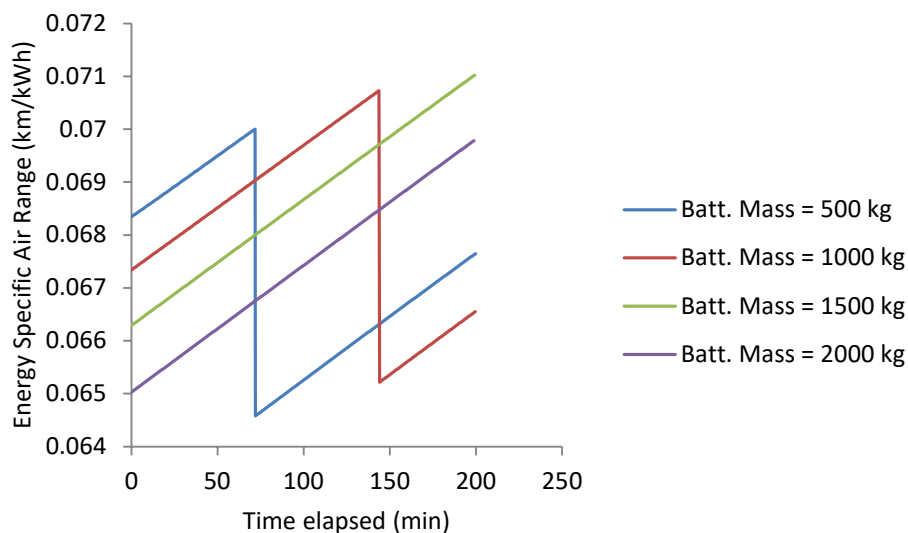


Figure 6-26 Energy Specific Air Range vs cruise time elapsed for different battery masses at the design range for batteries with power rating of 500 kW that have a specific energy of 1500 Wh/kg for regional transport

At increased battery masses, battery power rating (or power hybridisation) can be increased so that more of available energy stored in the batteries can be used. With Figure 6-27 as a reference, at relatively lower battery masses, in terms of block fuel there is no benefit of increasing battery power rating (see Figure 6-28) as block fuel increases due to electric transmission mass increase. At relatively lower battery power rating (using 500 kW battery power rating which corresponds to a power hybridisation of 0.073 as an example), increase in battery mass initially reduces block fuel. This reduction results in a lower

block fuel than baseline block fuel for hybrid electric aircraft with a battery that weighs 1500 kg with power rating of 500 kW. However the rate of block fuel reduction with respect to battery mass increase reduces when cruise duration is not long enough for battery to fully discharge. At battery masses where this is the case, unused stored available energy in batteries during cruise causes a deadweight effect on the aircraft in addition to the reserve fuel. Battery mass increase at constant battery power output while the battery does not fully discharge exacerbates the deadweight effect. Exacerbated deadweight effect eventually increases block fuel at 500 kW battery output when battery mass increases beyond 1500 kg. When battery power rating is increased at higher battery masses from power rating that resulted in unused stored available energy in battery, more energy is consumed from the battery at a lower GT fuel flow rate. Increased energy consumption from the battery coupled with lower engine fuel flow rate reduces block fuel as battery power increases at increased battery masses. This allows for hybrid electric aircraft configurations with battery masses greater than 1500 kg with power ratings of 1000 kW or higher to achieve lower block fuel than baseline. However, once the battery fully discharges at or just before the end-of-cruise, similar to the effect at lower battery masses, increasing battery power rating slightly increases overall block fuel due to higher electric transmission mass.

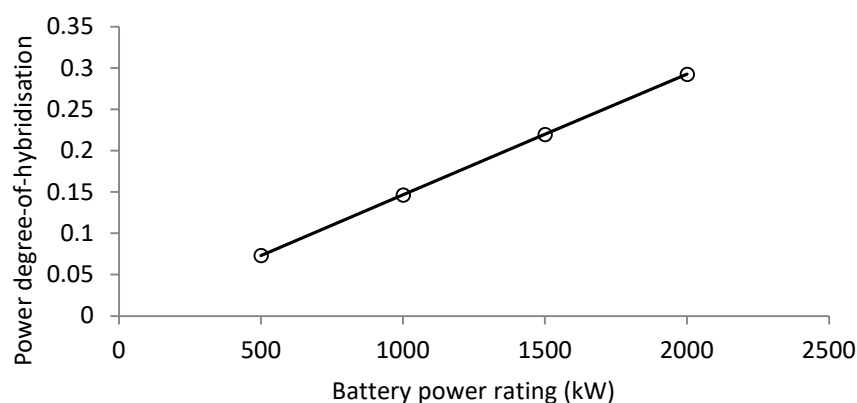


Figure 6-27 Power degree-of-hybridisation versus battery power rating

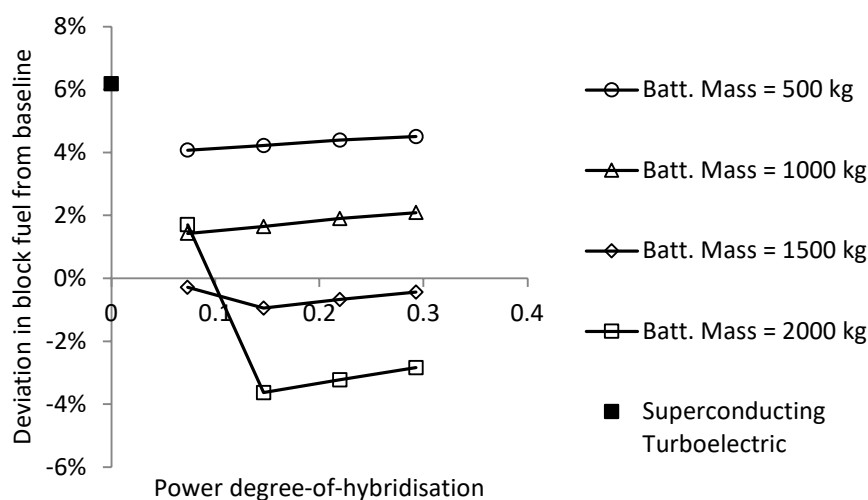


Figure 6-28 Deviation in block fuel from the baseline aircraft for hybrid electric aircraft configurations versus power hybridisation for a range of battery masses at the design range for 1500 Wh/kg batteries

Considering total energy consumption, hybrid electric aircraft configurations consume more energy than the baseline due to additional energy expended by the battery. This also means that for a given hybrid electric aircraft configuration the deviation in total energy consumption from the baseline configuration is going to be higher than the deviation in block fuel from the baseline configuration (see Figure 6-29). Additionally, the deadweight penalty at increased battery masses when battery is not fully discharged before the end of cruise at 500 kW battery power rating (0.073 power hybridisation) coupled with additional energy expenditure from battery, increases the difference between deviation in block fuel and deviation in total energy consumption even further at battery masses of 1500 kg and 2000 kg. This is due to extra fuel needed to carry heavier battery that is not fully utilising all of its available stored energy.

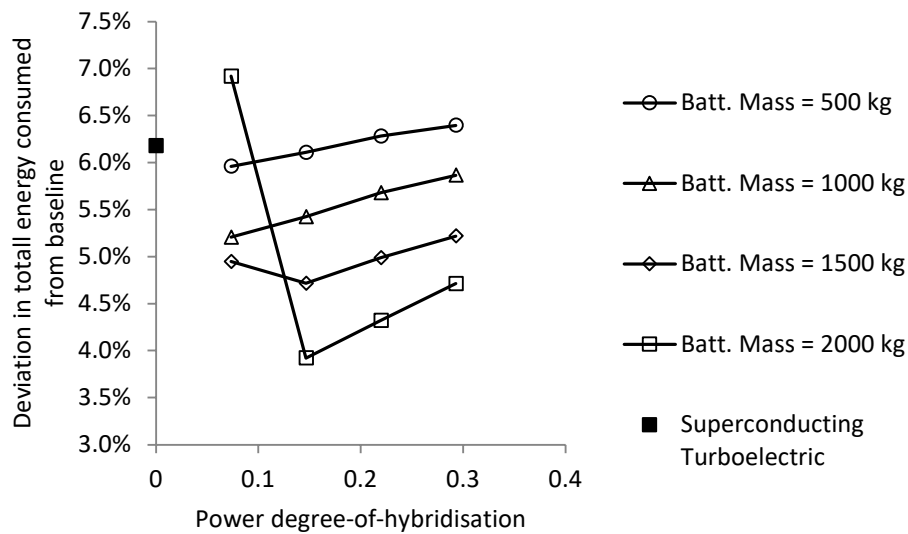


Figure 6-29 Deviation in main mission total energy consumed from baseline aircraft for hybrid electric aircraft configurations versus power hybridisation for a range of battery masses at the design range for 1500 Wh/kg batteries

6.6.4.5 Performance at Reduced Aircraft Range

600 n.mi

At 600 n.mi and batteries with specific energy of 1500 Wh/kg, for a battery that weighs 500 kg with a power rating of 500 kW, the deviation in block fuel from baseline for hybrid electric aircraft is lower than what is the case for the same battery set-up at the design range (see Figure 6-28 and Figure 6-30). With flight time shorter due to reduced range, the effects of a heavier aircraft make less of an impact on block fuel as aircraft structural fraction increases. Also, the reduced flight time means the ratio of time the battery spends discharging to overall cruise time increases from what the value of this ratio is at the design range (76% of cruise time at 600 n.mi compared to 36% of cruise time at 1,114 n.mi). Thus increased aircraft structural fraction and higher ratio of time the battery spends discharging to overall cruise time are the reasons for lower deviation in block fuel from baseline for hybrid electric aircraft with 500 kg/500 kW battery at 600 n.mi than corresponding value at the design range. As battery power increases for batteries that weigh 500 kg, block fuel increases slightly due to increased electric transmission mass (see Figure 6-30).

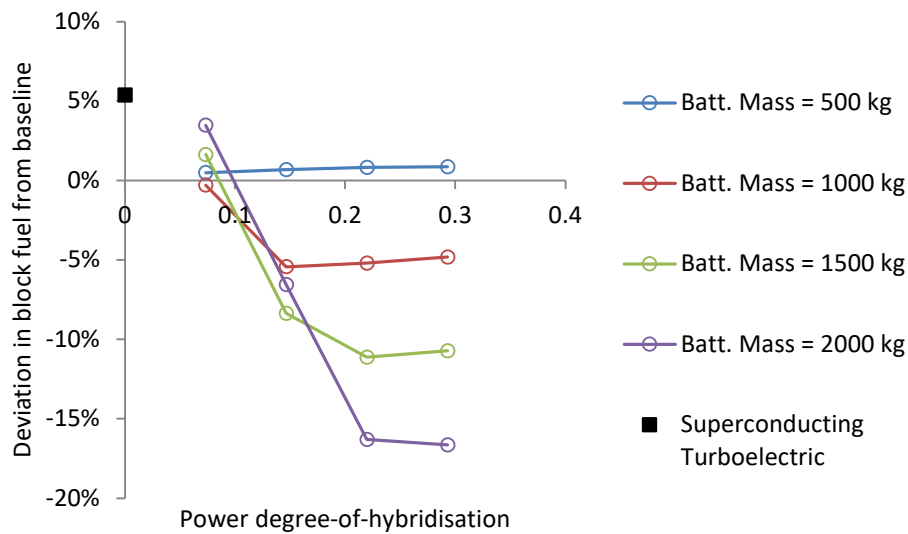


Figure 6-30 Deviation in block fuel from the baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 600 n.mi for 1500 Wh/kg batteries

For battery power rating of 500 kW, as battery mass increases, the amount of available usable energy in the battery increases and thus the ratio of time the battery spends discharging to overall cruise time is increased. This ratio increases to 100% of cruise time at battery masses of 1000 kg and higher. However at battery masses of 1000 kg and higher, due to increased amount of available stored energy in the battery, battery use during cruise is characterised by batteries not fully discharging stored available energy. As a result despite an initial slight reduction in block fuel as battery mass increases to 1000 kg, deadweight effect caused by batteries not fully discharging available energy increases block fuel as battery mass increases beyond 1000 kg. As battery power rating is increased for battery masses greater than 500 kg, more energy is consumed from batteries at lower GT fuel flow rates. Increased energy consumption from batteries coupled with lower GT fuel flow rate reduces block fuel as battery power rating increases at higher battery masses. Once batteries fully discharge at or before end-of-cruise, similar to effect at 500 kg battery mass, increasing battery power rating increases overall block fuel due to higher electric transmission mass.

Considering total energy consumption (see Figure 6-31), additional energy expended by batteries results in the deviation in total energy consumption from baseline configuration being higher than the deviation in block fuel from baseline. Also, similar to the case at the design range, deadweight penalty at battery masses greater than 500 kg when battery does not fully discharge stored available energy coupled with additional energy expenditure from the battery increases the difference between deviation in block fuel and total energy consumption as battery mass increases.

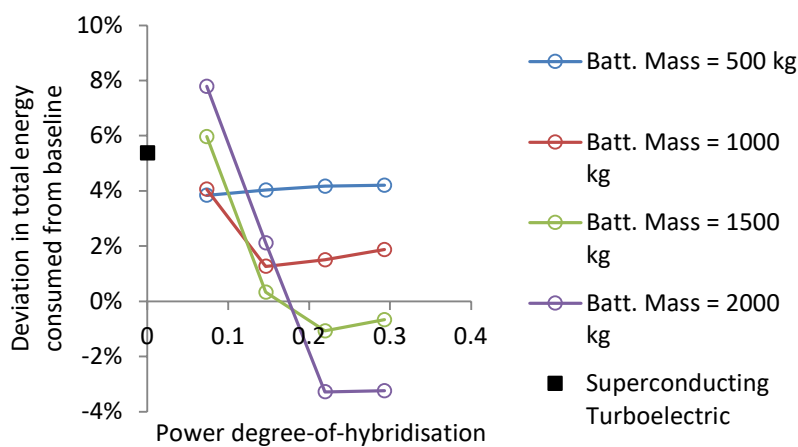


Figure 6-31 Deviation in main mission total energy consumed from baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 600 n.mi for 1500 Wh/kg batteries

200 n.mi

At 200 n.mi with operation of the aircraft in high speed cruise mode, overall flight time is shorter than what is the case at 600 n.mi. As such for hybrid electric aircraft with battery that weighs 500 kg and has a power rating of 500 kW the battery remains discharging throughout cruise. However even at 500 kg the amount of stored available energy in the battery exceeds the electrical energy consumed during cruise. As such, the deviation in block fuel from the baseline for hybrid electric aircraft configuration with battery that weighs 500 kg and has a power rating of 500 kW is higher than the deviation at 600 n.mi for the same hybrid electric configuration. Due to this, increasing the battery power rating reduces block fuel as more energy is consumed at lower GT fuel flow

rates (see Figure 6-32). For range of battery power ratings considered (from 500 kW to 2000 kW), block fuel continuously decreases as power rating increases because even at the higher power ratings considered, batteries do not fully discharge stored available energy. Also, as battery mass increases, block fuel increases due to battery deadweight penalty. In considering total energy consumption (see Figure 6-33), deviation in total energy consumption from baseline is higher than the deviation in block fuel from the baseline due to additional energy expended by the battery.

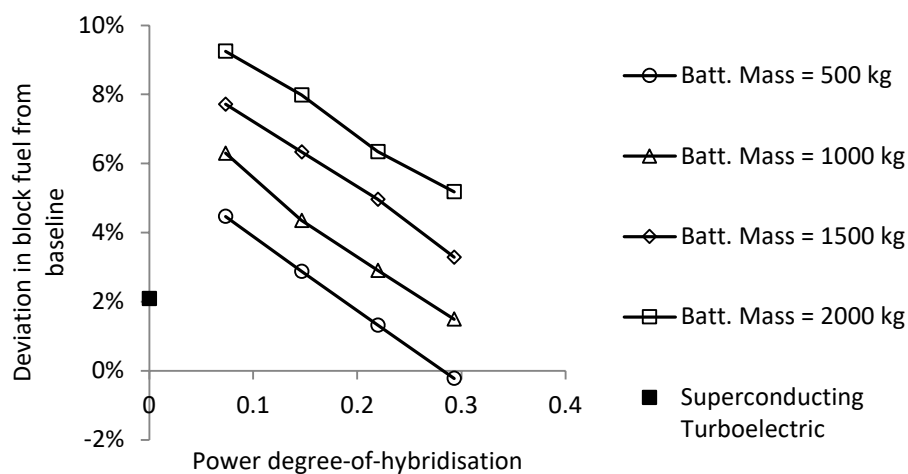


Figure 6-32 Deviation in block fuel from the baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 200 n.mi for 1500 Wh/kg batteries

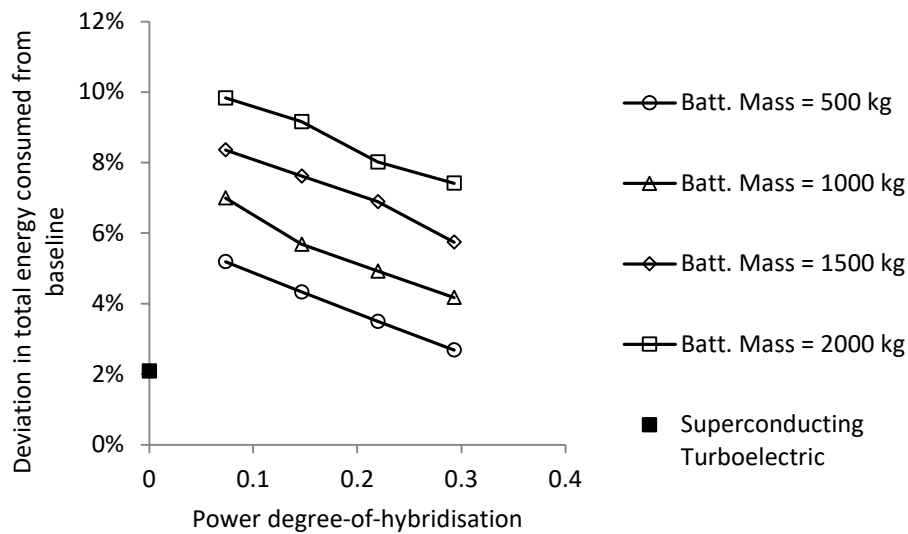


Figure 6-33 Deviation in main mission total energy consumed from baseline aircraft for hybrid electric aircraft configurations versus power hybridisation and battery mass at 200 n.mi for 1500 Wh/kg batteries

6.6.4.6 Configuration Summary

Top-level performance summary for minimum fuel hybrid electric aircraft configuration in long range cruise mode at the design range compared to baseline and superconducting turboelectric aircraft is shown in Table 6-23. The hybrid electric aircraft achieves 3.62% lower block fuel than the baseline configuration primarily due to extended periods of flight the propulsion system is operated at higher overall efficiency. However, total energy consumption is 3.84% higher than the baseline due to additional energy expenditure by the battery. Also heavier propulsion system and battery weight penalty result in hybrid electric aircraft having a higher MTOW, time-to-height and reserve fuel than baseline and superconducting turboelectric aircraft.

Table 6-23 Top-level performance summary for hybrid electric aircraft in long range cruise mode compared to baseline and superconducting turboelectric aircraft at the design range

Parameter	Baseline	Superconducting Turboelectric	Hybrid Turboelectric – minimum fuel configuration
Battery specific energy (Wh/kg)	N/A	N/A	1500
Battery mass (kg)	0	0	2000
Battery power rating (kW)	0	0	1000
Take-Off Field Length (m)	1364	1385	1568
Time-to-height (min)	16.98	17.06	19.76
Block fuel (kg)	2654	2818	2558
Reserve fuel (kg)	947	996	1031
Electrical energy consumed (GJ)	0	0	8.64
Main mission total energy consumed (GJ)	114.5	121.5	118.9
MTOM (kg)	29401	29881	31698

Top-level performance summary for minimum fuel hybrid electric aircraft configuration in long range cruise mode at 600 n.mi compared to the baseline and superconducting turboelectric aircraft is shown in Table 6-24. For hybrid electric aircraft, the minimum fuel configuration is achieved at a higher power rating than at the design range due to increased power rating needed to discharge all of stored available energy during cruise. Hybrid electric aircraft configuration achieves significantly lower block fuel (16.7% lower) than baseline primarily due to extended periods of flight the battery is discharging at a higher power rating than what is the case at the design range. This block fuel reduction more than offsets additional electric energy expenditure to an extent that is not achieved at the design range and results in an overall total energy consumption reduction of 3.23% from baseline aircraft energy consumption for hybrid electric aircraft. Additionally heavier propulsion system and battery weight penalty result in hybrid electric aircraft having a higher TOW, TOFL, time-to-height and reserve fuel than baseline and superconducting turboelectric aircraft. At 600 n.mi, for all configurations TOFL is lower than the associated value at the design range.

Table 6-24 Top-level performance summary for hybrid electric aircraft in long range cruise mode compared to baseline and superconducting turboelectric aircraft at 600 n.mi

Parameter	Baseline	Superconducting Turboelectric	Hybrid Turboelectric – minimum fuel configuration
Battery specific energy (Wh/kg)	N/A	N/A	1500
Battery mass (kg)	0	0	2000
Battery power rating (kW)	0	0	2000

Take-Off Field Length (m)	1252	1264	1431
Time-to-height (min)	15.47	15.43	17.70
Block fuel (kg)	1494	1574	1245
Reserve fuel (kg)	888	932	965
Electrical energy consumed (GJ)	0	0	8.64
Main mission total energy consumed (GJ)	64.43	67.90	62.35
TOM (kg)	28181	28574	30360

Top-level performance summary for minimum fuel configuration hybrid electric aircraft in high speed cruise mode at 200 n.mi compared to baseline and superconducting turboelectric aircraft is shown in Table 6-25. Despite the battery not fully discharging all of available stored energy and deadweight penalty as a result of this, hybrid electric aircraft achieves a slightly lower block fuel (0.32% lower) than baseline aircraft due to operation of the propulsion system at higher efficiency. However, total energy consumption is 2.68% greater than the baseline due to additional energy expenditure by the battery. Also heavier propulsion system and battery weight penalty result in hybrid electric aircraft having higher TOW, TOFL, time-to-height and reserve fuel than baseline and superconducting turboelectric aircraft. It should be noted that for the hybrid electric aircraft Also, at 200 n.mi, for all configurations TOFL is lower than the associated value at the design range.

