

CRANFIELD UNIVERSITY

BJORN LEE CLETON

VARIABLE GEOMETRY EXHAUST SYSTEM FOR LOW SPECIFIC
THRUST TURBOFANS

SCHOOL OF AEROSPACE, TRANSPORT & MANUFACTURING
(SATM)

PHD THESIS
Academic Year: 2017

Supervisor: Professor Vassilios Pachidis
February 2017

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Gas Turbine Performance Engineering Group

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“All our knowledge has its origins in our perceptions”

Leonardo da Vinci

EXECUTIVE SUMMARY

This research was undertaken within the context of the Rolls-Royce University Technology Centre (UTC) in Aero System Design, Integration & Performance at Cranfield University, hosted by the Centre for Propulsion Engineering.

Aero-engine designers are moving towards lower specific thrust and higher bypass ratio engines in an attempt to improve mission fuel burn. When going to Ultra-High bypass ratio and very low specific thrust engines there may be a possibility for further mission fuel burn improvement equipping such engines with a 'Variable Geometry Exhaust System' (VGES). This system might also be required when looking at the possibility of surge of the compressor components in very low specific thrust engines. The performance improvement of the VGES could arise from the aspect that this system includes a mini-mixer, where part of the bypass flow is mixed with the core flow. The idea behind the mini-mixer is the possibility to gain the benefits of a mixed exhaust turbofan while reducing the losses these systems introduce. This is because the mini-mixer will not mix the complete exhaust flows. Therefore, the mixing chamber will not require the same length as a regular mixed exhaust system. Next to the mini-mixer the VGES presents the possibility to vary four flow areas within the engine's exhaust system, hence potentially increasing the available 'handles' for optimum performance and preventing surge of the compressors. Two of these variable areas are exhaust nozzles and the other two are the inlets to the mixer. The main focus of this PhD programme was the representative simulation of the VGES with a mini-mixer, taking all important pressure losses, as well as mixing efficiency into account. The work reported herein discusses the engine cycle optimisation, weight and mission fuel burn studies undertaken.

For this research two different turbofan engine architectures are examined, the 'direct-drive' and the 'geared' configuration, where the fan is coupled to the Low

Pressure Turbine (LPT) via a reduction gearbox. The mission profile investigated is for an A330-size aircraft with a range of 3000 nm. Technology limitations are set for a potential entry into service in 2025-2030. The results of this study showed that regardless of how many areas were allowed to vary of the four variable areas, no significant performance benefit was present. In regards to surge possibility with the current assumption for the very low specific thrust engine configurations there was no need for the variable areas. The results of the study also demonstrate that the mini-mixer itself can improve engine performance and mission fuel burn. With the current assumptions the maximum reduction in mission fuel burn observed is 0.5% at the lower fan diameter range investigated for the geared configuration. For the direct drive the maximum reduction found was 0.1%. The reason for the direct-drive configuration having less improvement is due to the limit the LPT lets the fan FPR be reduced, which influenced the thrust gain from the mini-mixer, compared to the geared configuration. Furthermore, for both architectures going to higher fan diameters, there is no benefit anymore present by using the mini-mixer. The reason for this is that the thrust gain is numerically associated with the reduction in FPR and LPT pressure ratios when comparing to an unmixed engine with the same Specific Thrust. When the FPR of an unmixed is already very small the amount of thrust gain when using a mixed exhaust system can achieve is very small at this point the losses start outweighing the benefits.

This research programme has made a useful contribution to knowledge in various areas of engineering and scientific relevance. First of all the viability of a novel piece of technology was investigated, with potential benefits as well as practical limitations identified. The influence of mixing efficiency and pressure loss in a complex, mini-mixer arrangement were studied and a useful methodology was established to identify performance trade-offs. Also, the study looked into the effect on high-bypass ratio turbofan engine performance of using multiple variable nozzle areas simultaneously. The results are reported herein for the very first time.

ACKNOWLEDGEMENTS

The author would like to thank the supervisor Prof Vassilios Pachidis, for the supervision of this research project. Also for all the support he gave over the three years and the faith he had in my ability to bring this research project to a successful finish.

The following people I would also like to thank.

- Mr Alvis Pellegrini for all the time and effort he spent supporting and actively contributing to the research group and especially to this project. The countless hours he spent on the development of the MATLAB based *Genetic Algorithm* optimiser are greatly appreciated.
- Industrial supervisors for their support throughout this research project and I would like to give special thanks to Mr Andrew Rolt who developed the patent and whose advice was always greatly appreciated.
- My fellow research group and work colleges Mr David Bentley, Dr Christos Mourouzidis and Mr David Bosak. For making the time spent at Cranfield more enjoyable and fun and for the countless hours spent supporting and discussing the different research topics.
- Mrs Maria Negus the UTC administrator for the support throughout this Thesis.
- The rest of the UTC team and academic staff I have not mentioned, I am grateful for any help you have given me throughout the research project.

I would also like to offer my greatest gratitude to my wife with her constant love and support throughout this three year endeavour. Without her, I would have not

been able to complete my work. Last but not least I would like to thank my mother, father, sister, brother-in-law, mother-in-law and her boyfriend, for all their support and help.

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LIST OF ABBREVIATIONS

ACARE	Advisory Council for Aviation Research & innovation in Europe
ACO	Ant Colony Optimisation
BOV	Blow-Off Valves
BPR	Bypass Ratio
CFD	Computer Fluid Dynamic
CLEAN	<u>C</u> omponent <u>V</u> alidator of <u>E</u> nvironmentally Friendly <u>A</u> ero- <u>E</u> ngine
COT	Combustor Outlet Temperature
CR	Cruise
DoF	Degrees of Freedom
EFFAE	Efficient and Environmentally Friendly Aero-Engine
EOR	End of Runway
EPIDOSYS	Engine Preliminary Design Optimisation System
EU	European Union
FPR	Fan Pressure Ratio
GA	Genetic Algorithm
GasTurb	Gas Turbine Performance Software
GBP	Great Britain Pound
GSP	Gas Turbine Simulation Programme
HPC	High Pressure Compressor
HPT	High Pressure Turbine
IPC	Intermediate Pressure Compressor
IPT	Intermediate Pressure Turbine
ISA	International Standard Atmosphere
LEMCOTEC	Low Emissions Core engine Technology
LPT	Low Pressure Turbine
MFB	Mission Fuel Burn

NEWAC	New Aero Engine Core Concepts
NLR	Netherlands Aerospace Centre
OPR	Overall Pressure Ratio
PSO	Particle Swarm Optimisation
SLA	Sea Level Altitude
TET	Turbine Entry Temperature
UTC	University Technology Centre

NOMENCLATURE

Latin Symbols

A	Area
A_{IF}	Interface area function
B	BPR
BPR	Bypass Ratio
C	Sutherlands constant
CO_2	Carbon dioxide
C_p	Specific heat capacities
C_{pg}	Specific heat capacities after combustor
d	Dynamic head
D	Diameter
D_a	Core average diameter
D_C	Nacelle highlight diameter
D_h	Hydraulic diameter
D_{out}	Mixer chamber outer diameter
F	Friction factor
F_N	Thrust
F_{Nh}	Thrust core exhaust

F_{Nc}	Thrust bypass exhaust
F_{Nmixed}	Thrust mixed exhaust
$\hat{F}_N$	Specific Thrust
$(FPR)_o$	Fan pressure ratio optimum
K	Kelvin
KE_{jet}	Jet kinetic energy
k_{nac}	Density of nacelle material
L_a	Length of the core cowl
L_b	Length of the bypass duct
L_C	Length of the nacelle,
L_i	Length of the intake
L_m	Mixer length
M	Mach number
N	Rotational speed
NO_x	Nitrogen oxide
P_c	Total pressure mixer inlet pressure cold stream
P_{er}	Wetted perimeter
P_{er_m}	Mixing plane perimeter
P_h	Total pressure mixer inlet pressure hot stream
P_s	Static pressure
P_t	Total pressure
Q	Flow capacity
R	Gas constant
Re	Reynolds number
t_{pr}	Thickness of the pressure vessel wall
T	Temperature
T_0	Reference temperature
T_{blade}	Blade mean metal temperature

T_c	Temperature bypass mixer inlet
T_{ci}	Cooling flow inlet temperature
T_{gi}	Gas inlet temperature
T_h	Temperature core mixer inlet
T_s	Static temperature
V_a	Flight velocity
V_c	Exit velocity of bypass exhaust flow
V_h	Exit velocity of core exhaust flow
V_j	Exit jet velocity
V_{ideal}	Exit velocity of the ideally mixed flow
V_m	Mean jet velocity
W	Mass flow
W_c	Bypass Mass flow
W_f	Fuel mass flow
W_h	Core mass flow

Greek Symbols

γ	Heat capacity ratio
γ_g	Heat capacity ratio after combustor
ϵ_o	Cooling effectiveness
η_c	Compressor efficiency
η_f	Fan efficiency
η_{NB}	Bypass nozzle efficiency
η_{mix}	Mixing efficiency
$\eta_{overall}$	Overall engine efficiency
η_{Prop}	Propulsion efficiency
η_{Th}	Thermal efficiency
η_{tr}	Power transfer efficiency

λ	Lambda
μ	Viscosity
μ_o	Reference value for viscosity
ρ	Density
σ_y	Yield stress
ψ	Fraction of core flow used for turbine cooling

1 Introduction

1.1 Background

Gas turbines are the dominant propulsion system in the air transport industry. A gas turbine is an internal combustion engine. However, unlike the better known piston engine where combustion is intermittent, gas turbines use continuous combustion to operate. The modern gas turbine engine is one of the most complex pieces of machinery ever designed. It has taken a thriving industry over 70 years to reach the very high levels of performance and reliability which are achieved today. Thanks in part to the gas turbine engine, the world now relies heavily on air transport for passengers and goods, in what is becoming an ever-more globalised planet.

In 2014 there were a total of 38 million registered flights^[1], equating to more than 100,000 flights each day. Of these, 83% used turbofans^[1]. In 2016, the air transport industry was responsible for 2% of the all anthropogenic CO₂ emissions. In 2016, airline companies spent a total of 180 billion GBP on fuel^[1] which equates to 286 billion litres of jet fuel. This is an almost incomprehensible amount but can be visualised as a tank of 1 km square with a height of 286 meter. It can be concluded that any reduction in the fuel consumption of aircraft engines will result in a significant reduction in operating cost for the aircraft operators. As this industry keeps growing, goals have been set by the Advisory Council for Aviation Research and innovation in Europe (ACARE) for 2050 to reduce fuel consumption, CO₂ emissions, and noise^[2].

These goals can be summed up as follows;

- Aircraft fuel use reduced by 70%
- Aircraft noise pollution reduced by 75%
- Smoke and smell eliminated
- CO₂ emissions per passenger kilometre reduced by 75%
- NO_x emissions per passenger kilometre reduced by 90%

Engine manufacturers are continuously investigating methods to further reduce fuel consumption, engine noise, and emissions in future gas turbines. A clear trend emerges from Figure 1, where it can be seen that, as the fan diameter increases, fuel burn and emissions decrease. This means that due to the goals set out for 2050, aircraft manufactures need to move to larger fan diameter gas turbines. This also implies higher BPR engines.

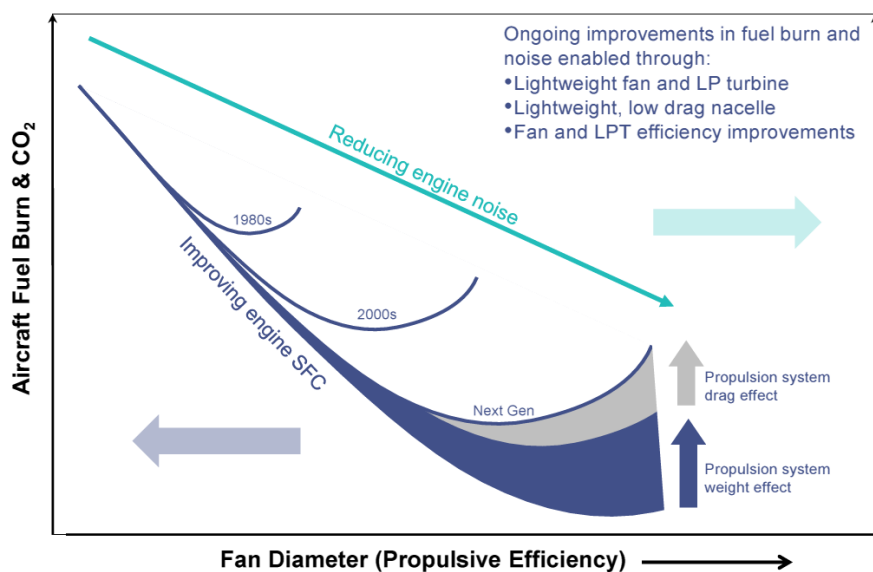


Figure 1 – Large turbofan engine technology trend^[3]

Figure 2 shows the design space of current gas turbines (blue area), and where future gas turbines expected to reside (red area). It can be seen that this area

has higher thermal and propulsion efficiencies. The increase in thermal efficiency is due to the improvement in technology limitations, which are expected to be available to future generations of engines. The increase in propulsion efficiency is due to the ability to increase the BPR, in part also due to the improvement in technology and better integration between the engine and the aircraft.

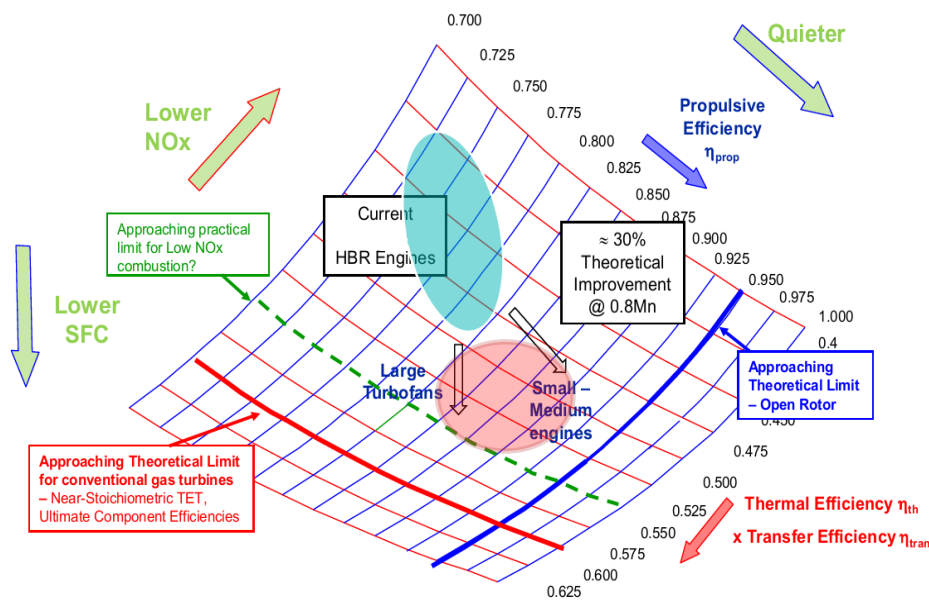


Figure 2 – Schematic of design space for current and future gas turbines^[3]

It is considered that future large turbofan engines, such as the UltraFan™, which will have larger fan diameters, and higher BPRs than current turbofan engines, may offer reduction in mission fuel burn. It is also believed that through the integration of a 'Variable Geometry Exhaust System' (VGES), further improvements may be achieved. The VGES includes a partial mixer, where part of the bypass flow is mixed with the core flow. Mixed exhaust gas turbines are most commonly used in the military in combination with an afterburner to achieve maximum performance benefit. There are examples of civil aircraft using mixed exhaust like the RB211-524G/H and -535E4, but these had small BPR.

The performance benefit from mixing the exhaust of a turbofan engine, compared to separate exhausts for the fan and core streams, diminishes when increasing the BPR. This is due to the fact that the benefit is achieved by increasing the temperature of the cold stream. The potential temperature gain that the cold stream can experience decreases, when the mass flow of the cold stream increases and the core steam mass flow stays constant. Furthermore, when increasing the amount of cold flow, the size of the mixer increases. This in turn increases the frictional losses as well as the weight of the mixer. It is considered that by allowing mixing with just part of the cold flow, the benefit won't diminish as significantly when going to larger BPR mixed exhaust turbofan engines. Furthermore, variable geometry could be applied in the bypass nozzle, core engine exhaust nozzle, and/or in a 'mini-mixer' device positioned between the bypass and core nozzles. It is believed that allowing a certain amount of variability within the exhaust system, improves the stability of the fan during off-design performance. Furthermore, it is understood that by allowing exhaust nozzle area variation, further mission fuel burn reductions may be achieved.

1.2 Aim & objectives

This research was undertaken within the context of the Rolls-Royce University Technology Centre (UTC) in Aero System Design, Integration & Performance at Cranfield University, hosted by the Centre for Propulsion Engineering. The aim is to demonstrate the potential benefits of incorporating variable exhaust nozzle technology, and to investigate the various trade-offs between architectural and cycle parameters, in order to achieve an optimum solution in terms of mission fuel burn and/or operability in regards to fan surge. At the same time, the study needs to understand how a VGES could be implemented in practice, and whether it poses any significant aerodynamic issues with the rest of the engine. This research does not look into the effect a VGES may have on engine emissions or on engine noise. The VGES that will be investigated is shown below in Figure 3.

In this there are four different areas all of these areas could change independently. Furthermore, it can be seen this concept included partial mixing where part of the bypass flow is mixed with the core flow.

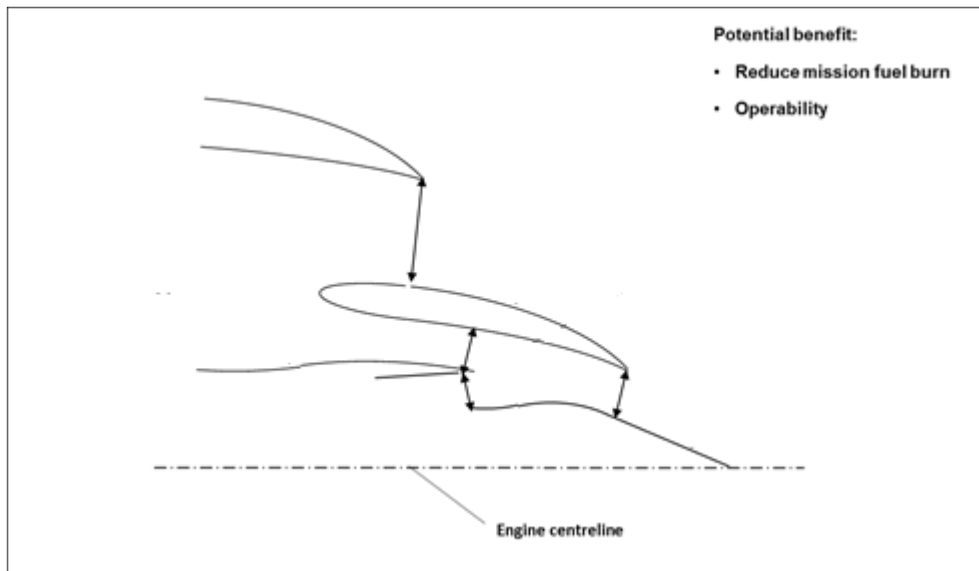


Figure 3 – Variable Geometry Exhaust System^[50]

The technology level of the low Specific Thrust turbofan engines considered in this study is appropriate for an engine/aircraft expected to enter service between approximately 2025 and 2030. The engine thrust range considered is between 70,000-80,000 lbf and the mission flight profile consistent with a 3000 nm range, A330-size aircraft. The aim of this study is to compare two engine architectures; the 3-spool geared, and 3-spool direct-drive. These two architectures have been chosen on the basis of their very different LP turbine sizes and stage counts, which impact bypass duct design, nozzle geometry and length of the engine. Both of these engine architectures are low Specific Thrust turbofan engines, incorporating a VGES.

There are three main objectives as shown below:

1. Create thermodynamic engine models of a 3-spool geared, and 3-spool direct-drive, low Specific Thrust turbofan engine, equipped with a VGES.
 - a) Take effects of mixing into account
 - b) Take pressure losses into account
 - c) Incorporate nozzle variability
2. The derivation of optimum exhaust system architectures and nozzle area schedules, to achieve best mission fuel burn (and/or improved operability in regards to fan surge), within the constraints of EIS, between 2025 and 2030 and the mission profile.
 - a) Create an optimiser to establish the optimum specific fuel consumption (SFC) within given technology limitations
 - b) Set all mixer and engine characteristics variable within the optimiser for a full exploration of the design space
 - c) Include mixer weight estimation in fuel burn study
3. The investigation of any major constraints at engine level resulting from aerodynamic interaction considerations.

1.3 Research scope & roadmap

This research is focused on the 'Variable Geometry Exhaust System' (VGES). The investigation centres on two different engine architectures. Both these engine architectures are considered to be good candidates for the UltraFan™. The UltraFan™ has been trademarked by Rolls-Royce and is intended to be a very low specific-thrust turbofan. The main difference between these architectures is the gearbox. However, this research does not focus on the gearbox, as previous research within the UTC has already focussed on that topic^[5]. The current research focuses only on the VGES, and how to correctly simulate its effects on engine performance and mission fuel burn. The VGES has 4 variable areas, and a mini-mixer which mixes part of the bypass flow with the core exhaust flow.

In regards to the VGES the following areas are included in the research scope:

- Mini-mixer loss prediction modelling
- Mini-mixer mixing efficiency thermodynamic modelling
- Mini-mixer weight estimation
- Effect of mini-mixer on engine fan surge
- Effect of variable areas

The areas which are not within the scope of this research are listed below. These are either irrelevant to the research aim, or have previously been researched.

- LPT and Fan efficiency methodology
- Cooling flow methodology
- Fan flutter
- Engine emissions
- Engine noise

Fan flutter could be considered part of the operability study of a gas turbine, but do to the nature of this research where the turbofan models that were created using the industrial partner data and without them providing the information regarding fan flutter it would be very difficult asses where fan flutter would occur on the compressors maps (which were also provided). Models for fan flutter prediction, applicable to this research, were not available. Creating these models would take more time than available and would require a dedicated study, beyond the scope of this research. It should also be noted that the focus of this study was mission fuel burn and part load or close to idle operations are not normally considered in this stage of the design process.

This was a three year research project. The project roadmap was established and agreed upon through discussions with the industrial partner and the academic supervisor. The project roadmap is shown Figure 4.

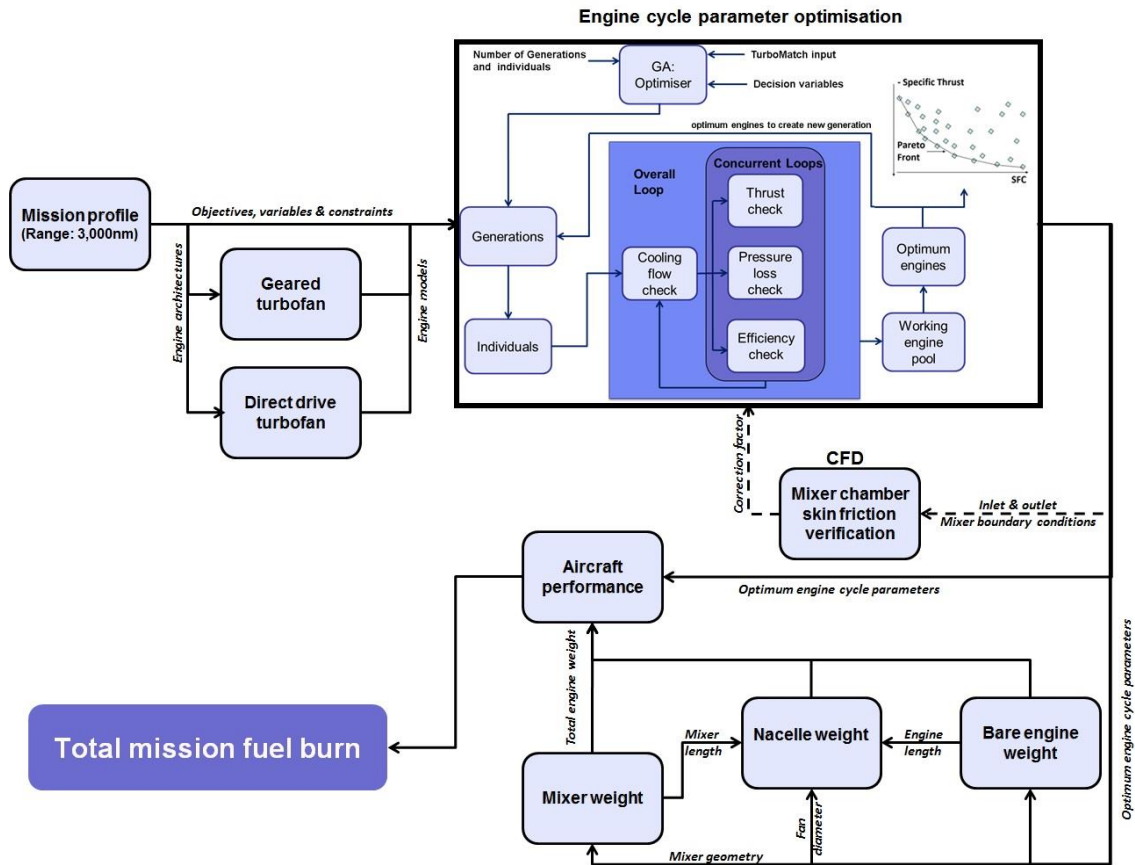


Figure 4 – Assessment Framework and Methodology

The first year of the research focused mainly on the creation of appropriate thermodynamic engine models for the geared and direct-drive turbofan, and their verification and validation. In addition to this, *Optimiser 1.0* was created to handle the performance optimisation of both engine architectures. The objective of the optimiser is to find the optimum engine architecture in regards to lowest mid-cruise SFC. The optimiser schematic can be seen in Figure 4 and the optimisation setup and requirements (objectives, variables and constraints) are given in chapter 3.3. *Optimiser 1.0* was only able to accommodate fixed values for losses and efficiency, and was designed to run on local machines. As a result the optimisation took a substantial amount of time to complete and did not have the level of accuracy that was desired.

The second year focussed on the exploration of the available design space, and the identification and shortlisting of optimised cycles for the two different engine architectures, namely the direct-drive and geared 3-spool turbofan. Furthermore, a significant amount of time was also spent on developing the methodology regarding modelling correctly and accurately the mixer in the two engine architectures, with particular focus on the losses experienced within the mixer. This step also involved developing *Optimiser 2.0*, capable of running on the high-performance computing facility at Cranfield, as well as including variable cooling flow, component efficiencies and mixing losses. Furthermore, a methodology was developed for the high-fidelity CFD analysis of the envisaged VGES, to satisfy thermodynamic and performance requirements.

The last year of the research involved an iterative loop between the high-fidelity CFD analyses and the optimiser. What this inclines is that first an optimisation was run using equations created to predict the skin friction losses within the mixing chamber. The results of that optimisation would then be used to create the boundary conditions for the CFD simulations. The outcome of the CFD simulation would then be compared against the optimisation results. From the comparison a correction to the equations were found and implemented into the optimisation. The optimiser would then be run using the newer equations and the results of the optimisation were used as inputs for CFD, this would continue until mixer skin friction losses showed agreement between the optimiser and the CFD simulations, the final cycle optimisation could then be carried out. Furthermore, a mixer weight estimation methodology was defined and included into the overall weight estimation this is discussed in chapter 3.5. The research ended with a full mission profile investigation for both engine architectures which is discussed in chapter 3.7. This made it possible to identify potential areas of benefit in the two architectures when using a VGES.

1.4 Contribution to knowledge

This research programme has made a useful contribution to knowledge in various areas of engineering and scientific relevance. In a nutshell, the viability of a novel piece of technology was investigated, with potential benefits as well as practical limitations identified. The influence of mixing efficiency and pressure loss in a complex, mini-mixer arrangement were studied and a useful methodology was established to identify performance trade-offs. Also, the study looked into the effect on high-bypass ratio turbofan engine performance of using multiple variable nozzle areas simultaneously.

More specifically, this research contributes to a new toolkit which was developed at Cranfield University. This toolkit is capable of optimising multiple engine configurations, for minimum mission fuel burn. Following this author's work, in this toolkit there are now options available to implement nozzle area variability, and the mixing of core and bypass airflows. This toolkit has been created in collaboration with other research groups within the Rolls-Royce UTC at Cranfield.

The performance effects of nozzle area variation have been investigated in the past, and data are available in the public domain. There is also data available in the public domain regarding mixing of core and bypass flow. However, there are no data available on the partial mixing of bypass flow, although there are patents for this concept. It is believed that engine manufactures have their own data and software to analyse this configuration, but no data have been made publicly available because of the sensitivity of this area.

In the public domain there is information available in regards to certain characteristics being affected by exhaust mixing. However, these results have never been investigated over a wide range of mixing efficiency. The results of this

research will contribute to a better understanding of how to simulate a more representative mixer, taking mixing efficiency, mixing inlet mixer Mach number and all the losses associated with mixing into account.

The findings of this research contribute to the understanding of the effects on turbofan engine performance of partial mixing of bypass flow with core flow. This research also sheds light into the potential benefits for future ultra-high bypass ratio engines.

1.5 Thesis structure

The contents of this thesis have been grouped into further chapters and subchapters. A brief description of each main chapter is given hereafter. Chapter 2 contains the literature review, summarising the knowledge required to understand the research that has been conducted. The contents of the literature search include theory on cycle design for separate exhaust and mixed exhaust gas turbines, with particular focus on the mixing section. Furthermore, Chapter 2 includes information on the flow mixing capability of current, state-of-the-art performance software, as well as on variable geometry nozzles, engine operability, and optimisation process.

Chapter 3 presents the methodology that was used or developed during this research. The focus of this chapter is on the thermodynamic models, the optimiser, the weight estimation tool and integrated aircraft simulation tool. A significant part of this chapter is devoted to the explanation on how the VGES mini-mixer was accurately simulated, while taking all the losses into account. The chapter concludes with how certain parts of the methodology were verified and validated.

Chapter 4 presents the results of the verification of the methodology. Furthermore, this chapter shows the results of various sensitivity studies conducted throughout the research.

Chapter 5 discusses the main results of this research. These include the performance optimisation results and comparisons between the baseline engines, and the engines with VGES. Additionally, a comparison is made between the direct-drive and geared engine architectures. This chapter also presents an engine operability study in regards to compressor surge, the weight estimation and finally the mission fuel burn results.

Chapter 6 summarises the key conclusions the author has drawn from the results. Additionally, the contribution to knowledge is highlighted. The chapter finishes with recommendations for future research and further interesting areas to explore.

2 Literature review

In order to compare a turbofan engine equipped with VGES to one without, one must first understand how to carry out a preliminary cycle design for aero gas turbine engines. Furthermore, a deeper understanding of mixed exhausts and the effects they have on an aero engine cycle design is required. The following chapter guides the reader through the knowledge required to understand the research topic, this includes turbofan performance optimisation and optimisation strategies. It clarifies certain choices and assumptions made in the later chapters. It identifies the knowledge gap from previous research as well as highlight previous R&D related to technologies for reducing the impact of civil aviation on the environment.

2.1 Recent research into environmental performance

Recently environmental performance has become a very important issue especially with the ACARE setting goals for the industry to follow. This research focusses on one particular technology which potentially reduces engine environmental impact by reducing SFC and overall mission fuel burn. There have been many other attempts towards this, focusing on other engine technologies but with the same end goal of reducing the environmental impact of aviation.

In a paper published in 2015^[6] an overview from the last 15 years is given of the technologies developed primarily in EU and UK programmes. Furthermore, this paper also discusses the possible future technological developments that are most likely to be available by 2035, when the current worldwide aviation fleet reaches its scheduled renewal cycle. One of the main conclusions of the paper is that the NO_x reduction required by 2020, to satisfy the ACARE 2020 goals, will

most likely be achieved. It also states that by 2035 the CO₂ reduction will be about 25%, this is of course lower than what is desired by 2050 as the FP2050 target has been interpreted by RR as a 30% CO₂ reduction from engine technologies alone (which means the airframers still need to do a lot to meet the overall 75% reduction). Furthermore, the paper predicts that the goal set out for noise reduction will not be achieved either. It needs to be noted that the prediction as to whether the current environmental targets will be achieved by 2050 is not an exact science and future 'unforeseen' developments may make these targets possible.

To give the reader a better understanding of some of the most recent research in the EU, aiming at improving aero engine environmental performance, a couple of case studies are discussed hereafter. The first one is the 'Efficient and Environmentally Friendly Aero-Engine' (EFFAE CLEAN) project^{[7][8]}. This research focused on the study of intercooled-recuperated aero engine cycles. This project also involved some experimental activity with a module of the recuperator tested alongside a gas turbine engine core. Follow-on research, namely in 'New Aero Engine Core Concepts' (NEWAC) and 'Low Emissions Core engine Technology' (LEMCOTEC) studied advanced cycles even further. The Aristotle University of Thessaloniki continued with the development of the recuperator and in particular with the reduction of pressure losses^[9] across it. The intercooler section of the cycles was investigated by a multitude of universities and companies^[10]. The overall conclusion of NEWAC was that for a long-range aircraft there can be a reduction of CO₂ emissions and mission fuel burn. However, the improvement predicted from technologies researched in that programme was lower than the original goal of 6%^[11]. The largest benefit was found when combining geared fan configurations with intercooling. However, there were some unforeseen effects with increased weight and drag and reduced core component efficiencies which resulted in the amount of mission fuel burn reduction to be lower than targeted.

Another recent example worth mentioning is the LEMCOTEC research project focused on the reverse-flow-core intercooled engine^[13]. Within LEMCOTEC, Cranfield University assessed the feasibility of a very high overall pressure ratio, intercooled, geared turbofan with a reverse flow core. For this configuration the low pressure shaft does not pass through the high pressure spool. This means that the efficiencies of the turbine and compressor of the two spools are not compromised by high hub/tip radius ratios and low rotational speeds. Subsequently, this configuration could benefit from mixed exhaust arrangements^{[14][15]}. It was concluded that there was some SFC improvement at cruise and reduction of mission fuel burn. However, like the previous studies discussed above, the improvement of 0.7% reduction in SFC at cruise was less than originally expected.

In general, the available literature makes it very clear that a bigger effort and bigger investment in new, evolutionary or revolutionary technologies are needed towards achieving the very ambitious ACARE 2050 goals. The author's work was instigated based on this premise.

2.2 Turbofan with separate exhaust

Since the advent of the gas turbine engine, engine cycle performance analysis has been researched and discussed countless of times. The goal of a turbofan engine cycle design is to achieve the required thrust, with the minimum fuel consumption. This can be done by choosing the correct values for specific parameters. There are many textbooks that comprehensively cover the practical and theoretical design principles required for a turbofan engine cycle. A good description of the major design and performance optimisation principles is provided by Saravanamuttoo^[16]. There is an extensive design point parametric

analysis taking different constraints into account, which is described in Pilidis 2012^[17].

The most critical operating points throughout the flight envelope are taken into consideration when designing an aero engine cycle. These are typically the End of Runway (EOR), the Top of Climb (TOC) and, Cruise (CR). Furthermore, limitations and constraints based on engine stability and technological limits need to be taken into consideration. There are multiple combinations of engine parameters that will produce an aero engine cycle. However, to produce the cycle which gives the lowest specific fuel consumption (SFC) and maximum thrust, one can follow the methodology outlined in the papers of Guha^{[18][19]}. The paper states that the parameters shown below are key to achieving optimum cycle design, however, it should be noted that the pressure ratio split between the booster and HPC were not considered for the optimisation study.

- Fan Pressure Ratio (FPR)
- Bypass Ratio (BPR)
- Overall Pressure Ratio (OPR)
- Combustor Outlet Temperature (COT)
- Specific Thrust ($\hat{F}_N$ or SpT)

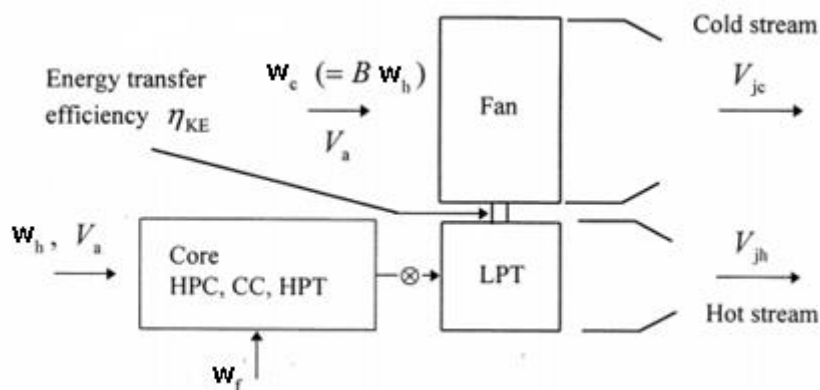


Figure 5 – Schematic description of separate-stream turbofan engine^[18]

To support the following analysis, a schematic description of a turbofan engine is shown in Figure 5. From the figure it can be seen that when fuel is added to the core of the engine, it increases the available energy of the gas in the core engine. The available energy is then partially used by the high pressure turbine (HPT) to power the high pressure compressor (HPC). Another part of the available energy is used by the low power turbine (LPT) to rotate the fan. The fan will transfer the energy to the bypass airflow. This flow is then expanded through the bypass nozzle to create bypass thrust. The remaining energy in the core flow is also expelled through the core nozzle to produce core thrust.

2.2.1 Separate exhaust turbofan cycle design

From a simplified analytical point of view there are two methods for finding the optimum design point^[18]. The first method is to fix a value for Specific Thrust which allows one to vary the BPR, and the second method is to fix a value for BPR which allows varying the Specific Thrust. By using these methods it can be seen that for every specific value of fixed BPR or Specific Thrust there will be a corresponding value for FPR, OPR and COT that gives the lowest SFC. To determine the correct FPR, the equation (2-16) has been developed by Guha^[18], which this analysis will work towards.