Table 6-25 Top-level performance summary for hybrid electric aircraft in high speed cruise mode compared to baseline and superconducting turboelectric aircraft at 200 n.mi

Parameter	Baseline	Superconducting Turboelectric	Hybrid Turboelectric – minimum fuel configuration
Battery specific energy (Wh/kg)	N/A	N/A	1500
Battery mass (kg)	0	0	500
Battery power rating (kW)	0	0	2000
Take-Off Field Length (m)	1172	1178	1228
Time-to-height (min)	17.36	15.54	16.18
Block fuel (kg)	623	636	621
Reserve fuel (kg)	843	884	897
Electrical energy consumed (GJ)	0	0	0.78
Main mission total energy consumed (GJ)	26.85	27.41	27.57
TOM (kg)	27266	27587	28168

6.6.4.7 Comparison with Works from Public Literature

In considering how results from current studies compare to results from existing studies carried out on hybrid electric propulsion, a comparison is made in Table 6-26. In Table 6-26 results from the minimum fuel hybrid electric configuration simulated at the design range of the current study is compared with relevant configurations studied in public literature. The configurations from public literature chosen for comparison are all propeller-powered regional hybrid electric aircraft and this makes them suitable configurations for comparisons. From Table 6-26 all the selected configurations achieve significantly lower block fuel and energy consumption reductions over a baseline configuration. However, there are a few caveats. Firstly, in the studies considered in [33], [34] & [35]; the propulsion system architectures assessed are different from the propulsion system architecture assessed here in the present study. The propulsion system architectures in [33], [34] & [35] are all parallel hybrid or geared hybrid electric architectures (see Figure 4-1(e)). As such, the electric transmission system weight for all the propulsion systems from public literature works are considerably reduced compared to what is the case in the current study. This allows for lighter propulsion system and reduced weight penalties at the aircraft level. Secondly from sizing paradigm in the works in [33], [34] & [35]; the mass of the battery is sized so that batteries are continuously used during segments of the mission where hybrid operation is configured to be utilised. This feature, coupled with assessment at higher values of power hybridisation, means configurations in public literature achieve much higher block fuel reduction and total energy consumption reduction than works in present study. However, if battery mass and power rating was increased in present study, from performance trends in Figure 6-28 and Figure 6-29, greater reductions would be achieved. Additionally, even though DC hybrid turboelectric architecture use is characterised by greater electric transmission weight penalty, the capability of the architecture for propulsor distribution to take advantage of propeller-wing slipstream lift augmentation could result in a further reduction in block fuel and energy, reductions in wing size and a reduction in TOM.

Comparing the present study to works in [33] & [34]; resizing of the wing, empennage and gas turbine was carried out in these works and not in the present study. This is an effect that further reduces block fuel and total energy consumption in [33] & [34] beyond simply employing the hybrid electric architecture. Resizing of the aircraft and the gas turbine for present study at the design range will yield greater reductions in block fuel for hybrid electric aircraft considered in present study. Comparing the present study to works in [35], part of the reason for the dramatic reduction in block fuel over a baseline configuration in [35] is the overly ambitious technology level considered with batteries with a specific energy of 1500 W.h/kg assessed. This specific energy falls out even the most optimistic forecasts for specific energy of lithium-ion batteries. Also, studies in [33], [34] & [35] only carried out hybrid electric assessment at a design range and not for off-design ranges. In the present study, there is the potential for even greater reductions in block fuel and total energy consumed at reduced mission ranges due to battery weight having less of an impact on mission performance. At an off-design range of 600 n.mi, a 16.7% block fuel saving and 3.23% energy consumption reduction was obtained.

Table 6-26 Comparison of minimum fuel hybrid electric configuration simulated at design range with results from relevant propeller-powered regional hybrid electric aircraft simulations carried out in public literature

Parameter	Current Study	Antcliff et al [33]	Voskuijl et al [34]	Cinar et al [35]
Design number of passengers	74	48	70	31
Design Range (n.mi)	1114	600	825	1102
Cruise Mach	0.48	0.48	0.46	0.51

TOM (kg)	31698	22680	25470	13990
Propulsion system architecture	DC hybrid turboelectric	Parallel hybrid electric/geared hybrid electric	Parallel hybrid electric/geared hybrid electric	Parallel hybrid electric/geared hybrid electric
Battery type	Lithium-ion	Lithium-ion	Lithium-air	Lithium-ion
Battery specific energy (W.h/kg)	1500	1000	1000	5000
Battery Mass (kg)/Energy Hybridisation	2000/8.92%	6323/40.4% ^a	2948/14.3%	625.3/19.8%
Battery Power (kW)/Power Hybridisation	1000/14.6%	1343/75%	633.8/34.0%	1626/50%
Deviation in main mission block fuel from a baseline configuration or advanced conventional configuration	-3.62%	-64.80%	-28.3%	-53.8%
Deviation in main mission total energy	3.84%	-30.95%	-17.0%	-42.3%

consumption from a baseline configuration or advanced conventional configuration				
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^aUse of block fuel for calculation

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7 MEDIUM HAUL TRANSPORT

7.1 Mission Requirements

For the current research effort, the baseline for the medium haul aircraft has been specified to be an aircraft similar to the Airbus A320-200 variant with IAE V2527E-A5 engines. With this baseline specified, mission requirements for TeDP and hybrid electric configurations are taken from the mission specifications obtained from public literature for the airbus A320-200. With the medium haul aircraft assumed to operate in ISA conditions, mission requirements for the medium haul aircraft are shown in Table 7-1. Similar to reasons explained for the regional mission, a present day state-of-the-art configuration is chosen for mission baseline configuration rather than the selection of an advanced tube-and-wing with advanced GTs.

Table 7-1 General mission requirements for medium haul aircraft

Design Range (n.mi)	2,546 ^a [1]
Maximum payload range at MTOW (km)	4,000 [1]
Maximum fuel range at MTOW (km)	5,333 [1]
Design number of passengers	150 [2]
Maximum number of passengers	180 [3]
Design cargo (kg)	3,500 ^b [1]
Initial cruise altitude (ft)	37,000 [2]
Maximum Altitude (ft)	39,800 [2]
Cruise Speed	M0.78 [2]
Climb CAS (kts)	250/300 [4]
Landing field length (m)	1,491[2]

^aPoint taken from A320-200 payload-range diagram in [1]

^bCalculated from A320-200 payload-range diagram and assuming mass per passenger is 95 kg and design range is 4715 km

For stepped cruise consideration, modelled aircraft are specified to cruise at three altitudes. The first altitude corresponds to the initial cruise specified in Table 7-1 and the third altitude is specified to be the maximum altitude specified in Table 7-1. The second altitude step is specified to be intermediate between initial cruise altitude and maximum altitude at 38,400 feet.

For the reserve mission, based on values from public literature requirements are listed in Table 7-2.

Table 7-2 Reserve mission requirements

Diversion range (n.mi)	200 [2]
Diversion cruise altitude (ft)	25000 ^a [5]
Diversion cruise speed	M0.6 ^b
Hold time (min)	45 [6]
Hold altitude (m)	457

^aIn absence of more data on this altitude from public literature, a typical height of 25,000 feet is chosen for diversion cruise altitude

^bDiversion is usually flown using a 'best-range' technique. A typical cruise speed of Mach 0.6 is nominally set

For assessment of take-off performance, the same approach used for the regional aircraft in determining end-of-runway Mach number is used here. With end-of-runway speed set equal to 1.2 multiplied by wing clean stall speed [7], aircraft wing loading at take-off for the A320-200 equal to 200 and clean wing maximum lift coefficient set equal to 1.5 [8], end-of-runway Mach number is estimated to be 0.30. For descent performance assessment, similar to the regional mission engines are set to operate at idle conditions and idle thrust and SFC simulated for the baseline aircraft are utilises as idle thrust and SFC for all integrated airframe-propulsion configurations considered.

In addition to aircraft design range, two additional aircraft ranges are considered for performance assessment at the aircraft level. The first range is 900 n.mi, an aircraft range which according to current market outlook for the year 2030 [9] represents the average distance a medium haul aircraft will fly in the year 2030. The second additional range is 1700 n.mi which represents an intermediate mission between the design range and average mission range.

7.2 Baseline Propulsion System

The Airbus A320-200 is equipped with a Garrett GTCP 36-300 APU and can be equipped with a number of different turbofan models [3]. Of interest here is the A320-233 configuration that is equipped with two IAE V2527E-A5 engines.

- For performance assessment here, turbofans similar to the IAE V2527E-A5 are simulated.
- APU is considered for mass estimation purposes. Weight of the APU similar to the Garrett GTCP 36-300 APU is 149.7 kg [10]. To account for generator attached to APU, total APU mass is estimated to be dry APU mass multiplied by two to account for 90 kVA generator coupled to APU.

GT configuration is defined from information in [11] for the IAE V2527E-A5 engine below

- Two spools, axial flow, high bypass ratio turbofan engine
- Single stage fan
- 4 stage low pressure compressor
- 10 stage high pressure compressor
- Annular combustion chamber
- 2 stage high pressure turbine
- 5 stage low pressure turbine
- Air bleed from HPC

It should be noted that from information in [11] the IAE V2527E-A5 is defined to have a mixed nozzle, however for faster computations, the baseline GT is

configured to have bypass and core nozzles be separate. With separate nozzles an advantage is reduced installation penalties. However, with mixed nozzles advantages include higher propulsive efficiency, higher aerodynamic efficiency and reduced noise.

In Turbomatch 2.1 based on provided information for the GT configuration from public literature, the turbofan is configured as shown in Figure 7-1. For this GT relevant characteristics of the GT are listed in Table 7-3.

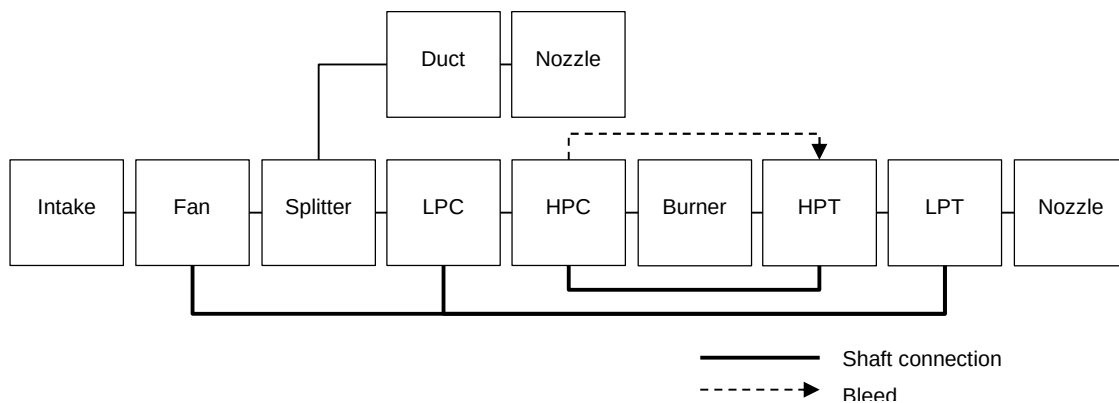


Figure 7-1 Medium Haul Turbofan Configuration

Table 7-3 Characteristics of medium haul GT obtained from public literature

Length	3.201 m [11]
Width	1.682 m [11]
Mass	2404 kg [11]
HPC bleed	6% [11]
Fuel	Kerosene [3]

With design point for the engine not specified from public literature obtained, design point for the turbofan is specified for initial cruise because it is expected that the aircraft will spend the greatest proportion of its flight time at cruise. With design point set, based on typical values, a selection of GT performance parameters are set in Table 7-4.

Table 7-4 Medium haul aircraft gas turbine parameters set based on typical values

Duct pressure loss	1%
Burner pressure loss	4% [12]
Combustion efficiency	0.999 [12]
Fuel lower heating value (MJ/kg)	43.12 [12]

In simulating design point and off-design performance of the GT; the mass flow rate, compressor PRs and component isentropic efficiencies still need to be defined in addition to GT design parameters already specified. To tackle this; design point mass flow rate, design point BPR, design point FPR, design point compressor PRs, design point component isentropic efficiencies, design point LPT polytropic efficiency, design point TET and off-design TETs are iterated within suitable ranges to match GT performance with available data from public literature on the performance of the IAE V2527E-A5 engine at given operating points. Using Turbomatch, a summary of the comparison of the simulated engine with the actual engine at selected off-design points for which performance data is available is shown in Table 7-5. The final values of the GT performance parameters that were iterated to determine engine performance in Table 7-5 are shown in Table 7-6.

Table 7-5 Comparison of simulated engine for medium haul transport mission with values obtained from public literature of engine for with similar architecture

	Literature	Turbomatch Simulation	Variation between simulated and quoted values (%)
<i>10,670 m, M = 0.8</i>			
Thrust (N)	25600 [13]	25449	-0.59 %

SFC (mg/Ns)	16.26 [13]	16.31	0.31 %
<i>Static maximum take-off rating at sea-level without bleed</i>			
Thrust (N)	110310 [11]	109364	-0.86 %
Exhaust Gas Temperature (K) ^a	918 [11]	926.0	0.87 %
FPR ^b	1.6 [14]	1.61	0.63 %
OPR	27.2 [14]	27.37	0.63 %
BPR	4.8 [14]	4.76	-0.83 %
<i>Static maximum continuous rating at sea-level without bleed</i>			
Thrust (N)	98920. [11]	97493	-1.44%
Exhaust Gas Temperature (K) ^a	883 [11]	892	1.02 %

^aEGT measured aft of LPT first stage

^bAssumed to be the same for hub and tip

Table 7-6 Values of iterated GT performance parameters used to simulate medium haul GT performance. Design point flight condition is M0.78/37,000 ft.

Design point FPR	1.83
Design point fan isentropic efficiency	0.89
Design point BPR	4.28
Design point LPC PR	2.84
Design point LPC isentropic efficiency	0.87
Design point HPC PR	6.95
Design point HPC isentropic efficiency	0.86

Design point HPT isentropic efficiency	0.90
Design point LPT polytropic efficiency	0.89
Design Point TET	1250
Maximum take-off rating TET	1320
Maximum continuous rating TET	1275

An overview of simulated propulsion system performance at selected flight conditions in the flight envelope is shown in Figure 7-2. In Figure 7-2 the propulsion system exhibits typical turbofan performance with reduced momentum drag primarily responsible for lower TSFC at lower altitude and speed flight conditions. Plus captured in Figure 7-2 is the trend where TSFC reduces as altitude increases for flight conditions where cruise Mach number is reached (i.e. from 9144 m to 11,280 m) primarily due to increasing thermal efficiency.

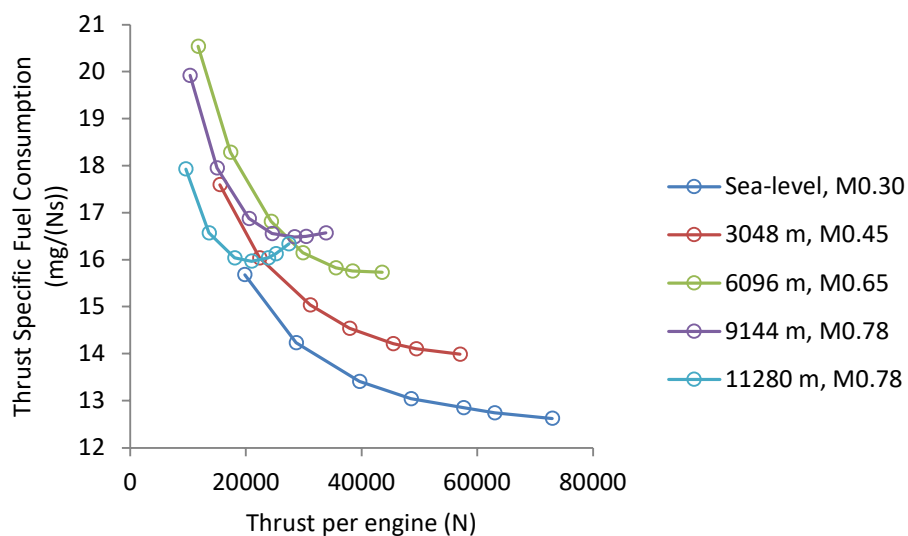


Figure 7-2 Medium haul baseline propulsion system performance at selected flight conditions

7.3 Baseline Aircraft

In addition to defined mission requirements and simulated propulsion system, design parameters are required to be set for sizing and performance assessment of baseline aircraft. Aircraft design parameters and characteristics are extracted from data available from public literature for Airbus A320-200 and are listed in Table 7-7.

Table 7-7 Medium haul aircraft design parameters and characteristics

Cabin pressure differential	0.563 bar [15]
Dihedral ^a	0°
Empennage material	Carbon-fibre-reinforced-plastic [2]
Empennage moment arm	0.36 [16]
Empennage type	Conventional tail [2]
Flaps type	Fowler [2]
Fraction of area of wing covered by slats	0.103 [2]
Fraction of trailing edge area of wing covered by control surfaces and high lift devices	0.172 [2]
Fuselage height	4.14 m [2]
Fuselage length	37.57 m [2]
Fuselage material	Aluminium alloy [2]
Fuselage width	3.95 m [2]
Horizontal tail aspect ratio	5 [16]
Horizontal tail average thickness-to-chord ratio	0.136 ^b

Horizontal tail quarter chord sweep angle	29° [16]
Horizontal tail taper ratio	0.256 [16]
Horizontal tail volume coefficient	0.799 [16]
Increment in take-off lift coefficient due to high lift devices	1.06 [16]
Mass per passenger	95 kg
Number of crew members	6 [2]
Relative spanwise distance from aircraft centre to engine centre	0.32 [1]
Ultimate load factor	5 ^c [7]
Vertical tail aspect ratio	1.82 [16]
Vertical tail average thickness-to-chord ratio	0.136 ^b
Vertical tail quarter chord sweep angle	34° [16]
Vertical tail taper ratio	0.303 [16]
Vertical tail volume coefficient	0.065 [16]
Wing average thickness-to-chord ratio	0.136 ^d [17]
Wing aspect ratio	10.47 ^e [2]
Wing inner section taper ratio	0.36 [1]
Wing kink relative span position	0.5 [1]
Wing quarter chord sweep angle	25° [2]
Wing taper ratio	0.24 [16]

Wing taper type	Double tapered [1]
Wing tips present?	Yes [2]

^aConsideration of dihedral is beyond the scope of current studies

^bData not available so setting thickness-to-chord ratio to be equal to wing thickness-to-chord ratio

^cLimit load factor multiplied by a safety factor of 1.5

^dEstimated from values in [17]

^eAspect ratio with winglets

From aircraft mission requirements set, aircraft design parameters set, aircraft design characteristics, simulated propulsion system and setting the turbofan to operate at maximum continuous rating during climb, the baseline aircraft was simulated in CADI. A summary of top-level simulation results from the simulation compared to the similar aircraft is shown in Table 7-8. From the simulation with the exception of take-off field length, top-level outputs of the simulated aircraft deviated from the A320-200 by less than 5%. Also, considering payload-range (see Figure 7-3), points on the payload-range diagram deviate from values for the A320-200 aircraft by less than 2%. Utilising geometry outputs from CADI, the baseline aircraft is visualised using OpenVSP in Figure 7-4.

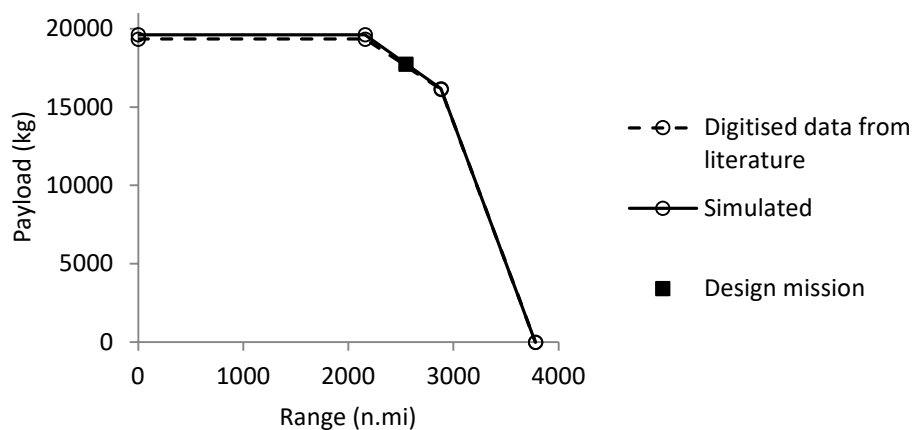


Figure 7-3 Payload-range diagram for baseline medium haul aircraft; digitised data obtained from [1]

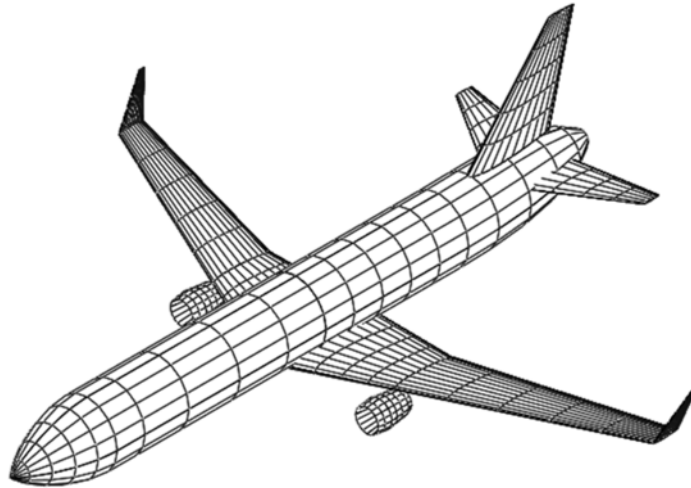


Figure 7-4 Geometry visualisation of baseline medium haul aircraft using OpenVSP

Table 7-8 Medium haul baseline aircraft simulation results compared with values obtained for similar aircraft from public literature at the design mission

	Simulated Aircraft	Literature	Deviation (%)
<i>Performance</i>			
MTOM (kg)	79451	78000 [3]	1.86
OEM (kg)	43896	42200 [2]	4.02
Summation of block fuel and reserve fuel (kg)	17805	18050 ^a [1]	-1.36
TOFL (m)	1994	1890 [2]	5.45
<i>Geometry</i>			
Wing area (m ²)	124.8	122.4 [2]	1.96
Horizontal tail area (m ²)	29.8	31.0 [2]	-3.87

Vertical tail area (m ²)	21.7	21.5 [2]	0.93
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^aDerived from definition of MTOM being equal to summation of OEM, block fuel, payload and reserve fuel

In considering the effect of aircraft range, reducing range lowers top-level performance parameters due to lower fuel burn (see Table 7-9). Performance at reduced aircraft ranges is considered because even though the aircraft is designed to fly a range of 2,546 n.mi with a total payload of 17,750 kg, medium haul aircraft typically fly multiple routes with ranges flown often less than stated design range.

Table 7-9 Aircraft top-level performance summary at different aircraft ranges for medium haul aircraft

Aircraft Range	2546 n.mi	1700 n.mi	900 n.mi
Block fuel (kg)	14030	9396	5277
Reserve fuel (kg)	3775	3538	3327
Time-to-height (min)	27.4	23.8	21.2
TOFL (m)	1994	1776	1598
TOW (kg)	79451	74579	70250

A breakdown of non-fuel aircraft masses for the baseline aircraft is shown in Table 7-10.

Table 7-10 Non fuel aircraft masses for baseline medium haul aircraft

Empennage	1264 kg
Equipment	15095 kg

Fuselage	7447 kg
Operational Items	2670 kg
Design Payload	17750 kg
Powerplant	7967 kg
Wing	9453 kg

7.4 Assessing the use of Hydrogen as a Fuel

7.4.1 Cryotank Performance

Hydrogen is considered as an aircraft fuel due to its attractive properties of zero CO₂ emissions and a gravimetric energy density that is about three times greater than kerosene [18], [19] & [20]. However as explained before in Sec. 5.6, hydrogen is stored in its liquid form in composite cryotanks to minimise volumetric penalties that arise through utilising it. With hydrogen utilised as a fuel; in addition to fuel density, tank ullage and clean tank gravimetric efficiency specified in Sec. 5.6.2.1, number of insulation layers, reference flight time and fuel amount are needed for the determination of fuel boil-off and insulated tank gravimetric efficiency. Utilising baseline aircraft block fuel and flight time as reference values, in Figure 7-5 insulated tank gravimetric efficiency reduces as the number of LRMLI layers increases due to increased insulation mass for a fixed tank volume. However, as the number of LRMLI layers increases, boil-off from tank reduces at a near exponential rate due to reduced heat leak from the cryotank (see Figure 7-6).

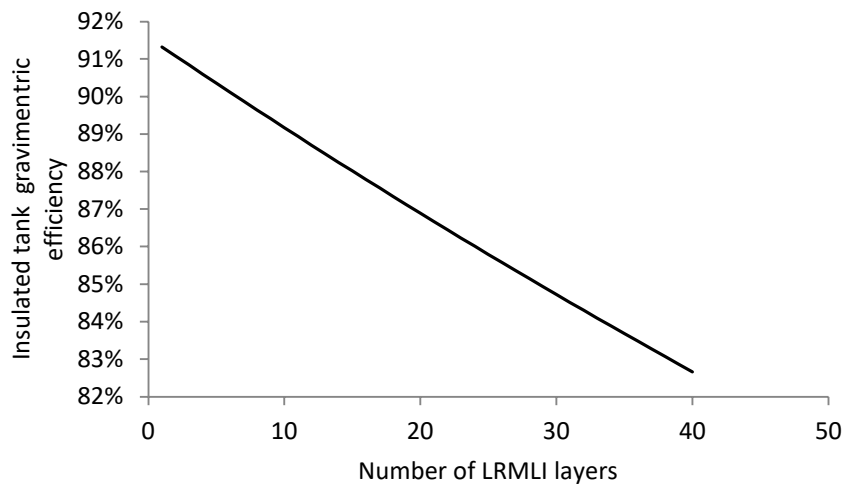


Figure 7-5 Insulated tank gravimetric efficiency versus number of LRMLI layers

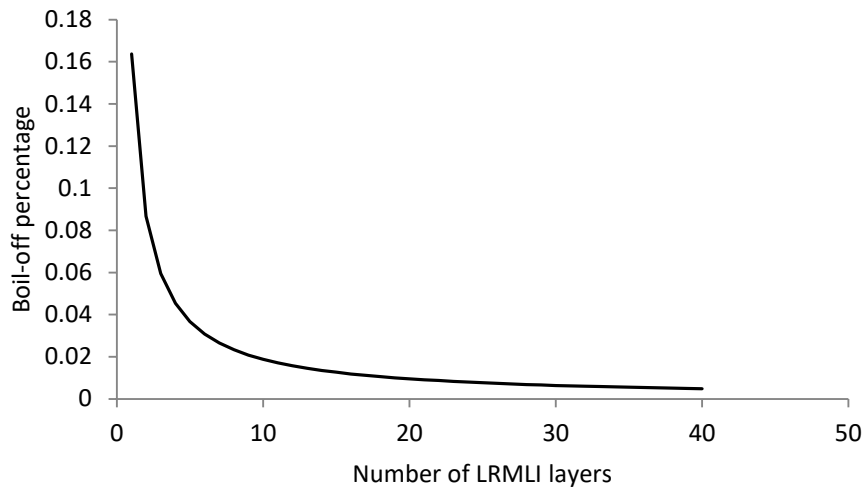


Figure 7-6 Boil-off from cryotank versus number of LRMLI layers

In considering the number of LRMLI layers to be utilised with the cryotank, one insulation layer could provide an acceptable amount of boil-off (approx. 0.164%). However practically ice build-up on tank surface due to boiled-off cold gas and redundancy reasons in the event that a layer of insulation layer is torn off (one layer of MLI is about only 0.5 mm thick and tensile strength is not high) will mean more than one layer is needed. Thus 10 LRMLI layers were nominally chosen for aircraft design for redundancy and to mitigate possible ice-build up on tank surfaces. For 10 LRMLI layers the boil-off due to vented gas is approximately 0.0188% and insulated tank gravimetric efficiency is 89.17%.

7.4.2 Propulsion System Performance

An overview of turbofan performance using hydrogen fuel simulated using Turbomatch at selected operated conditions compared to the kerosene baseline is shown in Figure 7-7. Keeping the same thermodynamic cycle (i.e. inlet air mass flow rate, OPR, TET & BPR are kept constant) as the baseline turbofan, the hydrogen turbofan achieves a lower TSFC than the kerosene baseline due to hydrogen having a higher LHV than kerosene (118.8 MJ/kg for hydrogen [20] as opposed to 43.12 MJ/kg for kerosene). Also, for the same throttle and flight condition the hydrogen turbofan produces slightly more thrust than the kerosene turbofan despite the same cycle parameters. This primarily because of the higher isobaric specific heat capacity and ratio of specific heats of the combustion products of hydrogen compared to what is the case for kerosene combustion. Considering emissions, although not assessed if only the thermodynamic cycle was considered, it would be expected that both the hydrogen turbofan and baseline turbofan will produced the same amount of NO_x since TET is the same. However, with hydrogen being typically injected into a combustion chamber in its gaseous form, its combustion is expected to be more uniform in the combustion chamber than for kerosene which is a liquid fuel. Due to this the combustion of kerosene is expected to produce more local hotspots in the combustion chamber which is likely to have the effect of more NO_x being produced by the baseline turbofan when compared to the hydrogen turbofan. Additionally for present studies, mass and size of the GT are assumed to remain constant as fuel changes from kerosene to hydrogen as an emphasis is centred on understanding physical system relationships. For more detailed studies, changes in GT size and mass are expected to be significant due to a likely big change in aircraft thrust requirements if aircraft resizing is carried out.

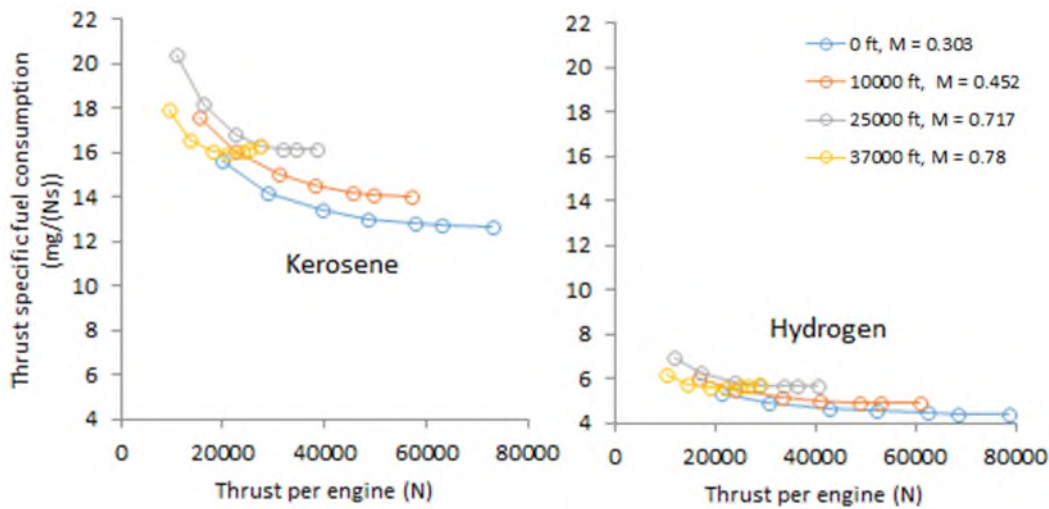
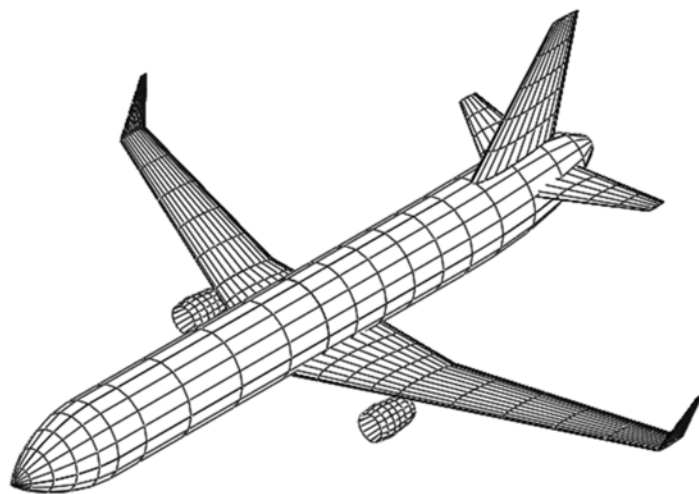


Figure 7-7 Comparison of hydrogen turbofan performance to kerosene baseline for medium haul aircraft at selected flight conditions

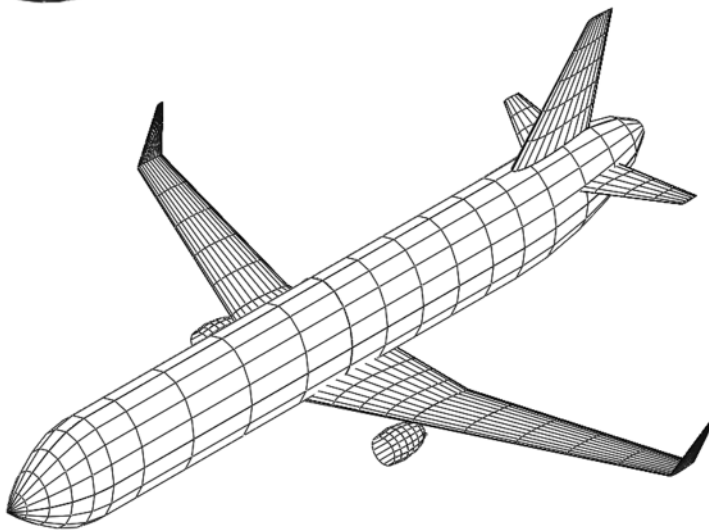
7.4.3 Hydrogen Turbofan Powered Aircraft Sizing and Performance Assessment

For assessment of hydrogen use at the aircraft level, the focus is on assessing how the propulsion system affects overall flight performance. With detailed analysis of aircraft aerodynamics, control and stability assessment beyond the scope of current work, sizing scenario for the aircraft utilised is for hydrogen configurations to have the same wing area and design range as the baseline aircraft. With this scenario, for fixed wing area aircraft stall speed changes as TOW changes. In considering the storage of hydrogen fuel in the fuselage, arbitrary cryogenic fuselage structural factors of 1.1 and 1.2 are set. Also, similar to the baseline aircraft, hydrogen turbofan powered aircraft's turbofans during climb are set to operate at maximum continuous rating.

A comparison between simulated hydrogen aircraft with cryogenic fuselage structural factor of 1.2 and the baseline aircraft is shown in Figure 7-8. For the hydrogen aircraft in Figure 7-8, its length and height are greater than the length and height of the baseline aircraft due to the storage of fuel within the fuselage. Additionally, increased fuselage length increases tail moment arm and this allows for a slightly smaller empennage.



Baseline



Hydrogen aircraft with
cryogenic fuselage
structural factor of 1.2

Figure 7-8 Aircraft geometry comparison between baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2 using OpenVSP

For both aircraft shown in Figure 7-8, due to both aircraft having the same wing area wing profile drag is the same when aircraft are flying at the same altitude. With hydrogen aircraft only flying at 38,400 ft throughout cruise, baseline aircraft spends time from initial cruise to a point just after mid-cruise flying at 37,000 ft. This results in baseline aircraft flying with a lower wing profile drag up to the point after mid-cruise where it is more efficient for the baseline to fly at the higher altitude of 38,400 ft. This means that for most of cruise wing profile drag of the hydrogen aircraft is higher than wing profile drag of the baseline configuration. In considering the profile drag of other airframe components, reduced empennage size for hydrogen aircraft results in hydrogen aircraft having a slightly lower overall empennage profile drag than the baseline aircraft.

With lift dependent drags (wave and induced) for hydrogen aircraft and baseline nearly matching each other over cruise due to equal wing areas, effect of increased fuselage profile drag due to increased fuselage size and higher wing profile drag dominates over reduced empennage drag and results in overall aerodynamic efficiency for hydrogen aircraft being lower than baseline aerodynamic efficiency (see Figure 7-9). For both aircraft as fuel is burned aircraft weight reduces. However, with aircraft lift coefficient lower than the lift coefficient for maximum aerodynamic efficiency just after initial cruise, aircraft weight falls at a faster rate than drag as the mission progresses and aerodynamic efficiency reduces.

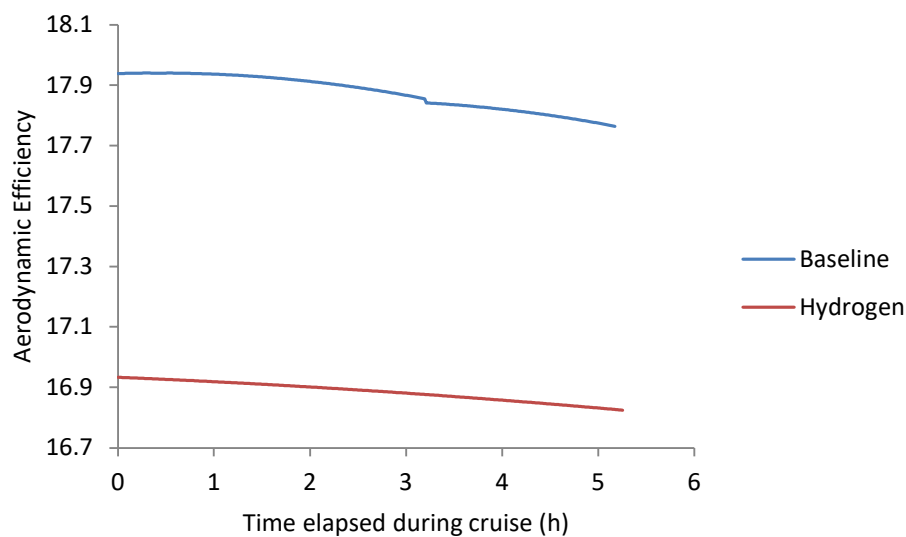


Figure 7-9 Aerodynamic efficiency versus cruise time elapsed for baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2

Considering mission performance at cruise further, at the start of cruise overall thrust requirement for the hydrogen aircraft's propulsion system is lower than the overall thrust requirement for the baseline aircraft due to lower overall aircraft weight with fuel on-board (see Figure 7-10). The reason for this is that despite the presence of cryotanks increasing vehicle OEW, the hydrogen aircraft burns less fuel than the baseline due to reduced TSFC throughout the flight envelope. With thrust requirement proportional to aircraft weight but inversely proportional to aerodynamic efficiency, as the mission progresses for

both aircraft overall thrust requirement falls due to fuel burn. However, because thrust requirement is also inversely proportional to aerodynamic efficiency, thrust requirements and fuel on board for the hydrogen aircraft falls at a slower rate than the baseline aircraft as cruise progresses. This eventually results in higher thrust requirements for the hydrogen aircraft from after approximately 1 hour and 36 minutes of cruise to end-of-cruise.

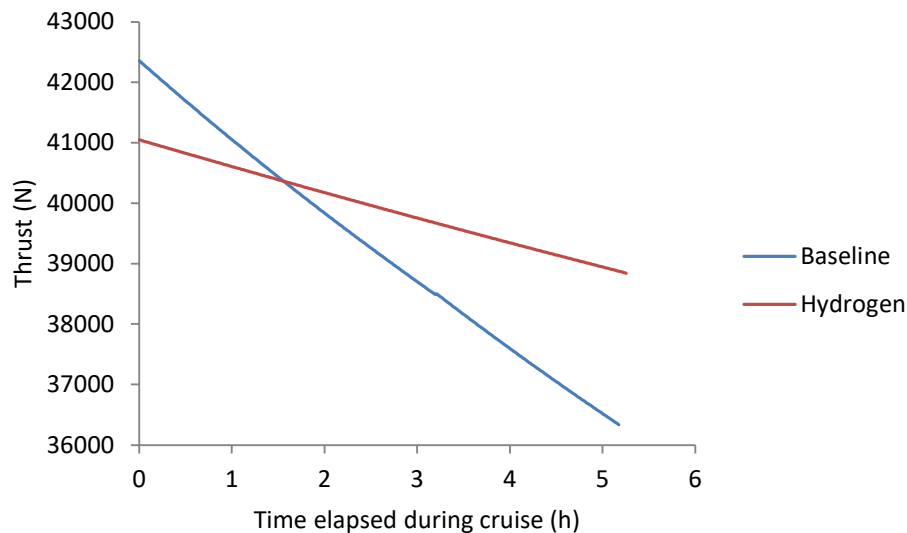


Figure 7-10 Overall thrust requirement versus cruise time elapsed for baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2

In assessing the effect on the propulsion system, TET for both the baseline aircraft and hydrogen aircraft falls as cruise progresses due to thrust requirement drop. For the hydrogen aircraft GT TET is lower at initial cruise for the hydrogen aircraft due to lower aircraft weight (see Figure 7-11). However, TET for the hydrogen aircraft falls at a lower rate due to lower vehicle aerodynamic efficiency. Lower rate of TET drop with cruise time elapsed combined with flight at a higher altitude results in TET for the hydrogen turbofan to be higher than TET for the kerosene turbofan from after approximately 37 minutes of cruise till about 3 hours and 13 minutes of cruise. At the point in cruise that corresponds to 3 hours and 13 minutes elapsed, the baseline aircraft's climb to the same altitude the hydrogen aircraft is flying at results in the GT for the hydrogen aircraft once again having a lower TET than the GT for

the baseline. TET increases for the baseline aircraft as it climbs to a higher altitude due to need for GT to increase throttle as density falls to meet thrust requirement. Then after 3 hours and 13 minutes of cruise the hydrogen aircraft has a lower TET than baseline configuration until about 4 hours and 12 minutes of cruise has passed. After this point the lower rate of TET fall of the hydrogen aircraft results in higher TET for the hydrogen aircraft from about 4 hours and 12 minutes of cruise elapsed to end-of-cruise.

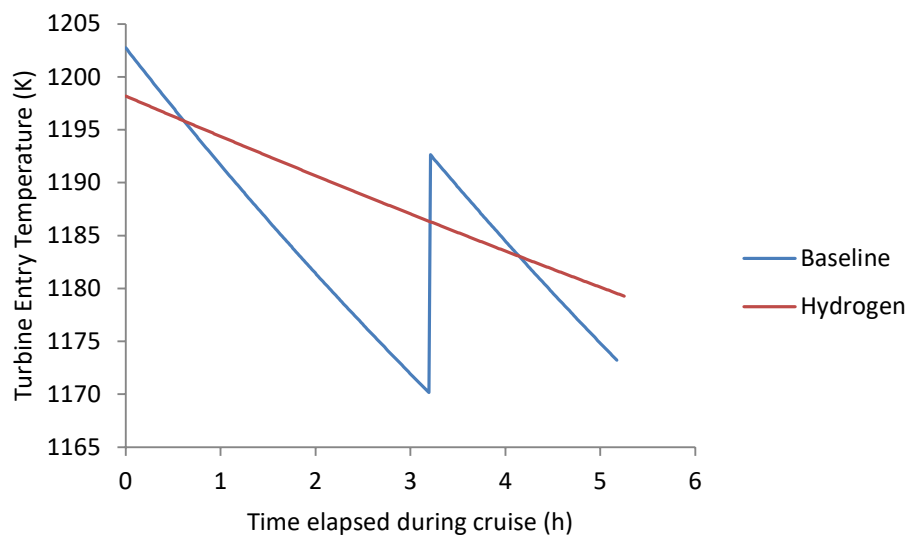


Figure 7-11 TET versus cruise time elapsed for baseline aircraft and hydrogen aircraft with cryogenic fuselage structural factor of 1.2

In Table 7-11 the mass breakdown of hydrogen aircraft for different fuselage cryogenic structural factors at the design range is shown. In Table 7-11 despite the presence of cryotanks the hydrogen aircraft burn less fuel than the baseline aircraft both during the main mission and reserve mission primarily due to reduced TSFC. Considering the mass of other aircraft items, storage of fuel in fuselage increases fuselage size which results in fuselage for hydrogen aircraft being heavier than the baseline aircraft's fuselage. With increased fuselage size increasing tail moment arm, empennage size is reduced for hydrogen aircraft and the result is in empennage of hydrogen aircraft being lighter than empennage of the baseline aircraft. Also, a reduced MTOM results in lighter wings for the hydrogen aircraft in comparison to the baseline. It should be noted

that in Table 7-11 the mass of the hydrogen cryotanks is relatively low due to the high tank gravimetric efficiency as a result of using composite tanks.