When designing an engine cycle one must always take the overall engine efficiency (η_{overall}) into account. This is defined in equation (2-1). It can be seen that overall engine efficiency is determined by the thermal efficiency (η_{Th}), and the propulsion efficiency (η_{Prop}). The η_{Th} is defined in equation (2-2), and η_{Prop} is defined in equation (2-3). One thing to note is that equations (2-2) and (2-3) are always in conflict. This is due to the fact that by increasing thermal efficiency the jet velocity increases, however this decreases propulsion efficiency. The equivalent happens in reverse: if the jet velocity decreases, the propulsion efficiency increases, but the thermal efficiency decreases. By going to higher fan

diameters the jet velocities are reduced. Therefore, this increases propulsion efficiency but reduces thermal efficiency.

$$\eta_{overall} = \eta_{Th} \times \eta_{Prop} = \frac{(Bypass\ thrust\ power + Core\ Thrust\ power)}{Fuel\ Power} \quad (2-1)$$

$$\eta_{Th} = \frac{Core\ power}{Fuel\ Power} \quad (2-2)$$

$$\eta_{Prop} = \frac{(Bypass\ thrust\ power + Core\ Thrust\ power)}{(Core\ Jet\ kinetic\ Power + Bypass\ Jet\ Kinetic\ Power)} \quad (2-3)$$

Taking equations (2-1), (2-2), (2-3), gives equation (2-4) for F_N and equation (2-5) for jet kinetic power, assuming full expansion in the nozzles. Where V_j is the exit jet velocity, the subscript c (for cold) represents the exit velocity of the bypass stream, the subscript h (for hot) represents the exit velocity of the core stream V_a is the inlet air velocity and W represents the mass flow. In equation (2-4) and (2-5) the fuel mass flow is neglected as it is only a very small portion of the total mass flow.

$$F_N = W_h (V_{jh} - V_a) + W_c (V_{jc} - V_a) \quad (2-4)$$

$$Jet\ Kinetic\ Power = \frac{1}{2} W_h (V_{jh}^2 - V_a^2) + \frac{1}{2} W_c (V_{jc}^2 - V_a^2) \quad (2-5)$$

The next definition required is for the bypass ratio. This is shown below.

$$BPR = \frac{W_c}{W_h} \quad (2-6)$$

It can be seen in Figure 5 that there is a transfer of kinetic energy between the core and the bypass. This is done through the LPT and the bypass section of the fan. The efficiency of the power transfer efficiency (η_{tr}) is given by the fan isentropic efficiency (η_f), the LPT isentropic efficiency (η_t), the bypass nozzle

isentropic efficiency (η_{NB}), flow losses in the duct, and the mechanical efficiency. To simplify this; it is assumed the flow losses in the duct, the mechanical efficiency, and the bypass nozzle efficiency are close to 1. Therefore, the power transfer efficiency can be defined as follows.

$$\eta_f \times \eta_t \cong \eta_{tr} \quad (2-7)$$

By combining equations (2-5), (2-6) and (2-7) the following equation for core power is defined.

$$Core\ power = \frac{1}{2} W_h (V_{jh}^2 - V_a^2) + \frac{1}{2} W_h \times \frac{BPR}{\eta_{tr}} (V_{jc}^2 - V_a^2) \quad (2-8)$$

It is necessary to determine the specific fuel consumption of the turbofan engine cycle. This can be done through equation (2-9). In this equation, W_f is the fuel mass flow.

$$SFC = \frac{W_f}{F_N} \quad (2-9)$$

To find the optimum design point with the lowest SFC, it is imperative to find the optimum velocity ratio between the hot and cold stream. To achieve this, the designer must maximise the net thrust while keeping the fuel flow fixed or minimise the fuel flow while keeping the thrust fixed. The maximum thrust condition is given by:

$$\frac{\partial F_N}{\partial V_{jh}} = 0 \quad (2-10)$$

Assuming a constant thermal efficiency, BPR and a fixed fuel flow will result in a fixed core power. This gives:

$$\frac{\partial (Core\ power)}{\partial V_{jh}} = 0 \quad (2-11)$$

Equations (2-8), (2-10) and (2-11) can provide a solution for the jet velocity ratio assuming a constant BPR. The references of Saravanamuttoo^[4] and Pilidis^[17]

show that equation (2-12) will result in maximum thrust and minimum fuel consumption.

$$\left(\frac{V_{jc}}{V_{jh}} \right)_{optimum} = \eta_f \times \eta_t = \eta_{tr} \quad (2-12)$$

Equation (2-12) can be satisfied through the choice of FPR. The optimum fan pressure ratio $(FPR)_{op}$ can be achieved by the choice of Specific Thrust ($\hat{F}_N$) and BPR. The reason for this is that $(FPR)_{op}$ can be expressed as a function of ($\hat{F}_N$) and BPR. However, first the mean jet velocity (V_m) needs to be defined, which is shown in equation (2-13). Subsequently, $\hat{F}_N$ needs to be defined, which is shown in equation (2-14).

$$V_m = \frac{1}{1 + BPR} \left(BPR + \frac{1}{\eta_{tr}} \right) V_{jc} \quad (2-13)$$

$$\hat{F}_N = \frac{F_N}{W_c + W_h} = V_m - V_a \quad (2-14)$$

By assuming that the total temperature rise in the bypass flow is isentropic the static temperature of the fully expended jet would be equal to the ambient temperature. When applying this in the steady flow energy equation gives equation (2-15).

$$\Delta T_{0,isen} = \frac{1}{2C_p} (V_{jh}^2 - V_a^2) \quad (2-15)$$

When combining equations (2-13), (2-14), (2-15) and with the isentropic pressure-temperature relationship the equation for $(FPR)_{op}$ can be obtained. This is shown in equation (2-16).

$$(FPR)_{op}^{\frac{\gamma-1}{\gamma}} = 1 + \frac{(\gamma-1)}{2 + (\gamma-1)M^2} \left[\frac{(1 + BPR)^2}{(BPR + \frac{1}{\eta_{tr}})^2} \left\{ \frac{\hat{F}_N}{\sqrt{\gamma RT_a}} + M \right\}^2 - M^2 \right] \quad (2-16)$$

It can be seen in Guha^[18] that the BPR effect is very small on the FPR, especially in regards to turbofan engines with BPR higher than 4. The Specific Thrust is the most dominant variable in determining the correct value for FPR, whereas the OPR does not have any effect on the $(FPR)_{op}$.

Taking the above equations into account, if one was to specify constant fuel flow and constant inlet mass flow and combustor outlet temperature (COT). This would result in only three variables available to control during the cycle design. These variables are: the Specific Thrust, OPR, and BPR. Engine cycles are designed for a net thrust requirement and to achieve minimum SFC. For a fixed specific thrust by choosing the OPR there is a specific combination of BPR and COT which will result in the lowest SFC value for that specific thrust. This means that there are only two parameters left for the designer OPR and Specific thrust. From the literature it was noted that there is a second design method: for fixed BPR. However, this method will not be discussed as it does not meet the requirements for this research of minimizing the SFC while maximizing the specific thrust. For comparison the two methods are compared for a set of specific component isentropic efficiencies, altitude, Mach number and OPR range, as shown in Figure 6.

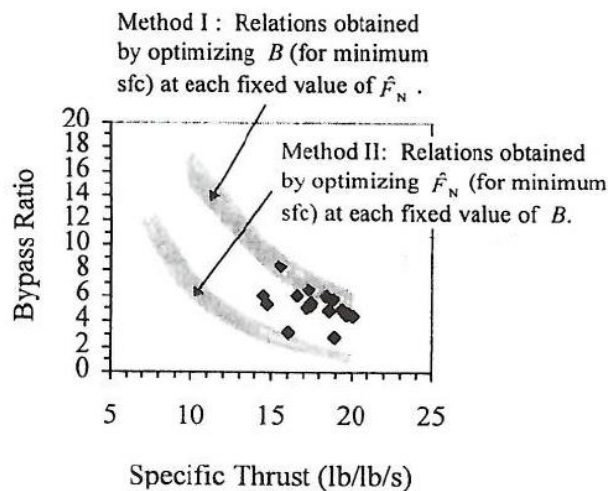


Figure 6 – Results from different turbofan engine design methods^[18]

From the figure it can be seen that method one produces engine cycles that have higher BPR for the same specific thrust value, which means that those engine cycles have higher propulsion efficiency. Furthermore, it can be seen that the symbols on the graph which represented current engines data shows that most current engines fall within or are very close to area of method one. From this graph it can be concluded that both methods produce different results. It can't be concluded which method produces the lowest SFC from this graph as it is not shown. However, dependent on the requirements the designer must recognise which method would best suit the engine that is being designed, as other aspects might influence his choice, like engine drag and weight.

2.3 Turbofan with mixed exhaust

For the mini-mixer in the VGES concept, an understanding of the mixed exhaust turbofan is needed. A mixed exhaust turbofan engine is defined as a turbofan engine where the bypass flow has been mixed into the core exhaust flow exiting through a common nozzle. The visual difference between a mixed exhaust and a separate exhaust configuration is shown in Figure 7.

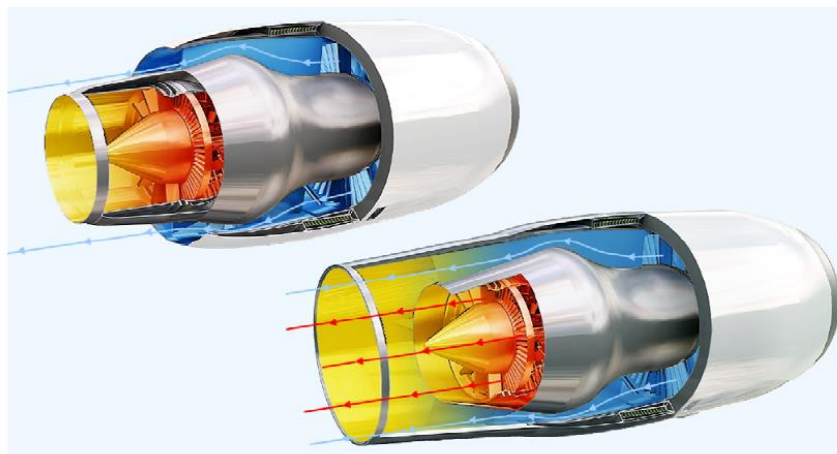


Figure 7 – Difference between mixed exhaust and separate exhausts^[20]

The mixed exhaust has commonly been used in gas turbines for military purposes, as well as a wide range of civil aero gas turbines, such as the RB211-524G/H, RB211-535E4, and Trent 700^[21]. The foundation of this type of gas turbine comes from research performed in the 1960s by Frost^[22], Pearson^[23] and Hartman^[24]. Over the years more research has been conducted into mixed exhaust gas turbines^{[25][26][27][28][29][30][31]}. The idea is that exhaust mixing enables the core exhaust stream (hot/fast) to transfer energy to the bypass exhaust flow (cold/slow) without the use of mechanical means. The goal is to increase in thrust in design point but also in other operational points. This is done by increasing the temperature of the bypass stream, therefore increasing the average exit velocity of the common nozzle^[32].

2.3.1 Advantages of mixed exhaust turbofans

Previous research has highlighted the reasons for considering the use of a mixed exhaust compared to separate exhaust jets. The advantages are as follows ^{[34][35][36]}:

- Using an afterburner in combination with a mixed exhaust gives a far greater thrust increase (military application)
- At cruise small Specific Thrust and SFC improvements may be achieved when cycle is specifically designed for mixing
- Optimum FPR is reduced (which reduced component weight of the fan)
- Reduction of jet noise is related to the jet velocity; mixed jet streams have lower jet velocities than unmixed core exhaust

For a given turbofan, the optimum FPR relates to a certain amount of energy conveyed between the core and the by-pass flow. In a separate-exhaust turbofan this energy needs to be transferred through the LPT. However, for a mixed exhaust turbofan some of the conveyed energy can be transferred through the mixer, increasing the transmission efficiency, and therefore reducing the LPT pressure ratio and the optimum FPR. Furthermore the reduction in FPR will cause

a reduction in fan weight, this is due to the fact that the fan component is under less stress because the RPM is lower, and this can be seen as an advantage because it can offset the weight that is added to the mixed exhaust system. The thrust increase achieved by the mixed exhaust is commonly referred to as theoretical thrust gain. Theoretical thrust gain is defined as the difference in thrust between a separate exhaust engine and a mixed exhaust engine of the same architecture. This is shown in equation (2-17). In this equation, F_{Nmixed} is equal to the thrust produced by the mixed exhaust engine, and $F_{Nc} + F_{Nh}$ is equal to the thrust produced by the same engine cycle but then as a separate exhaust engine.

$$gain_{theoretical} = \frac{F_{Nmixed}}{F_{Nc} + F_{Nh}} - 1 \quad (2-17)$$

Previous research has also shown that the theoretical thrust gain that can be achieved through the use of a mixer can be between 2-3% in ideal conditions^[22]. The amount of achievable theoretical thrust gain is highly dependent on several other factors that need to be taken into account, which are added weight, complexity and losses. These are also the disadvantages of the mixed exhaust system, and the reason why not all gas turbines are equipped with mixed exhaust.

The theoretical thrust gain that can be achieved depends on a number of factors. These will need to be considered when designing a gas turbine engine cycle with an exhaust mixer. The first factor that needs to be taken into account is that the static pressures at the inlet of the mixer need to be equal. The second factor that needs to be considered is the matching of total pressures at the inlet of the mixer at design point. The reason for this is that if there is a significant mismatch of total pressure at the inlet of the mixer there will be a reduction instead of an increase in thrust, which can be seen in Figure 8. There is also the possibility that if the total pressures are too different at design point, the mixer will not work as intended at off-design conditions. The last factor that needs to be considered is the effect of the exhaust temperature ratio T_h/T_c . When this ratio increases, the

theoretical thrust gain that can be achieved also increases, which can be seen in Figure 9. Changes in off-design fan flow will also affect fan efficiency and overall performance.

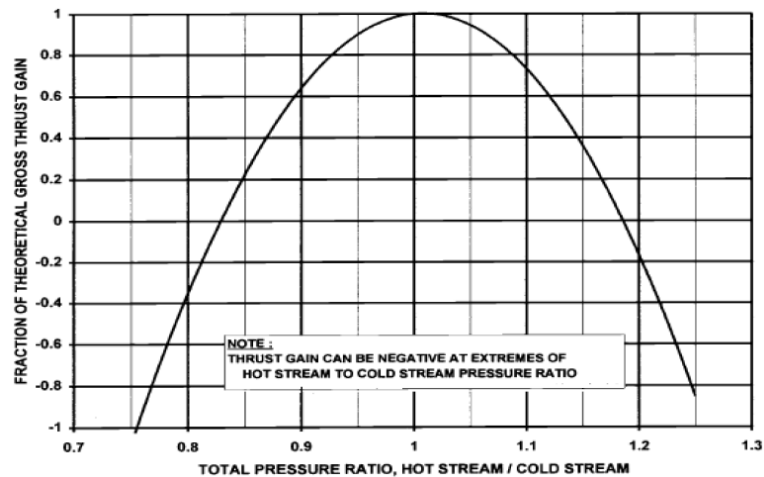


Figure 8 – Effect of inlet mixer total pressure ratio on theoretical gross thrust gain^[36]

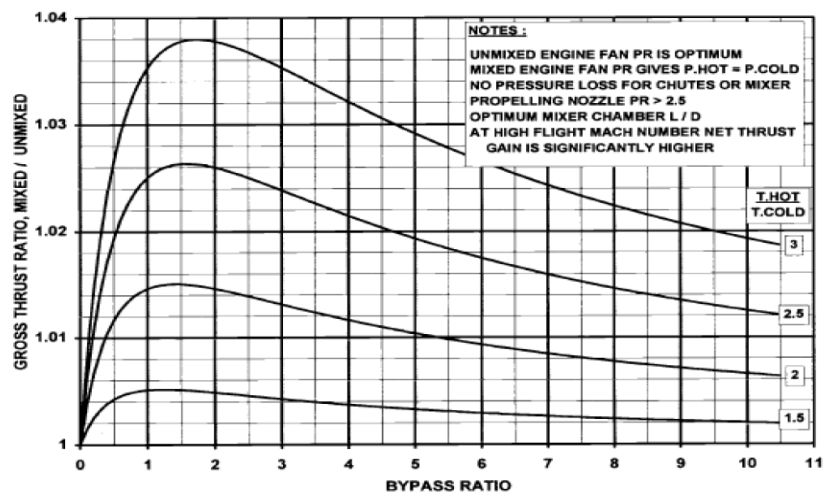


Figure 9 – Effect of bypass ratio and temperature ratio T_h/T_c ^[36]

With all these considerations the cycle design for the mixed exhaust turbofan engine is slightly different than that of a regular turbofan engine. These methods

have been extensively explained by Frost^[22], Pearson^[23] and Hartmann^[24]. The method for finding the correct FPR is slightly different from an unmixed exhaust engine cycle, which is extensively explained in Guha^[37] and Sanghi & Lakshmanan^[38]. The following section will work towards the set of performance equations that are necessary when taking mixed exhausts into account.

2.3.2 Calculating the flow conditions in the mixing chamber

Mixing of exhaust flows is achieved through the use of a mixing chamber; a schematic representation of one is shown in Figure 10. This shows that the bypass flow starts to mix with the core flow from the chute exit line. It can also be seen that the flow is not instantaneously mixed, and it takes a considerable length of the chamber for the two flows to become mixed. However, one can also note that some extra mixing takes place beyond the mixing chamber.

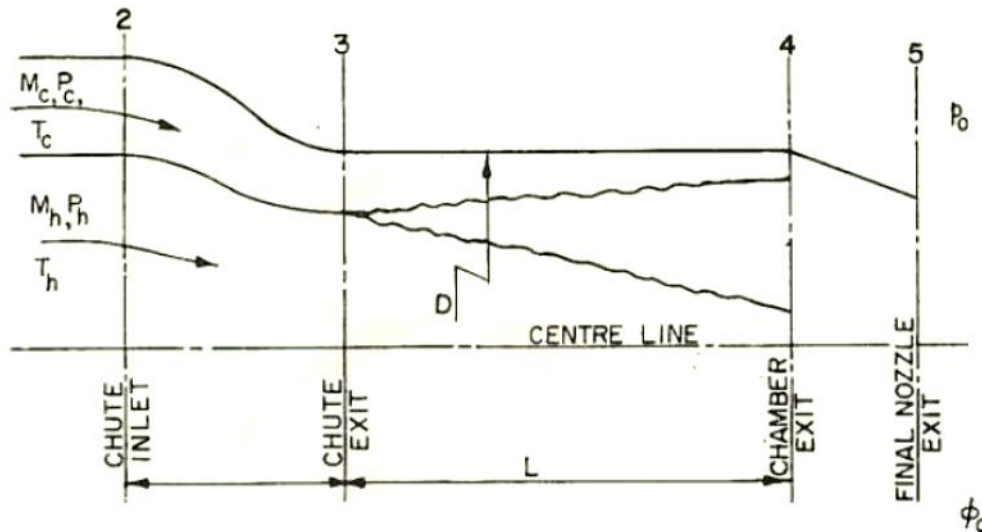


Figure 10 – Schematic of mixing process^[22]

The flow conditions of exit of the mixer can be calculated, if the inlet conditions, and areas of the mixing chamber are known, and the following assumptions are made^{[22][33]}.

- One-dimensional axial flow conditions
- Wall friction is disregarded
- Streams fully mixed
- Static pressure constant at the inlet of the mixer
- Total pressure constant in plane 4
- Total pressure constant in each stream in plane 3

Using the conservation of momentum for the fully mixed flow at plane 4, the following equation results;

$$W_c V_{c3} + A_3 p_3 + W_h V_{h3} = (W_c + W_h) V_{4m} + A_4 p_4 \quad (2-18)$$

Where W is equal to the massflow, A is equal to the area, p is equal to static pressure and V is equal to velocity. The subscript of c refers to the cold stream, and the subscript h refers to the hot stream as previously. The last subscript m refers to the mixed stream. By means of the flow momentum equation (2-19) and dividing equation (2-18) by $M_c \sqrt{T_c}$, equation (2-20) is created, where T equals to total temperature.

$$I = \frac{(WV + Ap)}{W\sqrt{T}} \quad (2-19)$$

$$I_{c3} + \frac{W_h \sqrt{T_h}}{W_c \sqrt{T_c}} I_{h3} = \left(1 + \frac{W_h}{W_c}\right) \sqrt{\left(\frac{T_m}{T_c}\right)} I_{4m} \quad (2-20)$$

With the use of the continuity equation the area of plane 4 is calculated.

$$A_{c3} + A_{h3} = A_4 \quad (2-21)$$

Dividing equation (2-21) by $M_c \sqrt{T_c}$, and using equation (2-22), where Q is equal to the flow capacity and P_t is equal to total pressure, will result in equation (2-23).

$$Q = \frac{W\sqrt{T}}{AP_t} \quad (2-22)$$

$$\frac{1}{Q_{c3}} + \frac{1}{Q_{h3}} \frac{W_h \sqrt{T_h} P_{c3}}{W_c \sqrt{T_c} P_{h3}} = \frac{1}{Q_{4m}} \left(1 + \frac{W_h}{W_c}\right) \sqrt{\left(\frac{T_m}{T_c}\right)} \frac{P_{c3}}{P_{4m}} \quad (2-23)$$

Using the heat balance equation yields the following;

$$W_c \overline{C_{pc}} T_c + W_h \overline{C_{ph}} T_h = (W_c + W_h) \overline{C_{pm}} T_m \quad (2-24)$$

Dividing the heat balance equation with $\overline{C_{pc}} M_c T_c$, where $\overline{C_p}$ is equal to specific heat at constant pressure, results in the following equation.

$$1 + \frac{M_h \overline{C_{ph}} T_h}{M_c \overline{C_{pc}} T_c} = \left(1 + \frac{M_h}{M_c}\right) \frac{\overline{C_{ph}} T_m}{\overline{C_{pc}} T_c} \quad (2-25)$$

Additionally, a connection is needed between $\frac{W_c}{W_h}$ and $\frac{A_{c3}}{A_{h3}}$, which will result in equation (2-26).

$$\frac{W_h}{W_c} = \frac{Q_{h3}}{Q_{c3}} \frac{A_{h3}}{A_{c3}} \frac{P_{h3}}{P_{c3}} \sqrt{\left(\frac{T_c}{T_h}\right)} \quad (2-26)$$

With the previous equations, an initial guess for $(M)_{4m}$, the compressible flow table and one of the following dataset $\frac{A_{c3}}{A_{h3}}, \frac{P_{h3}}{P_{c3}}, \frac{T_h}{T_c}$ or $\frac{W_h}{W_c}, \frac{P_{h3}}{P_{c3}}, \frac{T_h}{T_c}$, the designer can iterate to obtain all the flow conditions.

2.3.3 Mixed exhaust cycle design

The following analysis will show how to calculate the optimum FPR for a mixed turbofan engine with fixed BPR, or fixed Specific Thrust. Figure 11 shows the standard numbering used for mixed exhaust turbofan engines. This numbering will be used in the following analysis.

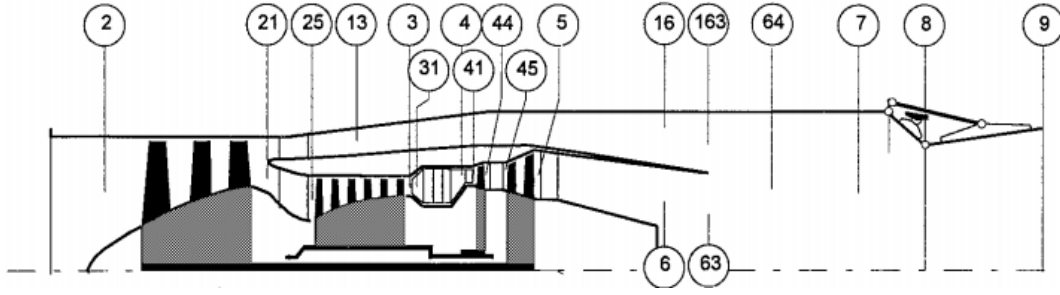


Figure 11 – Schematic for mixed exhaust turbofan engine^[37]

For the subsequent analysis these values for air and turbine exhaust gasses are assumed: $R = 287 \frac{\text{J}}{\text{kg} \cdot \text{K}}$, $\gamma = 1.4$, $C_p = \frac{\gamma \times R}{(\gamma - 1)}$, $\gamma_g = 1.33$, and $C_{pg} = \frac{\gamma_g \times R}{(\gamma_g - 1)}$. The total temperatures for specific station numbers in the schematic can be calculated using equations (2-27), (2-28), (2-29), and (2-30). These are needed to calculate the optimum FPR.

$$\frac{T_{02}}{T_a} = 1 + \frac{1}{2} \times (\gamma - 1) \times M^2 \quad (2-27)$$

$$T_{03} - T_{02} = \left(\frac{1}{\eta_c} \right) \times T_{02} \times \left[OPR^{\frac{(\gamma-1)}{\gamma}} - 1 \right] \quad (2-28)$$

$$T_{016} - T_{02} = \left(\frac{1}{\eta_f} \right) \times T_{02} \times \left[FPR^{\frac{(\gamma-1)}{\gamma}} - 1 \right] \quad (2-29)$$

$$T_{04} - T_{06} = \eta_t \times T_{04} \times \left[1 - \left(\frac{OPR}{FPR} \right)^{\frac{(1-\gamma_g)}{\gamma_g}} \right] \quad (2-30)$$

Using these equations and the schematic shown in Figure 11 and assuming $T_{05} = T_{06}$, $T_{013} = T_{016}$ gives the following equation.

$$(C_{pg}/C_p) \times (T_{04} - T_{06}) = (T_{03} - T_{02}) + BPR \times (T_{016} - T_{02}) \quad (2-31)$$

Equations (2-27), (2-28), (2-29), (2-30), and (2-31) can be combined to give a single equation to find $(FPR)_{op}$. This equation is given below.

$$\begin{aligned} (FPR)_{op}^{\frac{\gamma_g-1}{\gamma_g}} & \left[\frac{BPRT_{02}}{\eta_{fan}} + \frac{\left(\frac{C_{pg}}{C_p}\right)\eta_t T_{04}}{OPR^{\frac{\gamma_g-1}{\gamma_g}}} \right] \\ & = \left(\frac{C_{pg}}{C_p}\right)\eta_t T_{04} - \left(\frac{1}{\eta_c}\right)T_{02} \left[OPR^{\frac{(\gamma-1)}{\gamma}} - 1 \right] + \frac{BPRT_{02}}{\eta_f} \end{aligned} \quad (2-32)$$

To calculate the specific thrust (2-40) first three other equations are needed; namely equations (2-33), (2-34), and (2-35). In these equations V_j is the mixed exhaust exit jet velocity, T_j is the mixed exhaust exit static temperature and T_{064} is the mixed exhaust exit total temperature.

$$V_j = \sqrt{2C_{pg}(T_{064} - T_j)} \quad (2-33)$$

$$T_{064} = \frac{(T_{06} + BPR \times T_{016})}{(1 + BPR)} \quad (2-34)$$

$$T_j = T_{064} \left(FPR_{op} \times \left[1 + \frac{1}{2} \times (\gamma - 1) \times M^2 \right]^{\frac{\gamma}{(\gamma-1)}} \right)^{\frac{(1-\gamma_g)}{\gamma_g}} \quad (2-35)$$

It has to be pointed out that although losses from fans, compressors and turbines have been taken into account, other losses have been neglected for this analysis, for example duct losses. It is possible for equation (2-35) to incorporate incomplete mixing. This is done by introducing a parameter η_{mix} that takes into account three different streams with different total temperatures, but exiting through the same nozzle with the same pressure ratio. The total thrust is then a summation of the thrust of the three individual streams. The equations needed to calculate the incomplete mixed streams are shown below.

$$V_{j16} = \sqrt{2C_p(T_{016} - T_{j16})} \quad (2-36)$$

$$V_{j6} = \sqrt{2C_p(T_{06} - T_{j6})} \quad (2-37)$$

$$T_{j16} = T_{016} \left(FPR_{op} \times \left[1 + \frac{1}{2} \times (\gamma - 1) \times M^2 \right]^{\frac{\gamma}{(\gamma-1)}} \right)^{\frac{(1-\gamma)}{\gamma}} \quad (2-38)$$

$$T_{j6} = T_{06} \left(FPR_{op} \times \left[1 + \frac{1}{2} \times (\gamma - 1) \times M^2 \right]^{\frac{\gamma}{(\gamma-1)}} \right)^{\frac{(1-\gamma_g)}{\gamma_g}} \quad (2-39)$$

$$\hat{F}_N = \eta_{mix} V_j + \frac{(1 - \eta_{mix})(V_{j6} + BPR \times V_{j16})}{(1 + BPR)} - V_a \quad (2-40)$$

Through this analysis the $(FPR)_{op}$ can be calculated, this is a very important parameter in a mixed exhaust turbofan engine. The reason for this is because FPR controls the pressure ratio between hot and cold flow in the mixer zone, P_{016}/P_{06} . According to previous research this ratio should always be close to unity which is achieved when $(FPR)_{op}$ is used. However, when P_{016}/P_{06} is greater than unity at cruise, then serious mixing losses will appear at off-design conditions. Therefore, the goal should be to get a value between 0.95 and 1 as the ratio increases at higher non-dimensional off-design conditions^[37]. However, $(P_{016}/P_{06})_{op}$ is also dependent on mixer inlet Mach number associated with the hot flow, which is shown below in Figure 12. It can be seen that by increasing the mixer inlet Mach number $(P_{016}/P_{06})_{op}$ increases.

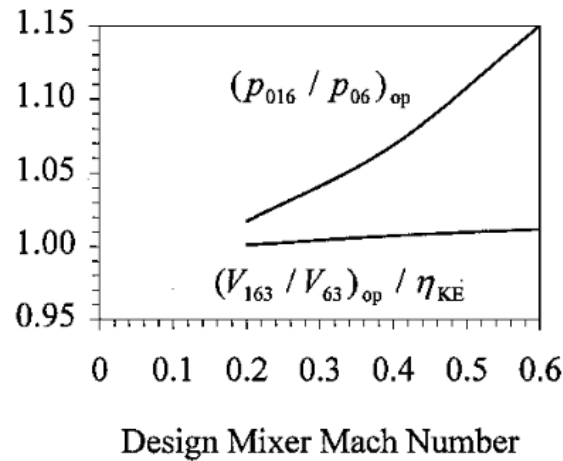


Figure 12 – Effect of inlet mixer Mach number of the hot flow at $(P_{016}/P_{06})_{op}$ ^[37]

Comparing the equation for $(FPR)_{op}$ for a mixed exhaust turbofan engine and the equation for $(FPR)_{op}$ for a separate exhaust turbofan engine a few differences can be noticed. The first point of interest is that when keeping the Specific Thrust and BPR fixed for a separate exhaust turbofan engine, changing the OPR will not affect the $(FPR)_{op}$. This does not apply in a mixed exhaust turbofan engine; when keeping the Specific Thrust and BPR fixed, any change in OPR and mixer inlet Mach number influences the $(FPR)_{op}$ ^[37]. The second point, as mentioned before, is that for a separate exhaust turbofan engine, changes in the BPR have a smaller effect on the $(FPR)_{op}$ than changes in Specific Thrust. For a mixed exhaust turbofan engine, the effect that BPR has on the optimum FPR is even smaller^[37].

2.3.4 Different mixer configurations

There are different types of mixed exhaust configuration to choose from. The most common are listed below;

- Forced injector mixer
- Lobed annular mixer
- Plain annular mixer

The forced injector mixer is shown in Figure 13 and shows that the chute of the bypass is angled into the mixing chamber and therefore forces cold air into the hot air flow. The advantage of this configuration is that a shorter length is required to achieve a complete mixing of the airflow. However, this creates a significant amount of turbulence, which increases the losses in the mixing processes. This is the main reason why injector mixers are not regularly used.

The most commonly used configurations are the annular mixer and lobed annular mixer, shown in Figure 14. The lobed annular mixer is an alteration of the annular mixer. The difference is when a lobed mixing wall at the end of the chute is used opposed to a circular mixing wall. This alteration affects the perimeter where mixing is initiated, which can be increased by up to three times compared to a plain annular mixer. This has substantial influence on the mixing length and the mixing efficiency; the reason for this will be shown in the following subchapter.

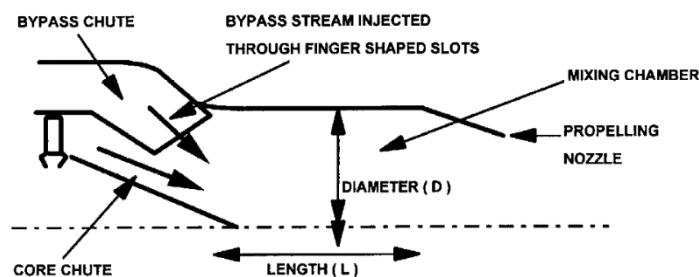


Figure 13 – Forced injector mixer^[36]

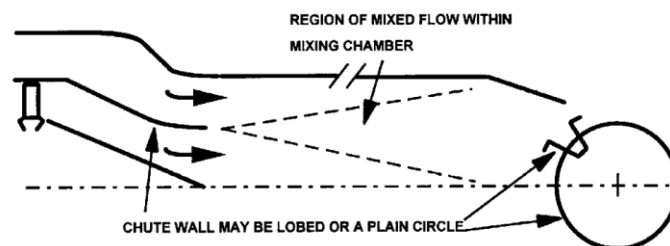


Figure 14 – Lobed annular and plain annular mixer^[36]

When considering mixing, it is unrealistic to assume that complete mixing will take place within the mixer chamber^{[22][23][36][39][40]}. This depends on the geometry of the mixer, and the mixer inlet Mach number. If there is incomplete mixing, there exists a section of hot and cold streams that are not mixed. To simplify this, the mixing model can be considered to have three different exit velocities expanding through a common nozzle. This is shown in Figure 15.

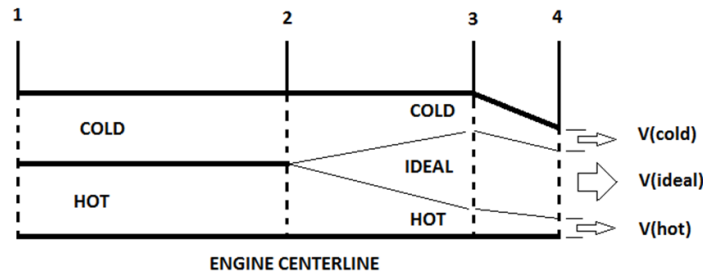


Figure 15 – Illustration of the effect of incomplete mixing on mixed bypass flow

With three different exit velocities, the theoretical thrust gain equation will need to be modified to incorporate this as shown below. However, to use this equation one must assume that the part of the flow that is not mixed is fully expanded as if through a separate nozzle.

$$gain_{theoretical} = \frac{F_{N_{mixed}} + F_{N_{unmixed\ cold}} + F_{N_{unmixed\ hot}}}{F_{N_c} + F_{N_h}} - 1 \quad (2-41)$$

Furthermore, the following equation can be used to calculate the Specific Thrust of a mixed exhaust engine with incomplete mixing taking place.

$$\hat{F}_N = \eta_{mix} \times V_{ideal} + \frac{(1 - \eta_{mix})(V_h + BV_c)}{(1 + B)} - V_a \quad (2-42)$$

In this equation η_{mix} is the mixing efficiency, V_{ideal} is the velocity of the ideal mixed stream, V_h is the velocity of the core exhaust stream, V_c is the velocity of the bypass exhaust stream, and V_a is the velocity of the aircraft.

An analytical study to investigate the effects of incomplete mixing & mixing efficiency on theoretical thrust gain, has been conducted by Rayee, T^[39]. Figure 16 shows the results of the analytical approach. This was conducted on an engine with a BPR of 5, $T_h/T_c = 3$ and $M_{mixed} = 0.4$.

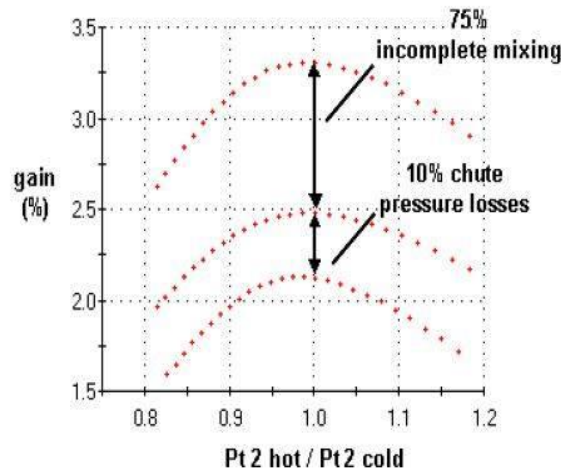


Figure 16 – The effect of incomplete mixing on the theoretical thrust gain^[39]

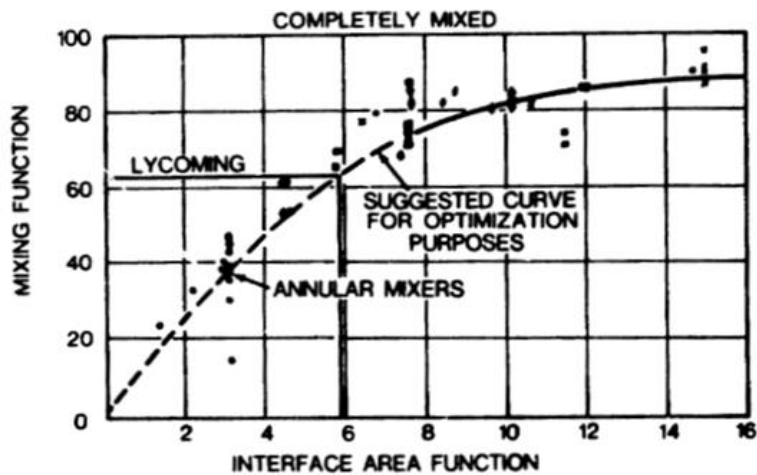


Figure 17 – Mixing efficiency against interface area function^{[22][40]}

As mentioned before the mixing efficiency is dependent on many factors. To illustrate this, Figure 17 shows a suggested curve for optimisation purposes, in

regards to choosing the correct mixing chamber geometry to achieve the desired mixing efficiency. The curve suggested for optimisation purposes was created by correlating experimental test data. In this figure 'mixing function' represents mixing efficiency. The interface area function can be calculated using equation (2-43).

$$A_{IF} = \frac{P_{er_m} L_m}{D_h^2} \quad (2-43)$$

In this equation P_{er_m} is equal to the mixing perimeter, L_m is equal to the mixer length, and D_h is equal to the hydraulic diameter. It was previously stated that when using a lobed mixer, the mixer length would decrease due to the increase in mixing perimeter. Using Figure 18 the differences between the annular mixer and the lobed mixer can be observed. Using the interface area function, it can clearly be seen that when the perimeter is increased, but the length and the diameter remain constant, a higher mixing efficiency will be achieved, as also suggested by the curve in Figure 17.

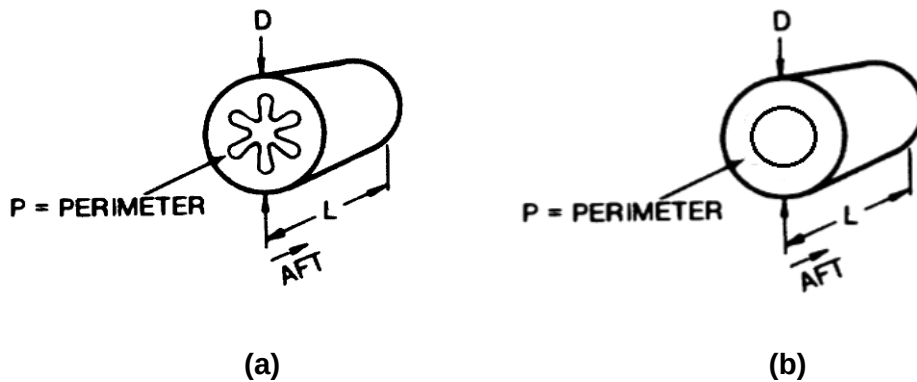


Figure 18 – Schematic mixer, (a) lobed mixer, (b) annular mixer^[40]

2.3.5 Losses due to mixing

Losses caused due to mixing are considered to be a very important aspect in the decision to use a mixer in a turbofan engine. This is because, if not carefully designed, the mixing losses can cancel out any potential thrust gain, and even

degrade engine performance, when compared to a separate exhaust configuration. Even though this aspect is important, there seems to be very little information in the public domain which describes how to estimate these losses.

In the literature that is available regarding losses due to mixing, Frost^{[22][31]} hypothesises that the losses are caused by four different types of loss. The four types of loss are:

- Pressure losses due to skin friction
- Chute pressure loss
- Losses due to incomplete mixing (mixing effectiveness)
- Fundamental mixing losses due to the mixing of two different streams

There is no method which describes pressure losses due to skin friction in a mixer. However, the mixer is very similar to a duct therefore the following method could be used. The pressure losses due to wall skin friction are related to the geometry of the mixer, the geometry of the plug, the inlet Mach number of the two streams, and the length of the mixer. To calculate the skin friction of the mixer chamber, the following equation (2-44) can be used^[36]. The equation relates to total pressure loss in a duct, but can be adapted to be used for the mixing chamber.

$$\frac{\Delta P_t}{P_{t \text{ inlet}}} = \lambda \left(1 - \left(\frac{1}{\left(\frac{P_{t \text{ inlet}}}{P_{s \text{ inlet}}} \right)} \right) \right) \quad (2-44)$$

In this equation the inlet flow conditions are used, also the influence of Mach number is captured. The geometry is taken into account through the use of λ . The equation for λ is presented below.

$$\lambda = F * \frac{L_m}{D_h} \quad (2-45)$$

In this equation, F represents the friction factor, L the length of the duct, and D_h is the hydraulic diameter. This can be calculated using the flowing equation. In this equation A is the area required for the mass flow of the duct at the given flow conditions, and P_{er} is the wetted perimeter. Walsh and Fletcher^[25] state that a representative number for friction factor of a duct is 0.04.

$$D_h = 4 \frac{A}{P_{er}} \quad (2-46)$$

If one does not assume the representative number of 0.04, the friction factor can be determined using the Moody diagram^[41], which is shown in Figure 19. Using the Reynolds number and the relative pipe roughness, one can determine the friction factor.

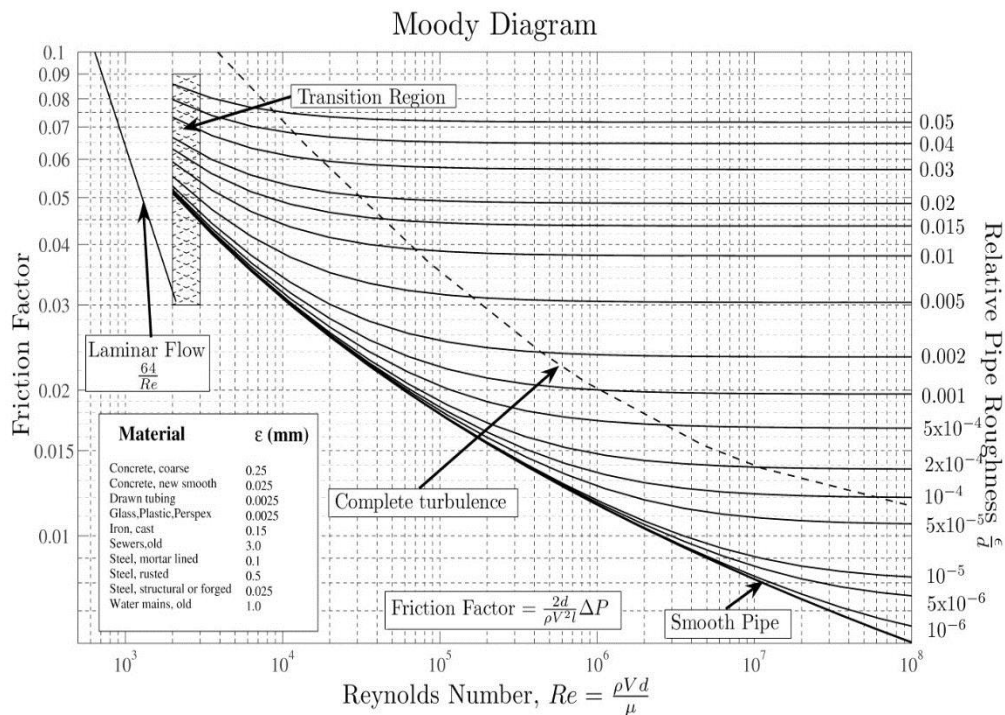


Figure 19 – Moody diagram^[41]

To calculate the Reynolds number, the following equation can be used.

$$R_e = \frac{\rho VL}{\mu} \quad (2-47)$$

V is the airflow velocity, ρ is the density of the air, μ is the viscosity, and L is the length of the duct. The relative pipe roughness is determined by the choice of the material of the duct.

The second type of loss is chute pressure loss, which represents the loss experienced due to the distance travelled between the cold stream and the inlet of the mixer. This loss could be determined through the previous method, or as described as a fixed percentage of loss in a simplified method. The third loss is the reduction in theoretical thrust due to mixing efficiency. This has already been described in the previous section.

The final type of loss is due to the fundamental loss associated with mixing two streams of different mass flow & temperature. Previous research^[42] has shown there is a relationship between these different flow conditions, and pressure losses that are experienced. The associated graph, which was produced from that research, is shown in Figure 20. This graph shows the effect of temperature and mixing mass flow ratio, against total pressure loss over dynamic head.

From the graph it can be seen that as the cold stream increases in size, the losses increase. However, there is a critical point, where losses begin to decrease again as the size of the cold stream increases further. It can also be seen that as the temperature difference between the two streams increase, the losses increase as well.

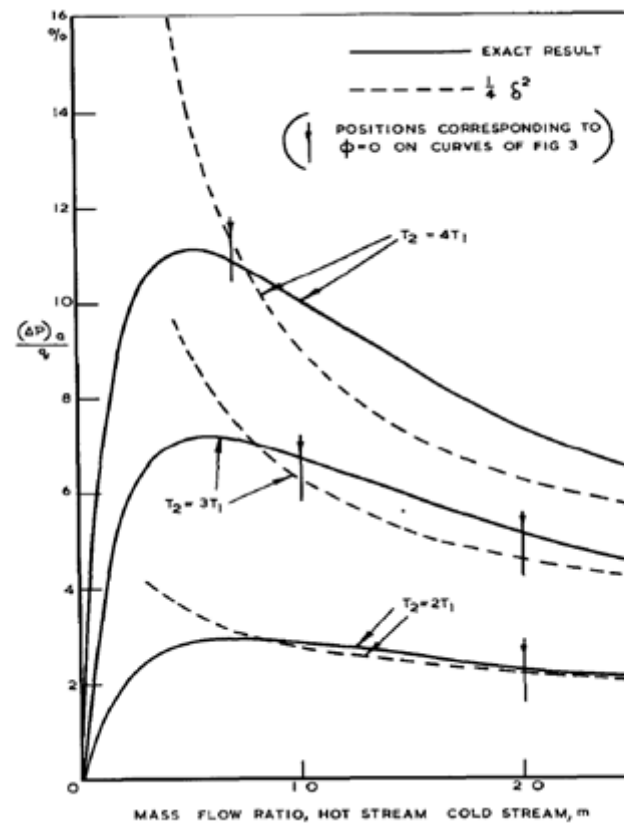


Figure 20 – Mixer pressure loss vs. mass flow ratio and temperature ratio^[42]

Now that all the losses have been defined, it can be seen that all the losses are related to the geometry of the mixer, the geometry of the plug, the inlet Mach number of the two streams, and the length of the mixer. For example, for a mixer designed with a higher mixer inlet Mach number, the pressure losses will be higher due to the high level of dynamic head. With a lower mixer Mach number the dynamic head will be smaller, therefore the losses due to the dynamic head will be less. However, the cross sectional area of the inlet of the mixer will be higher due to the lower mixer inlet Mach number. This will increase the diameter and have a negative effect on the wall friction losses. Furthermore, it will also increase the length of the mixer to achieve the same mixing efficiency. This means the designer will need to iterate between all the different variables; the target mixing efficiency, inlet mixer Mach number, and length of the mixer. Only then can the possible benefits for a mixer be properly considered.

2.3.6 Mixer weight

There is very little literature available in the public domain regarding mixing chamber weight estimation. To still be able to do a mixer weight estimation an investigation was conducted towards components which show similarities with the mixing chamber. From this investigation, two methods have been identified. The first is a weight estimation method proposed by Jackson^[43] for nacelle weight. In this method, he uses the basic turbofan nacelle dimensions to estimate the weight of the nacelle. The schematic for this method is illustrated in Figure 21.

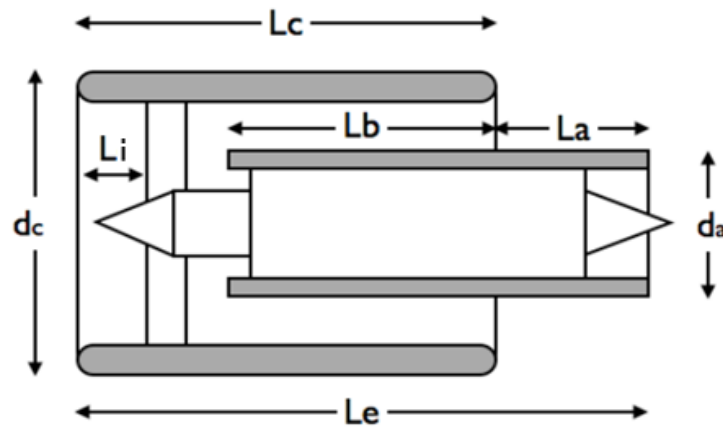


Figure 21 – Schematic of basic turbofan nacelle dimensions^[43]

In the figure, L_a is equal to the length of the core cowl, L_b is the length of the bypass duct, L_c is the length of the nacelle, L_e is the overall engine length, L_i is the length of the intake, D_a is the core average diameter, and D_c is the nacelle highlight diameter. Using equation (2-48), which is shown below, the nacelle weight can be calculated. In equation (2-48), k_{nac} is the density of the material of the nacelle.

$$WT_{nac} = \pi * k_{nac} * (2 * L_c * D_c + L_b * D_a + 2 * L_a * D_a) \quad (2-48)$$

The second method is proposed by Lolis^[44]. This method refers to the weight estimation for a duct component. To estimate the weight of a duct the following is required: The geometry of the duct, the thickness of the duct wall, and the materials used for the duct. Using this method, and from the equation for the volume of a cylinder, one can estimate the volume of the materials that are used in the duct. Using this estimation and the density of the materials used in the duct weight estimation can be made for the duct.

The thickness of the duct will need to be estimated, and this can be done using the assumption that the duct is very similar to a pressure vessel. Therefore, one can follow the pressure vessel theory which uses equation (2-49) to estimate the thickness of the pressure vessel wall to withstand the required pressure. In this equation t_{pr} is equal to the thickness of the pressure vessel wall, and σ_y is equal to the material yield stress.

$$t_{pr} = \frac{P_s D}{2\sigma_y} \quad (2-49)$$

The investigation showed that with the methods mentioned above a basic weight estimation can be carried out for the mixer. However, verification has not been conducted on this method and might not be possible without data in the public domain on the actual weight of a mixer.

2.4 Variable geometry nozzles

A mixed exhaust engine configuration may benefit from using variable geometry. The benefit arises from the more adequate matching of the mixer inlet pressure ratio between the hot and cold stream in off-design conditions. This could potentially reduce the core size required at critical operating points and therefore

have a positive effect on the SFC at cruise. Reducing core size but keeping the temperatures constant results in a reduction in fuel, this is because the amount of mass flow going through the core is reduced. To be able to achieve this improvement in performance (if even possible) a better understanding of existing variable geometry exhaust systems is needed. Although many military jet aircraft are equipped with variable nozzles for afterburner use, in the civil aviation industry for many years the drawbacks (weight, complexity) of a variable nozzle outweighed any benefits. Recently however, there has been a renewed interest in variable nozzles, with a view to use them on future engines, as part of an effort to reducing mission fuel burn. Two examples are the PurePower PW1000G^[45] (an early version of it), and the CFM56-7B demonstrator^[46], both equipped with a VAFN system. However, it needs to be noted that after extensive testing, neither engine came into service with a VAFN. Both examples are shown in Figure 22(a) & 21(b), respectively.

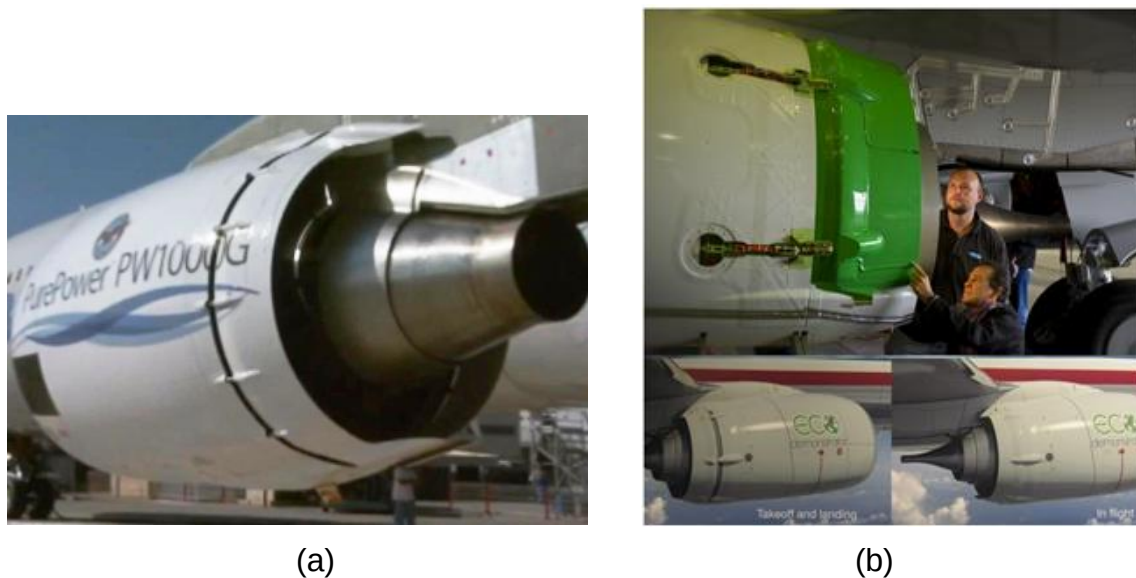


Figure 22 – (a) PurePower PW1000G^[45], (b) CFM56-7B Demonstrator^[46]

In general, separate exhaust turbofan engines have two nozzles, one installed at the core engine exhaust and another at the bypass exit. From a performance point of view, varying the area of these nozzles will have different effects on the

engine. Apart from potentially improving fan operability, the benefit of using a VAFN is in the possibility of reducing fuel burn. This is due to the fact that the engine may be able to operate at higher BPRs due to the VAFN, if the TET (Turbine Entry Temperature) at an operating point was limiting the BPR. Studies at Cranfield^{[43][47]} have shown that engines with VAFN can achieve the same thrust requirements with a smaller core size, therefore resulting in lower SFC at cruise. With a fixed core engine (the size of the core engine does not decrease further due to technology limitations) the only method of increasing BPR is by increasing the fan diameter. However, there are some significant challenges that arise from this. One of these challenges is the limited fan tip speed due to fan noise considerations. Therefore, in order not to exceed this limit, the fan rotational speed needs to be reduced⁵. Another challenge is the fact that as fan blades increase in size they add more weight to the system. The disc design needs to be improved to withstand additional loads because of the increase in blade size. The above engine design considerations results in a lower optimum FPR for turbofan engines with higher BPR. This effect becomes more noticeable when going to higher and higher BPR, as can be seen in Figure 23.

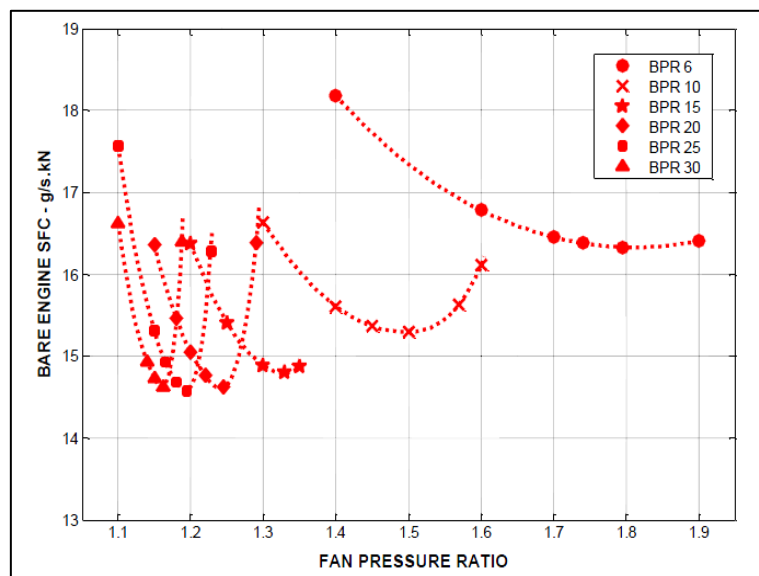


Figure 23 – Fan pressure ratio plotted against SFC for different BPRs^[43]

The increase in BPR typically results in a reduction of transfer efficiency (how efficiently the kinetic energy can be transferred from the core to the bypass), and therefore is not desirable. However, the benefit comes from increasing fan mass flow which improves propulsive efficiency and/or from reducing core size which increases thermal efficiency and reduces core engine weight. When going to lower FPR other challenges arise that relate to the operability of the fan at off-design conditions. For example, at the end of runway the bypass nozzle is typically unchoked (the nozzle pressure ratio is below the 'critical' pressure ratio to achieve sonic conditions) due to the low intake ram compression. As a result, nozzle mass flow capacity reduces and therefore the fan will operate at a reduced capacity as well. This moves the fan operating point towards surge, if the capacity drops significantly. A solution to this problem is to increase the fan nozzle area at off-design. By increasing the fan nozzle area, the mass flow capacity of the nozzle increases. This in turn increases the mass flow capacity of the fan, therefore bringing the operating point back into the fan map. Higher bypass ratios would typically require greater increase in the fan nozzle area.

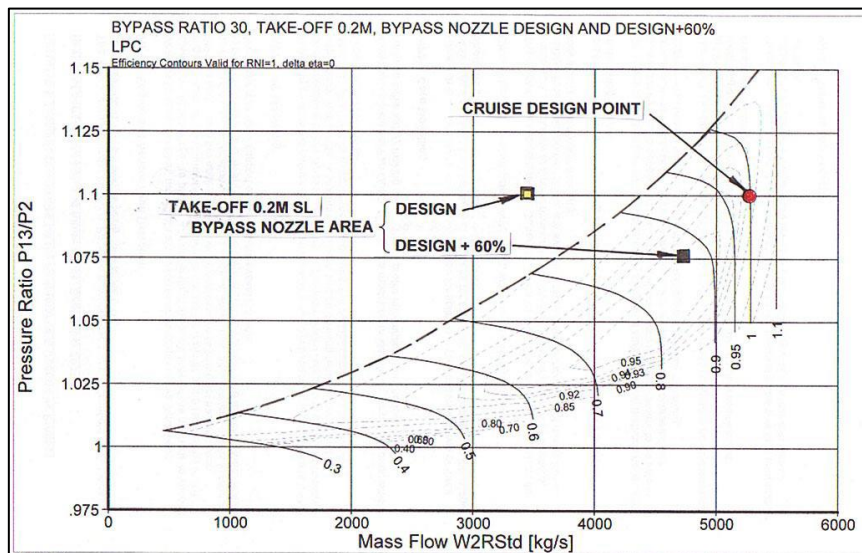


Figure 24 – Effect of BPR on fan compressor map^[43]

Jackson^[43] investigated the effect of nozzle unchoking on fan operating point as shown in Figure 24. In the figure, it can be seen that although the DP (cruise) is within the fan map, the operating point goes far beyond the surge line at take-off (for a BPR of 30 in this case, it should also be noted that the core size was fixed to the original design which was not BPR 30). To compensate for this an increase of the fan nozzle area by 60% compared to DP is needed. This moves the operating point back into the fan map. Furthermore, it can be noticed that by increasing the fan nozzle area, the FPR drops. When looking at this study a few notes should be made, first this was only conducted with 0D performance software and did not look at if this was even a conceivable option for turbofans to increase the nozzle area by 60%. Second the study assumed a fixed core area size this means by increasing the BPR the overall size of the engine increased and this also means the thrust produced by the engine increased significantly. This means when going to such high bypass ratio but by keeping the same thrust requirement the 60% area variation required is likely to be a signification over estimation. The last point is that the compressor maps used for this study was a scaled old compressor map, which means that were likely not representable for engine with a BPR of 30.

There is very little research reported in the public domain regarding Variable Area Core Nozzles (VACN), however, what is known from previous research^[47] is that any change of the core nozzle area from its design point, will affect the Low Pressure Turbine (LPT). Depending on whether the core nozzle area is increased or decreased the expansion ratio of the LPT will increase or decrease. This will translate into a change in FPR and bypass mass flow. Subsequently this will augment or reduce the thrust of the turbofan engine. This is dependent on whether the jet velocity ratio moves towards or away from its ideal value. Also a too big an increase may result in flow separation in the core nozzle and significant loss of performance, vibration and other problems.

2.5 Engine surge control

One of the most important requirements for a gas turbine is to be able to operate in a safe and stable manner throughout the whole flight envelope. Within the design process of a gas turbine, this is a separate discipline^[48]. However, due to design choices and the drive to get the lowest possible fuel consumption, gas turbines often suffer instability during non-critical operations. This is the case for most aircraft engines^{[43][48]}, as doing this will often result in a more efficient gas turbine. These non-critical operating conditions include start-up, part load, and shutdown. There are multiple methods of preventing instability during these non-critical operation conditions. The most commonly used methods are the Variable Stator Vanes (VSVs) and the compressor Blow-Off Valves (BOVs).

When looking at a multi-stage compressor it is required that all the stages have the same rotational speed and allow the same amount of mass-flow to pass through. This leads to some difficulty in off-design conditions. Important to note is that there is a pressure rise through the stages and that to keep the axial velocity at the same fraction of blade speed throughout the stages, the flow area will need to be reduced in each stage. This can be accomplished through different methods, changing hub radius, tip radius or both^[48]. This might be the correct choice at design point. However at low speeds where the pressure ratio over the compressor is lower, the chosen areas will not be correct. Compared to the centre stage the rear stage flow area will be too small and the front stage flow area will be too large. The incorrect flow areas in low speeds will lead to “windmilling” in the later stage and stall in the front stages. Figure 25 shows the situation schematically through velocity triangles. (a) Is the design point with associate pressure ratio. (b) Shows the low speeds setting with associate pressure ratio. (c) Shows the design point but with lower pressure ratio.

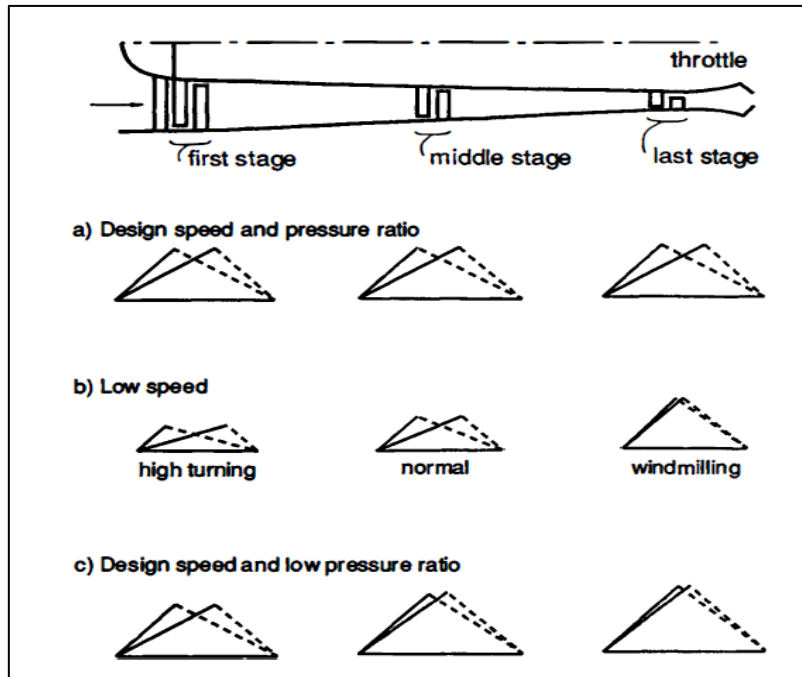


Figure 25 – Behaviour multi-stage compressor through velocity triangles^[48]

A solution to this problem is using variable stator vanes; this has been applied in the past through many methods. Figure 26 shows a method to vary the stator vanes to change the flow area available between stages moving the operating point away from the surge line. VSVs were simulated in this author's thermodynamic engine model and therefore the effect of the VSVs has been taken into account in all studies.

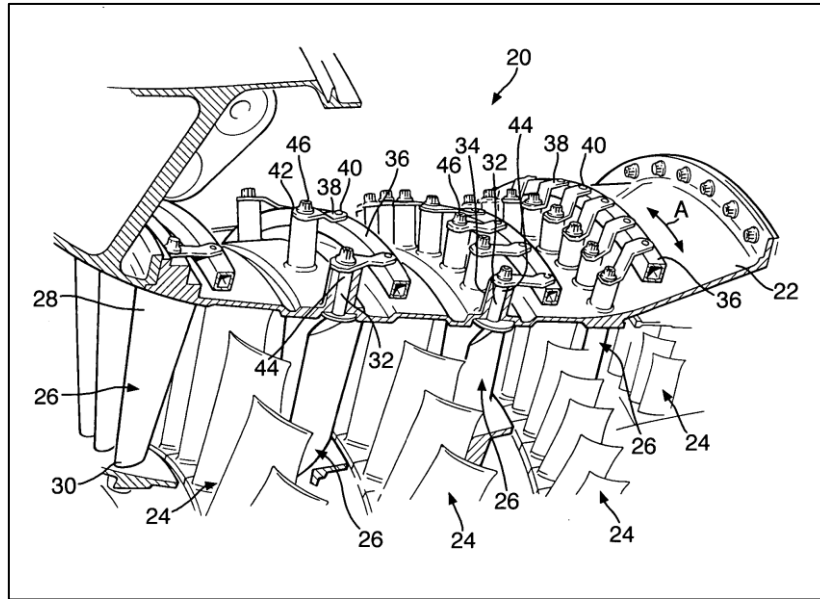


Figure 26 – Variable stator vanes patent US20050135926 A1^[49]

The other method of preventing compressor surge is the use of BOVs downstream of the compressor. Contrary to what happens with the variable stator vanes, this method allows for an increase in mass flow through the compressors. This results in the lowering of the pressure ratio within the compressor, therefore moving the operating point away from the surge line. However, if the BOVs are placed between compressor stages, it can change the compressor characteristic by affecting both affecting the operating and surge line changing the compressor characteristic. It should also be pointed out that there is no effect on the compressor downstream of the BOVs.

BOVs were not used in the thermodynamic models of this study as a visual inspection of compressor characteristics revealed no necessity for them, with adequate surge margins maintained throughout.

2.6 Variable Geometry Exhaust System (VGES)

With the understanding of the mixed exhaust cycle and the effect variable geometry has on the turbofan, the VGES can be explained in detail. A new patent application filed by Rolls-Royce in July 2014 is currently released to the public as US Patent Application Publication, US2016/0010590^[50]. This patent has not been granted at the time of this Thesis submission. However, the patent does describe the concept of a VGES in great detail. A potential application of this patent could be the UltraFan[™] engine that Rolls-Royce is currently developing.

A schematic of the VGES is shown in Figure 27. In this schematic, there are four different areas; defined A1 to A4. It is considered that all of these areas could change independently throughout the operating range. To achieve the area variation, it is considered that two components are able to move. The first component is the tail-cone (plug) which should be able to move axially rearwards. The other component is the mini-mixer cowl, which should be able to pivot or slide rearwards. With both variable geometry components, all four areas can be controlled. Through this, the fan mass flow can be varied in order to potentially reduce mission fuel burn. The arrangement also includes a mini-mixer which will enable partial mixing in the engine, where part of the bypass mass flow mixes with the core mass flow. This is believed to allow for even further reduction of mission fuel burn. The current idea of mounting the mini-mixer cowl is through pylons. This will of course affect the airflow exiting the VGES, because of the reduction of the available area over the full diameter of the VGES.

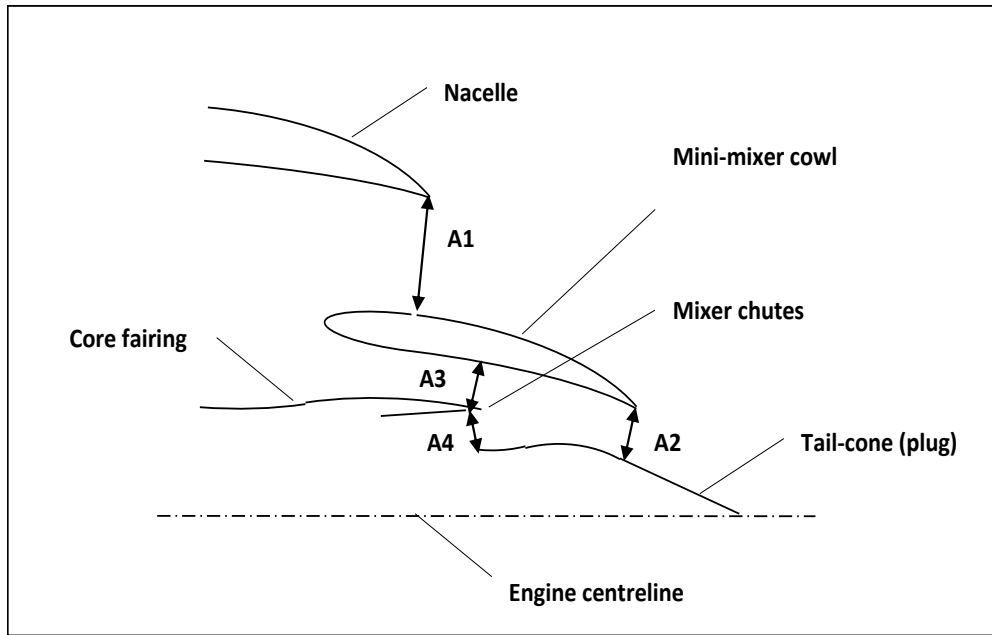


Figure 27 – Schematic Variable Geometry Exhaust System^[50]

Several related patents have been researched, to try and identify similarities of design characteristics which could be implemented to improve the Rolls-Royce concept. One patent that shows similarities to the current Rolls-Royce concept is the earlier Rolls-Royce patent 6,070,407^[51] as shown in Figure 28. It can be seen that part of the bypass flow is mixed into the core flow. This is a key feature of both concepts. Another point of interest is the ability to vary multiple areas, which include the bypass nozzle area (which is not illustrated in this figure), and the core nozzle area (through the tail-cone). These variable areas are also present in both concepts.

The main difference between the two concepts is how they control the mixer inlet area. Patent 6,070,407^[51] featured doors at the mixer inlet, which could effectively shut off the mixer. Conversely, the current Rolls-Royce patent is able to vary the mixer inlet area, rather than shut it off completely.

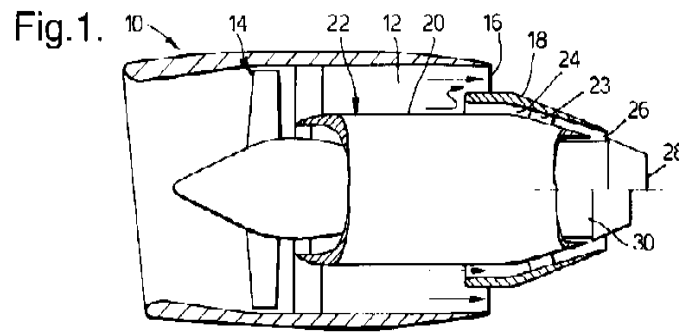


Figure 28 – Patent 6,070,407^[51]

Another patent which showed similarity to the Rolls-Royce concept is US 7,762,057 B2^[52], shown in Figure 29. A key feature of this patent design is its ability to vary the amount of bypass flow mixed with the core flow. However, the method of doing this is completely different to that in the Rolls-Royce patent. Whilst the two patents show similar features, the functions and objective of the patents are different.

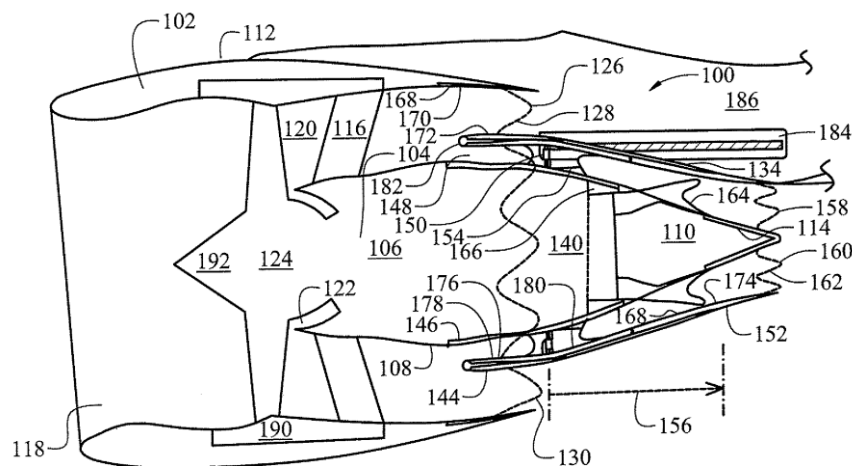


Figure 29 – Patent 7,762,057 B2^[52]

2.7 Current mixer modelling capabilities in 0D performance software

In gas turbine engineering, 0D performance simulation is a very important part of the preliminary design phase of a gas turbine. The performance calculations are done mostly by software developed specifically for that task. Although there are software packages available, a lot of companies and universities choose to develop their own in-house programmes. In this research, the Cranfield in-house 0D thermodynamic cycle performance modelling software TurboMatch was used.

2.7.1 TurboMatch

TurboMatch was first developed in 1974 by a PhD student^[54]. The software programme is capable of simulating a wide range of different engine configurations; this is achieved through a modular component system. This means every component is calculated separately and information is passed on between the different modules. TurboMatch uses a text file input system for the user to interact with the programme. This means that altering the configuration can be quite time consuming. When looking at how mixed exhaust flow is calculated in TurboMatch, it follows the same principles described in chapter 2.3.2. A schematic from an example mixed exhaust turbofan from TurboMatch is shown in Figure 30. A point to note is that the splitters, that are separate modules, are not shown here. These are needed to split any flow into multiple flows with different mass flows.