Table 7-11 Mass breakdown of hydrogen aircraft compared to baseline aircraft at the design range

	Baseline	Hydrogen – 1.1× structural factor	Hydrogen – 1.2× structural factor
MTOM (kg)	79451	69891	71452
Block fuel (kg)	14030	5071	5173
Reserve fuel (kg)	3775	1491	1511
Cryotanks (kg)	0	775	789
Empennage (kg)	1264	1126	1123
Equipment (kg)	15095	13278	13575
Fuselage (kg)	7447	10550	11545
Operational Items (kg)	2670	2670	2670
Design Payload (kg)	17750	17750	17750
Powerplant (kg)	7967	7967	7967
Wing (kg)	9453	9212	9349

In Table 7-12 top-level sizing and performance summary of simulated hydrogen aircraft compared to the baseline aircraft is shown. For the hydrogen aircraft in addition to lower block fuel, smaller empennage and larger fuselage in comparison with baseline aircraft; lower MTOM results in hydrogen aircraft having lower reserve fuel, TOFL and time-to-height than the baseline.

Considering the effect of structural penalties, a higher structural penalty on the fuselage is characterised by increased MTOM, reduced empennage size, increased fuselage size, increased TOFL, increased time-to-height, increased block fuel and increased reserve fuel. When main mission energy consumption is considered, increased fuselage structural penalty increases overall energy consumption. However if fuselage structure can be designed to be light enough (represented by reduced fuselage structural factor), a total energy consumption of 0.43% less than the baseline aircraft can be achieved.

Table 7-12 Top-level sizing and performance summary of hydrogen aircraft compared to baseline aircraft at the design range

	Baseline	Hydrogen – 1.1× structural factor	Hydrogen – 1.2× structural factor
MTOM (kg)	79451	69891	71452
Wing area (m ²)	124.8	124.8	124.8
Horizontal tail area (m ²)	29.82	27.07	27.00
Vertical tail area (m ²)	21.66	19.66	19.61
Fuselage height (m)	4.14	5.65	5.67
Fuselage width (m)	3.95	3.95	3.95
Fuselage length (m)	37.60	41.42	41.52
TOFL (m)	1994	1499	1582
Time-to-height (min)	27.41	20.45	21.76

Block fuel (kg)	14030	5071	5173
Reserve fuel (kg)	3775	1491	1511
Main mission energy consumption (GJ)	605.0	602.4	614.6

At the design range (2546 n.mi) the fuselage structural factor affects whether there is a reduction or increase in energy consumption for hydrogen aircraft over kerosene baseline. From Figure 7-12 for a structural penalty of 1.1, there is a slight reduction in main mission energy consumption (approximately 0.43%). However if the structural penalty is increased to 1.2 potential energy savings are eliminated. Considering the effect of a reducing aircraft range, as range reduces cryotanks on board increasingly act as deadweight. This is because flight time is not long enough for use of hydrogen fuel to make an overall impact on mission performance and cryotanks sized for the design mission are oversized for operation at reduced aircraft range. The impact for hydrogen aircraft is larger than what would be the case for a kerosene-powered aircraft due to the greater aircraft weight fraction of cryotanks compared to kerosene tanks. At reduced aircraft range this increases the deviation in total energy consumption for hydrogen aircraft from baseline aircraft. At all aircraft ranges increased fuselage structural penalty causes an increase in overall aircraft energy consumption.

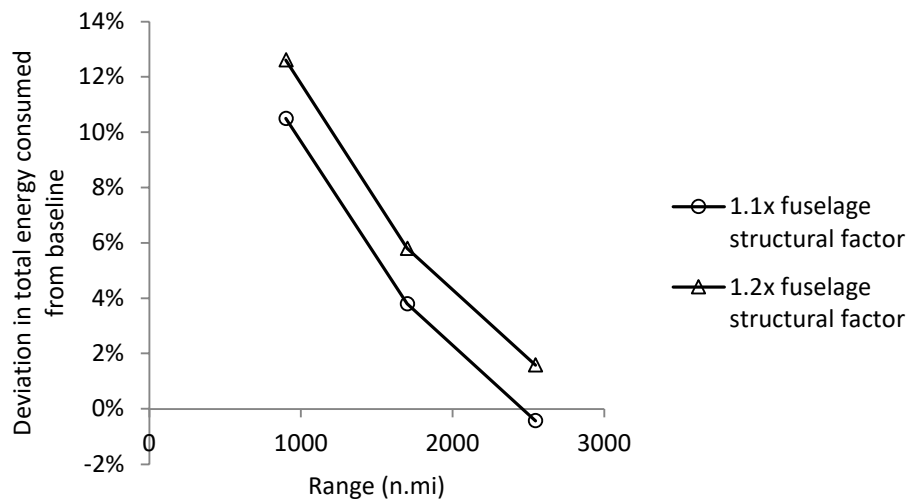


Figure 7-12 Deviation in main mission total energy consumption from baseline for hydrogen aircraft at different fuselage structural penalties

7.4.4 Hybridising the Powerplant

With utilisation of liquid hydrogen as fuel, superconducting turboelectric architecture can be employed with liquid hydrogen cooling superconducting machines and cryogenic power controllers. Additionally the employment of turboelectric architecture allows for the employment of hybrid electric operation for a potential fuel burn reduction. Considering the hybrid turboelectric architecture in Figure 4-1(d), with an emphasis on block fuel burn and energy consumption reduction, assessment of hydrogen DC hybrid turboelectric architecture is carried out. In assessing the hybrid hydrogen powerplant turbojet cores are considered for GT architecture with cores supplying power to ducted fans in conjunction with batteries, with power offtake to fans from GTs taken from the LP turbines. For comparison purposes with baseline propulsion system, with the exception of transmission efficiency, thermodynamic cycle parameters of the ducted-fan-GT combination of the DC turboelectric architecture are set to be equal to the thermodynamic cycle of the baseline turbofan at the design point (when not accounting for hybrid power input to propulsors). Also, during climb when battery is not operational, GT is set to operate at the same maximum continuous rating as the baseline GT.

7.4.4.1 Electric Component Assumptions

With LH₂ used as a machine coolant, MgB₂ superconducting generators and motors are considered ahead of BSCCO machines due to higher transmission efficiency. Also, for higher power density and transmission efficiency, cryogenic power controllers are considered ahead of ambient operating power controllers and HTS transmission lines considered ahead of copper transmission lines. With these technology assumptions, component power densities and transmission efficiencies utilised are shown in Table 7-13.

Table 7-13 Medium haul transport electric transmission component power densities and transmission efficiencies for hydrogen DC turboelectric architecture

Component	Power Density (kW/kg)	Transmission Efficiency
Generator	41.59 [21]	0.9998 [21]
Motor	22.03 [21]	0.9997 [21]
Power controller	32.88 [22]	0.9993 [22]
Transmission lines	227.1 [23]	1.0000 [23]

7.4.4.2 Battery Pack

For assessment of the battery pack, the model used for the regional transport aircraft in Sec. 6.6.2 is used for the medium haul transport. For assessment of stored energy in the battery pack for a given battery mass, a future technology level for batteries designed for year 2035+ EIS is considered. For this technology level battery specific energy is set at 1500 Wh/kg [24]. For design space exploration purposes, battery masses ranging from 1000 kg to 4000 kg are considered and battery power ratings ranging from 1000 kW to 4000 kW are considered for performance simulation.

7.4.4.3 Propulsion System Mass Breakdown

With electric installation factor set as 1.4 [8], an overview of propulsion system mass breakdown for hydrogen DC hybrid turboelectric architectures at selected battery power ratings compared to baseline and hydrogen turbofan configurations is shown in Table 7-14. Comparing hybrid turboelectric propulsion systems to hydrogen turbofan and baseline kerosene turbofan, overall propulsion system mass is increased by the incorporation of electric machinery. Also, with a DC/DC converter integrated into the propulsion system architecture to interface battery with main electric network of the hybrid turboelectric architecture, increased battery power ratings increase DC/DC converter mass and this has the effect of slightly increasing overall DC hybrid turboelectric system mass. For mass estimation of the turbomachinery, it was assumed that the mass of the turbojet coupled with a ducted fan combination has the same mass of the hydrogen turbofan. With BPR, intake air mass flow rate, TET and OPR the same for both configurations it is expected the actual mass of the turbojet-ducted fan combination would not vary greatly from hydrogen turbofan mass. Additionally since GT resizing was not carried out for the hydrogen turbofan, it is assumed its mass is equal to the mass of the baseline turbofan. Cryotank mass is not accounted for as propulsion system mass and is instead considered as a standalone aircraft component.

Table 7-14 Medium haul transport hydrogen DC hybrid turboelectric propulsion system mass breakdown

Item	Baseline	Hydrogen Turbofan	1 MW battery hybrid turboelectric	2 MW battery hybrid turboelectric	3 MW battery hybrid turboelectric	4 MW battery hybrid turboelectric
Turbomachinery (kg)	4808	4808	4808	4808	4808	4808
APU (kg)	299	299	299	299	299	299
Electric transmission (kg)	0	0	4203	4233	4263	4292

Installation (kg)	2860	2860	4541	4553	4565	4577
Total (kg)	7967	7967	13851	13893	13935	13976

7.4.4.4 Aircraft Performance

For performance assessment of DC hybrid turboelectric architecture at the aircraft level, a focus for studies is on how the DC hybrid turboelectric architecture can minimise fuel burn and energy consumption. With this as a focus, battery pack operational characteristics for a given mission can be fixed. For the hybrid mission the battery is configured to operate at full charge from the start-of-mission and to be employed during cruise. With cruise fuel burn for the baseline aircraft at 2,546 n.mi representing 83.3% of the block fuel of the baseline aircraft, it is expected that the largest reduction in block fuel by using batteries will happen during cruise.

For sizing of airframe, similar to hydrogen powered turbofan aircraft configuration, for focus on how the propulsion system affects aircraft performance, hybrid aircraft wing area and range are fixed by baseline aircraft wing area and range. For this sizing scenario the combination of battery mass and power rating within the previously mentioned parametric range (1000 kg to 4000 kg and 1000 kW to 4000 kW) that yield minimum block fuel for a given aircraft range are then determined.

Considering aircraft with a cryogenic fuselage structural factor of 1.2 (see Figure 7-13), while the battery is operational, power transmission from batteries to ducted fan propulsors reduces power required from GTs for a given thrust requirement. As power required from GT reduces, GT TET and fuel flow rate reduce. Although GT thermal efficiency drops with reduced TET, overall propulsion system efficiency is increased as a result of discharge of power from a more efficient battery. Also, despite transmission losses, this results in flight at lower TSFC while the battery is operational. However, while the battery is not operational, increased propulsion system mass and transmission losses increases climb fuel burn and fuel burn during battery-inoperative-cruise. For

the minimum fuel hybrid electric configuration at the design range (battery mass of 1000 kg and power of 1000 kW), the battery power rating is not high enough to cause a TSFC reduction that offsets increased fuel burn at other segments of the mission (climb and battery-inoperative-cruise) where the battery is not operating. In Figure 7-13 this results in a higher block fuel for hybrid electric aircraft than both hydrogen turbofan configurations even though overall block fuel is lower than block fuel for the baseline aircraft. Additionally, although a reduction in fuselage structural penalty reduces block fuel from the higher fuselage structural factor block fuel, the reduction is not high enough to result in a block fuel lower than hydrogen turbofan configurations. Also, as aircraft range reduces, deadweight penalty of cryotanks not being used at full capacity reduces deviation in block fuel from baseline value for all hydrogen configurations.

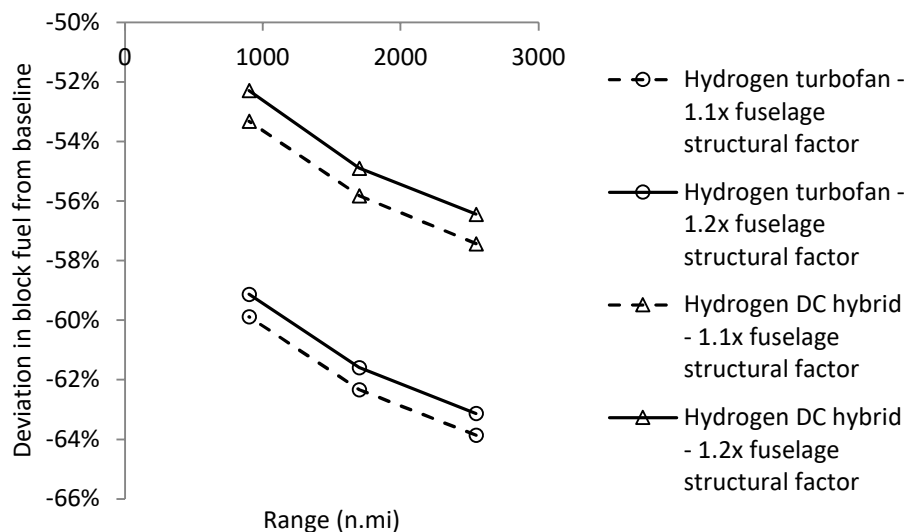


Figure 7-13 Medium haul transport deviation in block fuel from the baseline aircraft for minimum fuel hybrid electric configurations with 1500 Wh/kg batteries and hydrogen turbofan configurations versus aircraft range

Considering battery combinations for minimum fuel at a given design range, increased energy consumed by the batteries results in the deviation in total energy consumed for hydrogen hybrid electric configurations from baseline be higher than the deviation in block fuel (see Figure 7-14). Additional energy

expended by batteries and higher fuel burn result in total energy consumption by hydrogen DC hybrid turboelectric aircraft configurations being higher than the total energy consumption for hydrogen turbofan aircraft configurations.

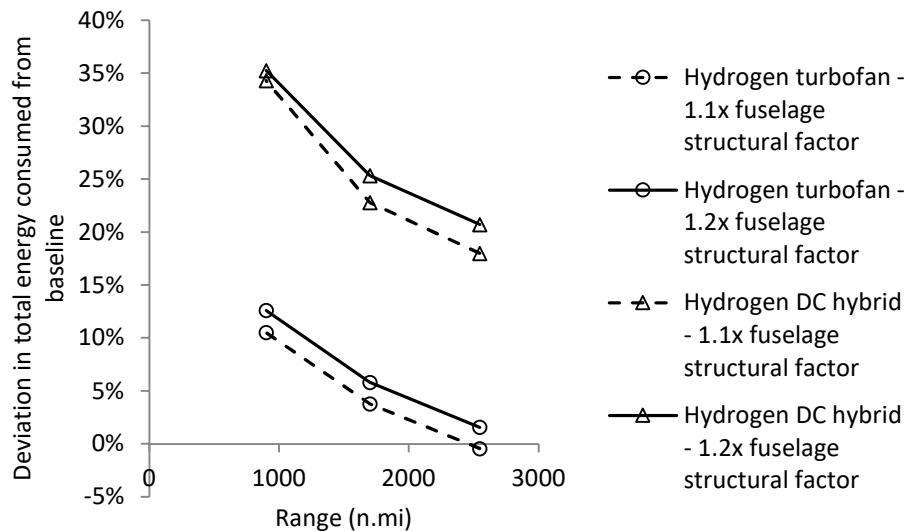


Figure 7-14 Medium haul transport deviation in total energy consumed from the baseline aircraft for minimum fuel hybrid electric configurations with 1500 Wh/kg batteries and hydrogen turbofan configurations versus aircraft range

7.5 The Employment of a Geared Hybrid Electric Architecture

For the application of hydrogen powered DC hybrid turboelectric architecture for medium haul transport, it was found that electric transmission weight penalty dominated the effect of reduced TSFC during mission segments where the battery is operational. This resulted in block fuel burn and energy consumption being higher than a hydrogen turbofan configuration. For reduced electric transmission weight penalty the geared hybrid electric architecture shown in Figure 4-1(e) is employed. Additionally, with the use of hydrogen not providing significant benefits (in terms of total energy consumption) over kerosene baseline without aircraft and propulsion system resizing, geared hybrid electric architecture employed here is assessed with kerosene as fuel.

7.5.1 Retrofit Hybrid Electric Aircraft

For purpose of assessing the effects of how the propulsion system affects overall aircraft performance, retrofit cases are considered. For the retrofit cases at the propulsion system level, the baseline turbofan is fitted with a gearbox on its LP shaft and power from installed battery pack is transmitted to the electric motor and then into the turbofan through this gearbox. For current studies resizing of GT through incorporation of a gearbox although expected to result in an improvement in GT performance has not been considered and is beyond the scope of current studies. In considering retrofitting at the aircraft level, the baseline airframe is utilised and installation of electric machinery and gearbox increases vehicle TOW for a given range. For installation of electric machinery at the aircraft level, the electric installation factor from Eq. 5-14 is taken to be 1.4 [8]. Since the baseline airframe is utilised for hybrid cases; masses and sizes associated with the empennage, aircraft equipment, fuselage, operational items, payload and wing remain constant.

7.5.2 Transmission Assumptions

For electric components of the transmission sub-system of the propulsion system, HTS motors and cryogenic power controllers are considered over present day state-of-the-art electric machinery for reduced weight and increased transmission efficiency. Also, HTS transmission lines are considered ahead of copper transmission lines for the same reasons. With the incorporation of a gearbox coupled with electric machinery technology assumptions set, component power densities and efficiencies of the transmission sub-system are shown in Table 7-15. In Table 7-15 power density for the gearbox is not listed as its mass is determined using the procedure in appendix B after the carrying out of propulsion system performance assessment.

Table 7-15 Medium haul transport transmission power densities and efficiencies for geared hybrid electric architecture

Component	Power density (kW/kg)	Transmission efficiency (%)
Motor	20.0 ^a [22]	99.5 ^a [22]
Power controller	20.0 ^a [22]	99.5 ^a [22]
Transmission lines	227.1 [23]	100.0 [23]
Gearbox	N/A ^b	99.0 [12]

^aIncludes penalty that arises from use of cryocooler to cool machinery

^bMass of gearbox estimated using procedure described in appendix B

7.5.3 Battery Pack

For modelling of the battery pack, the model used for the regional transport aircraft in Sec. 6.6.2 is also used for medium haul transport application. For assessment of stored energy in the battery pack for a given battery mass, a future technology level for batteries designed for year 2035+ EIS is considered. For this technology level battery specific energy is set at 1500 Wh/kg [24]. With overall propulsion system mass for geared hybrid electric architecture expected to be lighter than the hydrogen DC hybrid turboelectric architecture assessed in Sec. 7.4.4 due to a lower number of electric components sized at lower power ratings, higher battery masses and power ratings are considered for design exploration for geared hybrid electric architecture. Thus battery masses ranging from 1500 kg to 6000 kg are considered and battery power ratings ranging from 1500 kW to 6000 kW are considered for performance simulation.

7.5.4 Propulsion System Performance

7.5.4.1 Effect of Increasing Battery Power

Considering geared hybrid electric powerplant performance, for the same TET as the baseline propulsion system, as battery power input into the LP shaft of

GT via the electric motor and gearbox increases, the energy imparted into both the core flow and bypass flow of turbofan increases despite increased transmission losses. Also, as power is transferred into the LP shaft, its speed increases as battery power input increases. In addition to increasing core jet velocity and bypass jet velocity, increase in LP shaft speed causes mass flow rate through bypass and core to increase. This causes an increase in net thrust produced by turbofan as battery power rating increases. This is represented in Figure 7-15 with the maximum thrust produced by propulsion system at a given flight condition increasing as battery power rating increases. For a given flight condition and throttle, the increase in thrust produced by the addition of electric power is greater than what would be the case if engine throttle is increased. This is because despite increased transmission losses and lower propulsive efficiency, the efficiency of transmitting power from battery to LP shaft is higher than what GT thermal efficiency would be if GT throttle is increased. With increase in thrust produced by addition of electric power being greater than what would be the case if engine throttle is increased, overall system efficiency increases as battery power rating increases. Also, in Figure 7-15 for all configurations, pressure ratio increase with altitude causes GT thermal efficiency to increase as flight condition moves from end-of-runway (sea-level, M0.30) to initial cruise (37000 ft, M0.78). Up to the tropopause this results in higher system efficiency as flight altitude increases.

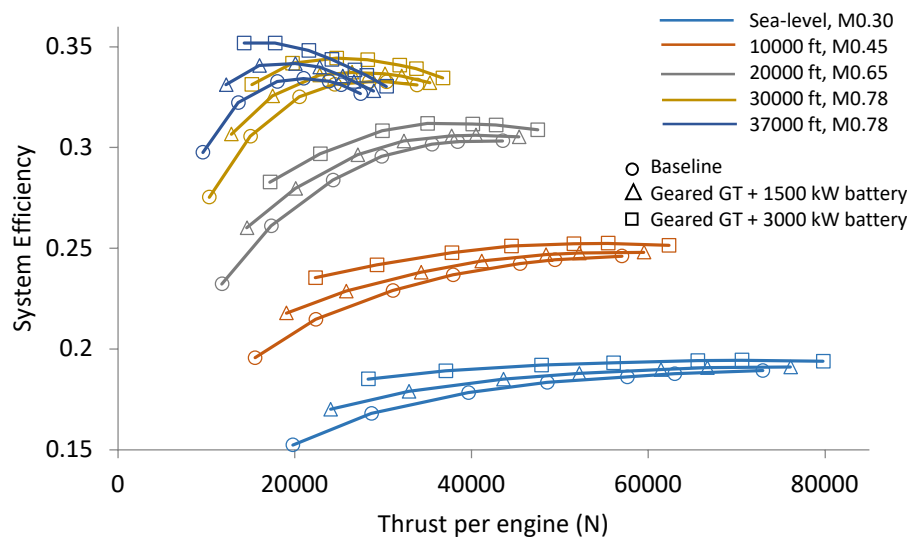


Figure 7-15 Comparison of overall system efficiency between geared hybrid electric powerplant at a SOC of 1.0 and baseline powerplant (power supplied to GT by battery is half of battery power rating since it is assumed that two turbofans power the aircraft)

It should be noted that in the performance simulation for the geared turbofan with power intake from batteries, consideration has not been made to using the gearbox to reduce fan speed. It is expected that use of a gearbox will allow for LPT to run at higher rotational speed which enables operation at higher isentropic efficiency for fewer stages [25]. Higher LPT efficiency is expected to improve overall GT efficiency and should increase powerplant system efficiency even further for hybrid electric powerplant than what is the case here; however assessment of this effect is beyond scope of current studies.

7.5.4.2 Engine Surge Prediction at High Battery Power Ratings

At higher battery power ratings, the minimum TET the propulsion system can operate at during a given mission at cruise flight conditions is increased due to LPC surge. Considering performance at the aircraft level and battery utilisation during cruise, weight reduction of the aircraft during cruise will reduce thrust required of the propulsion system, provided aircraft maintains level flight and constant speed. This effect coupled with battery use for a given thrust requirement is likely to reduce turbofan operating TET during cruise to within a

margin where surge of the turbofan's LPC becomes likely, even though this effect is likely to also reduce NO_x emissions. LPC surge TET is calculated in Turbomatch by accounting for battery power input into turbofan at off-design and then reducing TET until LPC surge margin approaches one. The results in Figure 7-16 are indicative of how surge TET is calculated for a given flight condition. In Figure 7-16 as TET reduces operation of the LPC moves towards surge as TET reduces. For turbofans simulated with power intake from batteries with power ratings of 4500 kW and 6000 kW, calculated surge TET's are shown in Table 7-16. Using the baseline aircraft as a reference, end-of-cruise TET is 1,173 K at a flight altitude of 38,400 ft. For a 6000 kW battery this TET is only 0.51% greater than surge TET. With use of non-resized GT, this difference is likely to be unacceptable for hybrid electric aircraft application if the battery is still discharging power by end-of-cruise. This is because during hybrid operation of the propulsion system, GT TET will be even lower than the baseline aircraft's end-of-cruise TET at the same flight condition due to battery operation reducing throttle required of GT to meet the thrust requirement.

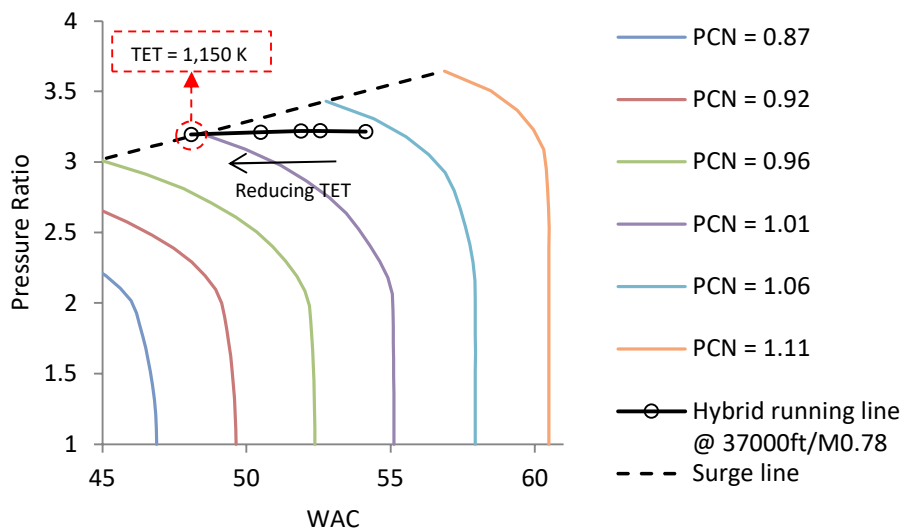


Figure 7-16 Zoomed in LPC map for geared turbofan showing hybrid electric running line for 6000 kW battery power rating at SOC = 1.0 at initial cruise flight condition

Table 7-16 LPC Surge TET at cruise flight conditions for 4500 kW and 6000 kW battery power ratings at SOC = 1.0

Flight condition	Idle TET of baseline powerplant (K)	LPC surge TET @ 4500 kW battery power rating (K)	LPC surge TET @ 6000 kW battery power rating (K)
37000 ft, M0.78	760	1002	1150
38400 ft, M0.78	760	1040	1167
39800 ft, M0.78	760	1081	1186

7.5.4.3 Variable Geometry to Offset Surge at High Battery Power Ratings

To reduce the TET at which the LPC surges at sufficiently high enough battery power ratings at cruise flight conditions, variable geometry nozzles can be employed. With variable geometry nozzles, LPC surge TET at cruise flight conditions can be reduced to within an acceptable margin through the following options:

1. Reducing bypass nozzle area
2. Reducing core nozzle area
3. Reducing both bypass nozzle area and core nozzle area

With option (1) there is a limit to how much bypass nozzle area can be reduced due to any reduction in bypass nozzle area having a counter effect of causing the operation of the fan to move closer to surge. Thus option (3) is chosen for maximal surge TET reduction by having the reduction of core nozzle area counteracting the effect of reduction in fan surge margin due to bypass nozzle area reduction. Setting a nominal minimum surge TET of 1000 K, bypass and core nozzle area reductions at cruise to achieve this TET for hybrid powerplants is shown in Table 7-17 for battery power ratings of 4500 kW and 6000 kW. Due

to the variable nozzles, the usable thrust range for the hybrid powerplant at battery power ratings of 4500 kW and 6000 kW increases. In Figure 7-17, at the initial cruise flight condition for 6000 kW battery power rating, due to variable nozzles the hybrid powerplant is able to generate thrusts from 23,550 N to 16,710 N (note: power supplied to GT by battery is half of battery power rating since it is assumed that two turbofans power the aircraft). This is important because using the baseline aircraft at the design range as a reference, mid-cruise thrust is 19,784 N. Thus variable geometry enables operation of the GT at this flight point. This procedure of reducing bypass nozzle area and core nozzle area for a given flight condition and battery power input to reduce LPC surge TET, is repeated for a range of SOC's ranging from minimum specified SOC for the battery to SOC at start-of-discharge (in this case a SOC of 100%).

Table 7-17 Bypass and core nozzle area reductions at cruise flight conditions for power intake into geared turbofan from 45000 kW battery and 6000 kW battery to achieve a nominal minimum surge TET of 1000 K

	<i>4500 kW @ battery SOC = 1.0</i>		<i>6000 kW @ battery SOC = 1.0</i>	
Flight Condition	Core nozzle area reduction (%)	Bypass nozzle area reduction (%)	Core nozzle area reduction (%)	Bypass nozzle area reduction (%)
11280 m, M0.78	1.23	1.75	12.4	14.6
11705 m, M0.78	3.21	2.24	14.8	18.3
12130 m, M0.78	5.08	6.45	28.9	18.7

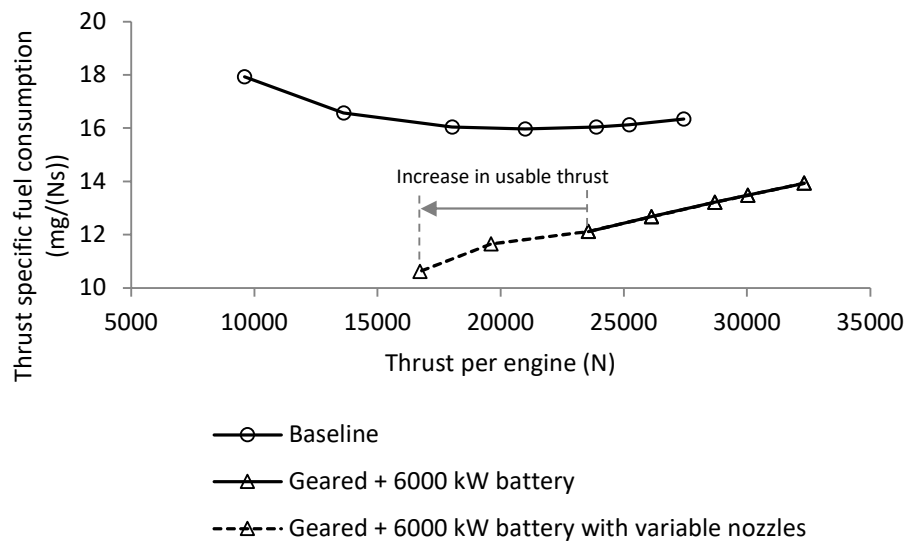


Figure 7-17 TSFC versus thrust comparison at initial cruise flight condition between baseline turbofan, geared turbofan with power input from 6000 kW battery and variable nozzle geared turbofan with power input from 6000 kW battery

7.5.5 Propulsion System Mass Breakdown

With electric installation factor set as 1.4 [8], an overview of the mass of components of the propulsion system for the geared hybrid electric architecture compared to baseline is shown in Table 7-18. Comparing hybrid electric powerplants overall propulsion system mass is increased by the incorporation of electric machinery and a gearbox. In assessing propulsion system mass of the hybrid electric powerplants as battery power rating increases, while gearbox mass remains constant, electric machinery mass (DC/AC converter, motor and transmission lines) is increased. Increase in electric machinery mass increases propulsion system mass as battery power rating increases. For hybrid electric powerplants with battery power rating of 4500 kW or higher, expected increase in turbomachinery mass due to incorporation of variable geometry nozzles is not taken into account in current study.

Table 7-18 Medium haul transport geared hybrid electric propulsion system mass breakdown compared to baseline

Component	Baseline	1500 kW battery hybrid electric	3000 kW battery hybrid electric	4500 kW battery hybrid electric	6000 kW battery hybrid electric
Turbomachinery (kg)	4808	4808	4808	4808	4808
APU (kg)	299	299	299	299	299
DC/AC Converter (kg)	0	73.1	146	219	292
Motor (kg)	0	72.7	145	218	291
Transmission lines (kg)	0	6.60	13.2	19.8	26.4
Gearbox (kg)	0	112	112	112	112
Installation (kg)	2860	2984	3045	3106	3166
Total (kg)	7967	8356	8570	8783	8996

7.5.6 Aircraft Performance

7.5.6.1 Design Objective and Hybrid Mission Scheme

Similar to the hydrogen DC hybrid turboelectric architecture assessed in Sec. 7.4.4.4, the design objective is on how use of hybrid electric architecture can minimise fuel burn. With cruise fuel burn for baseline aircraft at design range representing 83.3% of total block fuel, it is expected that the largest reduction in block fuel by using batteries will happen during cruise. As such, batteries are set to be employed during cruise. With recharging of the battery not considered, batteries are assumed to be fully charged at initial cruise to enable hybrid operation.

7.5.6.2 Accounting for Operational Penalties

For the assessment of aircraft performance for geared hybrid electric architecture, the following operational penalties are considered:

- GT operation at decreased efficiency when propulsion system is in hybrid mode due to GT operation at significantly reduced throttle from design point value when battery is operational
- Accounting for gearbox transmission efficiency and weight penalties
- Assessment with an electric machinery weight installation penalty greater than one

A summary of how these penalties affect aircraft block fuel and total energy consumption for a more representative scenario over “ideal” operation is shown in Table 7-19. Considering a battery pack that weighs 6000 kg with a power rating of 6000 kW, at the design range (2546 n.mi) the deviation in block fuel from baseline aircraft block fuel increases by about 3.28 percentage points as operation shifts from an ideal scenario to a more representative one. Despite GT operating at an efficiency not penalised by hybrid operation, at the design range for ideal scenario overall block fuel is higher than the baseline aircraft due to increased TOW increasing climb fuel burn and fuel burned during battery-inoperative-cruise. For more representative scenario, operational penalties increase block fuel further due to even higher climb fuel burn, higher fuel burn during battery-inoperative-cruise and lower TSFC reduction during battery-operative-cruise.

As aircraft range reduces, the fraction of cruise time the battery pack spends discharging increases. As a result for both ideal and representative scenario, increasing fraction of cruise time battery pack spends discharging lowers the deviation in block fuel from baseline aircraft as range reduces. At 900 n.mi both scenarios achieve a block fuel lower than the baseline aircraft at the same range. However at reduced aircraft ranges, the fraction of overall mission time that is spent at climb increases. With operational penalties increasing climb fuel burn due to higher TOW, coupled with higher TSFC during battery-operative-

cruise; at reduced aircraft ranges the discrepancy between block fuel under ideal operational scenario and more representative scenario increases.

Since battery mass is constant for both operational scenarios and batteries fully discharge before cruise is completed at all aircraft ranges considered, the percentage points increase in total energy consumption as operation shifts from an ideal scenario to a more representative one for a given aircraft range, is the same as the increase in block fuel. At all aircraft ranges for both scenarios, the deviation in total energy consumption from baseline aircraft total energy consumption is greater than the deviation in block fuel due to the additional energy expended from the batteries.

Table 7-19 Block fuel and total energy consumption for hybrid electric aircraft operating in a (1) “ideal” scenario and (2) scenario with operational penalties accounted for. Battery mass for hybrid electric aircraft is fixed at 6000 kg and battery power rating is fixed at 6000 kW

Range (n.mi)	Ideal operating scenario		Accounting for operational penalties	
	Deviation in block fuel from baseline (%)	Deviation in total energy consumed from baseline (%)	Deviation in block fuel from baseline (%)	Deviation in total energy consumed from baseline (%)
900	-6.84%	4.55%	-1.73%	9.67%
1700	0.75%	7.15%	4.61%	11.00%
2546	4.74%	9.02%	8.02%	12.30%

It should be noted that for performance of the representative scenario to achieve the block fuel and total energy consumption of the ideal scenario, the thermal efficiency of the GT can be improved by engine resizing. For the

resizing, by setting engine design point to be mid-cruise, cruise TET and thus thermal efficiency would be increased by the increasing of design point TET. Then to meet mid-cruise thrust requirement either mass flow rate and/or pressure ratio would have to be reduced. For this scenario performance penalty on take-off and climb performance would have to be assessed, however performance penalty could then be overcome by use of battery pack at these flight conditions to meet thrust requirements. Use of the battery this way could also reduce component maximum thermal power demand from the gas turbine core and hence improve component life [26].

7.5.6.3 Performance at Design Range

From results for geared hybrid electric aircraft configurations at the design range in Figure 7-18, for batteries that weigh 1500 kg block fuel increases as battery power rating increases. This increase in block fuel is driven by the increase in electric transmission mass that happens as battery power rating increases. With batteries fully discharging available stored energy, this trend is repeated at increased battery masses. Considering how block fuel changes as battery mass increases, for a given battery power rating as battery mass increases the amount of stored available electric energy on-board the aircraft increases. With configurations fully discharging available stored energy on-board during cruise the time the aircraft spends during battery-operative-cruise increases as battery mass increases. Increased duration of battery-operative-cruise extends the time the propulsion system spends in hybrid mode during cruise. With hybrid mode operation more efficient than flight with no power intake from the battery, as battery mass increases the aircraft spends a longer fraction of cruise time with propulsion system operating at a TSFC lower than what would be the case if the propulsion system is operating in turbo mode. However, at the design range increase in battery mass increases vehicle weight and reduces aerodynamic efficiency. During battery-operative-cruise for the same time interval, this effect coupled with battery power rating not being relatively high enough (with 6 MW batteries, battery power output represents 36.8% of total fan power at initial cruise flight condition and design TET) to cause significant reductions in propulsion system TSFC, means that at

increased battery masses more fuel is burnt per unit time interval. Also, during battery-inoperative-cruise, although duration is shorter as battery mass increases, the fuel burn per battery-inoperative-cruise time interval is greater than what is the case at lower battery masses. Thus although overall cruise fuel burn is reduced as battery mass increases due to increased battery-operative-cruise time; significant reductions in cruise fuel burn are limited by high vehicle weight, aerodynamic efficiency penalty and limited reduction in TSFC during battery-operative-cruise. Additionally as aircraft gets heavier with battery mass increase, time-to-height increases and this has the effect of increasing climb fuel burn. Increased climb fuel burn, increased battery-inoperative-cruise fuel burn and only a small reduction in battery-operative-cruise fuel burn increases overall aircraft block fuel as battery mass increases.

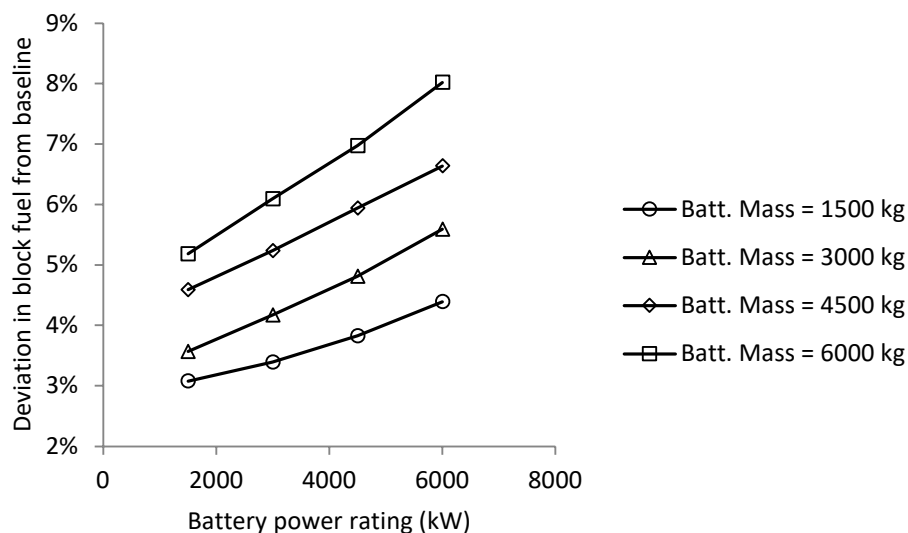


Figure 7-18 Deviation in block fuel from baseline aircraft for geared hybrid electric aircraft configurations versus battery power and mass at the design range for 1500 Wh/kg batteries

7.5.6.4 Performance at 900 n.mi

At 900 n.mi and batteries with specific energy of 1500 Wh/kg, for battery pack that weighs 1500 kg with a power rating of 1500 kW, the deviation in block fuel from baseline for hybrid electric aircraft is lower than what is the case for the same battery set-up at the design range (see Figure 7-18 and Figure 7-19).

With flight time shorter due to reduced aircraft range, the ratio of time the battery spends discharging to overall cruise time increases from what this ratio is at the design range (78% of cruise time at 900 n.mi compared to 23% of cruise time at 2546 n.mi). Also, with aircraft being lighter at 900 n.mi the aircraft weight penalty to battery-operative-cruise and battery-inoperative cruise fuel burn is reduced. Also, this means due to reduced thrust requirements that the fraction of battery power output to total fan power is increased which allows for a greater reduction in TSFC for the same battery power rating. Thus reduced aircraft weight penalty and increased battery-operative-cruise time are reasons for lower deviation in block fuel from baseline for hybrid electric aircraft with 1500 kg/1500 kW battery at 900 n.mi than the corresponding value at the design range. As battery power increases for batteries that weigh 1500 kg, block fuel increases due to increased electric transmission mass (see Figure 7-19).

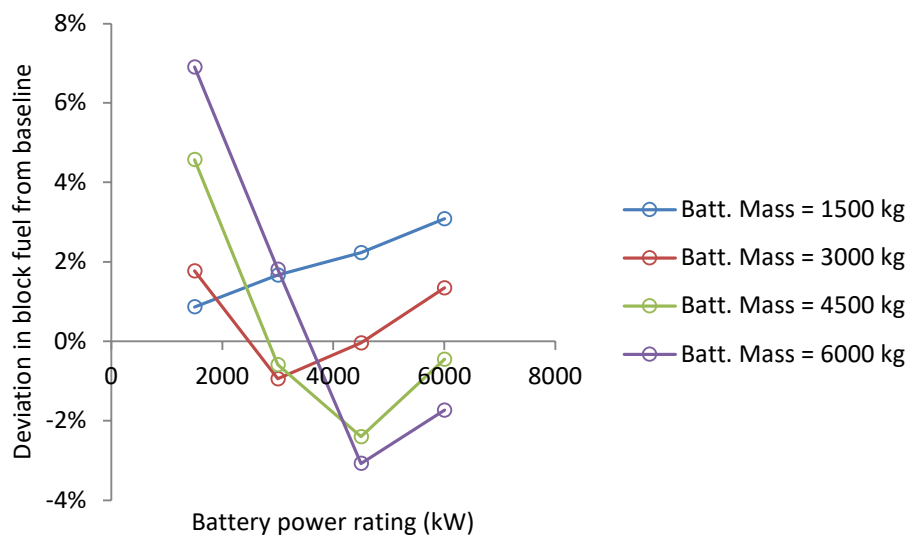


Figure 7-19 Deviation in block fuel from baseline aircraft for geared hybrid electric aircraft configurations versus battery power and mass at 900 n.mi for 1500 Wh/kg batteries

For battery power rating at 1500 kW, as battery mass increases, the amount of available usable energy in the battery increases and thus the ratio of time the battery spends discharging to overall cruise time is increased. This ratio

increases to 100% of cruise time at battery masses of 3000 kg and higher. However at battery masses of 3000 kg and higher, due to increased amount of available stored energy in the battery, battery use during cruise is characterised by batteries not fully discharging stored available energy. As a result deadweight effect caused by batteries not fully discharging available energy increases block fuel as battery mass increases beyond 1500 kg. As battery power rating is increased for battery masses greater than 1500 kg, more energy is consumed from batteries at lower GT fuel flow rates. This effect reduces block fuel as battery power rating increases for battery masses of 3000 kg and higher when batteries do not fully discharge available energy before the end-of-cruise. Once the batteries fully discharge available energy at or just before end-of-cruise, similar to effect at 1500 kg battery mass, increasing battery power rating increases overall block fuel due to higher electric transmission mass. Additionally at the power rating for minimum block fuel for a given battery mass, the rate of block fuel reduction with respect to battery mass reduces as battery mass increases. This is due to increasing battery mass increasing the aircraft weight penalty. The weight penalty increases climb fuel burn and reduces achievable fuel burn reduction during battery-operative-cruise. If battery mass is increased further by a sufficient margin, aircraft weight penalty would dominate the effect of hybrid operation at higher power ratings and eventually cause block fuel to increase at sufficiently high enough battery masses, even if batteries discharge all of available energy during cruise.