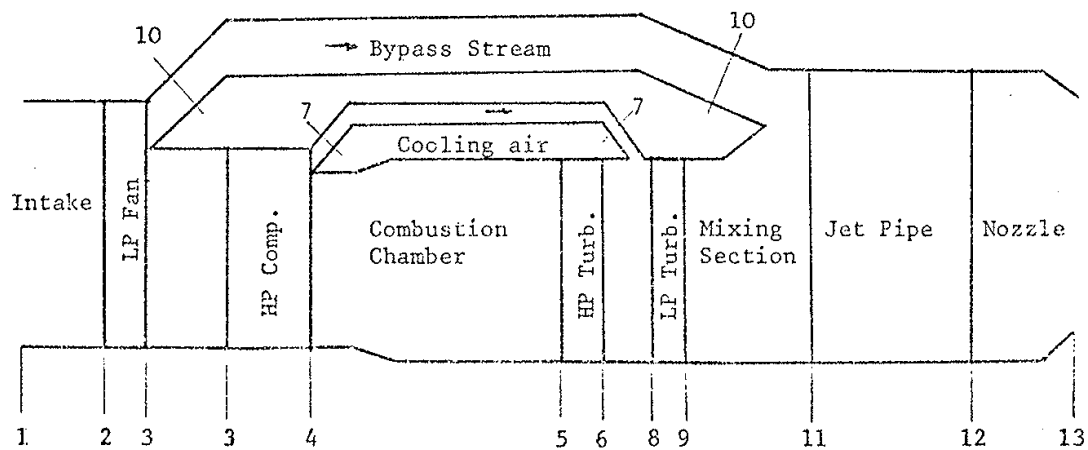


Figure 30 – Example mixed exhaust turbofan schematic in TurboMatch^[54]

The flow conditions of the mixer itself are calculated in the mixer module. The user has to set the static pressure or area of one of the mixer inlet streams. With the rest of the flow conditions the mixer section is able to calculate the outlet flow conditions. The following limitations of the mixer module should be highlighted.

- Ideal mixing assumed (complete mixing of both stream)
- No losses taken into account but these could be added separately

This means that if losses need to be taken into account they will need to be applied in the duct (jet pipe) after the mixer. Furthermore, with the setup shown above, only a single value for all the losses can be applied. Also, a mixing efficiency cannot be specified in the input file. There is an advantage that is that TurboMatch is an in-house software programme, where users can make changes to the base programme when required and include new modules and additional capability.

2.8 Optimisation

Any design, manufacture and construction of an engineering system, will require engineers to make technological and managerial decisions at every stage in the development. The goal is to maximize the desired benefit while minimizing the effort required. In the preliminary design stage of a gas turbine these decisions involve choosing the best engine characteristics to achieve the desired thrust with minimum SFC. This could be done by hand by varying all the decisions individually; however this will take a substantial amount of time. If the problem is too complex, one might not find the best solution through this method. The process of identifying the best decisions or the optimum solution is referred to as optimisation. In the public domain there are multiple methods to solve these kind of problems^{[55][56][57][58]}. Most modern or non-traditional optimisation methods are based on the behaviour or characteristics of molecular, swarm or insect, biological or neurobiological systems. The following three systems will be described; particle swarm optimisation, ant colony optimisation, and genetic algorithms as these are nowadays the most commonly used optimisation strategies^[56].

2.8.1 Particle swarm optimisation

Particle Swarm Optimisation (PSO) is called a behaviourally inspired algorithm. It is based on the behaviour of a colony of living things. This could be a flock of birds, a school of fish, or a swarm of insects. The algorithm copies the behaviour of these social entities. If one particle determines a path to objective which is good, the rest will also follow this path instantly even if location is far away. The definition of a particle in this sense is a bee within a swarm. The PSO algorithm was first published in 1995 by Kennedy and Eberhart^[59].

This model simulates a random search for the maximum value of objective function in a given design space. Because the model is based on the following rules;

1. When a particle locates a target maximum objective function, it will relay this information immediately to all other particles.
2. All other particles will gravitate to the maximum objective function however, not directly.
3. Independent thinking as well as previous memory is a component of each particle.

All the particles are initialised at random locations within the design space. Once a particle has found the maximum objective function, all other particles will follow a general trend towards the maximum objective function. However, no particle will travel exactly the same route. The advantages and disadvantages of the PSO are as follows^[56].

Advantages:

- Algorithm calculation is simple compared to alternatives
- Convergence can be achieved faster due to simple calculation and no overlapping

Disadvantages:

- High chance of finding a local optimum when using multi-objective functions
- No general convergence theory when using multi-objective functions
- Due to the simplicity of the calculation there are many different types with different advantages and disadvantages
- Significant amount of parameters to pre-programme which can influence the results substantially

2.8.2 Ant colony optimisation

Ant Colony Optimisation (ACO) is also a behaviourally inspired algorithm. This algorithm is instead inspired by the cooperative behaviour observed within an ant colony. Ants are able to find the shortest route to their food source from their nest. This type of optimisation method was developed by Dorigo and his associates and published in 1992^{[60][61]}.

In Figure 31, the ACO process has been represented as a multi-layered network. The different layers represent the different design variables, whereas each node on a layer represents the distinct value associated with that design variable of that layer. On the first iteration all particle (ants) are send randomly down the path to find the objective. If the objective is found the particle will return over the path it has taken to find that objective back home. At the same time, it will leave some pheromone on that path indicating that it is a path to the objective. At the end of each iteration, the path with the best value for the objective will get a pheromone boost, therefore having more pheromone than other paths. This will attract more particles to travel closely to that path in the next iteration. After a certain amount of iterations all particles will travel exactly the same path, at this point an optimum has been found. The advantages and disadvantages of the AOC are as follows^[56].

Advantages:

- Clear convergence criterion
- Global optimum is achieved once convergence criterion is reached

Disadvantages:

- Amount of time to convergence is uncertain
- Very complex algorithm to programme and changes per problem
- No intermediary results

- No clear strategy for multi-objective optimisation

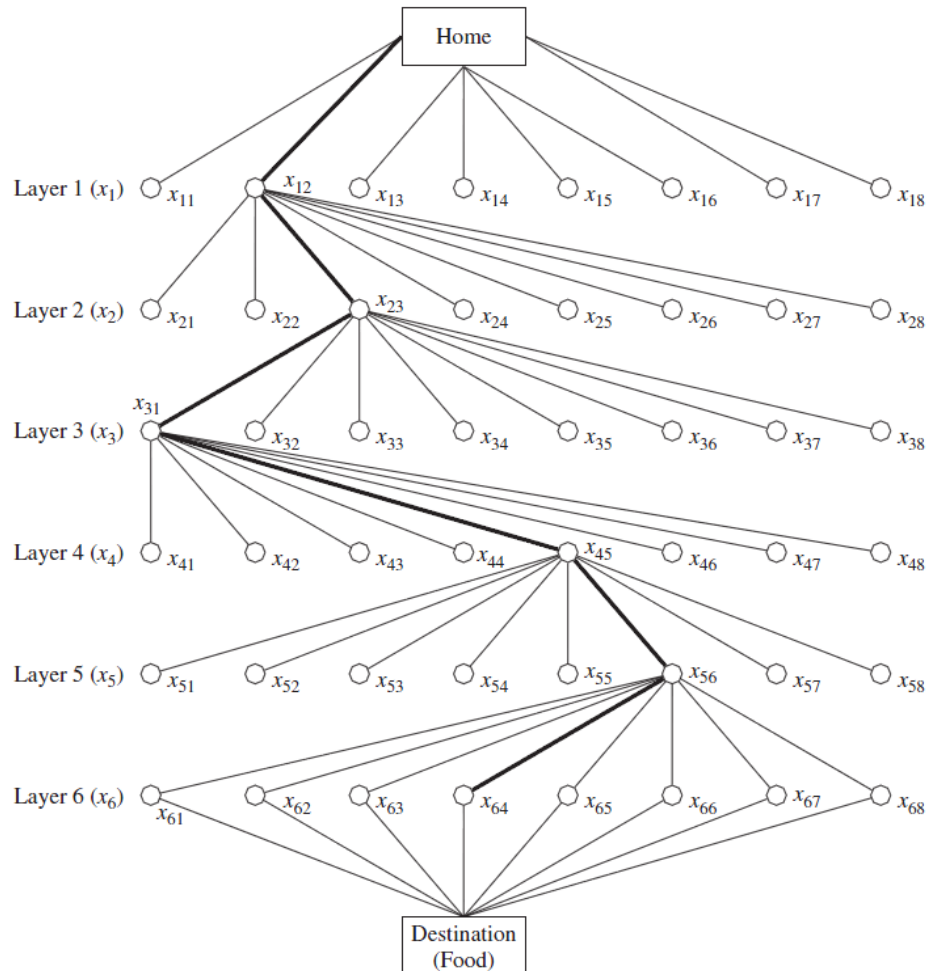


Figure 31 – A multi-layered network as a graphical representation of the AOC^[56]

2.8.3 Genetic algorithm

The Genetic Algorithm (GA) is considered to be an evolutionary algorithm. The GA was first presented by Holland^[63]. However, the basic design and analysis were based on biological evolution as described in the work of Rechenberg^[65]. Furthermore, the concept of basic evolution is older than that and it stems from Darwin's theory of survival of the fittest. GA is therefore based on the principle of natural selection and natural genetics. The basic concepts of reproduction,

crossover and mutation are therefore used in the GA. These are used to search the design space.

Traditional optimisation methods differ from the GA due to the following ^[56]:

1. A population of design points are used instead of a single design point. (The population size is determined by the amount of variables and the complexity of the problem).
2. In the GA, the design variables correspond to the chromosomes present in natural genetics. This is done by representing them as a string of binary variables.
3. In the search process the GA only uses the values of the objective function.
4. The GA can handle multiple objective functions. Therefore, it could minimise an objective while maximising another objective.
5. In every generation, as well as using the best solutions of the previous generation, it also produces a randomised selection of variables for the mutation as well as a selection of crossovers from older generations. Through this, the chance of finding a global optimum instead of a local optimum is increased.

This means the advantages and disadvantages of the GA are as follows^[56].

Advantages:

- Versatile and usable for multiple problems
- Clear strategy for multi-objective optimisation
- High chance of finding global optimum
- Intermediary results, which means optimisation can be stopped before completion and still find results close to optimum

- Clear indication of what each parameter does and how it influences the optimisation
- Pre-programmed in Matlab

Disadvantages:

- No clear convergence indication
- Slow to converge
- Trial and error for setup

Comparing all three above-mentioned algorithms one thing becomes clear. It is the nature of the problem that defines which algorithm one should choose to use. Some algorithms will be more suitable for certain problems, than other algorithms. Apart from the problem itself, other things need to be considered like computational time and computational power. On the basis that GAs had been used previously with success in various aero engine optimisation studies in Cranfield (for example in the TERA 2020 project^[64]) and the group's familiarity with this optimisation method, this author decided to adopt the same approach in his studies.

2.9 Identified knowledge gap

In the context of this particular patent, the turbofan engine configuration being investigated is of a partially mixed exhaust. There is no public domain literature available regarding partially mixed exhaust engines. Therefore the effect of partial mixing on the cycle design or theoretical thrust gain has not been quantified yet. One can assume that trends described in literature regarding mixed exhaust turbofan engines will be very similar to that of the partially mixed exhaust. However, this needs to be demonstrated and verified through analysis.

When reading through the literature regarding mixed exhaust turbofan engines, there seems to be a clearly defined path one can follow to design a cycle (subchapter 2.3.3). However, in this method, the mixer itself is not described in depth, only its effect on the overall engine cycle is considered. Investigating how mixers are optimised, taking the rest of the engine cycle into account, very little is specified on how this should be achieved. The paper from Frost^[22] and Pearson^[23] brings the most insight into the effects different characteristics have on mixer design and performance, with regards to finding the optimum mixer geometry for a given cycle. However, no method has been identified which simultaneously optimises both the cycle design and mixer geometry.

Further investigation into mixed exhaust turbofan engine performance simulation methods revealed another gap of knowledge, regarding how to realistically model this type of engine in a 0D performance simulation. This requires taking into account all the losses and performance benefits associated with the mixer. Most of the models in the literature take losses into account, however, they use a fixed value assumption. Furthermore, they do not show how this number is obtained. To the author's best knowledge no methodology has been available in the public domain up to now that describes how to take these losses into account.

When investigating mission fuel burn, the weight of the engine needs to be taken into account. There are multiple sources in the public domain on the subject of engine weight. However, these sources do not look at engines with a mixed exhaust system and therefore they do not include methods for estimating mixer weight.

The author's work reported hereafter attempts to improve the fidelity of 0D cycle performance simulation and preliminary design software by addressing all the issues reported above in this subchapter.

3 Methodology

Beyond achieving its industrial objectives, this research contributed to the further development of Cranfield's 'Engine Preliminary Design Optimisation System' (EPIDOSYS). This software toolkit is part of the Cranfield Techno-economic, Environmental & Risk Assessment (TERA) integrated simulation framework.

This chapter discusses in detailed the methodology that was established in the context of this doctoral programme. The first section gives a comprehensive explanation of the thermodynamic engine models developed, with the main focus on the modelling of the VGES architecture. This is needed in order to establish the complexity added to the baseline models due to the VGES. The following section describes the optimiser developed and the strategy that was followed in order to facilitate numerical convergence and the generation of optimal solutions. This task, given the complexity of the thermodynamic cycles modelled, proved to be extremely challenging. The subsequent section describes the method developed to take into account the mini-mixer weight, and the mission fuel burn study that was conducted. Immediately after, different verification studies are detailed to assure the reader that any new modelling techniques developed in the context of this work were sufficiently assessed for accuracy. The final section describes various case studies and trade-off analyses carried out before the final results for different engine configurations are presented and discussed.

3.1 Overall engine methodology

As mentioned before the study focused on turbofan engine architectures incorporating VGES that would potentially come into service between 2025 and 2030. As mentioned previously, two different types of engine architecture were

investigated. The first being a 3-spool direct-drive turbofan engine configuration with VGES, the second being a 3-spool geared turbofan engine configuration with VGES. The results of these two engine architectures were then compared against their 'baseline' counterparts without VGES. The mission profile and technological limits that were adopted are shown in Table 1. The T30 represent HPC exit temperature and T40 represent the turbine entry temperature. The technological limitations considered were based on expectations of technology improvements in that time frame. The upper limit of the OPR was determined using the T30 temperature limitations the exit blade height of the HPC was also considered by taking a hub/tip ratio into account. Furthermore, a wide range was chosen for the area variation of the four variable areas of the VGES, in order to identify any potential area-dependent benefits in mission fuel burn.

Table 1 – Mission profile and limitations

Aircraft size	A330
Take of thrust requirement	70,000 to 80,000 lbf
Range	3000 nm
T40	≤ 1970 K
T30	≤ 1030 K
Fan diameter	2.5 to 3.5 m
OPR at TOC	50 to 70
BPR	10 to 20
Area variation	Max 20% VAFN, 20% VACN and 20%VAM

3.2 Engine models

The first stage of this research involved several engine performance simulations, using Cranfield's in-house 0D performance software TurboMatch^{[66][67]}. The various operating points throughout the flight envelope investigated were predetermined and are shown in Table 2. It should be noted that it was decided to simulate performance at typical 'Kink Point' conditions (day temperature up to which the engine is flat-rated).

Table 2 – Operating points considered

Operating Point	Ambient conditions	Altitude (m)	Flight Mach Number	Net Thrust/engine (N)
Cruise	ISA + 0 °C	10670	0.82	XXXXX
Top of Climb	ISA + 10 °C	10670	0.82	XXXXX
End of Runway	ISA + 15 °C	0	0.25	XXXXX

Baseline engine models were created by a fellow PhD student^[5] in close collaboration with Rolls-Royce Bristol. What was included in the baseline engine models is discussed in chapter 3.2.2. The baseline engine models are shown in Figure 33.

3.2.1 Performance software

TurboMatch is the 0D performance software developed at Cranfield University. The software is capable of running design point, off-design, transient and degraded performance calculations. Therefore, this software was more than sufficient to be used in this study.

TurboMatch uses the Newton-Raphson method to solve a system of non-linear equations. It uses representative non-dimensional (hence 0D) component characteristics to solve the flow continuity and work compatibility equations. TurboMatch was designed to use a modular system to connect various turbomachinery components together. Through this method it is possible to simulate nearly every configuration of a gas turbine. TurboMatch has been programmed in FORTRAN, and therefore uses different subroutines to calculate the various turbomachinery components. These subroutines are called ‘bricks’. A sketch of a brick is shown in Figure 32. A full explanation of TurboMatch and a detail description of the code is given in references^{[66][67]}.

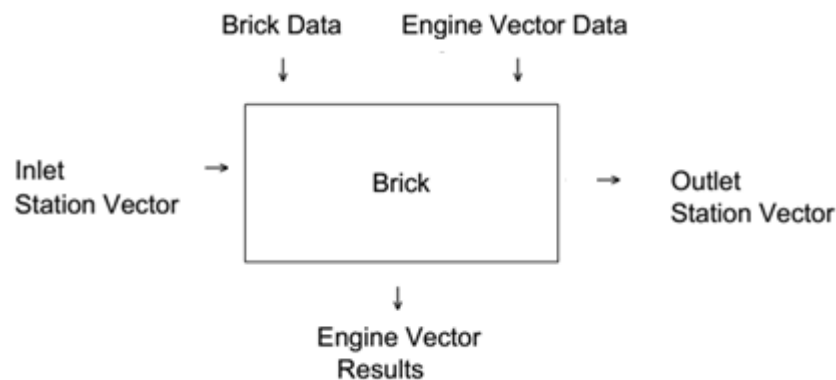


Figure 32 – TurboMatch subroutine representation as a “Brick”^{[67][54]}

3.2.2 Baseline engine models

To be able to make a comparison between geared and direct-drive engine cycles with a VGES, and without a VGES, two baseline performance models are required. As the potential application for the concept is the UltraFan™, this was also taken into consideration. The baseline performance models for these two architectures, and for the potential application, were created by a fellow researcher^[5]. The performance models include the correct application of cooling laws at their correct locations in the turbofan cycle, component efficiencies, and

feature a gearbox configuration option. These performance models also abide by the technology limitations set for the UltraFan™. Furthermore, they include power extraction for accessories.

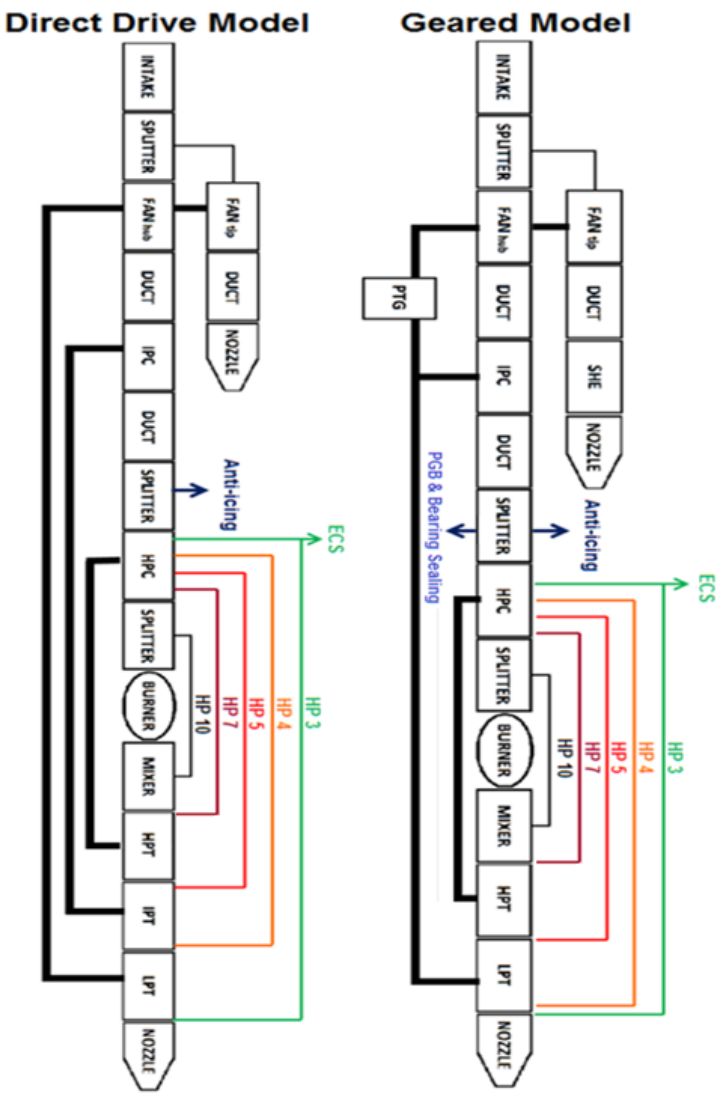


Figure 33 – Schematic baseline performance models^[5]

The schematics for these baseline performance models are shown in Figure 33. The following subchapter will explain which component polytropic efficiencies were taken into account, and following that, an explanation is provided of how the

cooling laws were implemented in these models, in order to simulate an accurate engine.

3.2.3 Component efficiencies

The component polytropic efficiencies assumed in this study were agreed upon with the industrial partner. All efficiencies as assumed at design point the efficiencies for the IPC, HPC, combustor, HPT and IPT are fixed and set as targets, therefore they are an input for the optimisation. This will have an effect on the final results as they might not be as accurate. However, because there is no optimisation done in regard to power distribution between HPC and IPC spool the impact will be minimal and the efficiencies that have been chosen have taken into account the fixed power distribution which was set by industrial partner.

The Fan efficiency is based on two assumptions. First the Fan tip loading is fixed at representative value depended on their configuration (geared or direct drive). Second Tip Mach number polytropic efficiency has a fixed correlation between them. The equation for the second assumption is shown below where a and b are constants and for this research were determined using data from the industrial partner.

$$\eta_{fan,poly} = a * M_{relative,fan,tip} + b \quad (3-1)$$

To determine the relative Mach number of the tip of the fan blade the following equations were used and a fan tip velocity triangle shown in Figure 34.

$$M_{relative,fan,tip} = \frac{V_{relative,fan,in}}{a_{sound}} \quad (3-2)$$

$$V_{relative,fan,in}^2 = V_{axial,fan,tip}^2 + U_{fan,tip}^2 \quad (3-3)$$

$$U_{fan}^2 = \frac{\Delta H_{fan}}{\psi_{fan}} \quad (3-4)$$

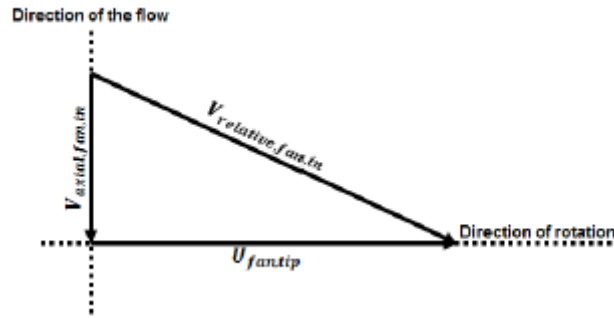


Figure 34 – Fan tip velocity triangle^[5]

With the above equation the polytropic efficiency for the fan can be determined. A more in depth look into how to determine the polytropic efficiency can be found in reference^[5].

The LPT stage efficiency is depend on the stage loading ψ and flow coefficient ϕ . For this approach it was assumed a fixed mean stage loading, this reduces the calculation problem and means on the flow coefficient will need to be calculated. To calculate the flow coefficient the following equations are used.

$$\phi_{LPT} = \frac{V_{axial}}{U_{axial}} \quad (3-5)$$

$$U_{LPT} = \frac{N_{LPT} * \pi * D_{LPT}}{60} \quad (3-6)$$

The axial velocity is dependent on the assumed inlet Mach number and temperature ratios, and the rotational speed is determined by the fan or the gear box. The diameter of the turbine is determined by the hub to tip ratio of the LPT as well as area required by the LPT turbine. Which are determined by basic thermodynamic principles. Now using the smith digraph shown below in Figure 35 an estimate for the polytropic efficiency of the LPT can be made. A more in depth explanation is shown in reference [5] which uses equations from reference [81].

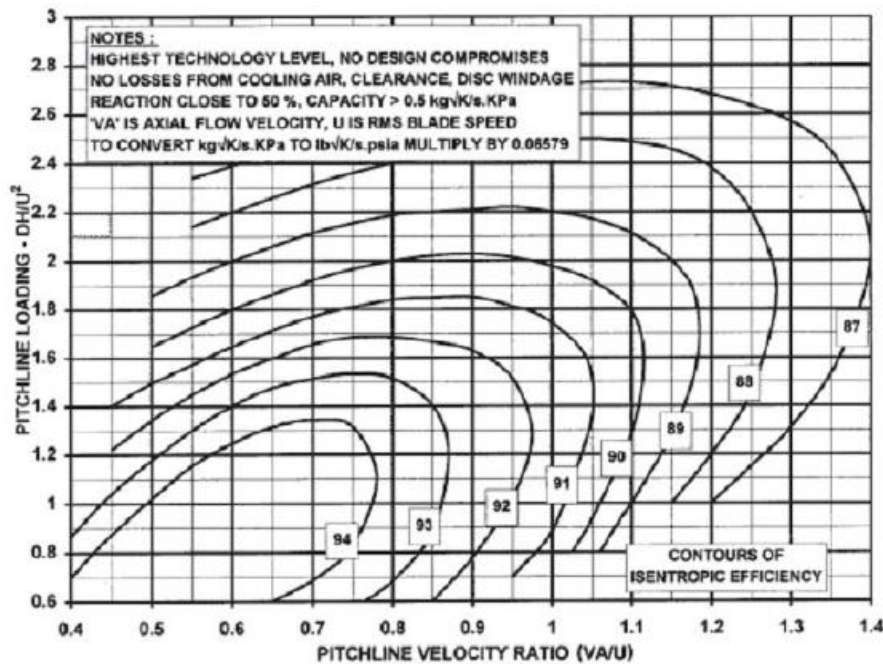


Figure 35 – LPT Smith diagram^[81]

For increased accuracy, the HPC is divided into five sections, with a total stage count of ten. Each of these sections can contain one or up to three stages. Each of these sections uses the same map but scaled. This is not as accurate as dedicated stage maps, however, these would need to be supplied by the industrial partner. But dividing the HPC into five sections each with their own scaled map is more representative than a single HPC map with all stages. For

the current purpose of this research this was considered to be reasonable. By dividing HPC into five sections the bleeds are able to be extracted at the correct location and not after the HPC which the most common practice in simulating turbofans with the software being used. This means that less work is carried out on the bleed flow making the simulated engine more realistic. The bleeds are then returned to the main flow at the correct location in the turbine section of the turbofan. To calculate the cooling/bleed requirements, the following subchapter will explain how the adopted cooling laws work, and which assumptions were made.

3.2.4 Cooling laws

The baseline performance engine models incorporate turbine cooling. The incorporation of the correct turbine cooling requirements in the preliminary design phase is a very complex undertaking. This is due to the fact that turbine cooling is highly dependent on the specific application and changes with each modification of the cycle design parameters. However, it is crucial that this is done correctly, as it affects the performance and design of the aircraft engine significantly. In open source literature many methodologies can be found to estimate turbine cooling flows^{[53][55][68][69][70]}.

To reduce the complexity of incorporating turbine cooling laws, the basic principles were simplified. More complex laws and correction factors can be implemented at a later stage if needed. The current simplified cooling equations are based on the methodology used in reference^[53]. Using empirical relationships, the turbine cooling requirements for the thermodynamic engine models can be expressed through the equations shown below.

$$\psi = \frac{w_{cooling}}{w_h} = 0.03 * \frac{\varepsilon_0}{(1-\varepsilon_0)} \quad (3-7)$$

(Convection cooling, ex. LPT)

$$\psi = \frac{w_{cooling}}{w_h} = 0.045 * \frac{(0.72*\varepsilon_0 - 0.12)}{0.7*(1 - \varepsilon_0)} \quad (3-8)$$

(Film + convection cooling, ex. HPT)

$$\varepsilon_0 = \frac{T_{gi} - T_{blade}}{T_{gi} - T_{ci}} \quad (3-9)$$

In these equations ψ is the fraction of core flow used for turbine cooling, ε_0 is the cooling temperature effectiveness, T_{gi} is the gas inlet temperature, which is equal to the turbine entry temperature, T_{ci} is the cooling flow inlet temperature, which is equal to the compressor outlet temperature, and T_{blade} is the blade mean metal temperature, which is determined by material limitation of the blade which are estimated based on industrial partner technology indicators. The arithmetic factors included in the equations were defined based on assumptions for technology levels of cooling efficiency, space to chord ratios, rotor angles of attack, exit swirl angles, C_p ratios and Stanton numbers.

The cooling air temperature is determined by the temperature of where the flow is taken from. All the cooling flow is taken from the HPC however as mentioned in the previous chapter the HPC is divided into 5 sections. After each section a specific cooling flow is taken of and put back into the system at a later point as shown in Figure 33. This means that each cooling flow will have its own cooling air temperature. Also to point out, due to the method of creating the OD model the cooling is always put back into the system before or after a component this means the cooling is calculated for the complete HPT and IPT and LPT and there is no individual stage cooling.

3.2.5 Incomplete mixing

One of the challenges regarding these thermodynamic engine models is to model incomplete mixing in the performance software that is being used. This is because the performance software was only designed to simulate ideal/complete mixing. The software calculates the mixing conditions with the subroutine MIXFUL^[54]. As mentioned in the literature review, with incomplete mixing three different flow characteristics exit through a common nozzle. To programme the subroutine to allow for this condition would require significant alteration to a substantial part of the code. Therefore, this was not seen as a viable option and an alternative modelling strategy was used.

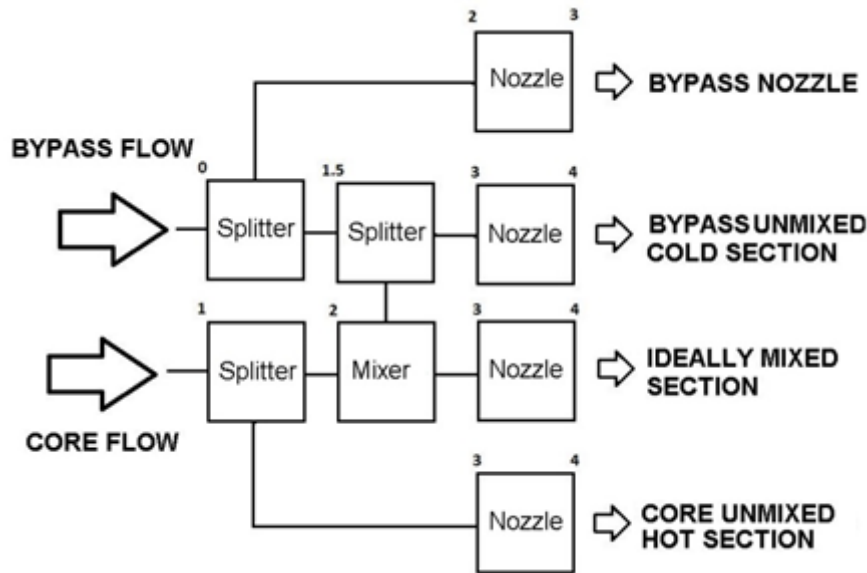


Figure 36 – Implementation of incomplete mixing bricking scheme in TurboMatch

According to the research of Frost^[22] and Pearson^[23], the theoretical approach to simulate incomplete mixing involves expanding the unmixed part of the hot and cold streams, through their own separate nozzles. To implement this, a number of splitter and nozzle ‘bricks’ in TurboMatch had to be used. The splitter brick is used to split the flow into multiple flows, with the same flow characteristics but different mass flow rates. A small improvement to the software code was made

to increase the number of nozzles allowed, which was required for this modelling strategy. Four nozzle bricks are required in total; one for the fan and three for the mixer/core nozzle. The resulting exhaust schematic that was implemented into TurboMatch is shown Figure 36. For this research the nozzle coefficients were determined by a single figure which combines C_v and C_d and put it against nozzle PR this figure was provided by the industrial partner. The same graph was used for the unmixed and mixed flow. It was also considered to use separate graphs for the C_v and C_d as well as for the unmixed and mixed cases. However, this would require massive changes within the performance software to accommodate multiple graphs. Furthermore, substantial amount of data from the industrial partners would also be required and this was considered outside of the scope of work.

3.2.6 Mixer pressure loss estimation

To make the most accurate comparison between the two architectures, with and without the VGES, the losses due to the VGES system had to be taken into account. These losses are associated with different effects taking place within the VGES.

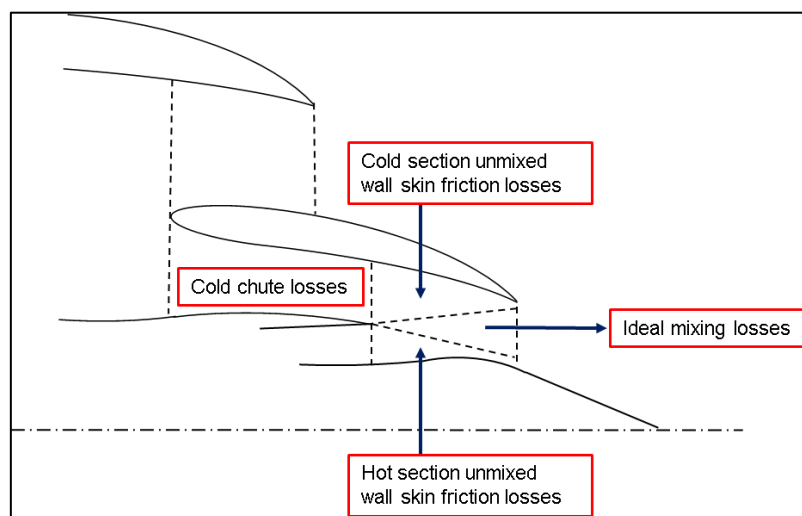


Figure 37 – Pressure loss locations in VGES

A total of four different loss mechanisms had to be taken into account. The first is the cold chute loss caused by the part of the bypass flow channelled into the mixer inlet. The second is the loss associated with the mixing of two flows. The third and fourth losses are associated with the wall skin-friction within the mixer. The locations of these losses are shown in Figure 37.

3.2.6.1 Cold chute losses

The first losses are the cold chute losses, which were set at a fixed value of 3% $= \frac{\Delta P}{d}$. This was decided after discussions with the project's industrial partner. To implement this loss in the thermodynamic engine model, the 'DUCTER' brick was used, which allows the user to input a total pressure loss. To investigate the effect of a 3% dynamic head loss in the cold chute, a sensitivity study was conducted of which the results are shown in chapter 4.2.

3.2.6.2 Ideal mixing losses

The second loss is associated with the ideal mixing section. In this section two streams with different total pressures, temperatures and mass flows mix. When these two streams are mixed, several flow phenomena occur, which are directly related to the loss in exit total pressure. Research conducted in 1962^[42] showed a relationship between the mixing of mass flow and temperature of two streams, and the associated losses. This is shown in Figure 20. The equations which govern this figure were extrapolated and then implemented within the brick 'MIXFUL'. This could easily be done as the subroutine already has all the required engine parameters as inputs. These characteristics are needed to calculate the exit condition of the mixer. The subroutine will then use the required characteristics to extrapolate the value for the losses associated with these characteristics. The losses are implemented after the 'MIXFUL' subroutine has completed the exit condition calculations instead of applying it within the calculation of the exit conditions. What this means is that these losses are calculated and applied at the exit of the mixer. This was done not to interfere with

the matching of exit conditions during the mixer calculations, as this was found to cause instability within the software compared to an unmixed engine configuration. What is meant with instability is that the program is not able to find any workable configuration, this does not mean the engine would not work. This is caused by the “MIXFUL” subroutine not being able to solve the exit conditions for the mixer, therefore, the software would produce an error and no output. What was found is that the MIXFUL has a very narrow design space it can work in as it requires certain conditions to be met to produce mixer exit values, without these conditions being met it would discard the engine configuration. One thing to note about this graph, is that the ratio used is Hot/Cold mass flow, whereas when normally referred to mixing ratio its Cold/Hot mass flow. It can also be seen that the amount of pressure over dynamic head losses associated with the ideal section peaks at a W_h/W_c ratio of about 0.5 approximately. Furthermore, the temperature ratio is also an influence, as higher temperature ratios result in greater losses.

3.2.6.3 Wall skin friction losses

The third and fourth sources of loss, that are associated with the wall skin-friction of the unmixed hot and cold section of the mixer, are harder to implement as they require more data and assumptions to be made. In particular the mixer geometry and the friction factor need to be calculated. This was done by using Figure 17, in combination with the interface area function, shown in (2-43). From this graph, if the curve for optimization/design purposes is used, the interface area function can read using the assumed mixing efficiency. From the interface area function the length can then be calculated. However, to calculate that length, additional assumptions need to be made, regarding other variables in the interface area function. An assumption of the mixer type and the plug diameter needs to be made. In combination with those assumptions, and the engine data for the areas required for the inlet of the mixer, the diameters of both the mixer and the mixing plane perimeter can be calculated. This can be done using the standard equations for circle geometry.

For the plug diameter there is very little information in the public domain which states how to determine the best geometry or material for the optimum plug design. Therefore, it was decided through discussion with the industrial partner, that the plug diameter would be fixed at 0.5 metres. Furthermore, the mixer type was assumed to be that of an annular mixer, as also suggested by the original patent design.

To calculate the wall skin-friction total pressure loss associated with the mixer, a multitude of equations is needed. The first step is to combine the following equations (2-44), (2-45) and (2-46). This will result in the following equation shown below.

$$\frac{\Delta P_t}{P_{t \text{ inlet}}} = F * \frac{L_m}{4 \frac{A}{P_{er}}} \left(1 - \left(\frac{1}{\left(\frac{P_{t \text{ inlet}}}{P_{s \text{ inlet}}} \right)} \right) \right) \quad (3-10)$$

It can be seen that the skin-friction loss is dependent on multiple aspects of mixer geometry namely the length of the mixer, the mass flow and associated mixer area required (A), as well as the wetted perimeter (P_{er}) of the area experiencing wall skin-friction. Along with these, the variable F (friction factor) needs to be calculated. Walsh and Fletcher^[25] state that a friction factor of 0.04 is a representative number. However, after extensive investigation through the use of CFD simulations, of which the results are shown in chapter 4.4.1, it was concluded that in regards to future gas turbine engines this value for friction factor would over predict the losses significantly. To calculate a more representative value for the friction factor, appropriate for the engines being investigated, the Moody diagram was introduced in the analysis, as shown in Figure 19.