7.5.7 Comparison with Works from Public Literature

In considering how results from current studies compare to results from existing studies carried out on hybrid electric propulsion, a comparison is made in Table 7-20. In Table 7-20 results from the minimum fuel hybrid electric configuration simulated at 900 n.mi of the current study is compared to relevant configurations studied in public literature. The aircraft studied in [27] & [28] are turbofan powered medium haul aircraft with parallel/geared hybrid electric propulsion system architectures. Comparing results from the current study with results in [27], block fuel reduction is about 9.93 percentage points lower but

due to lower battery mass the discrepancy in total energy consumption deviation is lower than the difference in block fuel deviation (the difference is about 2.44 percentage points). One of the reasons for the block fuel reduction difference is the higher range considered and as a result higher MTOM of the configuration in [27]. As a result, at the off-design mission of 900 n.mi a higher battery mass can be installed on-board on the aircraft for a reduced aircraft weight and drag penalty. Also, in the study in [27] the batteries were configured so that power is provided for the entirety of cruise where SOC at end-of-cruise effectively matches the 20% minimum SOC limit. In the current study with batteries of 6000 kg and 4500 kW battery power rating, end-of-cruise SOC is approximately 23.3%. If battery mass was slightly reduced or battery power output during cruise slightly increased, block fuel for geared hybrid electric case in current case would be slightly lower. The second reason for the different in block fuel reduction in current study and that in [27] is that in [27] gearbox transmission efficiency and weight penalties were not accounted for. In the current study this gearbox penalty is also a reason for the difference in total energy consumption deviation from a baseline configuration for the current study being slightly higher than what is the deviation in [27]. In [27] this factor offsets to some extent higher energy consumption from batteries with a higher mass.

Comparing the current study to the study in [28], although lower specific energy batteries in were assumed; GTs in [28] are downsized, TET is increased at cruise and BPR is increased at cruise. The net effect of these factors is a lighter GT which operates at higher efficiency during cruise even though the battery is not configured to operate during cruise. Additionally, with the batteries only configured to operate during take-off and climb, reduced energy requirements means battery mass in [28] is lighter. As a result of higher GT efficiency during cruise, a lighter battery and reduced fuel burn during climb and take-off; the hybrid configuration in [28] achieves a higher block fuel reduction than what is the case in the current study. Also in [28], the lower battery mass means that reduced energy consumption from the battery enables a net reduction in total energy consumption which is not the case for the current study.

Table 7-20 Comparison of minimum fuel hybrid electric configuration simulated at 900 n.mi economic mission with results from similar configurations assessed in public literature

Parameter	Current Study	Pornet et al ^a [27]	Lents et al ^b [28]
Number of passengers	150	180	154
Design Range (n.mi)	2546	3300	3149
Off-design mission range of interest (n.mi)	900	900	900
Cruise Mach	0.78	0.78	0.74
TOM (kg)	77106	78900	61875
Propulsion system architecture	Parallel hybrid electric/geared hybrid electric	Parallel hybrid electric/geared hybrid electric	Parallel hybrid electric/geared hybrid electric
Battery type	Lithium-ion	Lithium-ion	Lithium-ion
Mission segment(s) where battery is used	Cruise	Cruise	Take-off and climb
Battery specific energy (W.h/kg)	1500	1500	1000
Battery mass (kg)	6000	8400	1597
Battery power (kW)	4500	6000	4900
Deviation in main	-3.07%	-13.0%	-6.03%

mission block fuel from a baseline configuration			
Deviation in main mission total energy consumption from a baseline configuration	7.82%	5.38%	-2.01%

^aElectrification of cruise scheme selected for performance comparison

^b20-year horizon estimates

7.6 Configuration Summary

7.6.1 Summary at Design Range

Top-level performance summary for aircraft configurations considered at the design range is shown in Table 7-21. For hydrogen turbofan configurations reduction in propulsion system TSFC drives reduction in block fuel over baseline configuration. However due to installation of cryotanks to store hydrogen in fuselage of aircraft, increased drag and structural penalty have an effect on whether the hydrogen turbofan aircraft achieves lower total energy consumption than the baseline aircraft, with energy consumption assessment taking into account different energy content of hydrogen and kerosene per kilogram of fuel. However, with lower block fuel reducing aircraft weight; TOFL and time-to-height for hydrogen turbofan aircraft are lower than the TOFL and time-to-height of the baseline aircraft. The hybridising of the hydrogen turbofan powerplant resulted in increased vehicle weight, block fuel and energy consumption over hydrogen turbofan aircraft due to electric transmission weight penalty. Increased vehicle weight for hydrogen DC hybrid aircraft resulted in TOFL and time-to-height of the configuration being higher than the baseline and hydrogen turbofan configurations. With the geared hybrid electric architecture employed to reduce electric transmission weight penalty, although total energy

consumption of the configuration was less than the total energy consumption of hydrogen DC hybrid aircraft, it still burned more fuel and consumed more energy than the baseline aircraft. This was due reductions in TSFC during battery-operative-cruise not being significant enough to offset higher vehicle weight. Also, increased weight due to installation of battery and transmission components increased TOFL and time-to-height for the aircraft over the baseline aircraft's TOFL and time-to-height.

Table 7-21 Medium haul transport top-level performance summary for aircraft configurations considered at the design range

	Baseline	Hydrogen Turbofan – 1.1× structural factor	Hydrogen Turbofan – 1.2× structural factor	Hydrogen DC Hybrid – 1.1× structural factor – minimum fuel config.	Hydrogen DC Hybrid – 1.2× structural factor – minimum fuel config.	Geared Hybrid Electric – minimum fuel config.
Battery mass (kg)	0	0	0	1000	1000	1500
Battery power rating (kW)	0	0	0	1000	1000	1500
Take-off field length (m)	1994	1499	1582	1957	2032	2134
Time to height (min)	27.4	20.5	21.8	31.2	33.5	30.6
Block fuel (kg)	14030	5071	5173	5973	6112	14462
Reserve fuel (kg)	3775	1491	1511	1704	1728	3875
Electrical	0	0	0	4.32	4.32	6.48

energy consumed (GJ)						
Main mission total energy consumed (GJ)	605.0	602.4	614.6	714.0	730.4	630.1
MTOM (kg)	79451	69891	71452	81455	83122	81781

7.6.2 Summary at 900 n.mi

Top-level performance summary for aircraft configurations considered at 900 n.mi is shown in Table 7-22. For hydrogen turbofan aircraft although reduction in TSFC drives reduction in block fuel over baseline configuration, oversized cryotanks and short flight time result in total energy consumption for the aircraft being higher than baseline aircraft total energy consumption. However reduced TOW due to lower block fuel results in hydrogen turbofan aircraft having lower TOFL and time-to-height than the baseline aircraft. For the hydrogen DC hybrid aircraft, relatively high electric transmission mass results in increased vehicle weight, block fuel and total energy consumption over hydrogen turbofan aircraft. Increased vehicle weight for hydrogen DC hybrid aircraft also results in TOFL and time-to-height of the aircraft being higher than baseline and hydrogen turbofan aircraft TOFL and time-to-height. With the geared hybrid electric architecture, a reduction in block fuel of 3.07% over the baseline aircraft is achieved. Driving this benefit is extended period of time during cruise where the propulsion system is operating in hybrid mode at a relatively high battery power rating. However, due to the utilisation of energy from a battery pack, total energy consumption for the configuration is 7.82% greater than baseline aircraft total energy consumption. Also, increased vehicle weight due to installation of battery and transmission components increased TOFL and time-to-height for aircraft over the baseline aircraft's TOFL and time-to-height.

Table 7-22 Medium haul transport top-level performance summary for aircraft configurations considered at 900 n.mi

	Baseline	Hydrogen Turbofan – 1.1× structural factor	Hydrogen Turbofan – 1.2× structural factor	Hydrogen DC Hybrid – 1.1× structural factor – minimum fuel config.	Hydrogen DC Hybrid – 1.2× structural factor – minimum fuel config.	Geared Hybrid Electric – minimum fuel config.
Battery mass (kg)	0	0	0	3000	2000	6000
Battery power rating (kW)	0	0	0	4000	3000	4500
Take-off field length (m)	1598	1389	1441	1934	1936	1908
Time to height (min)	21.2	19.6	20.4	30.7	30.8	26.3
Block fuel (kg)	5277	2117	2157	2463	2518	5115
Reserve fuel (kg)	3327	1342	1359	1562	1564	3529
Electrical energy consumed (GJ)	0	0	0	12.96	8.64	24.84
Main mission total energy consumed	227.6	251.5	256.3	305.6	307.8	245.4

(GJ)						
TOM (kg)	70250	66788	68286	80941	80973	77106

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8 CONCLUSIONS AND FUTURE WORK

8.1 Conclusions

8.1.1 Research Summary

To achieve ambitious future environmental targets for aircraft by organisations such as NASA and the European Union, TeDP has been proposed as a novel concept that has the potential to achieve these targets by significantly improving integrated propulsion-airframe performance. Implementation of TeDP for aircraft brings into play a number of design and development challenges due to the highly-integrated nature of TeDP-airframe configurations, low TRLs of key enabling technologies and the new modes of operation such as HeP opened up by shift to more electric architectures. With the architectural shift from predominantly mechanical architectures in aero gas turbines to the more integrated electric nature of TeDP & HeP systems likely to open up the integrated airframe-propulsion design space and increase propulsion system complexity; gaps in knowledge arise in understanding system operation and developmental needs. As such the aim of the research was to assess the performance and feasibility of conceptual integrated aircraft-turboelectric and hybrid-electric configurations and to understand how the overall system changes for different aircraft mission roles.

In achieving the project aim, objectives set were (1) the development of a multidisciplinary and integrated method to assess the impact of TeDP & HeP sub-system system integration on aircraft and propulsion system performance as a function of configuration and mission requirements, (2) assess suitable propulsion system architectures to identify which architecture potentially performs best for a given civil transport mission, (3) assess the impact of novel technology integration on overall system performance and architecture, and (4) Consider the effect of the propulsion system on aircraft design, performance, aerodynamics and structures.

In meeting the first project objective, although DP aspects of turboelectric systems was not assessed in the case studies carried out, the method created allows for the assessment of TeDP system architectures. By considering turboelectric and hybrid electric propulsion system architectures as integrated entities and utilising a systems engineering approach; a methodology that assess TeDP system performance as a function of mission profile, the airframe, propulsor configuration, electric architecture, energy storage and gas turbines was developed. Through use of this method, application of turboelectric propulsion and hybrid electric propulsion for a regional and medium haul transport missions was able to be assessed. Although the method was applied for regional and medium haul transport missions, the methodology would be valid for assessing performance of turboelectric and hybrid electric propulsion for a long haul transport mission.

In meeting the second project objective synthesized propulsion system architectures for assessment were DC turboelectric architecture, geared architecture, DC hybrid turboelectric architecture and geared hybrid electric architecture. Not all possible propulsion system architectures possible for turboelectric and hybrid electric propulsion were assessed due to limited project time, with a notable architecture not being assessed being an AC turboelectric architecture which has the potential for reduced electric transmission weight and increased transmission efficiency. Also, the only airframe configuration considered was a tube-and-wing configuration. Although it is possible for other airframes such as BWB and triple-fuselage configurations to be assessed in synergy with TeDP and HeP, emphasis for studies was placed on acquiring a fundamental understanding on the physics of TeDP and HeP by first focusing on assessment for a tube-and-wing configuration. In studying how propulsion system architectures can be adapted to different mission roles, qualitative rationales were provided for which propulsion system architecture should be applied for a given mission role.

For accomplishing the third project objective; novel technologies such as utilisation of superconducting machinery, use of hydrogen fuel and hybrid

electric operation were assessed for impact on overall system performance for regional and medium haul transport missions. For models of technology at differing levels of fidelity; performance of propulsion system architectures with novel technologies was assessed at component level, propulsion system level and aircraft level. For the fourth project objective, at the aircraft level; through determination of propulsion system size, weight and location; the effect of the propulsion system on aircraft aerodynamics, structures, design and performance was able to be assessed. Considering aircraft structural assessment, detailed stress determination on parts of the airframe that the propulsion system was attached to, was not assessed. However with the main focus on assessing the effect of the propulsion system on aircraft performance, it was decided that taking into account bending relief effects of the propulsion system on the aircraft wing and accounting for increased weight of aircraft due to installation of the propulsion system, would be sufficient for conceptual assessment purposes.

8.1.2 Regional Transport Performance Concluding Remarks

With use of superconducting electric machinery for regional transport mission, the main benefit over use of conventional present day state-of-the-art electric machinery is significantly improved power density. Assuming a retrofitted aircraft with fixed wing area, at the aircraft level use of superconducting electric machinery with 2035-2050 EIS technology assumptions for turboelectric operation was reduced MTOM by 8.55% reduction in MTOM and block fuel by 15.9% block fuel at 1,114 n.mi range in long range cruise mode compared to what these values were with the use of present day state-of-the-art electric machinery was. However, compared to a baseline turboprop configuration, despite improved propeller efficiency, electric machinery power density not being high enough and cryocooler power losses resulted in superconducting turboelectric aircraft having 1.63% higher block fuel at 1,114 n.mi range in long range cruise mode, 5.35% higher block fuel at 600 n.mi range in long range cruise mode and 2.09% higher block fuel at 200 n.mi range in high speed cruise mode. Additionally for the superconducting turboelectric architecture for regional

transport application, the use of electric machinery was associated with increased vehicle mass when compared to the baseline turboprop aircraft. This meant longer TOFL and longer time-to-height for the superconducting turboelectric aircraft over the baseline turboprop aircraft. Although the superconducting turboelectric architecture performed worse than the baseline when comparing relevant top-level performance parametrics, the nature of the architecture allows for hybrid-electric operation and propulsor distribution for further performance improvement.

The hybridising of the superconducting turboelectric architecture for the regional transport mission was found to reduce aircraft block fuel over the baseline turboprop. However this is based on the assumption that highly ambitious specific energy targets for lithium-ion batteries are achieved in the 2035-2050 timeframe. This is because present day lithium-ion batteries still have very low specific energy compared to the specific energies of a hydrocarbon based fuel such as kerosene. Driving block fuel reduction for hybrid-electric configurations, was the improvement in overall propulsion system efficiency that occurs when there is hybrid power transmission from a battery and GT to propellers for a given thrust requirement. For range of battery masses and power ratings considered, maximum fuel and energy savings happened at 600 n.mi where hybrid turboelectric aircraft managed to achieve 16.7% lower block fuel and 3.23% lower total energy consumption than baseline turboprop aircraft. At 600 n.mi mission operation was characterized by extended periods of flight the battery is discharging at a higher power rating than what was the case at 1,114 n.mi and reduced deadweight penalty than what is the case at 200 n.mi where batteries were unable to fully discharge all of available stored energy. However, for the 200 n.mi mission, although with battery masses and power ratings considered the batteries did not discharge all of available stored energy, if higher battery power ratings were assessed this might not be the case. At sufficiently high enough battery power rating and as long as GT TET does not fall below idle TET, then even at 200 n.mi batteries will be able to discharge all of available stored energy for battery masses considered. This could allow for potentially a greater reduction in block fuel and energy consumption for a

hybrid-electric aircraft at 200 n.mi within the range of battery masses considered, if 200 n.mi was the desired mission range for performance optimisation. For the design range range of 1,114 n.mi further reductions in block fuel could be obtained if battery masses higher than 2000 kg were considered in the assessment.

From studies carried out, turboelectric architectures are unlikely to be used in the short-to-mid-term due to low component power densities when compared to using turbomachinery. Even for hybrid turboelectric architectures specific energies present day batteries are still too low for reduced aircraft fuel and energy consumption when compared to a present day turboprop configuration. Additionally, for hybrid turboelectric architectures modelled for the 2035-2050 timeframe benefits in terms of fuel reduction and are energy consumption reduction are dependent on overly ambitious technology performance targets being met. Even if these targets are met, meeting a target such as 75% reduction in CO₂ emissions relative to levels in the year 2000 is not going to be possible by just the employment of a hybrid turboelectric propulsion system alone. Other technologies will need to be employed in conjunction with a hybrid turboelectric propulsion system. Such technologies could include propulsor distribution, formation flying and more widespread use of airframe structures made of composites. However, it should be noted that in meeting the European Union target of 90% reduction in NO_x emissions for the 2050 timeframe, the reduction in GT TET during hybrid operation of a retrofitted DC hybrid turboelectric operation is likely to reduced propulsion system NO_x emissions. Thus, the DC hybrid turboelectric architecture is likely to help an aircraft meet NO_x emissions for the 2050 timeframe, even though from the present study the exact degree to which NO_x is reduced has not been assessed.

8.1.3 Medium Haul Transport Performance Concluding Remarks

For medium haul civil transport role novel technologies assessed included the utilisation of hydrogen as a vehicle fuel, use of superconducting machinery and employment of hybrid electric architecture. For utilisation of hydrogen as vehicle fuel, with detailed assessment of aircraft structural performance not carried out

a cryogenic fuselage structural factor was parametrically varied to represent possible airframe structural penalties due to installation of hydrogen cryotanks. At a range of 2546 n.mi the structural penalty had an impact on whether hydrogen turbofan aircraft achieved lower total energy consumption than baseline turbofan aircraft. A 0.43% reduction in total energy consumption over baseline configuration changed to a 1.59% increase in total energy consumption when the fuselage structural penalty increased from 1.1 to 1.2. At reduced aircraft range, shortened flight time and oversized cryotanks contributed to even higher deviation in total energy consumption from baseline aircraft than what was the case at the design range (10.5% increase in total energy consumption at 900 n.mi for hydrogen aircraft with 1.1 fuselage structural penalty). Additionally, hybridising of the hydrogen turbofan through the employment of a DC hybrid turboelectric electric architecture did not achieve total energy consumption lower than the baseline aircraft and hydrogen turbofan aircraft. This was because relatively high electric transmission mass and added energy consumption from batteries penalised total energy consumption. For hydrogen DC hybrid turboelectric aircraft with structural penalty of 1.2, within range of battery power ratings and masses considered, for minimum fuel configuration total energy consumption was 20.7% higher than baseline at 2546 n.mi and 35.2% higher at 900 n.mi.

Although not considered in the performance assessment, gas turbine resizing and aircraft resizing could have substantial benefits for hydrogen-powered aircraft. Additionally although a DC turboelectric architecture was considered for the medium-haul mission, an AC turboelectric architecture for reduced electrical machinery weight penalty coupled with the use of distributed BLI propulsors for increased propulsive efficiency and reduced propulsor weight could lead to aircraft level benefits for a hydrogen-powered aircraft. Considering gas turbine and aircraft resizing, due to reduced amount of fuel needed for mission operation when hydrogen is used as a fuel as opposed to kerosene, reduced aircraft weight allows for a downsized aircraft. A downsized aircraft means reduced thrust requirements for the propulsion system. For the same thermodynamic cycle as a kerosene gas turbine propulsion system, reduced

thrust requirements means a hydrogen gas turbine propulsion system will be smaller than the kerosene one. This allows for reduced propulsion system drag and weight penalties. Thus, the benefits of a smaller aircraft (reduced aircraft drag and weight) and reduced propulsion system installation penalties should allow for a greater reduction in total energy consumption for a hydrogen-powered aircraft that was predicted in current studies. If an AC turboelectric architecture with distributed BLI propulsors is utilised, potential energy savings over baseline kerosene turbofan configuration could potentially increase further.

With performance of hydrogen DC hybrid turboelectric architecture penalised significantly by relatively high electric transmission mass, for reduced electric transmission weight penalty a kerosene geared hybrid electric architecture was considered with the aim of reducing block fuel over baseline configuration. However when assessing performance of the architecture, within range of battery power ratings and masses considered, at 2546 n.mi even with reduced electric transmission weight penalty the geared hybrid electric aircraft still burned more fuel than the baseline aircraft. For the 2546 n.mi mission with the value for battery specific energy assumed and battery power ratings considered, it might be possible for a block fuel saving if battery mass is light enough since for all aircraft ranges assessed there appears to be an optimum battery mass for minimum block fuel when batteries discharge all of available energy before the end of a mission. At 900 n.mi a reduction of 3.07% was achieved over baseline configuration due to reduced aircraft weight penalty on fuel burn and increased battery-operative-cruise time at high battery power rating than what was the case at 2546 n.mi. However, due to utilisation of energy from battery pack for the minimum fuel configuration, total energy consumption was 7.82% greater than baseline aircraft total energy consumption.

Considering operation of the geared hybrid electric propulsion system assessed for medium-haul transport application, similar to the regional mission, operation of the GT at reduced TET during 'hybrid mode' could potentially allow for reduced NO_x emissions. Also at 900 n.mi, reduced block fuel allows for reduced

aircraft CO₂ emissions. However, similar to the regional mission these benefits are only realisable through meeting highly ambitious performance targets for key sub-system technologies and components, such as the development of fully superconducting motors and generators.

8.1.4 Hybrid Electric Operation for Civil Transport Aircraft

At all ranges considered for hybrid-electric operation for the medium-haul transport mission, potential block fuel savings and energy consumption reduction as a result of hybridisation for the medium-haul mission were lower than what is the case for regional transport mission. A primary reason for this is that the TSFC reduction possible by use of HeP is larger for a regional transport mission than what is the case for a medium-haul transport mission. For current studies, for the same battery power rating of 1500 kW and battery specific energy of 1500 Wh/kg, for the regional aircraft at its design range battery power output represented approximately 51.8% of total transmitted power from energy sources to propulsor shaft for a 500 kg battery pack at initial cruise. Meanwhile for the medium-haul aircraft at its design range for the same battery power rating and a battery mass of 1500 kg, a battery power output of 1500 kW represented only approximately 5% of power being transmitted to fans with the other 95% of power driving the fans coming from the GT core at initial cruise. Quantifying the effect of this discrepancy on TSFC, for the regional mission at initial cruise the 1500 kW battery reduced TSFC by approximately 37.4% compared to the value of the conventional propulsion system at initial cruise. Whereas for the medium-haul mission the 1500 kW battery reduced TSFC by only 6.77% compared to the value of the conventional propulsion system at initial cruise.

As such, the TSFC reduction possible during cruise is reduced as aircraft size, weight and range is increased for a given battery power output because of the reduced fraction of power being supplied to the propulsors by more efficient electric energy storage options such as batteries. Battery power could be increased but this will result in batteries discharging all of available energy much faster and thus the necessity of installing heavier batteries which will

impose a weight penalty on the aircraft. A consequence of this is that even with the very optimistic estimate of 1500 W.h/kg for battery specific energy for the 2035-2050 timeframe, with propulsion system architectures and hybrid operating schemes considered, is that energy savings will not be possible and fuel savings limited with batteries as electric energy storage option as mission operation shifts from regional to medium-haul and then to long-haul missions. This is unless there are significant improvements in battery technology that result in battery specific energies larger than the 1500 W.h/kg estimate. Possible options for electric energy storage options for future aircraft application that have higher theoretical specific energies of 1500 W.h/kg include lithium-air batteries and carbon nanotube flywheels.

8.1.5 Limitations in Research Undertaken

From work carried out, identified limitations of approach are listed below:

- Despite better identifying sub-system interrelationships and giving a good representation of overall system performance, for untested technology top-down nature of method employed can overlook some lower level details that could turn out to be critical to overall system performance
- Top-down nature of the method employed leads to possibility of identifying a large number of system architectures that can lead it to taking longer to identify best architecture for a given set of requirements
- Due to limited project time, models of some systems, sub-systems and components could not be carried out at higher levels of fidelity. This was due to main effort of work being spent on the formulation of a methodology that can assess TeDP/HeP system performance at the aircraft level that can take into account sub-system interrelationships for the wide variety of TeDP architectures possible
- For integration of the propulsion system with the aircraft to determine flight performance, look-up tables defined at small enough data intervals could be used ahead of 3rd order performance polynomials used in current study. Although 3rd order polynomials give a good general estimation of performance between two variables for a range of values,

use of look-up tables at small enough data intervals is likely to give better operating point performance estimation if used in conjunction with spline interpolation

8.1.6 Contribution to Knowledge

Although from literature read various system approaches have been utilised for the assessment of TeDP architectures, many of the studies focused on architecture and mission specific application. For studies here a focus was placed on creating a generic framework that could be used and adapted to assess TeDP system performance for different mission roles accounting for the number of system architectures possible with TeDP. Also, with the potential for the hybridisation of the TeDP system architecture, an emphasis was made on extending the framework so it can assess the performance of HeP systems. In achieving this; a multidisciplinary and integrated method to assess the benefits and challenges of turboelectric and hybrid-electric propulsion systems was created. For this method turboelectric and hybrid-electric propulsion system performance was assessed as a function of mission profile, airframe configuration, propulsor architecture, electrical system architecture, energy storage and gas turbine architecture. The method created was applied in case studies investigating the performance of turboelectric and hybrid-electric propulsion systems for regional civil transport and medium haul civil transport aircraft. With the case studies, physical behaviour of the system from component level all the way to aircraft level was assessed. The approach also allows for assessing the effect of mission operation on sub-system performance.

8.2 Future Work

From work carried out, a number of recommended areas for future work are listed below:

1. *Modelling Components of the Electric Transmission*: In current studies carried out a black box approach was used for the modelling of electric components with each having fixed efficiency and either fixed power

density or torque density. For better matching between GT and propulsor, and to understand better performance of electric architecture, future work to be carried out would centre on using mathematical models to represent performance of electric components. With this carried out suitable electric machinery component performance maps can be created and requirements of the electric transmission for preliminary design can be determined.

2. *Aerodynamic Synergy Assessment:* From studies carried out assessment of aerodynamic synergies possible with TeDP has not been assessed. For future work assessment of propeller-wing-slipstream-lift-augmentation effect in synergy with hybrid electric propulsion for greater reductions in fuel burn can be assessed. Exploiting propeller-wing-slipstream lift augmentation effect can potentially allow for reductions in wing area and vehicle drag for a given mission profile. To be identified is whether an AC, DC or DC hybrid turboelectric architecture performs best with distributed propellers. Additionally, identification of the most ideal turboelectric propulsion system architecture (whether AC, DC or hybrid) for exploitation of BLI can be assessed.
3. *Technoeconomic Analysis:* In the assessment of TeDP system performance at the aircraft level, consideration was mainly made to how the TeDP system affects physical performance parameters such TOW, TOFL, time-to-height, block fuel and energy consumption. However practical implementation is likely to come at a cost greater than present day state-of-the-art technology. In developing the technology to a high TRL Technoeconomic assessment would need to be carried out to determine at what year in the future TeDP technology is likely to become commercially viable.
4. *Assessment of GT Transient Performance:* For simulations of the GT carried out using Turbomatch, steady-state performance of engines was assessed. However with the employment of TeDP system architectures, electric energy storage in the architecture can be used to reduce engine

transients. Future work carried out investigating this can be done to determine how overall engine design and performance changes.

5. *Battery Thermal Management Systems*: In the assessment for hybrid electric propulsion architectures assessed in current research effort, consideration was not made to the effects ambient temperature and heat losses due to battery inefficiencies have on battery performance. Future work could entail the sizing and performance assessment of thermal management systems required for battery use on a hybrid electric aircraft.
6. *GT Hybrid Cycle Optimisation*: The presence of electric energy storage in hybrid electric architectures allows for resizing of the GT in such a way where TET is reduced at maximum operating condition for reduced component thermal power demand while TET is increased at cruise for improved GT thermal efficiency. Future work to be carried out could assess to what extent GT size can be changed due to the presence of electric energy storage to maximise benefits at the aircraft level.
7. *GT Optimisation for Geared Hybrid Electric Architecture with Fans as Propulsors*: For utilisation of geared hybrid electric architecture with fans as propulsors, due to the presence of a gearbox, the approach used in geared turbofans in which the run LPT is set to run at its optimum speed without deteriorating fan performance can be employed. With this scheme allowing for reduced LP system mass and improved LPT efficiency at the cost of gearbox mass penalty and gearbox cooling requirement, future work to be carried out would be centred on assessing the reduction in LP shaft weight and increase in LPT efficiency possible at the expense of gearbox installation. Performance would then be assessed at the aircraft level.
8. *Variable Nozzle Weight Estimation*: In the assessment of geared hybrid electric architecture for medium haul aircraft, it was found that at sufficiently high enough battery power output for a given thrust requirement, variable geometry nozzles needed to be employed to due to LPC surge when GT operates at reduced TETs. From studies carried out

although performance assessment accounted for use of variable geometry nozzles, the likely weight penalty to the GT by featuring variable geometry nozzles was not assessed. For future work studies carried out for medium haul aircraft can be repeated accounting for weight of variable geometry nozzle weight.

9. *Alternate Electric Energy Storage Options for Hybrid Electric Propulsion:*

It was mentioned previously in Sec. 3.3 that flywheels could be used as an electric energy storage option of choice for hybrid electric propulsion for the 2035-2050 timeframe. The reason for this was that even though at present flywheels don't have higher specific energies than lithium-ion batteries, development in material technologies such as carbon nanotube fibre technology maturation could lead to flywheels having higher specific energies than the highest forecasts for lithium-ion batteries in the 2035-2050 timeframe. As such for a fair comparison of hybrid turboelectric propulsion for the 2035-2050 timeframe when compared to conventional propulsion system architectures, the assessment of hybrid electric propulsion with a modelled flywheel should be assessed. If in the near future there are other electric energy storage technologies that might outperform batteries, they should be assessed too.

10. *Emissions Estimation:* For the 2050 timeframe, targets for CO₂ and NO_x emissions relative to levels in the year 2000 that future aircraft should meet have been set by the European Union. To adequately assess if turboelectric and hybrid-electric propulsion systems for future aircraft can meet these targets, the method utilised here should be extended so that CO₂ and NO_x emissions can be estimated. With this extension, whether turboelectric and hybrid-electric aircraft meet future emissions targets will be determined.

APPENDICES

Appendix A

The ratio of specific heats, γ , is determined in Eq. A-1 where C_P is isobaric specific heat capacity.

$$\gamma = \frac{C_P}{C_P - R} \quad \text{A-1}$$

C_P in SI units is estimated in Eq. A-2, where T_Z is the static temperature T_S divided by 1000 and $A_{i \rightarrow n}$ are constants used for the determination of gas thermodynamic properties. For dry air values of A_i are given in [A1].

$$C_P = 1000(A_0 + A_1 T_Z + A_2 T_Z^2 + A_3 T_Z^3 + A_4 T_Z^4 + A_5 T_Z^5 + A_6 T_Z^6 + A_7 T_Z^7 + A_8 T_Z^8) \quad \text{A-2}$$

The specific enthalpy, H , for a given station in SI units is determined with Eq. A-3.

$$H = 1000000 \left(A_0 T_Z + \frac{A_1}{2} T_Z^2 + \frac{A_2}{3} T_Z^3 + \frac{A_3}{4} T_Z^4 + \frac{A_4}{5} T_Z^5 + \frac{A_5}{6} T_Z^6 + \frac{A_6}{7} T_Z^7 + \frac{A_7}{8} T_Z^8 + \frac{A_8}{9} T_Z^9 + A_9 \right) \quad \text{A-3}$$

The entropy polynomial temperature function, F_T , for a given stations is determined in Eq. A-4.

$$F_T = A_0 \ln(T_S) + A_1 T_Z + \frac{A_2}{2} T_Z^2 + \frac{A_3}{3} T_Z^3 + \frac{A_4}{4} T_Z^4 + \frac{A_5}{5} T_Z^5 + \frac{A_6}{6} T_Z^6 + \frac{A_7}{7} T_Z^7 + \frac{A_8}{8} T_Z^8 + A_{10} \quad \text{A-4}$$

Reynold's number for computation of propeller blade aerodynamic characteristics is defined in Eq. A-5 where ρ is air density, U is flight speed, c is blade chord and μ is dynamic viscosity.

$$Re_{propeller} = \frac{\rho U c}{\mu} \quad \text{A-5}$$

The activity factor for a propeller blade is shown in Eq. A-6 where r_{hub} is blade radius at the propeller hub, r_{tip} is blade radius at the propeller tip and D is propeller diameter. Propeller diameter is equal to blade tip radius multiplied by two.

$$AF = \frac{100000}{16} \int_{\left(\frac{r_{hub}}{r_{tip}}\right)}^1 \left[\left(\frac{c}{D}\right) \left(\frac{r}{r_{tip}}\right)^3 \right] d\left(\frac{r}{r_{tip}}\right) \quad \text{A-6}$$

For a given blade element with a radius between r_{hub} and r_{tip} and a known propeller diameter, the Activity factor for a given blade element is determined in Eq. A-7.

$$AF = \frac{100000}{16} \left(\frac{1}{D}\right) \int_{\left(\frac{r}{r_{tip}}\right)_{i-1}}^{\left(\frac{r}{r_{tip}}\right)_i} \left[c \left(\frac{r}{r_{tip}}\right)^3 \right] d\left(\frac{r}{r_{tip}}\right) \quad \text{A-7}$$

If for a given blade element, i , blade chords are known at $(r/r_{tip})_i$ and $(r/r_{tip})_{i-1}$, then the average chord between i and $i-1$ is determined and the activity factor for a given blade element is determined in Eq. A-8. For a number of blade elements from propeller hub to propeller tip, blade activity factor is computed in Eq. A-9 for a number of blade elements from propeller hub to tip. Total propeller activity factor is determined in Eq. A-10 by multiplying activity factor for a single blade by the total number of blades (n_{blades}) the propeller has.

$$AF_{element,i} = \frac{100000}{16} \left(\frac{1}{D} \right) \left(\frac{c_i + c_{i-1}}{2} \right) \int_{\left(\frac{r}{r_{tip}} \right)_{i-1}}^{\left(\frac{r}{r_{tip}} \right)_i} \left(\frac{r}{r_{tip}} \right)^3 d \left(\frac{r}{r_{tip}} \right) \quad \text{A-8}$$

$$AF = \sum_{i=1}^{n_{elements}} AF_{element,i} \quad \text{A-9}$$

$$TAF = n_{blades} AF \quad \text{A-10}$$

Propeller advance ratio (λ) is defined in Eq. A-11 where ω is propeller rotational speed and U is flight speed.

$$\lambda = \frac{U}{\omega r_{tip}} \quad \text{A-11}$$

Propeller power coefficient (C_{PW}) is defined in Eq. A-12 where PW is propeller shaft power.

$$C_{PW} = \frac{2PW}{\rho (\omega r_{tip})^3 \pi r_{tip}^2} \quad \text{A-12}$$

References

[A1] WALSH, P. P. and FLETCHER, P. (2004) *Gas Turbine Performance*. 2nd ed. Oxford: Blackwell Science.

Appendix B

For parallel hybrid operation, a differential allows for the Gas Turbine and electric motor of parallel hybrid architecture to operate at different speeds during mission operation. The differential gearbox is assumed to be an epicyclic gearbox that has its ring gear, sun gear and planet carrier gear all being allowed to fully rotate. Advantages of using an epicyclic gear for a differential over a conventional architecture are that they are more compact and lighter [B1]. A

single stage epicyclic gear train differential is considered with a layout the same as the one shown in Figure B-1.

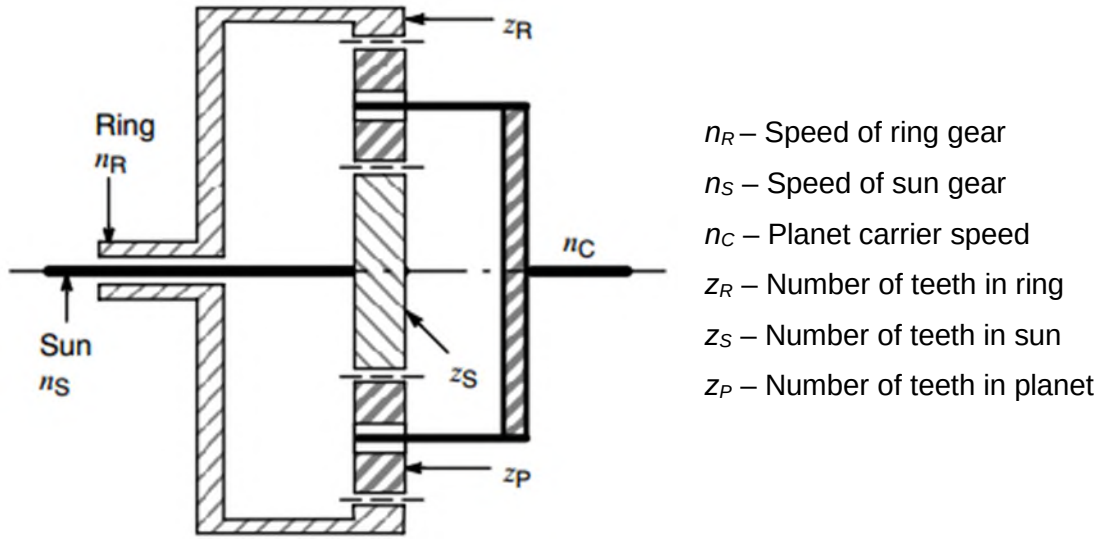


Figure B-1 Schematic of single stage epicyclic gear [B2]

With the ring gear set as the electric motor shaft output, sun gear set as gas turbine shaft output and planet carrier set as propulsor shaft input; planet carrier speed can be determined from Eq. B-1 with the shaft output from the electric motor and the GT serving as inputs to the epicyclic gear.

$$n_C = \frac{n_S}{1 + \frac{z_R}{z_S}} + \frac{n_R}{1 + \frac{z_S}{z_R}} \quad \text{B-1}$$

For a single stage epicyclic gear there are multiple gear ratios. For modelling purposes the gear ratio between the planet and the sun gear is denoted by U_P and the gear ratio between the ring gear and the sun is denoted as U_R . U_P is determined in Eq. B-2 and U_R is determined in Eq. B-3.

$$U_P = \frac{z_P}{z_S} \quad \text{B-2}$$

$$U_R = \frac{z_R}{z_S} \quad \text{B-3}$$

If U_R is set as a design parameter then ring speed is determined in Eq. B-4 through re-arranging Eq. B-1

$$n_R = n_C \left(1 + \frac{1}{U_R}\right) - n_S \frac{1}{U_R} \quad \text{B-4}$$

Initially assuming that $n_C = n_S$ (this means that speed from the output turbine of the GT = speed of the propulsor) to only consider the effects of how a change in U_R affects sizing of motor and gearbox. Otherwise considering the case where $n_C \neq n_S$ would mean that the effects of changing propulsor speed would have to be considered on overall propulsion system performance. For the case where the propulsor is a fan, changing fan speed affects design with an increase in fan efficiency likely as fan speed falls due to reducing tip Mach number. However the effect of this is not likely to impact overall system performance by more than a few percent, so this effect is not considered and is beyond the scope of current studies. With this assumption, Eq. B-4 simplifies to Eq. B-5.

$$n_R = n_S \quad \text{B-5}$$

Assuming that the pitch circle diameter (PCD) of the ring gear is the sum of the PCD of the sun gear plus twice the PCD of the planets, the relationship between the numbers of teeth on the ring, a planet and sun is determined in Eq. B-6 [B3]

$$z_R = z_S + 2z_P \quad \text{B-6}$$

With U_P defined in Eq. B-2 and U_R defined in Eq. B-3, Eq. B-6 can be manipulated to determine an expression for U_P in Eq. B-7.

$$U_P = \frac{1}{2}(U_R - 1) \quad \text{B-7}$$

To determine the mass of the gearbox the size of the gears needs to be determined first. The starting point for sizing of the gear is obtained from the expression for sun gear pitch diameter [B2] which is determined as a function of the applied nominal torque, gear geometry and number of planets in Eq. B-8 where d_s is the operating pitch diameter of the sun, T_{NOM} the total nominal torque, K_γ the mesh load factor, b the face width, D the assumed diameter to

set face width to diameter ratio (b/D), K the load intensity factor in mega pascals, C_G the gear ratio factor and N_{CP} the number of planets.

$$d_s = \frac{1}{1000} \left(\frac{2000 T_{NOM} K_Y}{\left(\frac{b}{D}\right) K C_G N_{CP}} \right)^{\frac{1}{3}} \quad \text{B-8}$$

The total nominal torque is evaluated at the rotational speed corresponding to the maximum power condition of the gearbox (see Eq. B-9).

$$T_{NOM} = \frac{\text{Maximum input power (W)}}{n_s @ \text{maximum input power (rad.s}^{-1}\text{)}} \quad \text{B-9}$$

The value for K varies depending on the application. Typical values for K are shown in Table B-1 [B2]. The values for K are recommended to be used for preliminary design calculations only. For more detailed calculations K is either calculated from the standards defined in ANSI/AGMA 2101-D04 or set based on previous design experience. Helical gears are considered ahead of spur gears for use in the differential because typically for aero applications the gearbox is going to be operating at high speeds. Spur gears at high rotational speed generate more noise than helical gears and this offsets the slight efficiency benefit that is realised from employing spur gears [B4].

Table B-1 Typical values for K [B2]

Application		Accuracy	Values (MPa)
Aerospace, (single pair)	helical	High precision	5.86
Aerospace, (epicyclic)	spur	High precision	4.14

C_G is expressed in terms of the number of the teeth of the sun gear and the planet gear (see Eq. B-10)

$$C_G = \frac{z_P}{z_P + z_S} \quad \text{B-10}$$

From the definition of U_P in Eq. B-2, Eq. B-10 simplifies to the expression in Eq. B-11.

$$C_G = \frac{U_P}{U_P + 1} \quad \text{B-11}$$

For the determination of N_{CP} , the approach in [B5] is employed. From studies in [B5], the weight of the gear train is reduced as the number of torque branches is increased. Considering an epicyclic gear train, this means that as the number of planets increases the lighter the gear train becomes. However there is a geometric limitation to the number of planets that can fit into differential epicyclic arrangement which corresponds to the case where adjacent planet gears are in contact with each other (see Figure B-2).

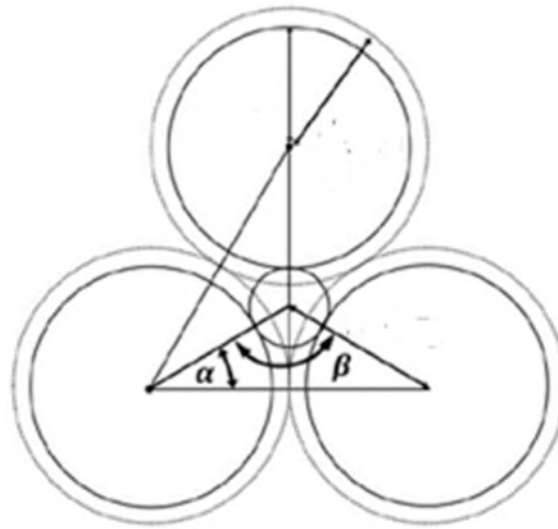


Figure B-2 Schematic showing the geometric limitation for planets around a sun
[B1]

From the gear arrangement in Figure B-2, the assumptions from [B1] are made:

- The tip diameter of the planet gears is the pitch diameter plus two times the gear module (denoted by 'm')

- Planets are equispaced within the mesh

Thus the cosine of α from Figure B-2 is determined in Eq. B-12

$$\cos \alpha = \frac{0.5 \left(\frac{d_p}{m} + 2m \right)}{0.5 \left(\frac{d_s}{m} + \frac{d_p}{m} \right)} = \frac{z_p + 2}{z_s + 2} \quad \text{B-12}$$

Also α can be determined from Eq. B-13

$$\alpha = 180^\circ - \beta = 180^\circ - \frac{360^\circ}{N_{CP,max}} \quad \text{B-13}$$

Re-arranging Eq. B-12 and Eq. B-13 yields the expression for the maximum number of planets for a given number of planet gear teeth and sun gear teeth in Eq. B-15.

$$N_{CP,max} = \frac{180}{90 - \cos^{-1} \left(\frac{z_p + 2}{z_s + z_p} \right)} \quad \text{B-14}$$

From the definition of U_P in Eq. B-7, Eq. B-14 changes to the expression in Eq. B-15

$$N_{CP,max} = \frac{180}{90 - \cos^{-1} \left(\frac{z_p + 2}{z_s + z_p} \right)} \quad \text{B-15}$$

$N_{CP,max}$ can thus be found for a known U_P and z_S , and z_S is determined from Eq.

$$z_S = \frac{d_S}{m} \quad \text{B-16}$$

For multiple path transmissions, the total tangential load is not quite evenly distributed between the various load paths (irrespective of design, tangential velocity or accuracy of manufacture). K_y takes into account the unequal tangential load distributed between the different planets and is expressed in Eq. B-17 [B2] where T_{Branch} is the torque in the planet branch with the heaviest load. Typical values for K_y are shown in Table B-2.

$$K_Y = \frac{T_{Branch} N_{CP}}{T_{NOM}}$$

B-17

Table B-2 Typical values for K_Y [B2]

Application level ^{1), 4), 5)}		Number of planets, N_{CP}								AGMA accuracy grade ²⁾	Flexible mounts ³⁾
		2	3	4	5	6	7	8	9		
1	$K_Y^{6)}$	1.16	1.23	1.32	1.35	1.38	1.47	1.60	~	A7 or worse	without
2	$K_Y^{6)}$	1.00 ⁷⁾	1.00 ⁷⁾	1.25	1.35	1.44	1.47	1.60	1.61	A5-A6	without
3	$K_Y^{6)}$	1.00 ⁷⁾	1.00 ⁷⁾	1.15	1.19	1.23	1.27	1.30	1.33	A4 or better	without
4	$K_Y^{6)}$	1.00 ⁷⁾	1.00 ⁷⁾	1.08	1.12	1.16	1.20	1.23	1.26	A4 or better	with

NOTES

1) Application level 1 typical for slow speed gears, mining mill drives, etc.;
2 moderate quality, i.e., commercial marine, non-military;
3 & 4 high quality, high speed, gas turbine/generator drives, military marine, wind turbines.