To use the Moody diagram, the Reynolds number of the flow is required, which can be calculated using equation (2-47). To calculate the Reynolds number the

following equations need to be used. The first is the density equation 3-11 and the second is the viscosity equation 3-12.

$$\rho = \frac{P_s}{287 * T_s} \quad (3-11)$$

$$\mu = \mu_o \frac{T_o + C}{T + C} * \left(\frac{T}{T_o} \right)^{3/2} \quad (3-12)$$

In the first equation, P_s represents the static pressure and T_s represents the static temperature at the given location. In the second equation μ_o represents the reference value for viscosity which is fixed, and in this study has a value of 1.789×10^{-5} Kg/m/s. T_o represents the reference temperature, which is associated with the reference viscosity. This is also fixed, with a value of 273.15 K. Furthermore, the C represents the “Sutherlands Constant” which is also fixed at 110.4 K.

Table 3 – Absolute roughness coefficient and density^{[71][72]}

	Absolute Roughness Coefficient		Density	Yield stress
Material/units	(m) 10^{-3}	ft.	10^3 kg/m ³	psi
Aluminium	0.01 - 0.02	(3.28 - 6.56) 10^{-6}	2.83	40000
Titanium	0.046	1.51 10^{-4}	4.5	128000

As well as the Reynolds number, the Moody diagram also requires the pipe relative roughness. This is dependent on the material used. The material of the plug will need to withstand high temperatures of at least 700K. Therefore, the material that needs to be chosen should be suitable to operate under these conditions. The literature that is available in the public domain regarding plugs does not mention the most commonly used material. Therefore, after a discussion with the industrial supervisor it was decided to use titanium for the plug assembly. The material chosen for the cold-side of the mixer is aluminum. Table 3 shows the relative pipe roughness for the given materials.

The pressure losses on the hot and cold sides of the mixer are calculated separately, and prescribed in the model at different locations. Figure 38 shows the updated VGES modelling scheme, now including the losses at the assumed locations. It can be seen that losses experienced due to wall skin-friction are applied before the flows are split for incomplete mixing, and before they enter the mixer.

To further elaborate this, a schematic is shown in Figure 39 where it can be seen that the losses are applied at the inlet of the mixer. This was chosen due to stability issues in the thermodynamic engine model caused by how the wall friction is calculated. As mentioned certain parameters are needed to calculate the wall friction length, diameter, mixing efficiencies, this entails that to get these values an engine configuration needs to run without losses first. Once these parameters are determined the losses can then be applied to the model. If they were then applied to the model after the mixer and only to the unmixed sections, these losses could be very large and reduce the pressure at the exit of the unmixed section significantly causing problems with the nozzle subroutine. However, if the losses are divided over the complete hot or cold mixed flow there the pressure drop would be less extreme and would not cause problems for the nozzle subroutine.

An investigation was conducted to the effect of applying these losses at the chosen location against the losses only affecting the unmixed sections of the flow. The difference in thrust generation is insignificant and can therefore be neglected.

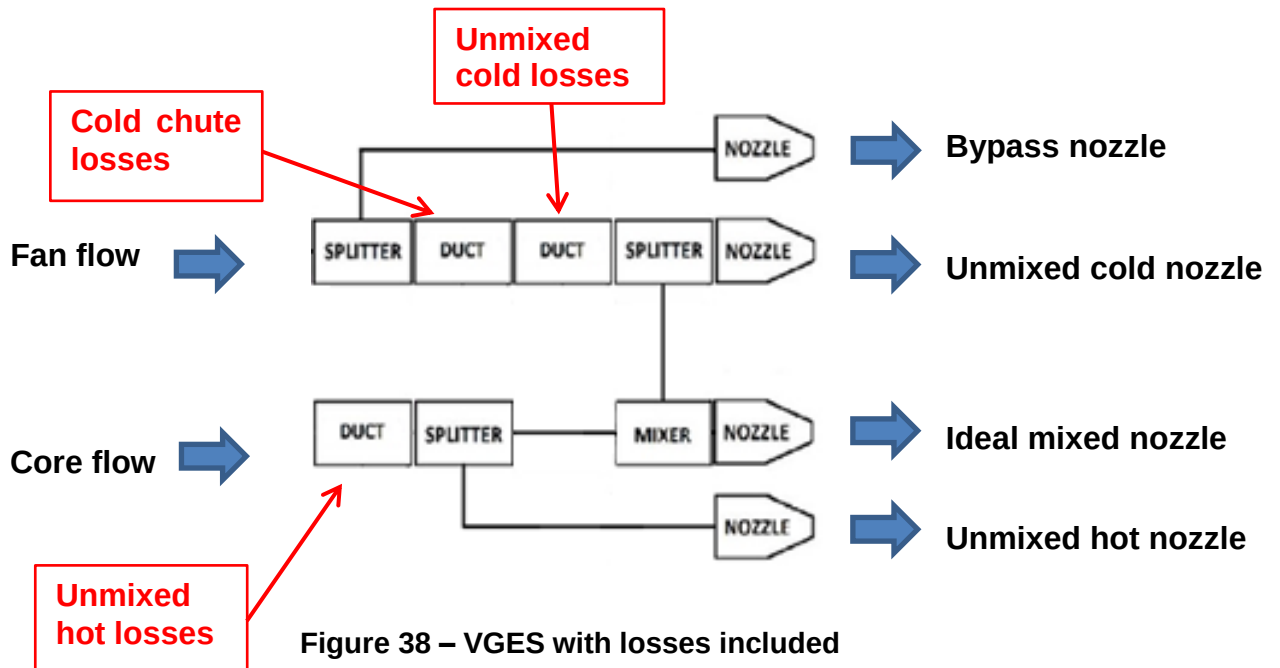
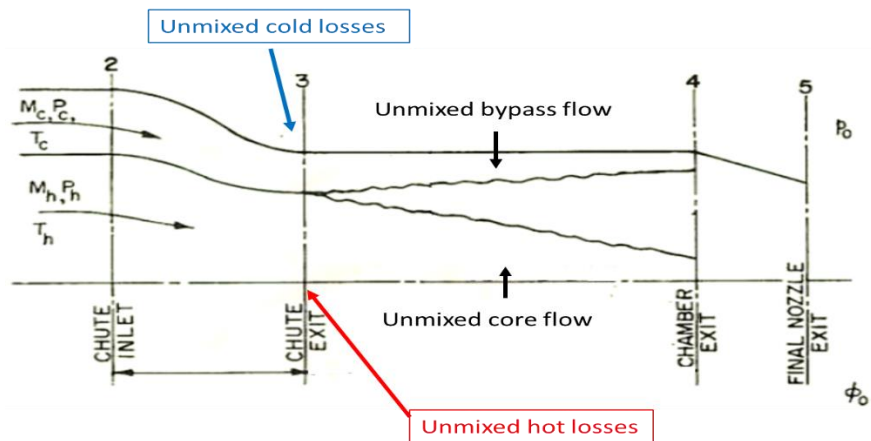


Figure 40 shows the final engine modelling scheme. This was done by replacing the nozzle section of the baseline models (shown in Figure 33) with the new mixer model (shown in Figure 38). These are the final engine models used to conduct this study. Two verification studies were carried out. The first was on the incomplete mixing methodology and the second on the skin-friction pressure losses in the mixing section. Both verification studies will be explained in the section below.

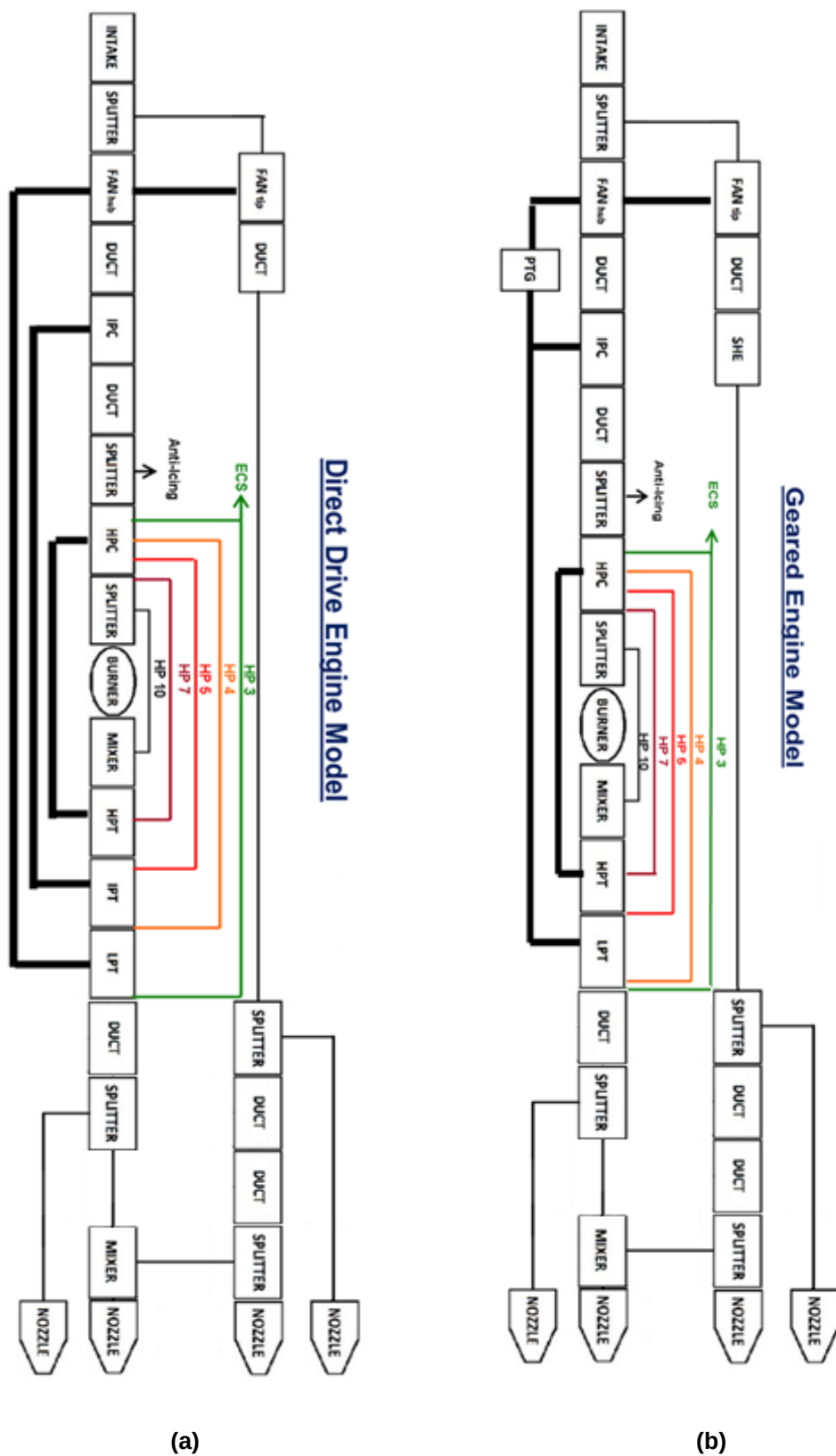


Figure 40 – Block diagrams of engine models with VGES (a) 3-spool direct-drive, (b) 3-spool geared

3.3 Cycle design optimisation

Due to the additional Degrees of Freedom (DoF) introduced by the VGES architecture, which increase from 4 to 15, it was found that the number of variables that need to be taken into account, in order to find the optimum engine cycle for the lowest mission fuel burn, is quite large. To find the optimum engine would be a very complicated and time consuming task, if this were to be done through a hand calculation. As a result, it was decided that the best option was to use an optimiser to find the optimum SFC at cruise which is expected to produce the optimum mission fuel burn (or to be very close to it).

As mentioned in chapter 2.8 there are different types of optimisation strategies to consider, the PSA, ACO and the GA were all considered for this study. For this study it was decided that the GA was the most suitable choice due to the nature of the problem where the idea was to have multiple-objective optimisation where one objective is the lowest SFC and the other objective is the highest SpT. In addition, the GA was previously used in the baseline engine study with success^[5] as well as other study which investigated performance of gas turbine using the GA^[13]. GA is also it is easy to setup as it is already pre-programmed into MATLAB. Furthermore, the code is very versatile and therefore does not require any changes to the algorithm like the other optimiser algorithms. The GA also has a very well defined strategy for multi-objective optimisation which coincides with the importance of this study to use multi-objective optimisation and even though it did not have a clear convergence criteria it was expected to produce results within the available timeframe and computational power. The results of the optimisation were expected to be the optimum or very close to the optimum objective functions.

3.3.1 Variables, constraints and objectives

The first step in using an optimisation process is to identify the different objectives, variables and constraints, which will guide the optimiser to find the optimum engine cycle. The total number of variables that need to be taken into account in this study is 13, listed in the Table 5 below.

The first thing to observe from the table is that the two off-design operating conditions require a separate variable for the area variation. This results in each added variable area within the VGES requiring 2 variables for the optimiser. Furthermore, the variable area of the mini-mixer has been combined to a single variation percentage. This is done to improve model stability, as the investigation into separate variability led to instability within the mixer subroutine, failing to find any workable configuration.

Two of the three constraints are mentioned in Table 1. It can be seen that the T30 and T40 upper limits were set to 1030 K and 1970 K respectively. The last constraint has to do with the fan diameter. Through discussion with the industrial partner, the axial inlet velocity of the fan, as well as the fan hub/tip ratio were fixed, which allowed for the maximum fan mass flow at TOC to be fixed. This resulted in a maximum fan mass flow at TOC of 720 kg/s.

The objectives set for the optimisation were the minimum SFC and maximum Specific Thrust, both at mid-cruise. Setting two contradictory objectives would not result in a single optimum engine cycle configuration but a set of optimum engines, each with the lowest possible SFC at cruise and maximum Specific Thrust possible. The reason for selecting these two objectives is because the optimisation only looks at cruise SFC. Therefore, the optimiser does not take weight or drag into account. These two contradictory objectives will assure that the engines found by the optimiser will be the smallest possible with the minimum

value for SFC. This is because greater Specific Thrust means smaller overall engine size.

Table 4 – Optimisation variables

	Range	
Variables	Min	Max
BPR	10	21
FPR at Mid-Cruise	1.1	1.5
OPR at ToC	35	70
TET at Mid-Cruise	1500	1700
Mixing efficiency	10%	45%
Mixing Mach number	0.3	0.5
Mixing ratio	0.01	20
Variable area mixer (VAM) TOC	-10%	10%
Variable area mixer (VAM) EOR	-10%	10%
Variable area fan nozzle (VAFN) TOC	-10%	5%
Variable area fan nozzle (VAFN) EOR	-5%	20%
Variable area core nozzle (VACN)TOC	-10%	5%
Variable area core nozzle (VACN) EOR	-5%	20%

3.3.2 Optimisation flow chart

The workings of the optimiser and the path each engine configuration takes will be explained in this subchapter. This will be conducted with the help of Figure 41. This figure illustrates the workflow of the optimisation process. Through trial and error, the best approach when considering computational power and time was determined.

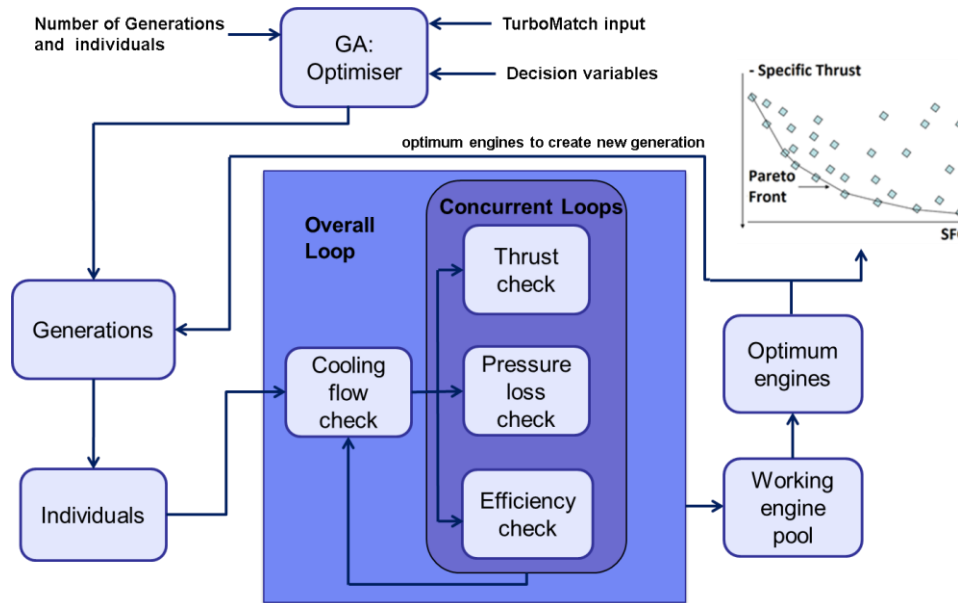


Figure 41 – Optimiser code flow chart

Individuals:

The optimisation starts with the given variables from Table 4. From this range of variables, the optimiser begins by creating a number of individuals. Each single individual will be assigned a random value for each variable. The amount of individuals that are created is set by the user. As mentioned before, the amount of individuals is dependent on the amount of variables. However, after testing it was found that the recommended population size for the amount of variables was not sufficient enough to produce a reliable result. The population size is determined by the amount of individuals and is case dependent. In general the rule of thumb is to multiply the variables by 10 and that will result in a population size adequate for the problem. This was the result of the complex nature of the problem and the instability caused by the mixer. Therefore, a much larger amount was chosen as the mixer requires a certain combination of engine characteristics to work, which is not always achieved when using randomisation of variables. The software was programmed in such a manner that if a mixer subroutine is used, mixing will always need to be present. If the configuration doesn't allow mixing it will not produce any results.

Iterative process:

Each individual with a random set of variables will then go into the iterative process. This starts with running the DP (Cruise) and the two OD (End of Runway and Top of Climb) simultaneously with the set thrust requirements for those conditions. This is an option in TurboMatch where the user is able to set the required thrust and the engines will be scaled to achieve the required thrust. The thrust requirements are shown in Table 2. In these initial calculations the cooling requirement and the components efficiencies have not been taken into account and an arbitrary value was chosen as initial starting point. It would also check if the T30 and T40 both at EoR and the mass flow at TOC had not exceeded the allowed value which are given in Table 1. Once it has been confirmed that the engine was able to run at DP and the two OD conditions and did not exceed any allowed values in these conditions an overall loop would take place where the correct value for cooling, component efficiencies and mixer pressure losses would be calculated. This would require a few iterations as it would calculate values and feed those values back into the system, then run TurboMatch, and validate the thrust was achieved and the temperatures and mass flows had not exceeded the allowed limitations. Afterwards the overall loop would calculate the cooling, component efficiencies and mixer pressure loss values again to see if there was a discrepancy between the value that was fed into the system and the value that is calculated once TurboMatch had run. If the difference for any of the values exceeded 1% the overall loop would not consider the engine configurations to be converged and would continue the iterative loop. The optimiser uses the new values as the initial values for the following loop.

Working engine pool:

Once it has been determined that an individual is able to produce the required thrust, at all operating points, and errors of the new calculated values for cooling flow, efficiencies and mixer losses are within 1%, a check is carried out to ensure none of the constraints set has been exceeded. These were the T30, T40 and mass flow at TOC. If any of these constraints has been exceeded, the individual's

objective function value for SFC is penalised by the percentage it has exceeded the constraint. Through this method the individual is not discarded and might still be used in future generations. The reason for not discarding this individual is that if it were to be used to create future generations it might eventually produce an engine which could meet the requirements. Once an individual (engine cycle) has proven that it can meet the set requirements, and has found values for the objective functions, the individual is added to the pool of working engines. The rest of the individuals are discarded.

Pareto front:

The best individuals from the working engine pool, that meet the objectives set for minimum SFC and maximum Specific Thrust, are plotted on a 'Pareto front'. The Pareto front provides a useful visualisation of the relationship between the 2 objective functions. An example of this is given in Figure 41.

Generations:

Once all individuals have been run through the optimiser, a single generation is complete. From the current Pareto front of engines, a new generation of individuals is created. Those individuals have values for the variables that reflect those found on the current Pareto front. Furthermore, there is some mutation added, as it was explained in the literature chapter 2.8.3. This cycle continues until one of the following three rules has been met:-

1. The specified number of generations has been reached
2. There has been no change in the Pareto front in a set number of generations
3. The computational time available has been reached

3.3.3 Computational power / time:

The first optimiser that was created was designed to run on a local machine and would require about 43 days to complete one case (this is based on an estimation and calculation as a full run was never complete). A total amount of 250000 individuals are run in each case, each individual averaged between working and non-working takes about 1 min to verify if it would work or not work as configuration. A total of 4 cores could be used on a local machine, leading to a total time of 43.4 days to complete a single case. However, this was never tested the longest it ran for was 10 days after which the optimisation was terminated. It was decided that the optimiser would need to be run on high-performance computing facility and on a single node with 16 cores to be able to access all the different cases within a reasonable time frame. Furthermore, the optimiser evolved and more functionality was added to it. The optimisation strategy changed from doing a single very large optimisation, to multiple optimisations, carried out sequentially. This means that the results of the first optimisation are used as initial values of the second optimisation, and so on. It was discovered that by doing this the computational time and power required would be reduced. Each optimisation would have its own specific task within the larger optimisation. This is shown in Table 5.

The first optimisation is used to establish a group of working individuals. The large amount of individuals required per generation is a result of the numerical instability of the mixer, which was mentioned previously. The amount of individuals reduces significantly after the first optimisation. However, the amount of generations increases. It was observed that the greatest improvement regarding the Pareto front's location, happens during the second and third optimisations. The last two optimisations are used to populate the Pareto front and improve the Pareto front further. Through this method an expected outcome is that the Pareto front will contain approximately between 500 and 700 different engine configurations.

Table 5 – Convergence strategy

Optimisation	Individuals	Generations	Task
1 st	18000	3	Create baseline of working individuals
2 nd	700	24	Push Pareto front closer to optimum result
3 rd	400	24	Push Pareto front closer to optimum result
4 th	300	99	Populate Pareto front
5 th	300	99	Populate Pareto front

3.4 Verifications methodology

3.4.1 Verification of incomplete mixing modelling

The verification of the modelling strategy to simulate incomplete mixing was conducted using data available in the public domain, to see if the same trends are present with the current approach, compared to the analytical approach used in the literature. To do this, two gas turbine models were created. The first model is a performance simulation of the RB211-524G/H-T and the second model is a performance simulation of the RB211-535E4. These engines were chosen as they have mixers and also there are data available in the public domain about these engines. The known engine characteristics are shown below in Table 6. These values have been acquired through the following reference^[21]. Unfortunately, not all data are available in the public domain, and therefore certain parameters had to be assumed. The assumptions made, used values which are considered reasonable for the expected technology level of those engines.

The mixing efficiency for both engines was set at 25%. This value was chosen with the help of the industrial partner. The engines were matched for SFC, thrust at cruise, and thrust at SLS. FPR was varied, while maintaining OPR, and the

power distribution between the IPC and HPC, to find the optimum FPR for these engines. Once the engines were matched, the mixer was removed, and the process was carried out again. This produces the optimum separate jet configuration, which provides a baseline to compare against. With the baseline determined, the mixing efficiency of the mixed engine was varied parametrically to establish the performance trends of the author's modelling strategy and compare against public domain data. The results of this verification are shown in chapter 4.1.1.

Table 6 – Engine model performance characteristics for the verification of incomplete mixing modelling^[21]

Engine	RB211-524G/H-T	RB211-535E4	Units
SFC mid-cruise	0.572	0.617	lb/h/lbf
Thrust cruise	11,813	8,700	lbf
Thrust at SLA	60,600	43,100	lbf
OPR	34.5	28.0	-
BPR	4.1	4.3	-
Mass flow rate	738.5	522	kg/s

3.4.2 Verification of mixer skin-friction loss modelling

To verify the methodology for the skin-friction losses within the mixer, a 2D axisymmetric CFD investigation was conducted. This investigation was conducted using the help of the optimiser described in chapter 3.3. Using the methodology described in the previous subchapter. Most of the assumptions regarding the CFD settings were derived from literature^{[73][74][75][76][77]}. The CFD code that was used was the commercial code FLUENT. The verification of the skin friction was conducted as followed. First an optimisation was run using the original assumption regarding skin-friction losses with a skin friction factor of 0.04.

Afterwards the results of that optimisation were used as the boundary conditions for the CFD study. The results of the CFD investigation were then used to verify the original assumption of skin friction factor. After this investigation an iterative investigation was started using first the optimiser to produce boundary conditions and then after CFD to confirm the skin friction losses. This would produce a correction factor for the skin friction equation. The correction factor would then be implemented into the optimiser and the optimiser would be run again to produce new boundary conditions. This was continued until the skin friction losses of the optimiser showed agreement with the results of the CFD. How the CFD simulations were created using the results of the optimiser is shown in the following subchapters.

3.4.2.1 Mixer geometry

From the result of the first optimisation, five engines were chosen with different Specific Thrust values, covering the entire span of the Pareto front. These five engines were used together with the assumptions of mixer geometry, to create a very simplistic representation of the mini-mixer for that specific engine. An example of the simplistic representation of the mini-mixer is shown in Figure 42.

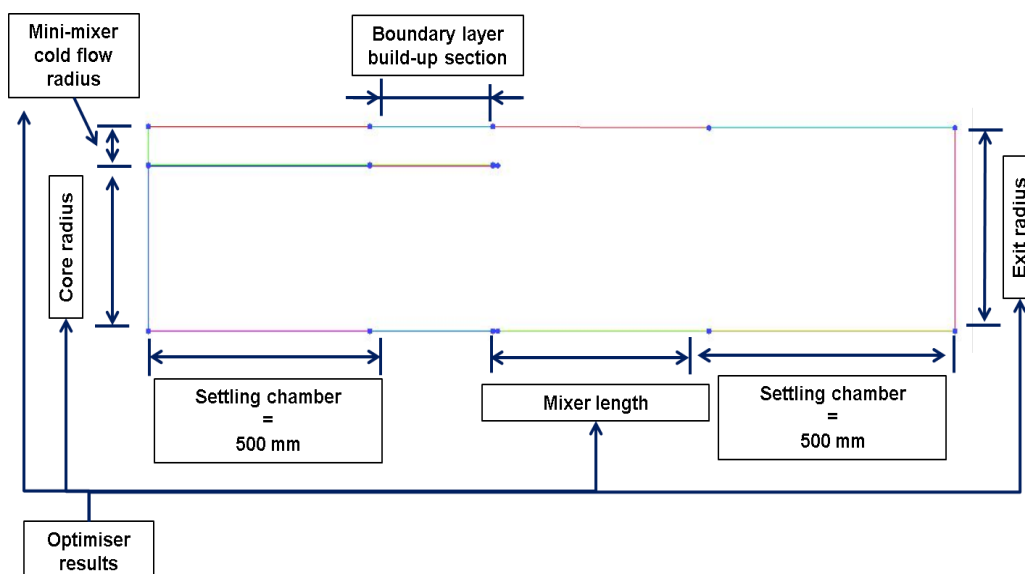


Figure 42 – Simplified mini-mixer geometry for CFD investigation

The model includes settling chambers, both upstream and downstream of the mixer. This was done to move the boundaries of the CFD simulation away from the measuring plane. Without these settling chambers, the boundaries themselves could influence the local flow conditions, and therefore the mixing performance. The settling chambers are 0.5m long each. Furthermore, there is a section before the mixer inlet, to allow the boundary layer formation on the mixer walls, prior to mixing. These factors together should allow for the model to provide realistic results.

3.4.2.2 Boundary conditions & blocking scheme

From the results of the Pareto front, the total pressure, temperature, and mass flow from the inlet and outlet of the mixer, for five different engines were extracted. These were used to set the boundaries for the CFD simulation. The types of boundaries that were chosen for this simulation are shown in Figure 43.

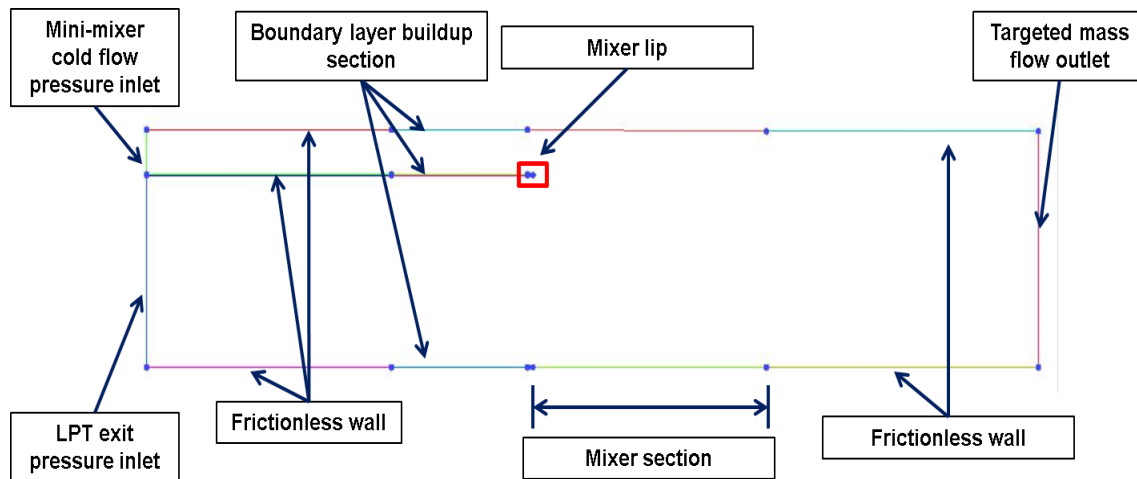


Figure 43 – Boundary conditions for the mini-mixer CFD investigation

It can be seen that the inlets of the model are pressure inlets, where total pressure is defined. In reality the turbulence level for the inlet boundary conditions should be determined by experimental results. However. Experimental testing was

outside the scope of this project. Furthermore, experimental data regarding turbulence level for an inlet of a mixer are not readily available in the public domain. Therefore, following the guidelines put out but experts in the field specify general turbulence intensity to use these were for a duct 5% and exit of LPT 15%. The domain outlet is a target mass flow type, as there will be multiple flows exiting, and therefore not appropriate to use a pressure outlet. Another reason for choosing target mass flow is that if a pressure outlet was chosen the simulation may converge, but on an incorrect value for mass flow. The boundary conditions for the walls of the settling chambers have been set to frictionless. The walls of the boundary layer build-up section, as well as the walls of the mixer, are set with the appropriate absolute pipe roughness value.

Heat transfer takes place between the hot and cold flow but there is also some extra heat transfer between the hot gasses and the solid structure like the mixer lip. This same structure will transfer some additional heat to the cold bypass flow on the other side of it. In the literature of Frost^[22] it has been mentioned that this may help to improve the mixing efficiency. To simulate this accurately is a very complex and time consuming task, with no significant benefit to the area of interest of this simulation. Therefore, to simplify the effect it was decided to assign a temperature to the mixer lip as well as the build-up section. This temperature is the same temperature as the hot gasses. The choice of temperature would ensure the hot gasses were not affected, while still influencing the bypass flow entering the mixer.

The blocking scheme chosen for the simulation, with regards to the mixer geometry, is a combination of straight blocking, with a C block around the mixer lip. This blocking scheme is capable of producing high quality meshes, which will help achieve convergence as fast as possible. Furthermore, it would produce the most accurate results and airflow behaviour, due to the amount of control that can be expressed on the nodes around the lip.

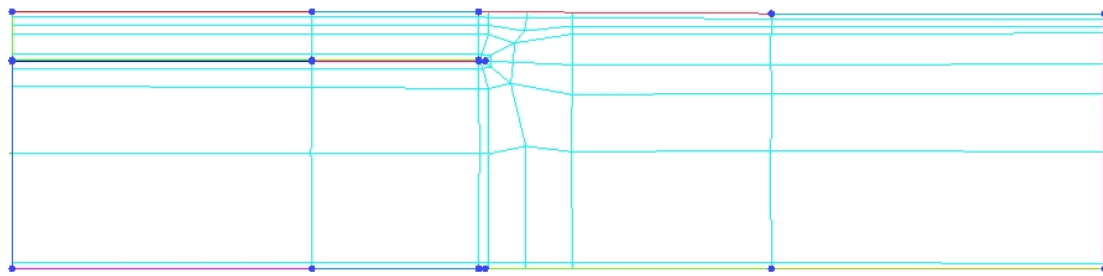


Figure 44 – Mini-mixer CFD model blocking scheme

3.4.2.3 Meshing

For the meshing of the mixer it was decided that a structured mesh would be most appropriate. The goal of the simulation is to measure the pressure losses due to wall skin-friction within the mixer. Therefore, it is imperative that the boundary layer be simulated accurately. This resulted in a choice of Y^+ equal to 1. Due to the different flow conditions within the mixer an accurate estimation of the actual distance associated with Y^+ equal to 1 will be different at each wall. Furthermore, the flow conditions may be slightly different from those in the thermodynamic software produced for that specific engine. Therefore, once the simulation is complete the Y^+ will need to be measured. If the Y^+ is not adequate, the mesh must be altered to achieve the required Y^+ . This needs to be done for each different mixer that is simulated. The size of the mixer mesh is roughly ~200,000 elements. Through a mesh independency study, this was found to be a suitable mesh density.

3.4.2.4 Solver

Within the literature available in the public domain there seems to be a lack of agreement on how to measure the portion of the flow that is mixed. This is because in the analytical models the mixed flow has a single defined value associated with the exit characteristics. In the computational model this is not the case. This is because the flow conditions gradually change between the hot and cold flow that is mixed. For this reason, it was decided that the main goal was not

to match exactly the mixing efficiency of the computational model with the performance model, but instead focus on determining accurate skin-friction losses associated with the unmixed section of the flow.

From the literature^{[73][74][75][76]} it was decided that the turbulence model to be used would be the K- ω SST model. This model is particularly suited to internal flow cases, and has been shown to accurately predict boundary layer formation. Furthermore, it shows it is capable of reasonably simulating mixed flows

3.4.2.5 Convergence

Achieving convergence with the computational model of the mixer can take a significant amount of time. This is due to the mixing of hot and cold flows, and significant differences between the flow characteristics of the two streams, which the exit boundary of the simulation needs to accommodate.

Through trial and error, a method was established that could achieve second order convergence. The Courant number was set at 50 for first order convergence and 10 for second order convergence. Further the convergence criteria residuals were set to 10^{-6} .

3.4.2.6 Post processing

Once the simulation achieved 2nd order convergence, the results from CFD were compared against the thermodynamic simulation for the same engine. Without post processing the CFD results, the exit flow conditions when measured would not only include the mixer skin-friction losses, but also the losses due to the mixing of two streams with different temperatures and pressures. Therefore, a back to back comparison was not possible. This is due to the modelling scheme

chosen in the thermodynamic simulations, where the skin-friction losses are applied before mixing takes place. As mentioned in chapter 3.2.6.3.

Figure 45 will show the difference between how 0D software applies the losses and how in CFD the losses can be measured. In this figure it can be seen that the 0D model applies the losses at the front of the mixer. However, for CFD the losses can only be measured at the exit of the mixer. If the complete flow was measured at the exit of the mixer in CFD then all the losses will be measured and given a single value however this study is only interested in the skin friction losses. This means that only the unmixed section should be measured in CFD. Which will result in a different value for the losses compared to the 0D software as it is a smaller section of the overall mass flow. To be able to compare the CFD losses against the 0D performance software losses equation (3-10) will need to be used. In this equation the hydraulic diameter and mass flow was changed to accommodate only the unmixed section. This will make comparison between CFD and 0D performance software results possible.

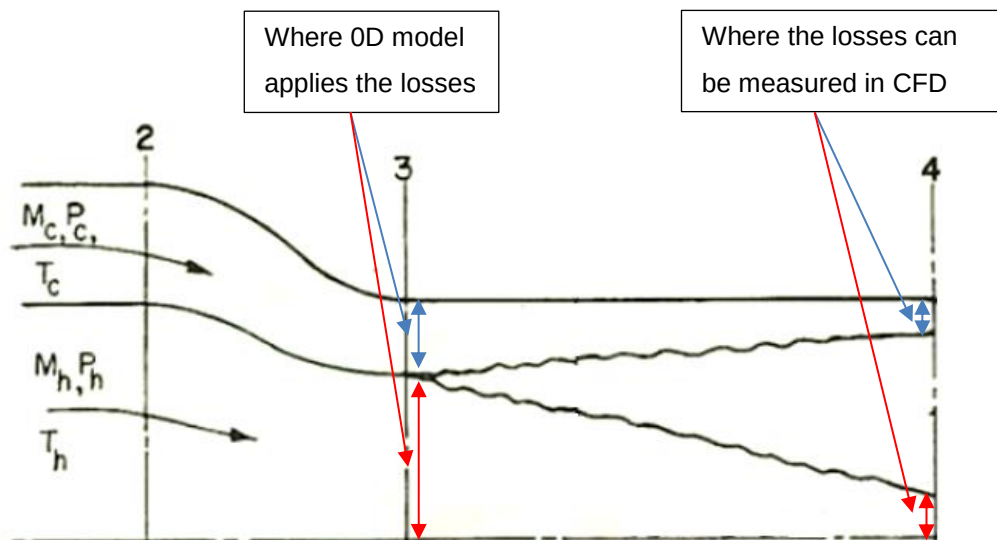


Figure 45 – Schematic measuring losses location difference between CFD and 0D model

The second part was the correction of the CFD results. Due to the complex nature of this simulation it was observed in multiple simulations that the static pressure at the inlet of both streams entering the mixer was not equal. This is a requirement for mixing, however, the deviation was very small and the rest of the flow conditions did show agreement with the thermodynamic models. As a result, the cold side mass flow was slightly smaller than specified in the thermodynamic results. The deviation in mass flow was between 0 – 1%. Once the deviation was measured, the correction was applied to the amount of mass flow that should be measured for the unmixed section. The results of these studies are shown in chapter 4.4.

3.4.3 Verification of CFD methodology

It was also required to verify if the chosen setup for the simulation with regards to boundary types, meshing, turbulence models and is accurate enough to predict the wall skin-friction of the mixer chamber. However, this type of verification to correctly assume skin-friction loss, but also take mixing into account was not found in the public domain. An alternative simulated case was found however, which focuses only on the skin-friction losses within a pipe. As the main concern of this model verification is the skin-friction losses, this method was deemed adequate enough. The verification case, called “Pressure Drop in Turbulent Flow Through a Pipe” can be found in the following references^{[74][78][79]}.

The test case that was originally simulated in the references had a Y^+ of 2.5. Furthermore, the test case also used a different turbulence model; the $k-\epsilon$. Both cases were simulated with the difference turbulence models. This made it possible to compare the variation in losses between two case and the verification data, in order to see if the setup for the current CFD investigation was adequate. The results are shown in chapter 4.3.

3.5 Weight estimation

The weight estimation of the derived engines is divided into three parts, the bare engine without the mixer weight, the mixer weight and nacelle weight. An automated process was created for the weight estimation for all the Pareto front engines, for all cases investigated. The bare engine weight estimation without the mixer was conducted using the Cranfield in-house software programme ATLAS. A general setup was chosen which was similar to the baseline engine weight estimation to make a comparison possible. Previous research^{[44][5]} has shown that this method for weight estimation produces results of acceptable accuracy compared to public domain data. In the following subchapter, the general setup of the weight estimation tool is discussed, as well as the added weight due to the mixer. The nacelle weight estimation procedure will also be explained.

3.5.1 ATLAS bare engine weight estimation

The software used for the bare engine (without the mixer) weight estimation is called ATLAS. This software is an in-house software design by P. Lolis^[44]. The software is able to estimate the weight for each component of the engine. The code uses an explicit method for component geometry generation, and has inputs for the component materials and their properties. The general setup for the geometry of the components is based on the results of the initial engine cycle and the annulus design procedure. An overview of which components are taken into account in the bare engine weight estimation is shown in Figure 46. It can be seen that for each engine component, all the relevant sub-components are taken into account in the weight estimation.

The turbomachinery components and the combustor are the major contributors to engine weight. However, there are other components that also need to be considered. These are the ducts connecting the turbomachinery components, the

components' casing, the shaft weight, frame and controls & accessories. More details about these individual components' weight estimation can be found in reference^[44].

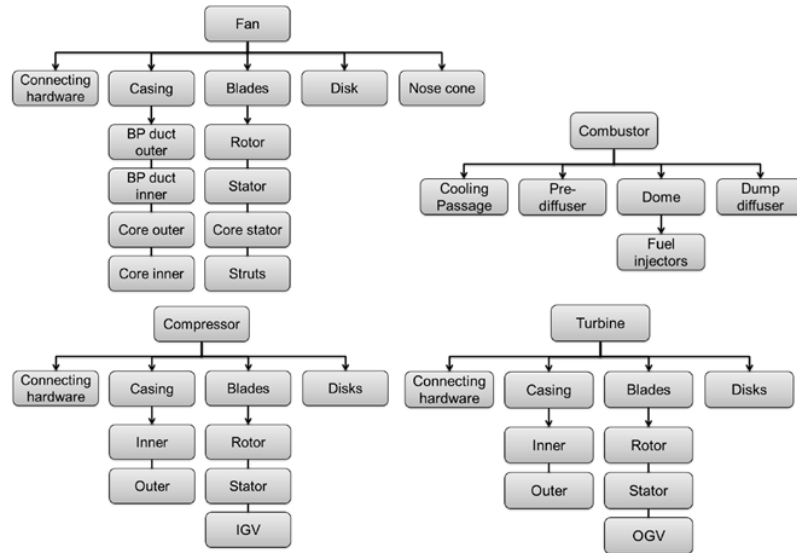


Figure 46 – Main engine modules in ATLAS weight estimation^[44]

Tables with list of what ATLAS requires for input are shown in Appendix A. Table 14 shows the input parameters for the fan, Table 15 shows the input parameters for the compressors, Table 16 shows the input parameters for the turbines and Table 17 shows the input parameters for the combustors. The majority of the required input parameters for this programme are obtained from the results of the Pareto front engines produced by the optimiser. These included the OPR, BPR, TR over component, PR over component, mass flows for each component and rotational speed of each component. This information would be read from each individual engine configuration on the Pareto front and would be used weight estimation for each engine. However, there are a number of design considerations that have become fixed assumptions for all the engines. The Mach numbers, stage loading and de Haller numbers, these were based on previous simulation done with the Trent 1000^[5] (which showed agreement between ATLAS

and the real weight of Trent 1000), technology estimates as well as information provided by the industrial sponsors.

This was done to allow for automatic weight estimation calculations, for all the engines from all the cases. By fixing some of the design considerations, the calculated annular gas path and weight estimation may not be the optimum solution for that particular engine. However, it is considered that the produced results are representative for this engine type and their variation is consistent across the design space. This allowed for a massive amount of engines to be simulated within ATLAS and the trends to be investigated. Individual configuration for each engine for the optimum gas path would not be visible. Once all input parameters have been set for each individual engine ATLAS will calculate the gas path and the weight of each component. ATLAS will use the work compatibility and mass flow continuity to designs all the components and the gas path through an iterative process. The process is finished once all critical parameters fall within the set targets.

When looking at the weight of the compressor ATLAS uses the parameters mentioned in Table 15. This will allow ATLAS to estimate the overall efficiency and the Outlet and inlet annulus dimensions. The velocity triangles in addition to the annulus dimensions for each stage can be calculated using the mass flow continuity and work compatibility principles. The design follows an iterative process and finishes when all critical parameters fall within their technological limitations or targets. Using the annulus dimension atlas can calculate the required volume for each component. Using that volume and the material information ATLAS is able to calculate the weight of each of these components.

The weight of the gearbox is calculated slightly differently. This calculation uses the power requirement as well as the torque requirement for the gear box, it is also based on a single-stage helical gearboxes. By calculating the required gear

size and the required teeth size the gearbox main weight can be determined. Furthermore using weight information for a catalogue the weight of the bearings can be determined. Using the weight of these two component the remaining weight can be estimated as percentage of the weight that has already been calculated. This methodology was validated and confirmed in the following reference^[5] which also provide a more in depth explanation. It should be pointed out that this method was established using an optimiser to find the optimum gearbox which tried to minimise the weight as well as maximise the efficiency of the component which lead to gear boxed which were small but wide. This does influence the overall engine length and therefore can have an effect on the nacelle weight and this was not accounted for when optimising the gearbox.

3.5.2 Mixer weight estimation

As mentioned before, no clear methodology was found about the weight estimation for the mixer. Therefore, a method has been developed that assumes the mixer is similar to a duct. When considering the mixer weight, there are three components which need to be assessed. These are the mixer casing, the mixer plug, and the mixer nacelle section. These three components are shown in Figure 47. The orange section is the mixer nacelle, the blue section is the mixer casing, the red section is the mixer plug. That the mixer nacelle and case are considered seperatly is because the mixer the mixer cowl which is what that section is called does not need to be solide and can be hollow with support struts. The mixer plug is the section of the tail-cone that is concidered to be the mixing chamber, the remaining section of the tail-cone would be concidered the nozzle plug.

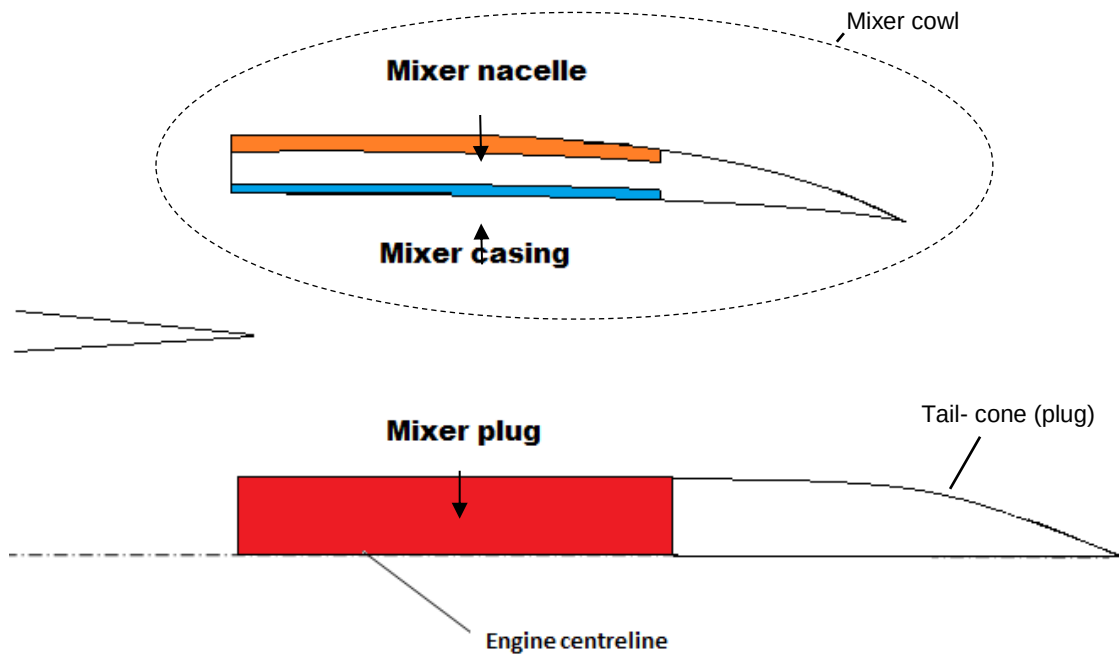


Figure 47 – Mixer weight components schematic

The mixer casing, as well as the mixer plug, follows the same method for weight estimation. The first assumption is that the mixer is essentially a duct. Therefore, it can be assumed that the weight of the casing and the plug can be estimated through the volume and density of the material of those components^[44]. The following equation can be used to estimate the weight of the components.

$$\text{Mass} = \text{Volume} * \text{density} \quad (3-13)$$

To determine the volume of the mixer casing and the mixer plug, it is assumed that both components are hollow cylinders with a specified wall thickness. The wall of the casing, as well as the plug, will need to be thick enough to withstand the pressures in the mixer during operation. These are the static pressures of either the cold and hot streams, depending on location. Equation (2-49) can be used to determine the minimum thickness required. The yield strength and density are shown in Table 3. A minimum wall thickness of 10 mm was decided following feedback from the industrial partner. The geometry of the flowpath of the mixer is also required. The plug diameter is set at 0.5 meters, as mentioned

previously in chapter 3.2.6.3. The length and diameter of the mixer are extracted from the results of the optimisation. With all the above data, a basic weight estimation can be carried out on the mixer casing and plug sections. The weight of the nacelle section, not yet covered, is discussed in the following chapter.

3.5.3 Nacelle weight estimation

The nacelle weight is the final piece of the mixer weight estimation. However, a detailed design falls outside of the scope of the project, as the engines with VGES follow a very similar overall design to those without. It was decided to use the same procedure as that for the baseline engine, which is based on reference^[43]. As mentioned in chapter 2.3.6, the nacelle weight estimation is conducted using only the bulk dimensions of the gas turbine. The basic concept of this design is shown in Figure 21.

To simplify the nacelle weight estimation, it was assumed that all engine sizes are very similar. It has been established in another method, which uses the Cranfield in-house design tool GEMINI^[82], that there is a relationship between the fan diameter (d_f) and other engine dimensions which is listed below;

- $d_a = \frac{d_f^{0.26}}{0.94} / 0.72$
- $d_c = d_f * 1.023$
- $L_c = d_c * 1.01325$
- $L_b = L_c * 1/2$
- $L_a = L_c * 1/4$

Adopting this assumption makes the overall nacelle weight to be a function of fan diameter. Therefore, by using equation (2-48) it allows for the process to be automated, as the fan diameter can be extracted from the optimisation results. It is important to clarify that the engine length for this calculation does not account for the mixer. The value for nacelle surface density was based on typical values

of the larger gas turbines^[43]. To account for the mixer length added onto the nacelle, equation (2-48) will need to be further developed into the following expression;

$$WT_{nac} = \pi * k_{nac} * (2 * L_C * d_C + L_b * d_a + 2 * L_a + L_m * d_a) \quad (3-14)$$

In this expression L_m represents the mixer length, extracted from the optimisation results. Furthermore, a verification check needs to be made, to ensure that d_a is substantially larger than the mixer diameter. If that is not the case the mixer outer diameter multiplied by 1.15 will need to be taken instead. This will allow for sufficient space for the casing around the mixer. As previously mentioned, the weight estimation method for the mixer could not be verified against comparable data. However, through discussions with the project's industrial partner, it was concluded that this method would suffice for the scope of the project.

The overall weight can now be calculated, by adding the individual components together. In order to make a fair comparison, the mixer plug weight was not included when comparing engine weights. This is because the baseline engine weight estimation does not account for plug weight.

The weight estimation for the VGES, in regards to the added weight due to the variability of areas, has not been carried out. There are two reasons for this; the first is the lack of information available in the public domain regarding how to estimate this, or data on comparable variable nozzle weights. The second and most important reason is that a study carried out by the author revealed that there is very little, if any, performance benefit using variable geometry areas throughout the mission profile of the aircraft. The results of this study will be shown in chapter 5.2

3.6 Engine operability studies

As mentioned previously, good gas turbine operability is very important. The goal in these studies was to verify that the gas turbine configurations derived do not experience dangerous aerodynamic instabilities throughout their operating envelope. A complete study falls outside of the scope of the project. However, it is required to verify that the engines created in the preliminary design stage are able to operate without significant instability issues. It should also be noted that not all the significant instabilities can be investigated currently, as not all information regarding these instabilities is available in the public domain.

3.6.1 Operability study

The only case of engine operability that was investigated assessed whether the compressors approach surge conditions. Other stability issues, such as fan flutter, were not taken into account. The control parameter that is available in our thermodynamic engine modelling tool to prevent the engine from surging is the surge margin. The surge margin has been discussed in multiple references in the public domain^{[4][17][25][80][48]}. To determine the surge margin, one can use the difference between surge point pressure ratio and the operating point pressure ratio. Figure 48 shows this surge margin on a generalized compressor map.

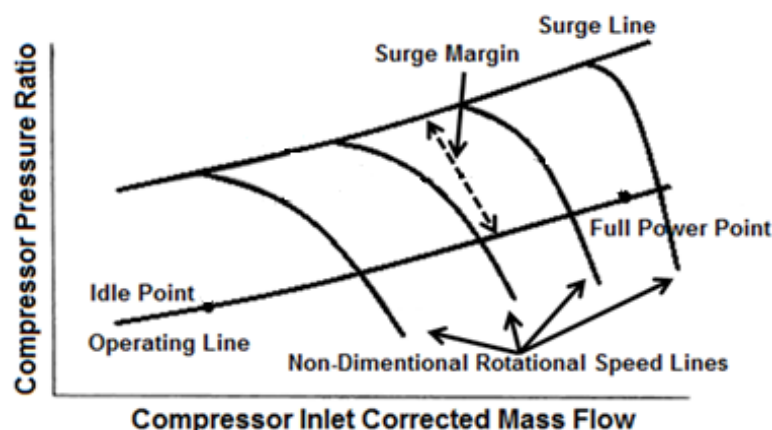


Figure 48 – Compressor characteristic and surge margin [4][17][25][80]

The thermodynamic engine modelling tool uses the following two equations to define the surge margin, which are based on a pressure ratio for a fixed non-dimensional rotational speed^{[67][54]}. Max and min refer to the maximum and minimum value of pressure ratio on the non-dimensional rotation speed line.

$$\text{Surge Margin : } z = \left(\frac{PR_{OP} - PR_{min}}{PR_{max} - PR_{min}} \right)_{CN} \quad (3-15)$$

$$\text{Relative Non – dimensional Rotational Speed : } CN = \frac{\left(\frac{N}{\sqrt{T_{in}}} \right)_{OP}}{\left(\frac{N}{\sqrt{T_{in}}} \right)_{DP}} \quad (3-16)$$

The thermodynamics performance software only allows these values to be set for design point. To ensure the stability of the engine, a limited minimum value was chosen for the surge margin based on the available data in the public domain. This surge margin was sufficient enough to allow for operation throughout the whole flight envelope accounting for transient and steady state operation^{[4][48]}.

The stability study was conducted as follow. All three operating points were investigated by increasing power settings beyond the designed operating point. Furthermore, the effect of decreasing the power setting to part speed was also investigated. Of the three operating points, the end of runway with ISA+15 °C and top of climb with ISA+10 °C are considered to be critical as these operating points are closer to the surge line in these conditions. All the running lines were plotted on associated compressor maps. This was done to visually observe if the surge line was exceeded and if a sufficient amount of surge margin was still left. The study was conducted on a number of engines on the Pareto fronts over a range of Specific Thrust. The result of this study are shown in chapter 5.3

To avoid potential fan operability issues, another solution can be used which has been described in great detail in chapter 2.4. It has been mentioned that a

Variable Area Fan Nozzle (VAFN) is able to increase or decrease the mass flow going through the fan, moving its operating point away from the surge line. The instability of the fan is expected to be a problem when moving to lower Specific Thrust turbofan engines, like the ones investigated in this study. This is why one of the objectives in this study was to verify if there is a need for VAFN for the engines being investigated. This investigation was carried out with special considerations for the operating conditions at both top of climb and end of runway.

3.7 Total mission fuel burn estimation

With the engine weight estimation completed, the final stage of the study is the total mission fuel burn. The aim of the mission fuel burn study is to estimate the total amount of fuel needed to complete a given mission profile with the optimum cycle engines. Comparisons are made against the baseline engines' total mission fuel burn and also between the two engine architectures. The aim of this study is to identify any potential areas where the VGES has reduced the mission fuel burn, and could therefore be a beneficial system.

The mission fuel burn studies were only performed after the cycle optimisation study. It is assumed that most of the time the engine will be flying at cruise, as the optimisation took this into account. This means that the mission fuel burn produced for each engine might not be the truly optimum mission fuel burn, but a very close to optimum solution. Additionally, as the same methodology was used for both the baseline and VGES engines, any discrepancies should have an equal effect in both mission fuel burn studies, and should not interfere with the outcome of this research.

3.7.1 Mission profile

The mission profile is determined by the aircraft size and aircraft range. These have both been mentioned in Table 1. The engine is designed for an A330- sized aircraft, and has to have a nautical mile range of 3000. For safety reasons extra fuel is carried on board. The mission profile selected was based on the available data in the public domain regarding A330 flight data^{[83][84][85][86][87]}. A general schematic of a mission profile is shown in Figure 49. The mission profile consists of multiple flight conditions which are shown below in Table 7

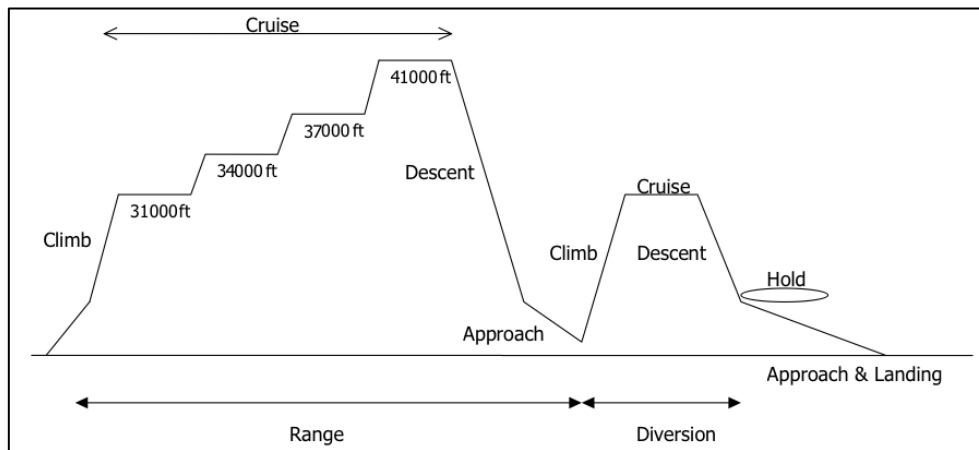


Figure 49 – Schematic of a typical mission profile^[88]

Table 7 – Mission profile segments^{[83][84][85][86][87]}

Mission segments	Altitude	Unites
Ground-idle Taxi	0	ft
Take-off and initial climb	0 – 1500	ft
Climb	1500 – 39,000	ft
Cruise	35,000	ft
Decent	35,000 – 1500	ft

Landing	1500 – 0	ft
Ground-idle Taxi	0	ft
Diversion climb	1500 – 20,000	ft
Diversion cruise	20,000	ft
Diversion decent	20,000 – 1500	ft
Diversion approach and landing	1500	ft
Hold	1500	ft

From the graph and table it can be seen that in this mission profile the cruise segment is carried out at a constant altitude, unlike the general mission profile schematic shown below. The main mission profile is simulated using 101 operating points. Out of these, 68 are used to simulate cruise, as the total weight in fuel is reduced through cruise. The remaining operating points are used to simulate the remaining segments respectively.

3.7.2 HERMES mission fuel burn estimation

To conduct the mission fuel burn study the Cranfield in-house software HERMES was used. HERMES takes into account the engine performance, the engine and fuel weight, and nacelle drag at different operating points. The input parameters for HERMES can be extracted from the optimisation results, geometry and weight estimation results. HERMES is linked to the thermodynamic performance software TurboMatch which has been described in chapter 3.2.1. This coupling allows HERMES to run the 101 operating points mentioned above automatically, for each engine configuration. The schematic on how HERMES works with TurboMatch is shown in Figure 50.

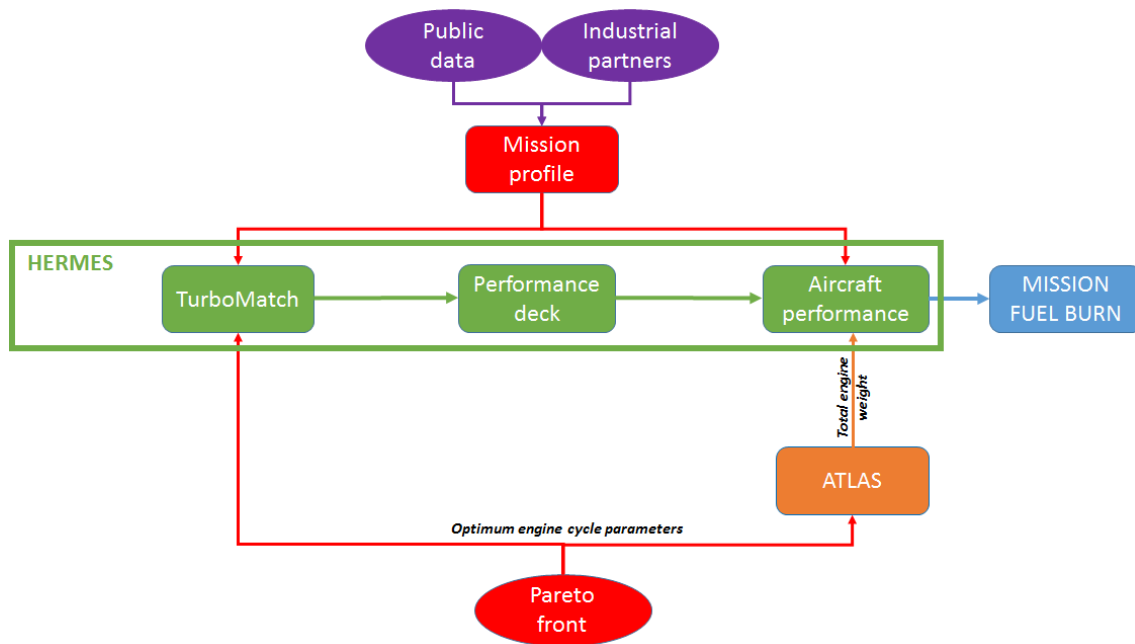


Figure 50 – Schematic HERMES and TurboMatch linked

From the figures it can be seen that HERMES requires engine configuration, this is a result of the optimiser cases that has been run. Furthermore, it will require a total of 101 operational points chosen to represent the mission profile which has been provided from industrial partners and public data. From this p TurboMatch runs to get the results of the 101 operating points. It than transforms the 101 operation points results in a single engine deck which will include all the relevant data from those operational points. With that single file and the weight information from ATLAS from the same configuration, as well as the length of the engine (which also comes from ATLAS), HERMES can perform integrated aircraft and performance simulations. The simulation uses an iterative process to find the correct values associated to the mission. It uses initial guesses estimated for mission duration and assigned tolerances for fuel weight, mission and diversion range, and climb and descent duration to reach a converged solution.

The code was not adapted to allow for variations in the flow areas of the VGES along the mission profile. The reason for not altering the code to allow for this

was a study carried out by the author which revealed that there is very little, if any, performance benefit using variable geometry areas throughout the mission profile of the aircraft. The result of this study are shown in chapter 5.2

3.8 Specific case studies

For each of the engine architectures considered, multiple configurations were investigated. This is due to the number of variable areas that the VGES can incorporate. The effect of each variable area option, as well as a combination of variable areas was investigated. The case number and respective engine configurations investigated are shown in Table 8. It should be noted that these are only the main case studies. Many more cases were investigated to assess the sensitivity of certain aspects of the VGES. In total there were 16 different configurations to investigate. In Table 8 “Fixed” represents that none of the areas are variable at any operating point. VAFN stands for variable area fan nozzle, VACN represents the variable area core nozzle and VAM is the variable area mixer. For these cases area variability was only allowed at EOR and TOC.

Table 8 – Specific case studies schematic showing different configurations.

Case No.	Architecture	Fixed	VAFN	VACN	VAM
1	Geared	X			
2	Geared		X		
3	Geared		X	X	
4	Geared		X		X
5	Geared			X	
6	Geared			X	X
7	Geared				X
8	Geared		X	X	X
9	Direct-drive	X			
10	Direct-drive		X		
11	Direct-drive		X	X	
12	Direct-drive		X		X
13	Direct-drive			X	
14	Direct-drive			X	X
15	Direct-drive				X
16	Direct-drive		X	X	X

4 Results verification & sensitivity

In this chapter the results of the various verification and sensitivity studies are presented. These results were created using various versions of the optimiser. Sensitivity studies were carried out to get a clearer understanding of the effect of certain designer choices regarding the performance of the mixed exhaust turbofan engine, with and without VGES.

4.1 Mixer verification and sensitivity study

4.1.1 Incomplete mixing

The first verification study that was conducted was for the applied bricking strategy for incomplete mixing or mixing efficiency. This was essential to assure that the chosen bricking strategy would produce accurate results when taking mixing efficiency into account. The bricking strategy is shown in Figure 36. The turbofan engines used for this verification study are shown in Table 6. Losses in these engines were considered fixed, therefore the mixing efficiency is independent from the losses in this verification.

The first engine to be simulated was the RB211-535E4. When the engine was matched with the data in the public domain the mixer was removed and the separate exhaust turbofan was created to find the optimum baseline thrust. As mentioned in chapter 3.4.1 the optimum baseline thrust was found by only varying FPR, while maintaining OPR, and the power distribution between the IPC and HPC, and a constant TET. Once the baseline thrust was established, the mixing efficiency of the mixed exhaust engine was increased, first to 75% and then to 100%. At each mixing efficiency level the FPR was used as handle and varied. For each engine the theoretical thrust gain was calculated by comparing the thrust achieved against the optimum thrust of the baseline turbofan engine. The

results of this investigation are shown in Figure 51. The second engine model was also simulated in the same manner. The results of that investigation are shown in Figure 52.

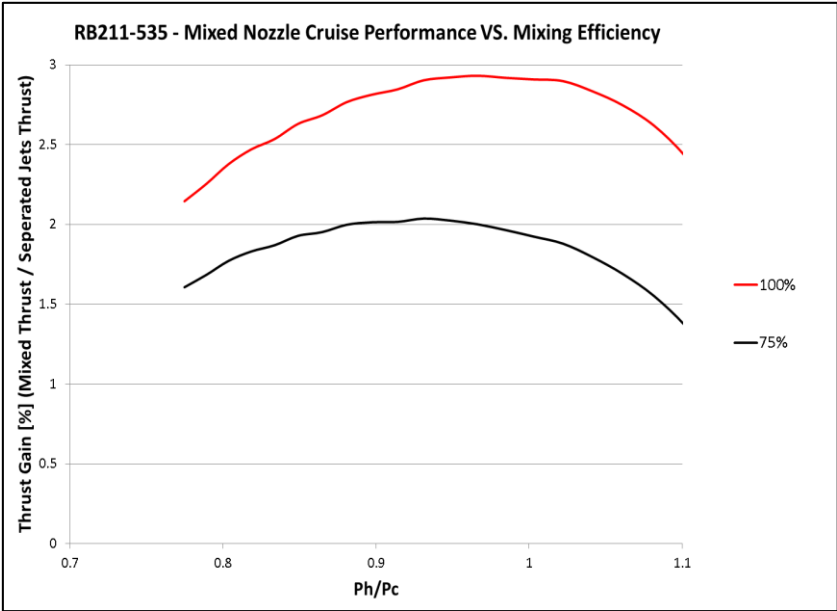


Figure 51 – RB211-535E4 mixed nozzle cruise performance vs. mixing efficiency

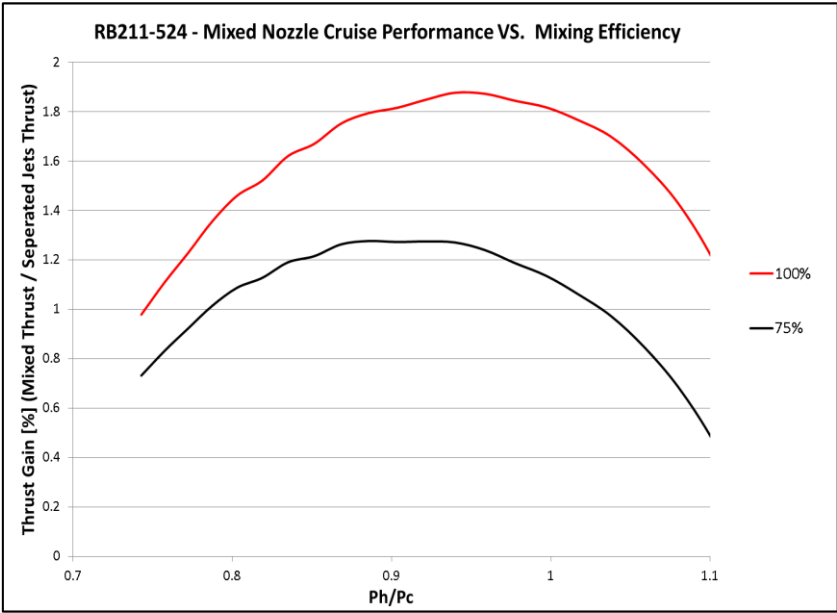


Figure 52 – RB211-524G/H mixed nozzle cruise performance vs. mixing efficiency

When analysing these figures and comparing them against Figure 16, a similar trend can be observed but also some differences. The first difference is that the maximum value for thrust gain is different for all three engines. This has to do with the effect of the bypass duct and the temperature which are shown in Figure 9. The fan diameter of the engine also plays a part as there is only a small difference between the RB211-535E4 and the RB211-524G/H in bypass ratio and temperature ratio. However there is a difference in fan size.

The second inconsistency is regarding the location of the optimum thrust gain compared to P_h/P_c (Total pressure hot stream / total pressure cold stream). In Figure 16 it can be seen that the optimum thrust gain is achieved when $\frac{P_h}{P_c} = 1$. However, this is not the case for the RB211-535E4 and the RB211-524G/H. There are many reasons for this, the first being that the engine analysed in Figure 16 was created using an analytical approach whereby flow characteristics at the inlet of the mixer could be set directly. Also the effect of inlet Mach number was not taken into account. The mixer inlet Mach number has an effect on the optimum mixer P_h/P_c ratio as it can be seen in Figure 12.

The similar trend that can be observed in the three figures is that engine performance degrades in the same manner. This is despite the fact the maximum value for thrust gain and the location of that value with respect to P_h/P_c are different for all three engines. Furthermore, when comparing the delta in thrust gain between the optimum values for 75% and 100% mixing efficiency, it can be calculated that there is a reduction of thrust gain of 25%. This is consistent for all three engines.

A final verification case was run using an engine derived through optimisation. This was done to establish that the same trends exist when using partial mixing. The results of this case are shown in Figure 53. It can be verified that the same

trend is present in this case. Furthermore, it can be seen that the thrust gain is very small compared to the previous case. The reason for this is that in the previous cases the full thrust for the mixed exhaust turbofans was produced by a single nozzle. However, in the UltraFan™ the major contributor to the thrust is the fan nozzle and not the core nozzle, therefore any thrust gain that the mixed nozzle experiences will have less impact on the overall thrust. This is of course dependent on how much of the bypass flow is mixed with the core flow.

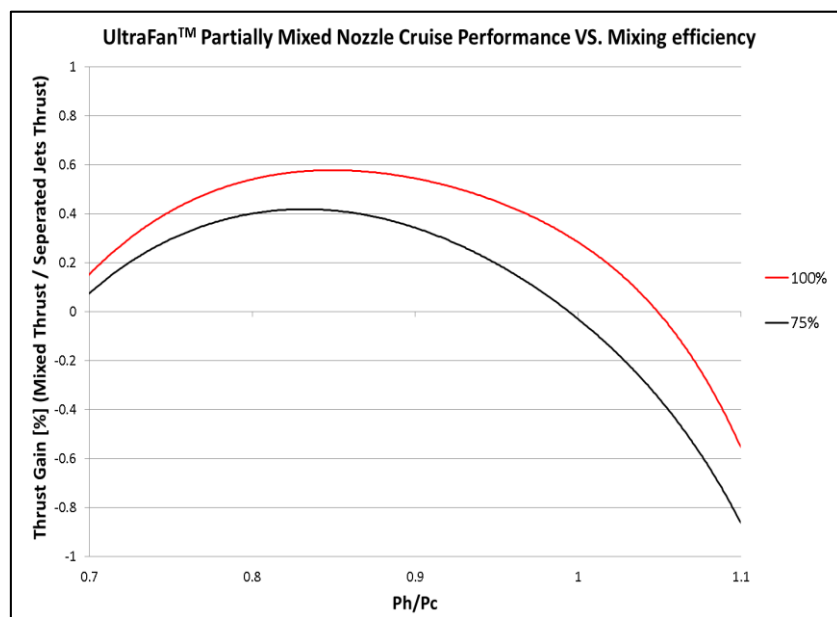


Figure 53 – UltraFan™ partially mixed nozzle cruise performance vs. mixing efficiency

4.1.2 Mixing efficiency sensitivity study

To get a better understanding of the design space available for the mixer and the effect the mixing efficiency has on the mixer inlet characteristics, a sensitivity study was conducted. The sensitivity study was on the effect of mixing efficiency on optimum mixer inlet pressure ratio, as this is directly controlled by the FPR. The sensitivity study was conducted using the same engines and methodology from the incomplete mixing verification study. For this a range of mixing efficiency

was simulated at different inlet mixer pressure ratios. The mixing efficiency ranged from 10 to 100%. The mixing Mach number chosen was 0.5. The results for the RB211-535E4 are shown in Figure 54. The results for the RB211-524G/H are shown in Figure 55.

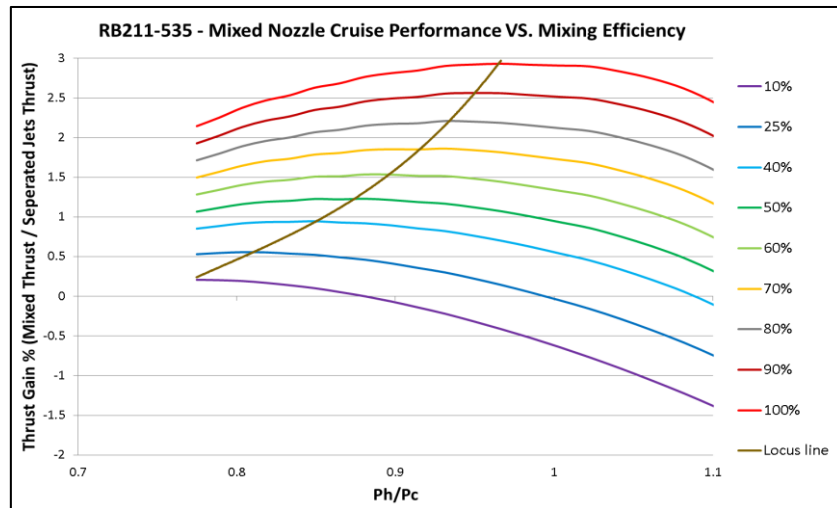


Figure 54 – Effect of mixing efficiency on % thrust gain for RB21-535E4

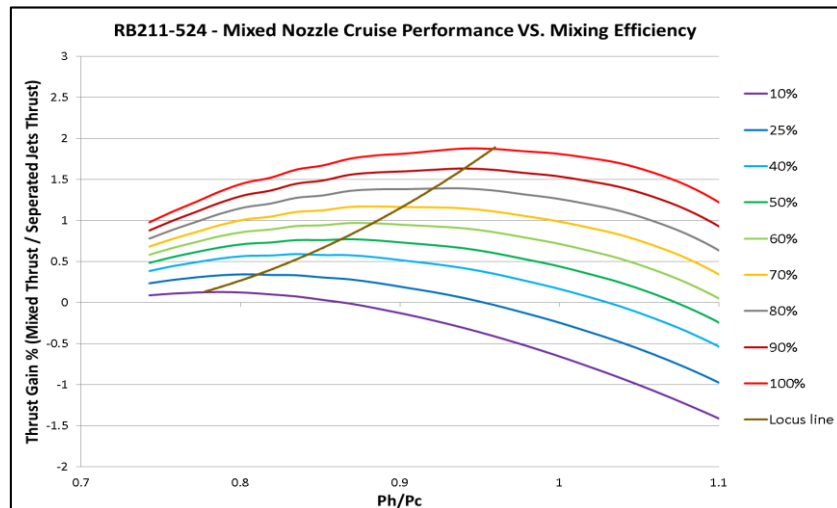


Figure 55 – Effect of mixing efficiency on % thrust gain for RB211-524G/H

When analysing these two figures multiple observations can be made. One observation is that the optimum mixer inlet pressure ratio is very similar for both engines at each point of mixing efficiency. However, this is not the case for the theoretical thrust gain which is substantially larger for the RB211-535E4 at 100% mixing efficiency than the RB211-524G/H. This cannot be a result of different BPRs as both engines have very similar BPR. The difference in potential thrust gain is due to the fan diameter, with the RB211-524G/H having a larger fan and about 40% more mass flow than the RB211-535E4. A closer look at the FPR associated with the optimum mixer inlet pressure ratio shows that, even though the mixer inlet pressure ratio is the same, the FPR_{op} for the RB211-535E4 is larger than for the RB211-524G/H.