2) Gear manufacturing quality has an influence on the load sharing performance of the planets. Higher gear quality results in reduced load variations through each planet mesh.

3) Flexible mounts of the planets, such as flexible shafts or pins, flexible planet couplings or flexible annulus improves load sharing. See 9.2.

4) Load sharing level 2 or higher requires at least one floating member.

5) Load sharing level 3 or higher requires a flexible ring gear.

6) Mesh load factor, K_Y , is the product of the number of planets times the L_f factor in ANSI/AGMA 6123-A88.

7) A value of 1.00 may not be conservative enough for applications where the sun mass is high in relation to the speed and the radial forces required to accelerate the sun are significant.

With the value for z_S initially unknown N_{CP} cannot be determined. If N_{CP} cannot be determined then d_S cannot be found. Thus an iterative method is employed where for a given gear module a value of d_S is initially guessed to determined z_S . With z_S determined, N_{CP} can now be found and then a new value for d_S is calculated from Eq. B-8. If the calculated value for d_S does not match the initially guessed value, the initial guess for d_S is updated with the calculated value and the calculation procedure is repeated until the calculated value of d_S matches guessed value of d_S to within a set convergent limit. This procedure is summarised in Figure B-3.

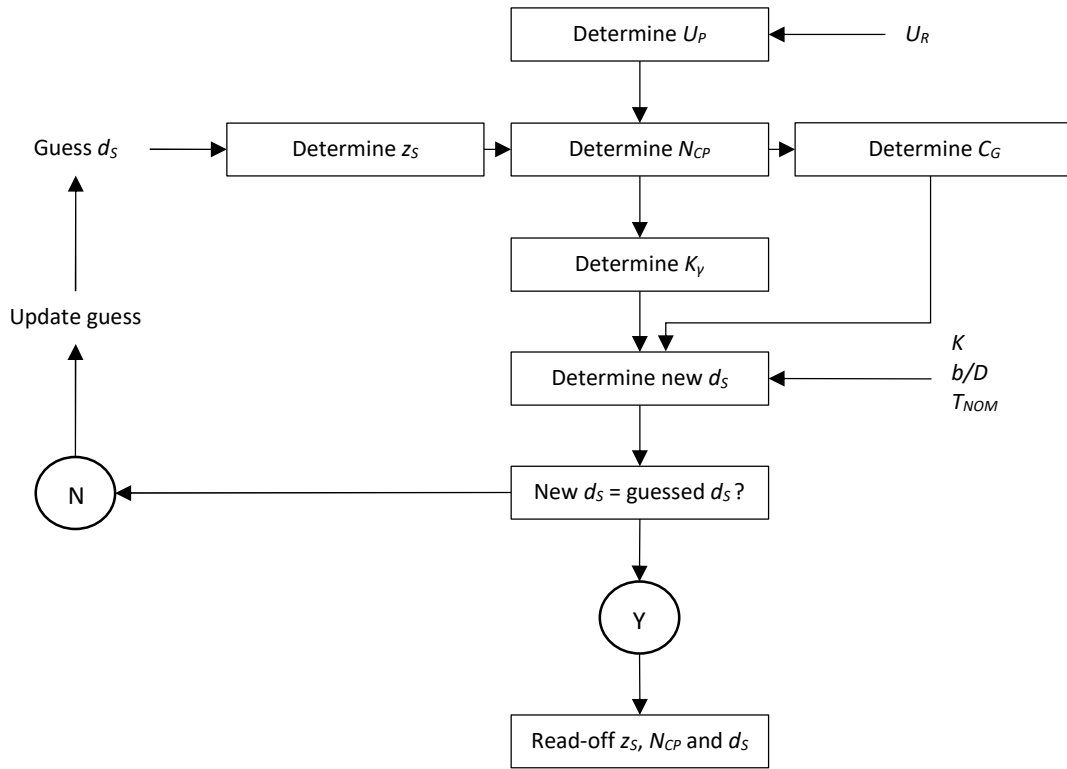


Figure B-3 Sun gear diameter, number of planets and number of teeth in sun gear determination procedure

Gear mass is estimated using the method described in [B6] and shown in Eq. B-18 where ρ is material density, V_1 is operating pitch cylinder volume of sun gear, K_{V1} is volume utilization coefficient of sun gear, K_{V2} is volume utilization of planet gear, K_{V3} is volume utilization coefficient of ring gear, K_{bw} is effective gear face width ratio in the epicyclic gear stage and m is total gear mass. The method is a semi-empirical approach that estimates gear weight by estimating the volume for each gear in an epicyclic set and multiplying it by material density. Gear volume is estimated by taking into account gear volume utilization i.e the ratio of the actual gear volume to pitch cylinder volume.

$$m = \rho V_1 (K_{V1} + U_P^2 N_{CP} K_{V2} + U_R^2 K_{V3} K_{bw}) \quad \text{B-18}$$

Mass estimation here does not include mass of accessories such as oil cooling, circulation, control and valve systems. The weight and sizing of these systems primarily depends on the heat dissipation requirement of the gearbox [B7]. Additionally the weights of gear supports and bearings are not accounted for. In

accounting for these it is assumed the propulsion system installation factor accounts for penalties of gearbox installation at the aircraft level.

V_1 in Eq. B-18 is determined in Eq. B-19 where b_w is effective gear face width in mesh

$$V_1 = \frac{\pi}{4} d_s^2 b_w \quad \text{B-19}$$

For a defined b/D , b_w is determined in Eq. B-20

$$b_w = \left(\frac{b}{D}\right) d_s \quad \text{B-20}$$

The volume utilisation coefficients K_{Vi} depend on gear body shape. For sun gears K_V varies from approximately 0.8-1.0; for planet gears K_V varies from 0.3-0.7; and ring gears the range is 0.05-0.1. K_{bw} typically has a value in the range of 0.7-0.9 [B5].

References

- [B1] Höhn, B-R., Stahl, K. and Gwinner, P. (2013) Light-Weight Design for Planetary Gear Transmissions. *Gear Technology*, September 2013, pp. 96-103.
- [B2] American Gear Manufacturers Association (2006) Design Manual for Enclosed Epicyclic Gear Drives. AGMA 6123-B06
- [B3] Kovacevic, A. (2007) Mechanical Analysis – Gear Trains. [Lecture Notes] City University London
- [B4] Miguel Ángel Valbuena Antón (2015) Preliminary Design and Optimisation of a Power Gearbox For the Ultrafan™ Geared Engine. Published thesis (MSc) Cranfield University
- [B5] Willis Jr., R. J. (1963) New Equations and Charts Pick Off Lightest-Weight Gears. *Production Engineering*, 34 (2), pp. 64-75
- [B6] Kapelevich, A. L. and Ananiev, V. M. (2011) Gear Transmission Density Maximization, *Gear Technology*, November/December 2011, pp. 46-52

Appendix C

The centrifugal compressor configuration employed is shown in Figure C-1. In the configuration flow enters through the inducer and is changed from an axial to radial direction in the impeller before entering a diffuser. Flow then passes through a set of axial straightener vanes to eliminate flow swirl before the flow enters the combustion chamber [C1].

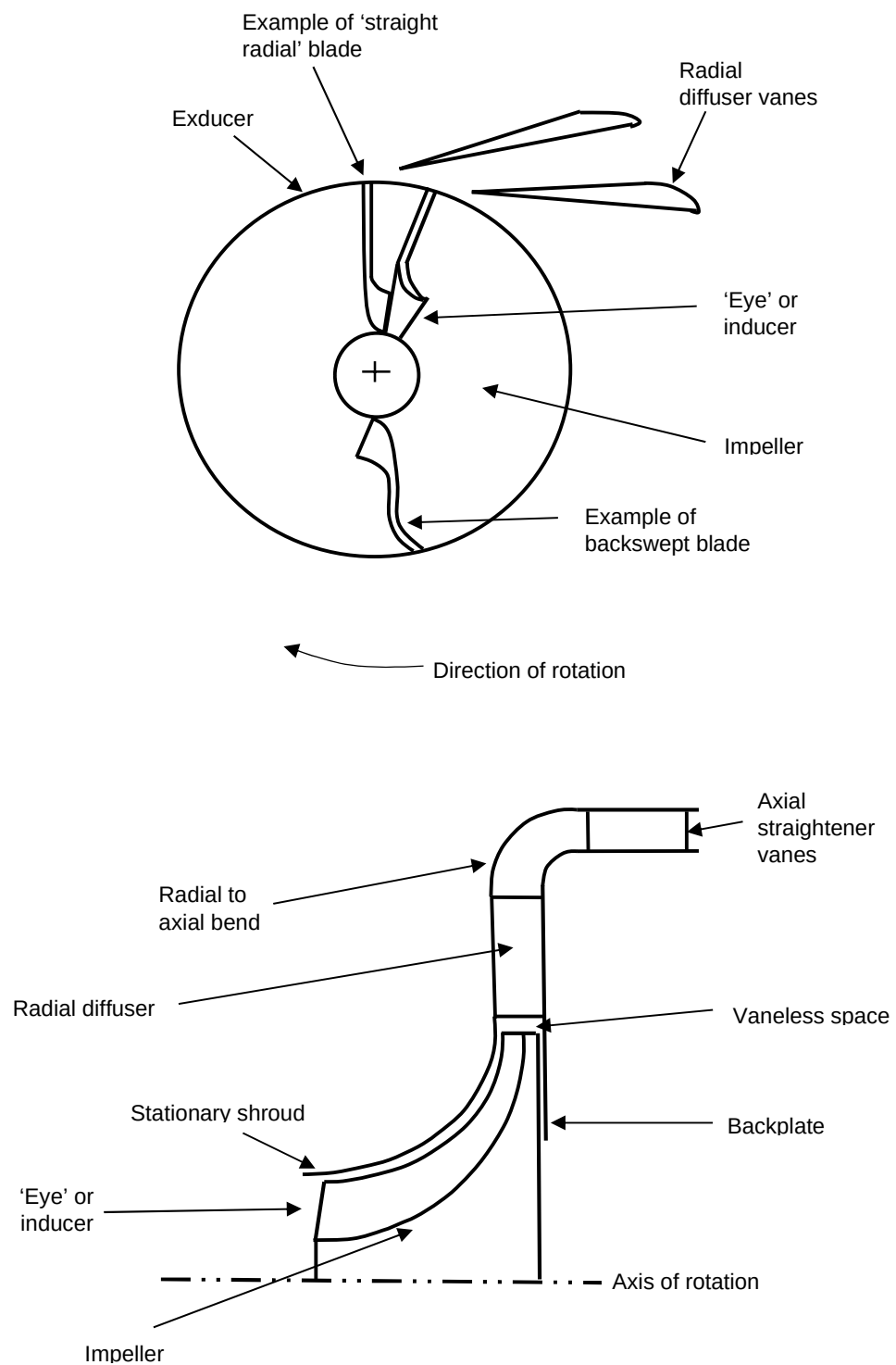


Figure C-1 Centrifugal compressor configuration [C1]

For size and speed estimation, the procedure in Figure C-2. is utilised. In the procedure for thermodynamic characteristics at compressor inlet and exit and design variables estimate compressor angular speed and compressor size.

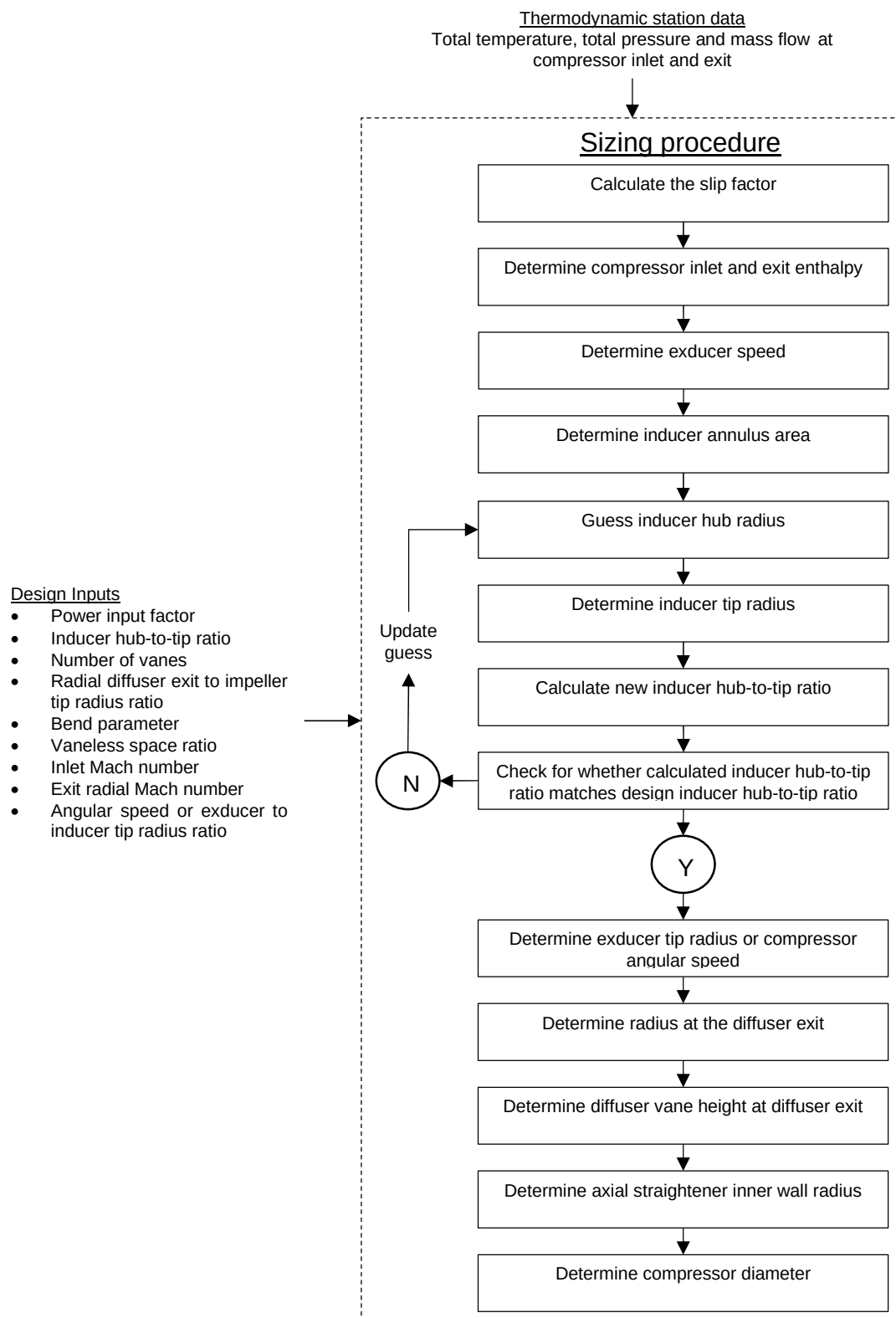


Figure C-2 Centrifugal compressor size and speed estimation procedure

The slip factor, F_{slip} , is estimated with the relation in [C1] and shown in Eq. C-1, where N_{vanes} is the number of impeller vanes

$$F_{slip} = 1 - \frac{0.63\pi}{N_{vanes}} \quad \text{C-1}$$

From defined total temperature at compressor inlet and exit; enthalpy at the inlet and exit are determined using the definition in Eq. . With inlet and exit enthalpy determined, flow speed at the exducer, U_{ex} , is determined using the relation in [C2] in Eq. C-2 where H_{out} is enthalpy at the outlet of the compressor, H_{in} is enthalpy at the inlet of the compressor and $F_{PowerIn}$ is the power input factor.

$$U_{ex} = \frac{\sqrt{H_{out} - H_{in}}}{F_{slip} F_{PowerIn}} \quad \text{C-2}$$

To find the annulus area of the inducer, the value of Q from Eq. 4-23 needs to be found for the inducer. Thus inlet static temperature, $T_{S,in}$, needs to be determined from defined inlet total temperature, $T_{t,in}$, and inlet Mach number, M_{in} (see Eq. C-3 where γ is the specific heat capacity ratio). With $T_{S,in}$ determined static inlet pressure, $P_{S,in}$, is found in Eq. C-4 for a given inlet total pressure, $P_{t,in}$, $T_{t,in}$ and $T_{S,in}$. With $P_{S,in}$ determined Q is found from Eq. 4-23. With Q determined, the annulus area of the inducer is determined in Eq. C-5 where W_{in} is inlet mass flow rate.

$$T_{S,in} = \frac{2T_{t,in}}{2 + (\gamma - 1)M_{in}^2} \quad \text{C-3}$$

$$P_{S,in} = P_{t,in} \left(\frac{T_{S,in}}{T_{t,in}} \right)^{\frac{\gamma}{\gamma-1}} \quad \text{C-4}$$

$$A_{ind} = \frac{W_{in} \sqrt{T_{t,in}}}{Q \frac{P_{t,in}}{1000}} \quad \text{C-5}$$

For a guessed inducer hub radius, $r_{hub,ind}$, inducer tip radius, $r_{tip,ind}$, is determined in Eq. C-6. With $r_{tip,ind}$ determined a new inducer hub-to-tip ratio is estimated (see Eq. C-7 where $htr_{ind,new}$ is calculated inducer hub-to-tip ratio) and

if this does not match design inducer hub-to-tip ratio the guessed value of $r_{hub,ind}$ is updated until calculated hub-to-tip ratio matches design hub-to-tip ratio.

$$r_{tip,ind} = \sqrt{\frac{A_{ind}}{\pi} + r_{hub,ind}^2} \quad \text{C-6}$$

$$htr_{ind,new} = \frac{r_{hub,ind}}{r_{tip,ind}} \quad \text{C-7}$$

Usually for designing a centrifugal compressor there will be an RPM requirement [C1]. With this RPM requirement exducer tip radius can be determined from Eq. C-8 where ω is angular speed of the compressor and $r_{tip,ex}$ is tip radius of the exducer. In absence of an RPM requirement however, the exducer tip radius is estimated from an arbitrary exducer to inducer tip radius ratio, σ , which is dependent on centrifugal compressor design. This ratio is then used to determine exducer radius in Eq. C-9. With exducer radius determined, Eq. C-8 is solved to determine ω .

$$U_{ex} = \omega \cdot r_{tip,ex} \quad \text{C-8}$$

$$r_{tip,ex} = \sigma \cdot r_{tip,ind} \quad \text{C-9}$$

With $r_{tip,ex}$ found, radius at the diffuser inlet, $r_{diff,in}$, is determined for a given vaneless space ratio, VSR , in Eq. C-10. With $r_{diff,in}$ found, for a defined radial diffuser exit to impeller tip radius ratio, DTI , radius at the diffuser exit, $r_{diff,exit}$ is determined in Eq. C-11.

$$r_{diff,in} = VSR \cdot r_{tip,ex} \quad \text{C-10}$$

$$r_{diff,exit} = DTI \cdot r_{diff,in} \quad \text{C-11}$$

In the determining of diffuser vane height, H_{diff} , the area of the annulus of the diffuser is first estimated using Q . For a defined exit radial Mach number, $M_{rad,out}$, and exit total temperature, $T_{t,out}$, exit static temperature, $T_{S,out}$ is determined in Eq. C-12. With $T_{S,out}$ determined static outlet pressure, $P_{S,out}$, is found in Eq. C-13 for a given outlet total pressure, $P_{t,out}$, $T_{t,out}$ and $T_{S,out}$. With

$P_{S,out}$ determined Q is found from Eq. 4-23. With Q determined, the annulus area at the diffuser exit, A_{diff} , is determined in Eq. C-14 where W_{out} is exit mass flow rate. With A_{diff} determined, H_{diff} is determined in Eq. C-15.

$$T_{S,out} = \frac{2T_{t,out}}{2 + (\gamma - 1)M_{rad,out}} \quad \text{C-12}$$

$$P_{S,out} = P_{t,out} \left(\frac{T_{S,out}}{T_{t,out}} \right)^{\frac{\gamma}{\gamma-1}} \quad \text{C-13}$$

$$A_{diff} = \frac{W_{out} \sqrt{T_{t,out}}}{Q \frac{P_{t,out}}{1000}} \quad \text{C-14}$$

$$H_{diff} = \frac{A_{diff}}{r_{diff,exit}} \quad \text{C-15}$$

With H_{diff} determined by the setting of a design bend parameter, BP , the radius of the axial straightener, $r_{axstraightener}$, is determined in Eq. C-16.

$$r_{axstraightener} = H_{diff}BP + r_{diff,exit} \quad \text{C-16}$$

With compressor radius assumed to be equal to the radius of the axial straightener, compressor diameter is determined in Eq. C-17.

$$d_{compressor} = 2r_{axstraightener} \quad \text{C-17}$$

References

- [C1] WALSH, P. P. and FLETCHER, P. (2004) Gas Turbine Performance. 2nd ed. Oxford: Blackwell Science.
- [C2] SARAVANAMUTTOO, H. I. H. et al (2009) *Gas Turbine Theory*. 6th ed. Harlow: Pearson Education

Appendix D

Maximum zero fuel weight or maximum zero fuel mass is defined in Eq. D-1

$$MZFW = OEW + M_{payload,max} \quad \text{D-1}$$

Maximum landing weight or maximum landing mass is defined in Eq. D-2

$$MLW = MZFW + M_{fuel,reserves,max} \quad \text{D-2}$$

Appendix E

E.1 Electric Energy Storage Level

Identified modes of operation of the battery during a hybrid electric flight are a discharge mode and a recharge mode. Descriptions for what these modes mean are given below

- Discharge mode: Operation of the electric energy storage when power is discharging from it
- Recharge mode: Operation of the electric energy storage when power is being transferred into it

E.2 Propulsion System Level

At the propulsion system level the propulsion system can operate in either a hybrid mode or a turbo mode. Descriptions for what these modes mean are given below

- Hybrid mode: Operation of the propulsion system when power is being transferred to the propulsor from both the GT and electric energy storage device
- Turbo mode: Operation of the propulsion system when power is being transferred to the propulsor from core GT only