The thrust gain a mixed exhaust engine experiences, is mostly caused by the reduction in FPR and LPT pressure ratio compared to the separate jets. For a given turbofan the optimum FPR relates to a certain amount of energy conveyed between the core and the by-pass flow. In a separate exhaust turbofan this energy needs to be transferred through the LPT and the fan where some losses take place. However, for a mixed exhaust turbofan some of the conveyed energy can be transferred through the mixer, where no losses take place regarding transfer efficiency (other losses take place), therefore reducing the LPT pressure ratio and FPR. The reason for the reduction in thrust gain for larger size engines with roughly the same BPR is that larger separate exhaust turbofan engines are designed for lower Specific Thrust and thus lower FPR. Therefore, in a mixed exhaust turbofan the impact on the LPT pressure ratio and FPR will be reduced.

Another important observation is that the optimum mixer inlet pressure ratio P_h/P_c increases as the mixing efficiency increases. This change in optimum mixer inlet pressure ratio has been depicted by the 'locus line' in both figures. As the FPR is directly linked to the mixer inlet pressure ratio it can be seen that as the mixing efficiency increases the FPR_{op} decreases. The reason for this is that the FPR is primarily determined by the mixer, and looking at what Guha^[37] states, the FPR_{op}

is greatly influenced by the Specific Thrust of a mixed exhaust engine. The Specific Thrust can be related to the size of the fan. Therefore, it can be said that as the size of the mixed exhaust engine fan increases the optimum FPR reduces. Increasing the mixing efficiency is very similar to increasing the size of a mixed exhaust engine fan and therefore this shows a very similar effect on FPR.

It can be concluded that as the engine fan size reduces the potential thrust gain available for the mixer increases. However, this influences the FPR_{op} and not the inlet mixer pressure ratio. Furthermore, the mixing efficiency plays a significant part in determining the optimum value for FPR as that is directly linked to the mixer inlet pressure ratio.

4.1.3 Mixer inlet Mach number sensitivity study

The second sensitivity study conducted was on the mixer inlet Mach number for the hot stream (LPT exit) entering the mixer. The reason for this sensitivity study was to gain a better understanding of the influence the mixer inlet Mach number has on the stability of the thermodynamic engine model. It is also important to understand the influence the mixer inlet Mach number has on the inlet flow characteristic of the mixer, in regards to mixer inlet pressure ratio. For this study, the same engines as in the previous sensitivity study were used. The engines were simulated using different inlet Mach numbers (for the hot stream entering the mixer) from 0.1 to 0.5. Furthermore, the engines were also run over a mixing efficiency range from 10 to 100%. It needs to be pointed out that the losses were fixed in this study. This means that the increased diameter of the mixer due to the lower Mach numbers do not affect the losses experienced in the mixer. From this study several interesting observations were made.

Figure 56 shows the mixer inlet Mach number against the optimum mixer inlet pressure ratio. In this figure P_c/P_h is plotted to be able to compare Figure 56 with

Figure 12. When comparing these figure the similarities between the two figures can be seen. The optimum mixer inlet pressure ratio is different between the two engines but this is what is expected as the engine for Figure 12 is significantly smaller than the RB211-535E4. It can be observed that the mixer is working in the same manner as in the analytical approach.

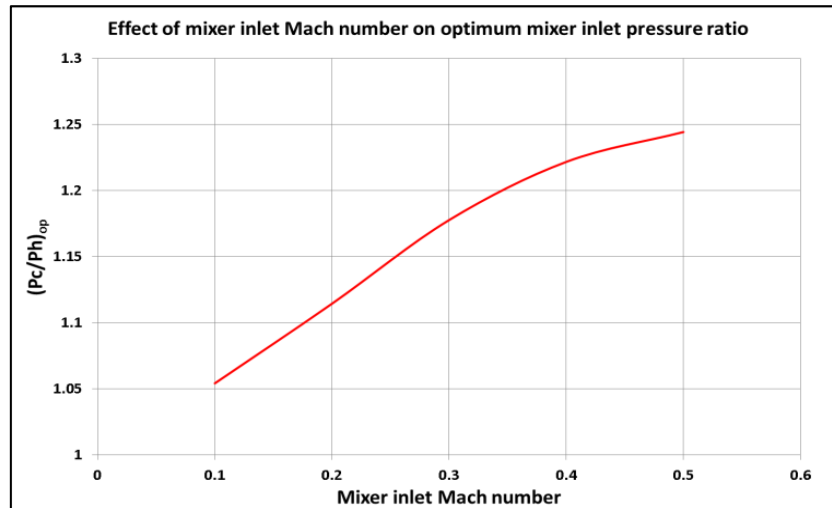


Figure 56 – Effect of mixer inlet Mach number on optimum mixer inlet pressure ratio for RB211-535E4 at a mixing efficiency of 25%

Expanding Figure 56 to include all mixing efficiencies produces Figure 57. It can be observed from this figure that increasing mixing efficiency changes the optimum mixer inlet pressure ratio. For the lowest Mach number, this optimum value is already close to one, from 60% mixing efficiency to 100%. However, going to higher inlet Mach numbers, say from 0.3 to 0.5, an inlet pressure ratio of 1 can never be achieved. It can be concluded that the mixing efficiency is dependent on the mixer inlet pressure ratio as well as the mixer inlet Mach number. Furthermore, for a fixed mixer inlet pressure ratio the mixing efficiency increases significantly with the increase of the mixer inlet Mach number.

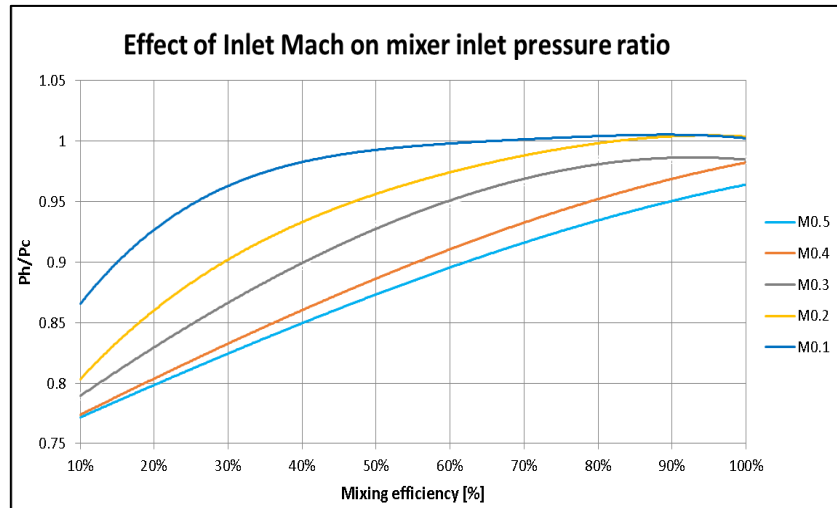


Figure 57 – Effect of inlet mach number and mixing efficiency on mixer inlet pressure ratio for RB211-535E4

The next case examines the effect of inlet Mach number on thrust gain for the RB211-535E4 model, with a constant mixing efficiency. Figure 58 and Figure 59 illustrate the effect of inlet Mach number on thrust case for a constant 25% and 100% mixing efficiency respectively.

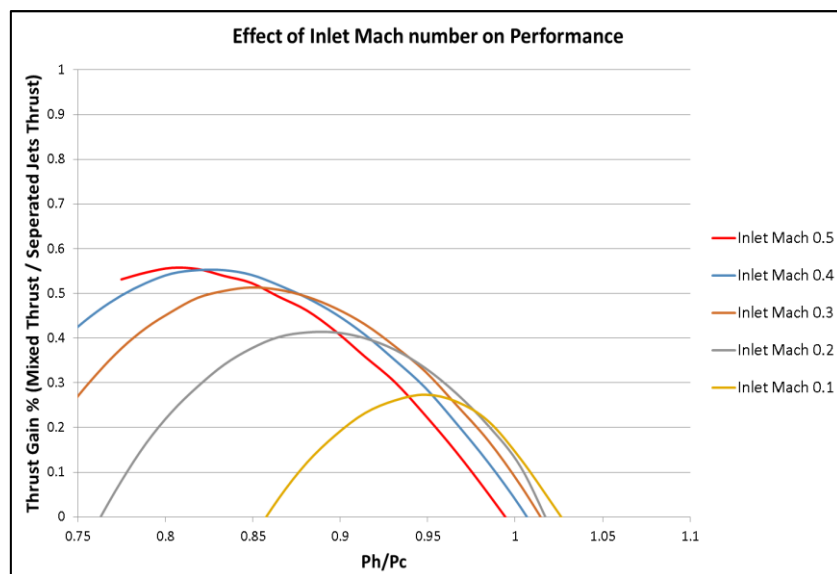


Figure 58 – Effect of mixer inlet mach number and pressure ratio on % thrust gain for RB211-535E4 (mixing efficiency 25%)

Looking at Figure 58, several observations can be made. The first is that there are no results for an inlet Mach number of 0.5 with a lower mixer inlet pressure ratio than 0.77. The reason for this is that the Mach number is only fixed for the hot mixer inlet. A fixed hot mixer inlet Mach number results in a static pressure for a given total pressure. This static pressure will need to be used in the cold side which will result in a Mach number for that inlet. When going to lower mixer inlet pressure ratios the cold section total pressure is significantly higher than the hot section. Thus using an inlet Mach number of 0.5 at the hot mixer inlet, will result in a significantly higher Mach number at the cold mixer inlet. Below 0.77 mixer inlet pressure ratio, the Mach number at the cold inlet goes supersonic. When that occurs other problems arise, such as shockwaves which cause significant losses. Therefore, any results beyond that point are considered non-viable.

Another observation is that the available design space seems to increase when going to higher Mach numbers. The positive thrust gain seems to be restricted over a narrow range of pressure ratios for an inlet Mach number of 0.1. However the range of pressure ratios resulting in a thrust gain increases substantially with inlet Mach number. It can also be seen that the maximum thrust gain achievable at the inlet Mach number of 0.4 and 0.5 is very similar.

When analysing Figure 59, different observations can be made compared to Figure 58. The first concerns the higher mixer inlet pressure ratios. It can be observed that there is a specific upper limit to the mixer inlet pressure ratio for each inlet Mach number. When the Mach number is lower than 0.5, this reduces the maximum achievable value for mixer inlet pressure ratio. The reason for this is that at that point the static pressure of the mixer hot inlet is too close or exceeds the total pressure of the mixer cold inlet. Therefore, no mixing can take place and the thermodynamic cycle simulation software cannot produce a viable engine.

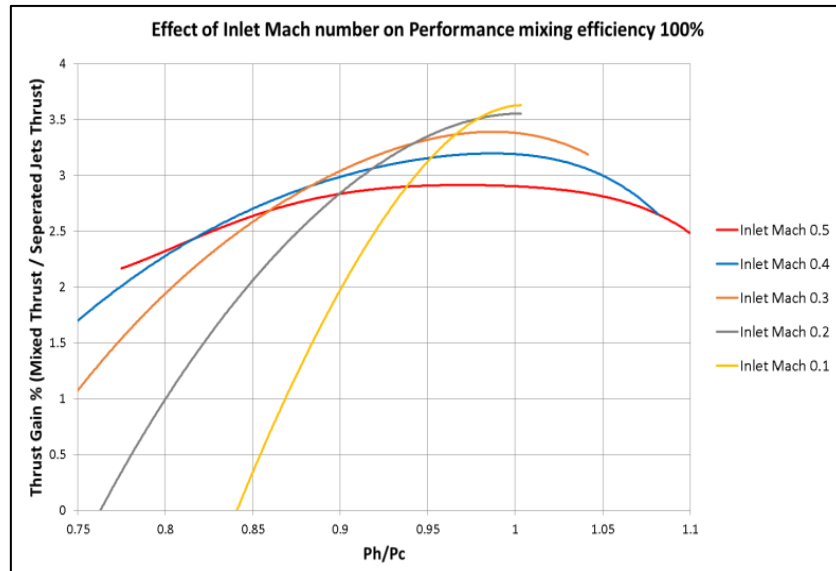


Figure 59 – Effect of mixer inlet mach number and pressure ratio on % thrust gain for RB211-535E4 (mixing efficiency 100%)

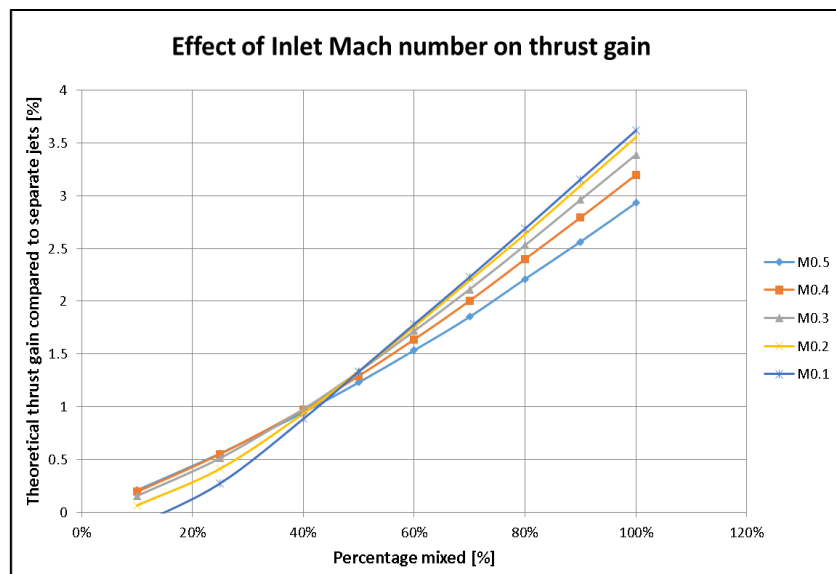


Figure 60 – Effect of mixer inlet mach number and mixing efficiency on % theoretical thrust gain for RB211-535E4

Figure 60 clearly presents the effect of mixing efficiency on the theoretical thrust gain. The trend presents a fairly linear fashion with the thrust gain increasing with

mixing efficiency. The mixer inlet Mach number affects the slope of the lines. As the mixer inlet Mach number increases, the slope decreases. Additionally, moving towards the separate jet case, the effect of increasing mixer inlet Mach number is positive regarding thrust generation. This is because greater mixer inlet Mach numbers lead to greater FPR_{op} which resembles the separate jets case. Another interesting observation is that the point where the slope of the lines begins to change is the design point value of 25% mixing efficiency. After that point, lower inlet Mach number lines present steeper slope and due to the fact that their thrust gain was lower at design point they intersect at a specific value of mixing efficiency. At the high region of mixing efficiency the increase in mixer inlet Mach number results in a negative effect on the thrust gain.

4.2 Critical pressure loss investigation

With the thermodynamic model completed and the skin friction methodology implemented within the optimiser, a number of optimisations were run to investigate which of the pressure losses that have been implemented into the thermodynamic model is the most critical. It should also be noted that these optimisations were using the assumption that the friction factor is a fixed value of 0.04. This investigation was conducted using the geared architecture with no variable areas for the VGES. The results of this investigation are shown in Figure 61. In the figure the mixed fixed configuration with no losses is considered as baseline. The percentage reduction of mid-cruise SFC is based on the SFC value of the baseline engine with a Specific Thrust value of 8.2 (lbf/(lb/s)) or around 80 m/s.

When examining the figure the first thing to be observed is that the penalty on SFC when including all losses is around 5%. From this 5% penalty, roughly 1% is due to the losses associated with the mixing of the two streams entering the

mixer, another 0.5% to 1% is associated with the cold chute pressure loss. The remaining 3% to 3.5% is associated with the skin friction losses within the mixer.

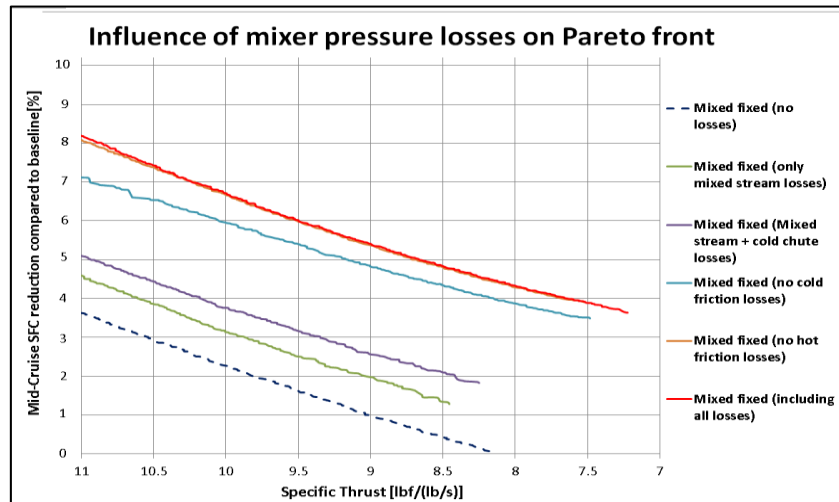


Figure 61 – Influence of mixer pressure losses on SFC vs Sp. Thrust Pareto front

Further investigation revealed that when removing the cold-section friction losses the Pareto front moves downwards and depending on the Specific Thrust level, SFC improves between 0.5% and 1% (blue line). However, when removing the hot-section friction losses, there is nearly no shift in the Pareto front (orange line). Subsequently, it can be concluded that the skin friction losses are considered the most critical loss mechanism, and will determine if the VGES patent will be viable. Furthermore, cold-section friction losses are more influential than the hot-section friction losses. It is essential that these losses are as accurate as possible.

4.3 Verification of CFD methodology

The verification of the CFD methodology was conducted using a test case prescribed in literature^[79] and section 3.4.3. In total two cases were simulated the first case was simulated with a turbulence mode k- ϵ against the data in the

literature. The second test case was nearly exactly the same the only difference was the turbulence model k- ω STT. The geometry of the cases is a straight tube with a diameter of 0.1 meter and a length of 8 meter.

Table 9 – CFD methodology verification results

	Units	Cases		
		1 st	2 nd	Delta
Turbulence model	-	k- ϵ	SST k- ω	-
Inlet conditions	m/s	50	50	0
W	kg/s	1.57	1.57	0
$P_{t-inlet}$	kPa	103.249	103.232	0.015
$\frac{\Delta P_{t(inlet-outlet)}}{P_{t-inlet}}$	%	0.59	0.57	0.02

The results of both test cases can be seen in Table 9. Even though both cases have the same inlet boundary condition there is a very slight discrepancy between the inlet total pressures calculated. This is so small that it will not influence the results for the verification of the skin friction methodology (also the inlet total pressure is set as an input in the model). Looking at the total pressure loss experienced in the pipe of the baseline case and the verification case, it can be seen that a very small percentage of pressure loss is experienced in the length of the pipe. The total pressure loss predicted is very similar comparing the two cases, despite the differences in the modelling and particularly turbulence model. With the results showing agreement between the two test cases and as the original test case was already verified against the moody diagram it can be concluded that the turbulence model k- ω STT is also suitable to estimate skin friction pressure losses.

4.4 Verification of skin friction methodology

Several computational fluid dynamic simulations were run to investigate if the value the overall loop within the optimiser calculated for the skin friction pressure loss within the mixer was accurate. This was an iterative process with each set of CFD simulations further improving the optimiser accuracy in calculating the skin friction pressure loss. The first set of CFD simulations was setup using the optimisation results which were produced with the assumption that the friction factor would be a fixed value of 0.04.

4.4.1 Mesh independency study

The first study was to investigate the influence of the mesh density upon the result of the CFD simulation in order to assure that the chosen node count was sufficient enough to capture the total pressure loss due to skin friction. This study was conducted using an engine from the Pareto front mentioned previously. The node count was varied from 25,000 to 300,000 nodes. Furthermore, Y^+ was the same for each simulation. Once each mesh had reached second order convergence, the total pressure at the outlet of the unmixed hot-section and unmixed cold-section was measured. The results for the cold section are shown in Figure 62 and for the hot-section in Figure 63.

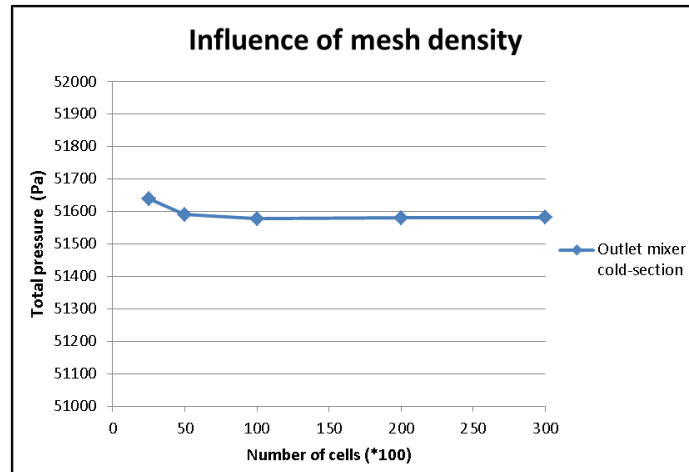


Figure 62 – Influence of mesh density of unmixed cold-section

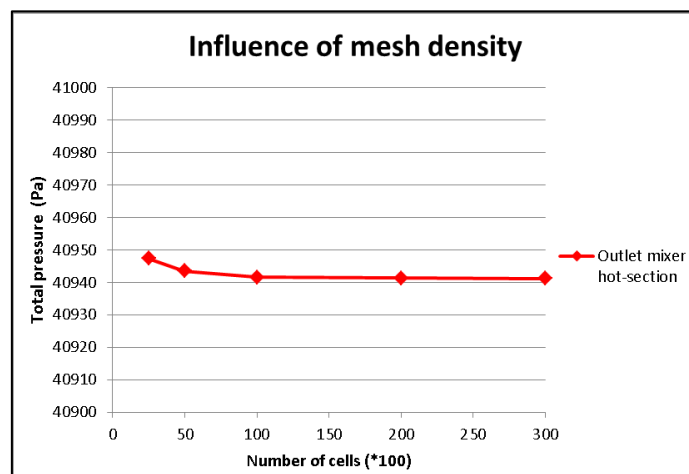


Figure 63 – Influence of mesh density of unmixed hot-section

The following can be observed when analysing the results. Firstly, it can be seen that the outlet total pressure of both the unmixed cold-section and hot-section becomes constant once the cell count is equal to or exceeds 100,000. Secondly, the deviation from the stabilised value that occurs at a node count of 50,000 is very small for both the cold- and hot-sections. However, once the mesh density is reduced further to 25,000, the deviation observed at this point has increased significantly for the cold-section but not for the hot-section. It should be noted that the losses observed for the unmixed hot-section are significantly less than for the

unmixed cold-section. However, the smaller deviations on the unmixed hot-section can still have significant effect on the accuracy when comparing against the thermodynamic model.

From a node count of 100,000 and above the current CFD simulation showed that the results were independent of the mesh. To ensure that a wide range of mixer geometries in regards to mixer length and diameter could be accommodated in with these CFD simulations, a higher node count of 200,000 was chosen. The chosen amount of nodes is sufficient in capturing the pressure losses due to skin friction within multiple sizes of mini-mixer.

4.4.2 Convergence

The convergence method was discovered through trial and error as direct convergence was not possible. To achieve second order convergence, the following steps needed to be taken; first the simulation was run with total pressure both inlets of the mixer equal to the hot-inlet total pressure and a lower mass flow at the exit. Thereafter, first the total pressure of the cold-section is increased until the correct value is achieved, then the mixer outlet mass flow is increased to achieve the correct value there. The last step is achieving second order convergence with the correct total pressure and mass flows. Figure 64 shows the typical residuals of the convergence process that are present in every case. From the figure it can be observed that the convergence took place a total of six times. Locations A to C are the first order convergence, achieving the correct pressure at the cold mixer inlet boundary. Locations D and E are first order convergence achieving the correct exit mass flow. Location F is the second order convergence.

Once second order convergence was attained, another check was carried out to see if mixing was achieved. A visual inspection of the total temperature distribution at the end of the domain confirmed whether the simulation had

achieved any mixing. A typical total temperature distribution in the simulation domain is shown in Figure 65. From this figure it can be concluded that mixing is taking place within the simulation. Once this was confirmed, a comparison between the flow characteristics of the CFD simulations against the results of the thermodynamic model was made to confirm the simulation results were accurate.

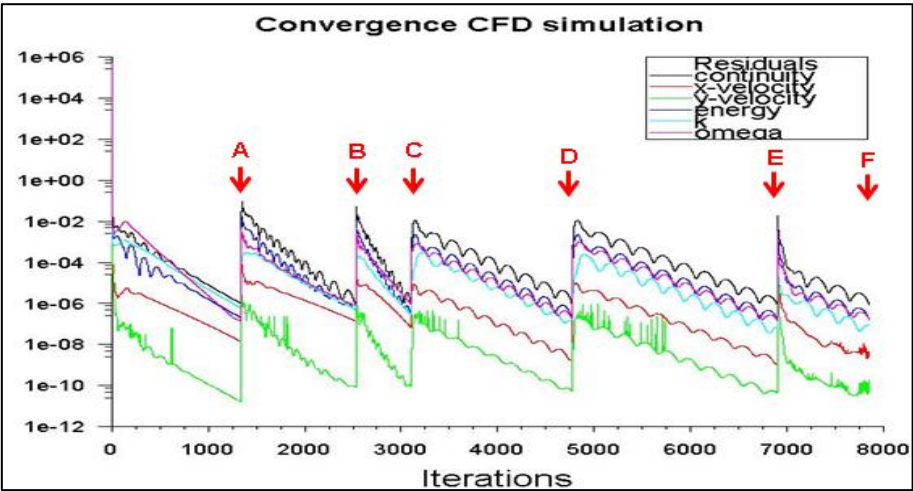


Figure 64 – Confirmation of convergence

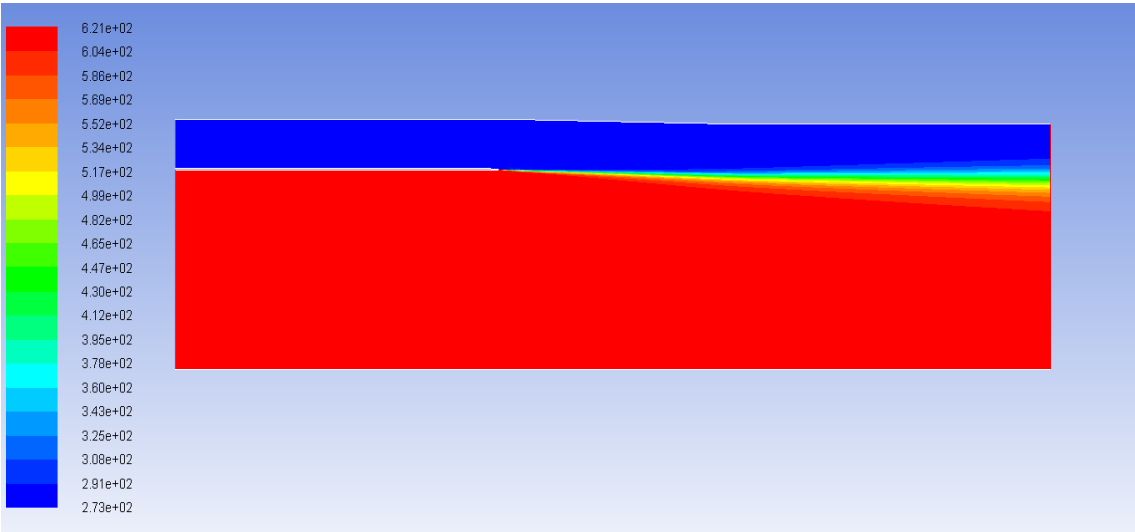


Figure 65 – Total temperature distribution for a CFD mini-mixer simulation

4.4.1 First comparison set

The first comparison set was created using the results of the first optimisation which assumed of a fixed friction factor 0.04. To assure that there was an even spread of the Pareto front and mixer dimensions five resulting engine configurations were chosen at different values of specific thrust on the Pareto front. These five engines were than simulated as per described in chapter 3.4.2.

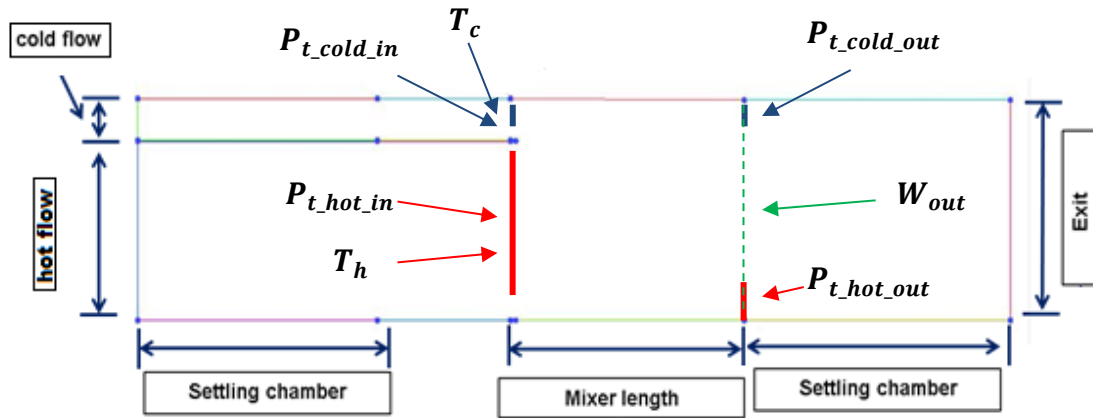


Figure 66 – Flow characteristic measuring location

All five CFD simulation showed similar results. Therefore, it was decided to only show the results of two simulations. Different mixer flow characteristic were measured according to what is shown in Figure 66. In this figure it can be seen that $P_{t_cold_in}$ and T_c were measured at the cold section inlet of the mixer chamber, $P_{t_hot_in}$ and T_h were measured at the hot section inlet of the mixer chamber. The location for $P_{t_cold_out}$ and $P_{t_hot_out}$ are shown as the outlet of the mixer chamber however only for the section of the unmixed flow. This is explained in greater detail in chapter 3.4.2.6. The W_{out} is measured at the exit of the mixer chamber. The two results shown below had different mixer chamber lengths and diameter, the first engine configuration had a mixer length of 0.51 metres, and a mixer outer diameter of 1.3 metres, of which the results are shown in Table 10. The second engine configuration had a mixer length of 0.38 metres, and a mixer outer diameter of 1.2 metres, with results shown in Table 11.

Table 10 – CFD simulation results for Engine #1

Engine number		Engine #1 (mixer length 0.51 m)		
	Units	TurboMatch	CFD	Delta [%]
$P_{t_cold_in}$	[Pa]	51,778	51,627	-0.29
$P_{t_hot_in}$	[Pa]	41,558	41,534	-0.04
T_c	[K]	278.0	278.1	0.04
T_h	[K]	620.7	620.7	0.0
W_{out}	[Kg/s]	72.29	72.29	0.0
ΔP_{t_cold}	[%]	3.05	1.00	-67.5
ΔP_{t_hot}	[%]	0.25	0.1	-61.5

Table 11 –CFD simulation results for Engine #2

Engine number		Engine #2 (mixer length 0.38 m)		
	Units	TurboMatch	CFD	Delta [%]
$P_{t_cold_in}$	[Pa]	51,497	51,345	-0.30
$P_{t_hot_in}$	[Pa]	41,457	41,443	-0.033
T_c	[K]	277.3	278.1	0.30
T_h	[K]	619.7	620.7	0.15
W_{out}	[Kg/s]	63.74	63.74	0.0
ΔP_{t_cold}	[%]	2.38	0.71	-70.1
ΔP_{t_hot}	[%]	0.16	0.05	-70.1

From the results the following observation can be made. The flow characteristics of the CFD simulations and the thermodynamic engine models show agreement. There is a very small discrepancy between the results but this is within reasonable levels. This is the result of a slight mismatch between the static pressures at the inlets of the mixers. The reason for this slight discrepancy is due to fact that the boundary conditions for the CFD are extract from TurboMatch. TurboMatch is a 0D software and does not take into account the geometry of the mixer in the general performance calculations. The specific geometry influence that is causing this discrepancy is that there is a slight divergence of the mixer chamber is visible in Figure 65. These dimension do influence the mixer performance slightly.

By confirming that the mini-mixer simulated in CFD shows agreement with the thermodynamic model, the pressure losses due to friction can be compared. It can be seen in the two tables above that both mini-mixers experience different values for skin friction losses. However, both show significant disagreement with the TurboMatch results. For the three engines that are not shown here, the difference in pressure losses was between 60% and 70%.

Further investigation revealed that the fixed friction value of 0.04 was not correct and was significantly too high. It was concluded that a more robust method was needed to estimate the friction value. The value should be dependent on the flow characteristics of the specific situation. To achieve this, the Moody diagram shown in Figure 19 was implemented into the optimiser. However, as mentioned before, TurboMatch does not take fully into account the mixer geometry, nor are the flow characteristics within the mixer fixed. Therefore, there will always be a slight deviation between CFD and TurboMatch results. To account for this a correction factor was implemented into the skin friction pressure loss calculation this will be explained in the following section.

4.4.2 Final comparison

To achieve convergence between TurboMatch and CFD required many iterations. Each iteration had a slightly altered correction factor to reduce the discrepancy between TurboMatch and CFD skin friction pressure losses. The results would only be considered converged once the difference between skin friction pressure loss measured in CFD and calculated in TurboMatch was less than 1%. This would need to be consistent on both hot and cold sections of the mixing chamber as well as consistent over the complete Pareto front.



Figure 67 – Spread of engines of which mini-mixer was simulated in CFD

Originally the chosen method for the spread of engines was regarding the mini-mixer geometry. However, with the implementation of the Moody diagram this changed. The reason for this was that all the engines on the Pareto Front had very similar mini-mixer geometry. Further discussion on why this happened will be conducted in chapter 5.1. A new method was chosen for this section of the validation and the spread was now chosen to incorporate a wide range of Specific Thrust. The location of the five chosen engines depicted by letter A to E are shown in Figure 67. All five mini-mixers showed agreement between the flow characteristics of the TurboMatch and CFD simulations. The results

comparing the percentage of total pressure losses due to skin friction within the mini-mixer for all five engines are shown in Table 12.

Table 12 – Comparison between TurboMatch and final CFD simulations

Engine	Characteristics	Units	TurboMatch	CFD	Delta
A	ΔP_{t_cold}	[%]	1.30	1.30	-0.01
A	ΔP_{t_hot}	[%]	0.08	0.08	0.91
B	ΔP_{t_cold}	[%]	1.01	1.00	-0.27
B	ΔP_{t_hot}	[%]	0.10	0.10	0.5
C	ΔP_{t_cold}	[%]	0.94	0.93	-0.69
C	ΔP_{t_hot}	[%]	0.12	0.12	0.35
D	ΔP_{t_cold}	[%]	0.95	0.94	-0.9
D	ΔP_{t_hot}	[%]	0.12	0.12	0.53
E	ΔP_{t_cold}	[%]	0.97	0.97	0.08
E	ΔP_{t_hot}	[%]	0.12	0.12	-0.19

From the table it can be seen that all five engines are within the allowed 1% discrepancy. Each friction factor needed an independent correction factor as well. This meant that two slightly different equations were needed to calculate the skin friction pressure losses. The cold-section skin friction pressure loss including correction is shown in (4-1). The hot-section skin friction pressure loss equation is shown (4-2).

$$\left(\frac{\Delta P_t}{P_{t \text{ inlet}}}\right)_{cold} = (F * 1.32)_{cold} * \frac{L_m}{4 \frac{A}{P_{er_cold}}} \left(1 - \left(\frac{1}{\left(\frac{P_{t \text{ inlet}}}{P_{s \text{ inlet}}}\right)_{cold}}\right)\right) \quad (4-1)$$

$$\left(\frac{\Delta P_t}{P_{t \text{ inlet}}}\right)_{hot} = (F * 1.09)_{hot} * \frac{L_m}{4 \frac{A}{P_{er_hot}}} \left(1 - \left(\frac{1}{\left(\frac{P_{t \text{ inlet}}}{P_{s \text{ inlet}}}\right)_{hot}}\right)\right) \quad (4-2)$$

From the equation it can be observed, that the correction factor for the cold-section at 32% is significantly higher than that for the hot-section at 9%. This is a result of the outer geometry of the mixer, which has a slight angle due to the mini-mixer lip. This pushes the mini-mixer cold-section inlet outer diameter slightly higher than the outer diameter of the exit of the mixer. Table 13 shows the friction factor used by the engine A to E of the final optimisation. This friction factor already has the correction factor applied to it.

Table 13 – Corrected friction factors values

Engine	Friction factor hot-section	Friction factor cold-section
A	0.0152	0.0139
B	0.0149	0.0138
C	0.0146	0.0135
D	0.0146	0.0135
E	0.0145	0.0134

From the table it can be seen that even with the corrections applied, the friction factor for the hot-section is higher than for the cold-section. This corresponds to what is shown in Figure 19, because the hot-section will have a lower Reynolds number as well as a higher value for pipe roughness. Furthermore, it can be seen that the new friction factor is significantly less than 0.04 which was the initial assumption. It can be concluded that the Moody diagram was correctly implemented in the optimiser and that the correction factor derived for optimiser shows agreement with the CFD simulations.

5 Results & discussion

This chapter focuses on the main results of this research, with regards to demonstrating the potential benefits of incorporating variable exhaust nozzle technology. This chapter is divided into five subchapters. The first subchapter shows and discuss the optimiser results for the cases described in the methodology. The following section assesses if there is a need for nozzle area variability to improve performance, and what the variability trends are. The next subchapter shows the results of the engine operability check. The subsequent subchapter discusses the results of the engine weight estimation, focusing in particular on the mixer weight. The final section discusses the mission fuel burn studies. For confidential reasons none of the actual values could be given in the analyses. The discussion of all the graphs is conducted using percentage differences with respect to a baseline value (from the baseline engine models). Stacked graphs, showing all the performance results, have been included in Appendix B.

5.1 Cycle design optimisation results

In the fuel burn study the cycle design process is the starting point. During the development of the tool, multiple optimisations were conducted. With each consecutive run the understanding of the optimisation process increased and the optimisation process was improved. Each optimisation would typically examine up to 200,000 engine configurations. However, due to numerical stability issues within the mixer module, and the convergence criteria, a significant amount of those engine configurations did not produce a viable solution. The amount of actual, working engines varied from one simulation to another but was always kept between 35,000 and 60,000.

5.1.1 Design space exploration

To show that the optimisation has explored the design space sufficiently for both architectures, the results are plotted on the following two graphs. Figure 68 shows the explored design space for the geared architecture, and Figure 69 shows the explored design space for the direct-drive architecture. For these graphs, SFC is plotted against Specific Thrust.

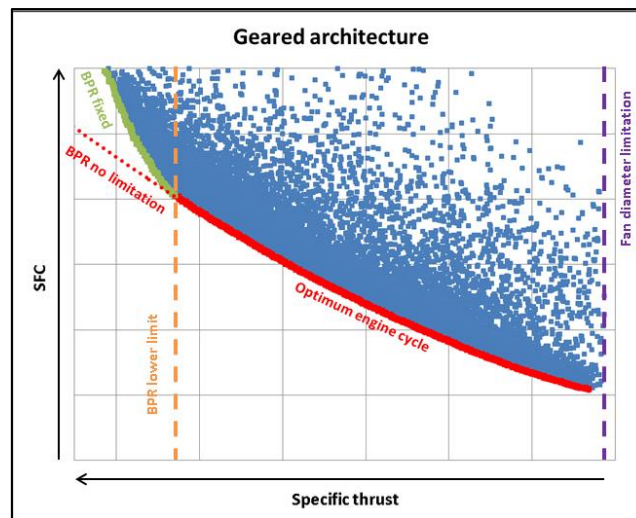


Figure 68 – Design space exploration for geared architecture

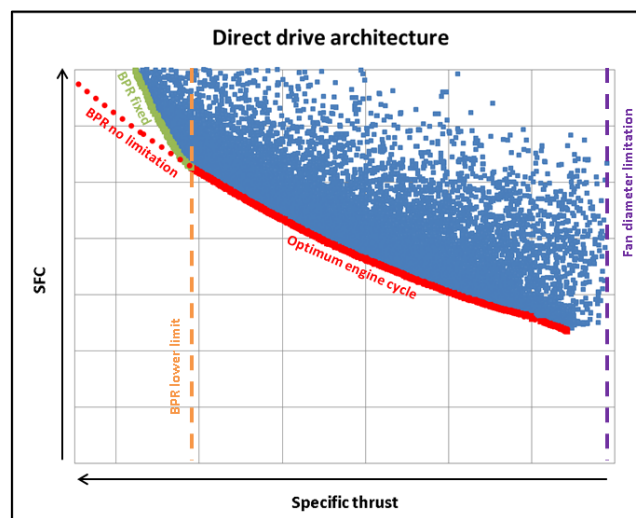


Figure 69 – Design space exploration for direct-drive architecture

In both figures the blue points represent working engines. The red line represents the optimum engine cycle - this is considered to be the Pareto front. What can be observed is that further from the Pareto front, less working engines were created. This does not mean there are less working engines at these locations, but relates to the way the optimiser works. Initially the optimiser finds results that have an increased potential to satisfy the objective functions and won't explore around engines with worse potential. Once the optimiser comes close to the optimum engine cycle, the amount of improvement gets smaller from one iteration to the next, until no improvement is found and the cycles are converged. This is enforced by the method used within the optimisation process, which is divided into multiple optimisation runs. This is described in chapter 3.3.3.

Another interesting effect that can be observed is the effect of the lower BPR limitation, which is represented by the green line. The gradient of the Pareto front changes completely once the lower BPR limit has been reached. It can be seen that the SFC increases significantly after this point. The effect of not having the lower BPR limit is also depicted by the dashed red line in both figures.

The final observation is that for both the geared and direct-drive architectures, the optimum fan diameter found, is lower than the maximum fan diameter limitation imposed. The maximum fan diameter is represented by the purple dashed line. It can be seen that the geared configuration is closer than the direct-drive configuration to that limit. It can also be seen that there are engine configurations which do achieve the maximum fan diameter. However, these have worse SFC than the smaller fan diameter engines. The reason for this is a combination the component efficiencies of the Fan and LPT systems and cooling requirement that are calculated in the optimiser and are not considered fixed. There is a point when the increased fan size causes of the LPT stage the efficiency to stagnate or even reduce. This is significantly more present in the direct drive case compared to the gear case due to the fact of minimizing the stage count and the assumption for stage loading. Additionally, the cooling

requirements for the cycle are limit the reduction of core size. This is because the cooling requirements are fixed for each cycle and determined at the critical design point which for engines being investigated (the EoR condition). This is where the temperature limitations are first hit. However the size of the fan is determined by the mass flow at ToC. These conditions as well as cruise have a set thrust requirement. If one would increase the size of the fan at ToC further this would also produce more thrust at EoR, this would reduce the temperatures at EoR to compensate for the increased thrust. However to still achieve the T30 and T40 limitation means the core would reduce in size, therefore, producing less thrust again and increasing the temperature. When the core is smaller and still achieves the same temperature, the percentage of cooling taken from the compressors to cool components will need to increase. This will negatively influence how much power the LPT could extract from the flow. There is a tipping point where the amount of energy to drive the fan exceeds the amount of energy available from the LPT. This limits the core reduction. In turn this will also limit the fan diameter reduction. The alternative would be to reduce the T30 and T40. It can be seen that those engines did meet the thrust requirements however their SFC at cruise have been negatively influenced by this.

5.1.2 Geared architecture analysis

With the optimisation completed, and having demonstrated that the design space was explored sufficiently, a thorough analysis was carried out on the effect of fan size on other engine characteristics. The optimisation was conducted using Specific Thrust as an objective function. Nevertheless, comparisons will be made hereafter on the basis of fan diameter which is directly related to Specific Thrust. Comparing fan diameter provides a clearer insight into the physical size of the engines explored.

The first analysis will be conducted on the geared architecture with VGES and no area variability: this is shown in Figure 70. It should be noted that the comparison

is made against a baseline engine, which in this case is the engine of the same configuration but with a fan diameter of 112 inches. It was decided to choose this fan diameter because a Rolls-Royce turbofan engine currently in service has the same fan diameter.

From Figure 70 the following observations can be made. First it must be noted that the thrust is the same for every engine configuration due to the set thrust requirement. When increasing the fan diameter at the same thrust it can be seen that the BPR increases significantly. Furthermore, the core size reduces, as well as the FPR. The core size is dependent on the technology limitation for the turbines, as well as the cooling requirements. If the core size were to reduce even further, the percentage of cooling flow required would reduce the overall engine performance more than the benefit that arises from reducing core size. It can be seen that the maximum fan diameter with the lowest SFC is achieved at just above 136 inches of fan diameter. Which is the same as previous research has shown for a gear architecture^[5].

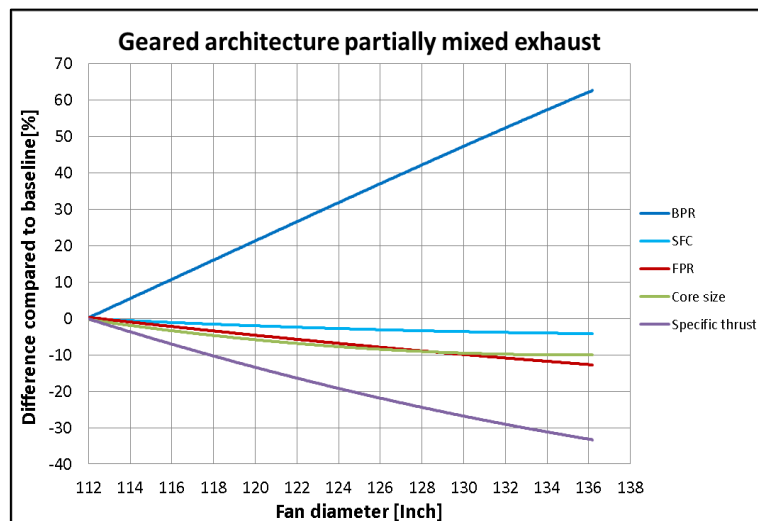


Figure 70 – Geared architecture partially mixed exhaust performance trends based on fan diameter

Figure 71, illustrates the percentage differences between the technology limitations imposed for T30 and T40, and the values each engine reached. This verifies that the produced engines were close to the set T30 and T40 limitations however, these were not surpassed by the optimiser as expected. The comparison in Figure 67 is made against the limitations mentioned in Table 1. The cause for not hitting exactly the limitations is due to how the optimiser was setup; using OPR and fixed thrust requirements. This means that the T30 and T40 correspond to the optimised solutions and are not set values. Furthermore, it can be seen that with the higher fan diameter range the rise in T30 variation is negligible and the maximum variation is only about 0.5 degree, however the T40 reduces. The reason behind this is complex and there are several factors which determine T40. An attempt to explain this trend in the results is made in the next paragraph.

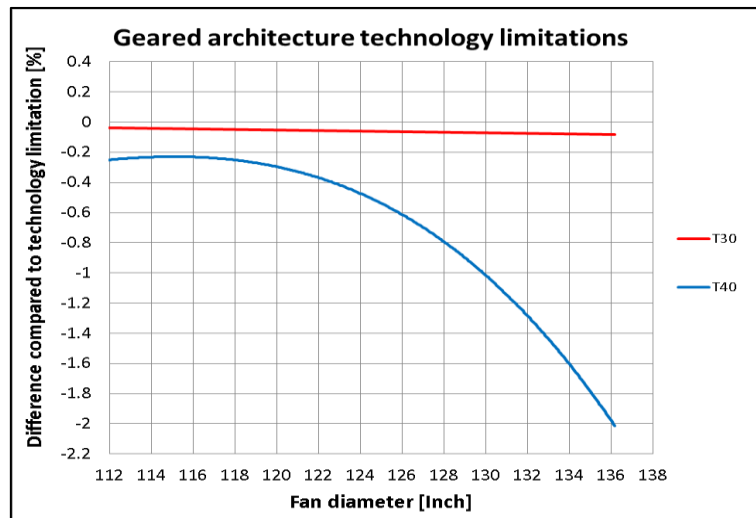


Figure 71 – T40 and T30 limitations plotted against fan diameter for the geared architecture

In engine design the turbine cooling requirement can significantly penalise the performance of the engine, meaning that minimising cooling is very important. The size of the cooling flow is determined by the core engine size, as well as T40.

The minimum core size is limited by T30 at EOR. With a higher T30 a smaller core can be achieved. The cooling flow size is determined by the T40 at EOR. The reason for this is because the cooling flow is not variable which means all operating conditions must have the same percentage of cooling flow requirement. This will have a negative effect when not as much cooling is required at other operating points such as cruise. This is unfortunate but unavoidable. The T40 at EOR is determined by thrust requirements at that operating point. All the above considerations influence results in the following manner; the core size cannot be reduced further due to the cooling flow requirements and the T30 limitation, but the thrust produced at the set T40 value at EOR is too large. The only method of reducing the thrust is by decreasing the T40. By reducing the T40, the cooling requirement is also reduced, therefore the core can slightly reduce again. It can be seen that the optimiser has done this. Another consideration is that the mixer may also require a larger core size, as the smaller core size could reduce the thrust gain. When comparing this against the unmixed engine the reduction in T40 is not present. The main difference is that there is a thrust gain due to the mixer, which results in excessively high thrust at the set T40 at EOR for the larger fan diameters. This same effect is present when using a VAFN due to the thrust increase achieved in EOR condition.

The following figure presents a comparison between a geared mixed exhaust against the geared separate exhaust configuration. This is illustrated in Figure 72. This figure shows increase or decrease in percentage of the geared mixed exhaust compared against the geared separate exhaust for different fan diameters.

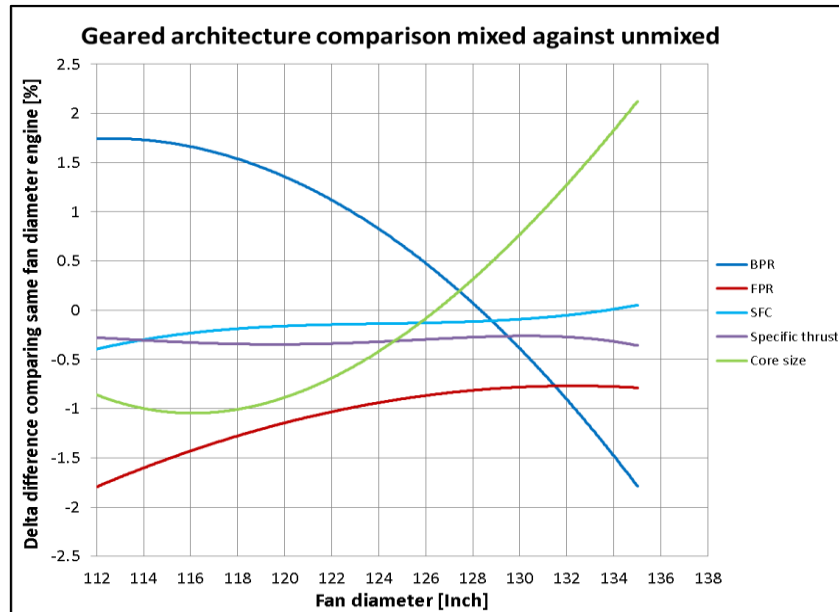


Figure 72 – Comparison of geared mixed against unmixed optimised engines over a range of fan diameter

From the above figure multiple observations can be made. The main point of interest is that in the lower fan diameter range there is a SFC reduction. The maximum reduction in SFC is shown to be very close to 0.5% compared to the baseline engine. This is achieved at a fan diameter of 112 inches. It also seems that at even lower fan diameters a further reduction in SFC can be present. From the 134 inch fan diameters it can be observed that there is a slight increase in SFC compared to the unmixed geared architecture. The reason for this is the potential thrust gain that a mixed-exhaust engine can have is numerically associated with the reduction in FPR and LPT pressure ratios when comparing to a separate-jets engine with the same Specific Thrust. For a given turbofan, the optimum FPR relates to a certain amount of energy conveyed between the core and the by-pass flow. In a separate-exhaust turbofan this energy needs to be transferred through the LPT and the fan where some losses take place. However, for a mixed exhaust turbofan some of the conveyed energy can be transferred through the mixer, increasing the transmission efficiency, and therefore reducing the LPT pressure ratio and the optimum FPR. The difference in magnitude between the optimum FPRs for both configurations will frame the potential thrust

gain. When the FPR of an unmixed is already very small the amount of thrust gain when using a mixed exhaust system can achieve is very small at this point the losses start outweighing the benefits. A solution would be to remove the mixing system however unfortunately this not possible with the current thermodynamic software programme. When looking at the other characteristics it can be seen that the FPR slope is flattening, when increasing the fan diameter. Furthermore, at small fan diameters the BPR is larger than the unmixed case, but when looking at the larger fan diameter the BPR is smaller than the unmixed case. The same trend but reverse is present for the core size. For lower fan diameters the mixer contributes meaningfully to the design, helping to reduce the core size further compared to the unmixed case, this is due to the thrust gain from the mixer.

But in the larger fan diameter range the core size is larger than the unmixed case. This has to do with the fact that the mixer requires certain parameters to achieve the most performance benefit. Looking at Figure 9 It can be seen that the mass flow ratio between hot and cold stream plays significant importance in achieving the best performance benefit. When taking into account Figure 20 which shows the losses associated with the mass flow ratio of the hot and cold streams and now also taking into account the other losses like skin friction there is a only a specific set of parameters where a mixed exhaust engine configuration will show performance benefits. This is where the benefits outweigh the losses. However once the core becomes too small the benefit of mass flow ratio disappears as shown in Figure 9. Therefore, there is a tipping point when the core size become too small and the losses start to outweigh the benefits for the mixed configuration. The optimiser is therefore preventing the further reduction of core size in the mixed case compared to the unmixed case which can still reduce the core size further.

Further evidence that the optimiser is trying to remove the mixer can be seen in Figure 73. This figure shows the mixer characteristics against different fan

diameters. It can clearly be observed that at the larger fan diameters the optimiser is reducing the mixing ratio to reduce the impact the mixer has on the design. It would preferably remove the mixer completely but this is not possible. However, as it can be observed at smaller fan diameters the optimiser is increasing the mixing ratio, to achieve the maximum benefit from the mixer.

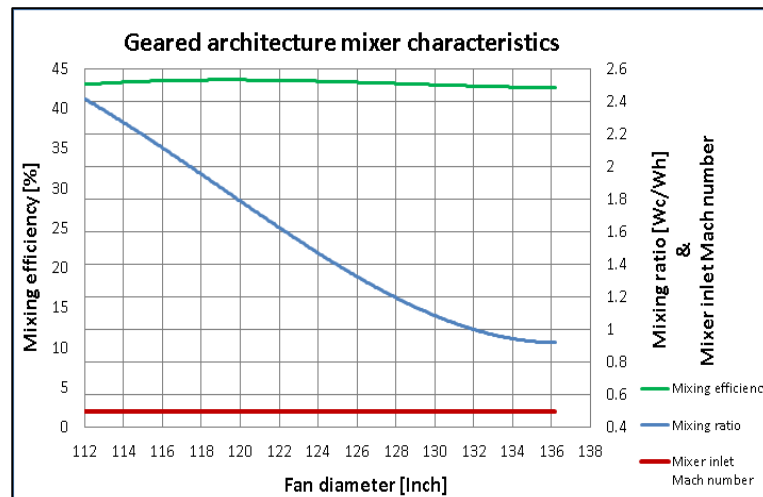


Figure 73 – Geared architecture mixer characteristics

Figure 73 also shows another point of interest; that the optimum mixing efficiency is between 40% and 45%. This is across the full fan diameter range. It should be noted that the maximum mixing efficiency was set to 45% as this is the maximum achievable for an annular plain mixer, without increasing the length of the mixer significantly. The mixing ratio as well as the achieved mixing efficiency has a particular effect on the mixer diameter and length, which is shown in Figure 74. From that graph it can be seen that mixer outer diameter slightly reduces when going to higher fan diameters. This slight reduction in diameter allows the optimiser to keep the mixer length constant. This would suggest that the optimiser has found the optimum length for this configuration and fan diameter range, whilst keeping the losses of the skin friction to a minimum.

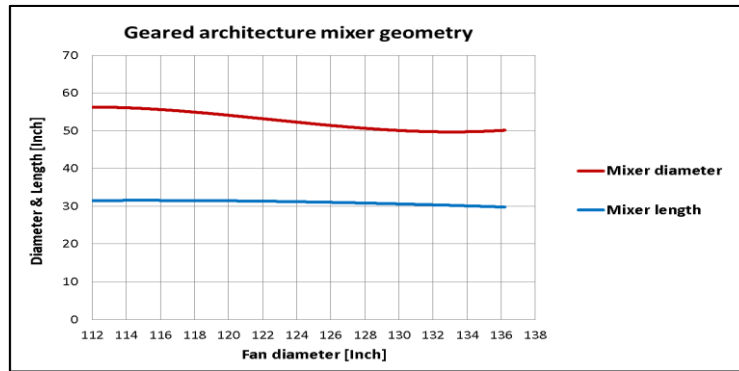


Figure 74 – Outer mixer diameter and length for the geared architecture

5.1.3 Direct-drive architecture analysis

The following analysis is focused on the direct-drive architecture with VGES but without any area variability. As with the previous case, this case compares against a baseline of the same engine configuration but with a fan diameter of 112 inches. It should first be noted that, like with the geared architecture analysis, the same thrust is required for every engine. The first results are shown in Figure 75. The figure presents the effect of different fan diameters on various engine characteristics.

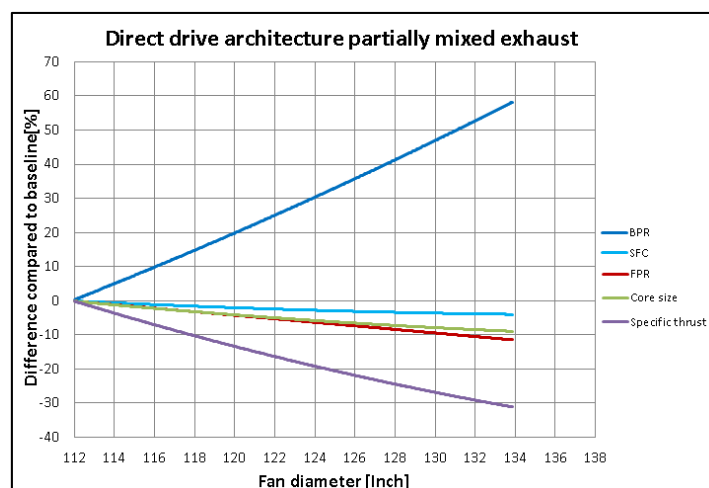


Figure 75 – Direct-drive architecture partially mixed exhaust performance trends based on fan diameter

From the figure above the same trends that are present in the geared architecture can be seen. When increasing the fan diameter, the BPR increases, whilst the FPR and core size reduce. It can be seen that the SFC decreases by around 5 percent at a fan diameter of 134 inches, compared to the baseline engine, which is very similar to the geared configuration. To verify that the direct-drive architecture is close to the limitations set for T30 and T40, percentage differences between the limitations and the results obtained are shown in Figure 76.

For the T30 it can be seen that the optimiser was able to get very close to the limit. However, the T40 limit is only reached at 112 inches of fan diameter. When moving away from the 112 inches, the difference in T40 increases substantially to about 2% at 134 inches. This is partly due to the same reason explained in the analysis for the geared configuration. However, another contributing factor is the LPT polytropic efficiencies. Due to the use of the gearbox the LPT in the geared configuration can achieve a higher efficiency as it can operate at its optimum rotational speed. This is one of the advantages of using a geared configuration, and has a major influence on the possibility for the engine to achieve the maximum possible value for T40.

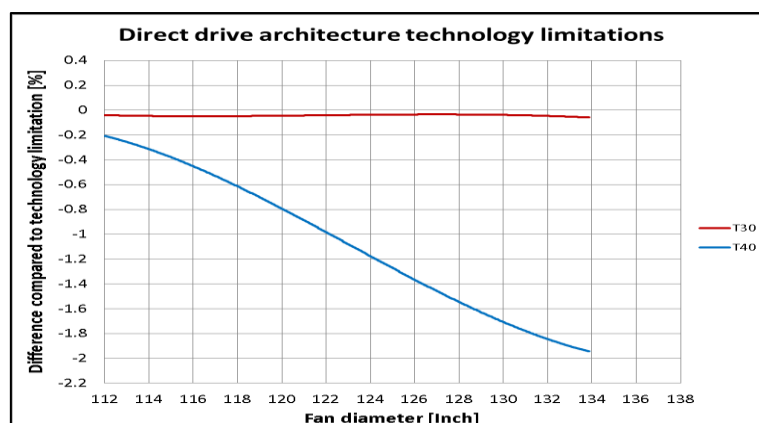


Figure 76 – T40 and T30 limitations plotted against fan diameter for the direct-drive architecture

Figure 77 compares in percentage the direct-drive mixed exhaust against the direct-drive separate exhaust configuration over a range of fan diameters. The most important observation that can be made from this figure is that at the 112 inch fan diameter the mixed case has 0.15% lower SFC compared to the unmixed case. This can also be seen by taking a closer look at the other characteristics at 112 inches; there is a reduction in core size, as well as a reduction in FPR. Furthermore, there is an increase in BPR compared to the unmixed case. This all indicates that the mixer is adding a performance benefit at 112 inches, but reduces the performance at fan diameters larger than 118 inches. The reason why the mini-mixer is not as beneficial to the direct-drive configuration as the geared configuration is due to the freedom of the LPT system from the fan in the geared configuration

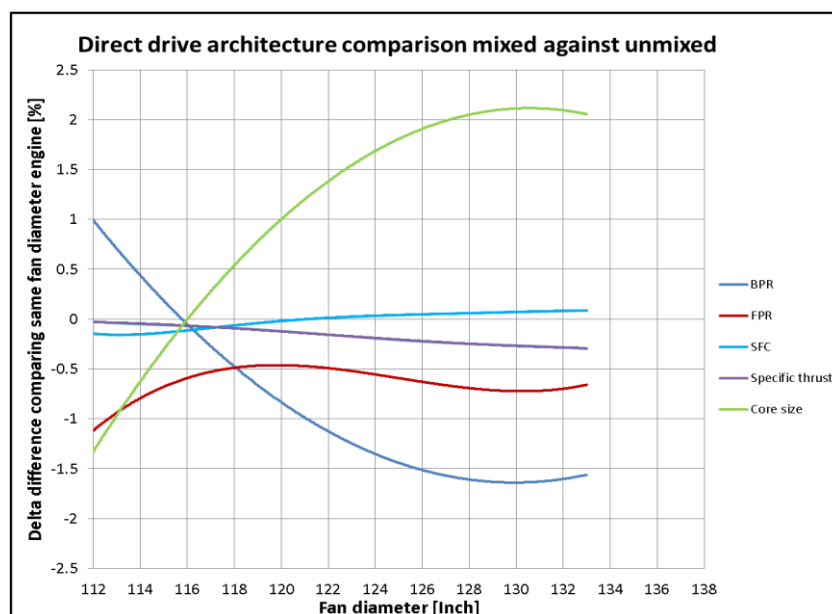


Figure 77 – Comparison of direct drive mixed against unmixed optimised engines over a range of fan diameter

Like in the geared mixed case at the higher fan diameter, the direct-drive mixed case is trying to remove the mixer from the engine. However, as this is not possible it is reducing the impact the mixer has on the engine. An interesting point

to notice is that the FPR is still lower for the mixed case compared to the unmixed case. The reason for this is that even though the performance is degraded the optimiser is still trying to get the most benefit from the mixer. Therefore, it is forcing the FPR to lower values to accommodate the mixer inlet pressure ratio. To get a better understanding of what is happening with the mixer a closer look into the mixer characteristics as well as the mixer geometry is required. The mixer characteristics are shown in Figure 78. The mixer geometry is shown in Figure 79.

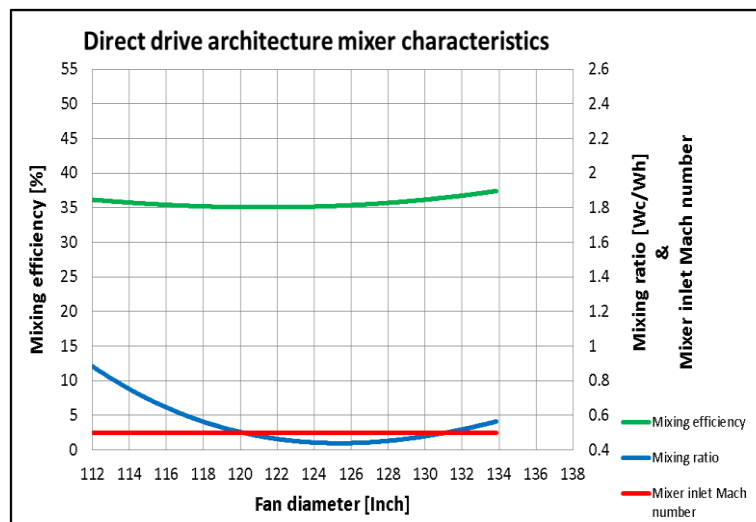


Figure 78 – Direct-drive architecture mixer characteristics

From the mixer characteristics the following can be observed; in comparison with the geared configuration, the direct-drive configuration mixing ratio only increases when going to fan diameters lower than 118 inches. Above 118 inches the optimiser is reducing the impact of the mixer on the model. At 112 inches there is only a very slight performance benefit and this trend would suggest that lower fan diameter engines may benefit more from a mixer. Another interesting observation is that the mixing efficiency is lower than the geared configuration. The mixing efficiency is between 35 and 40%. This would mean that the losses associated with mixing are higher for the direct-drive configuration. Therefore, the

optimiser has found that for this architecture the mixing efficiency should be lower. This has to do with the mixing skin friction losses which are higher, due to the higher fan pressure ratio, which generates higher loss in the cold-section.

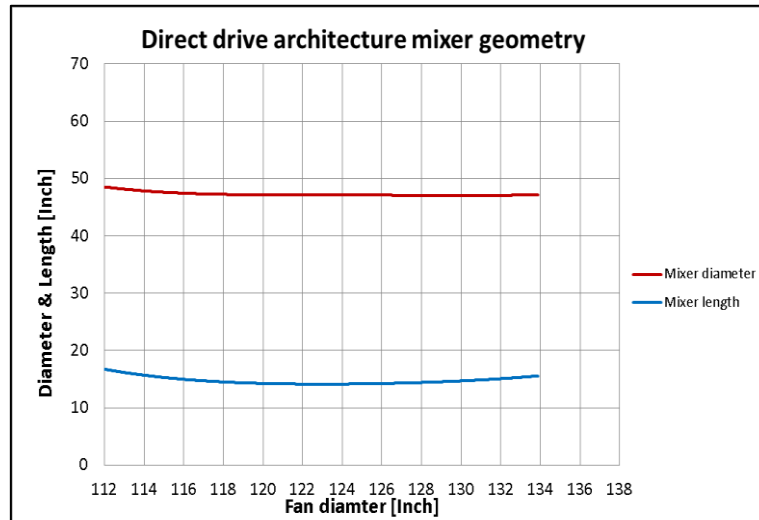


Figure 79 – Outer mixer diameter and length for the direct-drive architecture

When looking at the mixer geometry it can be seen that, as with the geared configuration, the direct-drive mixer geometry does not vary significantly, and all engine configurations have similar mixer geometries. The mixer length is substantially shorter than in the geared configuration, even though the mixer efficiency is only slightly less. Looking at the mixer diameters, the size seems to be consistent across the direct-drive and geared architectures. This would suggest, when taking Figure 17 into account, that the core size of the direct-drive is substantially larger than that of the geared configuration, which influences the perimeter of the mixing section, reducing the length of the mixer substantially. However, even with the increased length, there appear to be more losses due to the higher FPR than compared with the geared configuration, forcing the optimiser to reduce the influence of the mixer.

5.1.4 SFC performance comparison

With the overall trends presented, and the comparison between unmixed and mixed configurations discussed, the following step is to compare the geared and direct-drive engine architectures against each other. Figure 80 shows the SFC comparison between the two configurations. This is compared against a modern high bypass ratio turbofan in service today. This was done to show the amount of improvement these engine configurations can bring compared to today's technology.

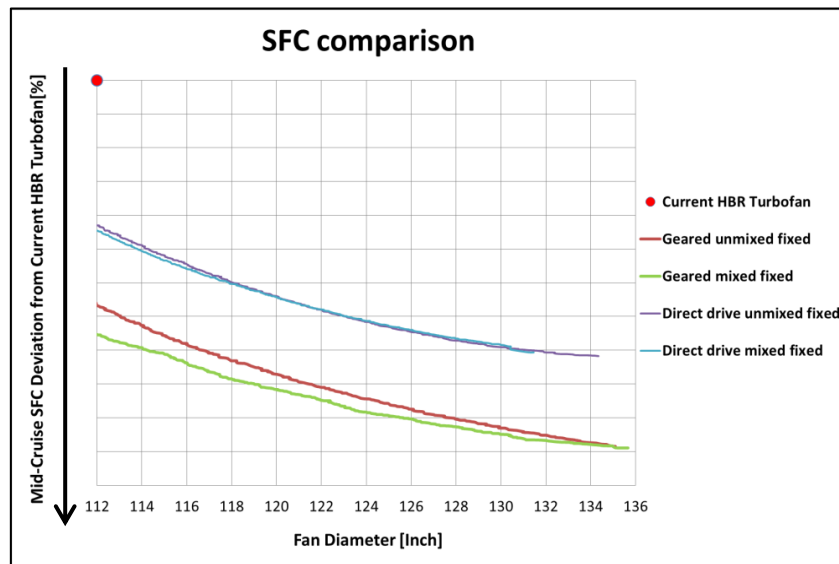


Figure 80 – SFC comparison between geared and direct-drive architectures

The first observation is that regardless of which engine architecture is chosen, the new engine configurations have substantially better performance compared to a current turbofan. It can be seen that even without the partially mixed exhaust system, the geared configuration has the lowest possible SFC. Furthermore, it can be seen that the direct-drive optimum engine with the lowest possible SFC has a smaller fan diameter than the optimum geared architecture. Both architectures are not able to reach the fan diameter limitation of 138 inches which is equal to 3.5 meters.

The next observation is that for the direct-drive architecture, the partially mixed exhaust system shows hardly any benefits in performance. Only at the very low fan diameters there is a small benefit. The reason there is nearly no benefit in the direct drive architecture, is due to the fact that the mixer losses in the direct drive are larger than in the geared configuration due to the higher FPR. The higher FPR is caused by the LPT and the fan being connected by a shaft and have to have the same RPM. Looking at the mixed exhaust geared architecture, fan diameters of 112 inches show significant benefits. However, this benefit reduces when going to higher fan diameters, and at 134 inches any performance benefits from a mixer configuration diminish completely. The optimiser at this point is trying to remove the mixer from the configuration, however this is not possible. Therefore, it reduced the impact the mixer has on the configuration.

5.2 Variable geometry trends

One of the main objectives of this research is to identify any variable geometry trends when using the VGES that could potentially bring additional performance benefits. To investigate this, the specific cases shown previously in Table 8 were run. Each case has a different area or combination of areas, which are allowed to increase or decrease. Allowing different areas of the VGES to vary presents different trends, which have various performance effects on the configurations. Optimisations for all the cases were run, and the results for both the direct-drive architecture and the geared architecture were found to be very similar. Therefore, only the results of the geared architecture are shown and discussed hereafter. Stacked performance graphs with all the results are shown in Appendix B.

Through a trial and error approach the numerical limitations of the variable area mixer were established. It was found that independent variability between the hot and cold inlets would cause significant convergence issues within the mixer; this

is because by changing the areas the static pressure changes. If the static pressure ratio at the inlets of the mixer are not close to unity, the configuration will not produce results. Therefore, it was decided to vary the mixer inlet areas by the same percentage each time to keep the static pressures in check.

5.2.1 SFC comparison of a geared mixed exhaust turbofan engine with VGES

Figure 81 shows the results of the optimisations for a geared mixed exhaust turbofan equipped with a VGES. Each configuration has one or several variable areas, except the fixed cases which have no variable areas. From the figure it can be seen that none of the various possible configurations of the VGES has a significant performance benefit at cruise, compared to the fixed exhaust geometry case.

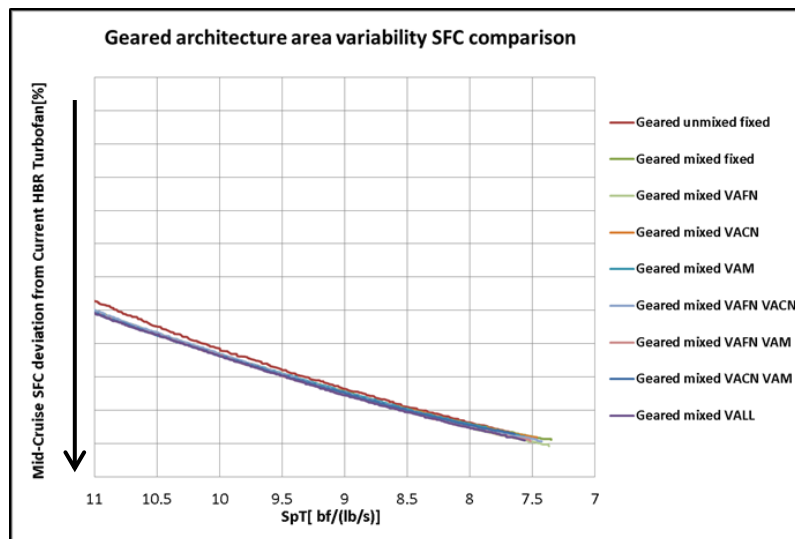


Figure 81 – SFC comparison of a geared mixed exhaust configuration engine with VGES

The goal of changing the areas of the VGES is to increase the thrust in critical operating points like EOR. Because there are set thrust requirements, for the

engine to accommodate the extra thrust the core size is reduced. This in turn increases the BPR, which increases the propulsive efficiency and thrust. By reducing the core size but still hitting the limitations of T30 and T40, a reduction in SFC should take place. This is because reducing the core size reduced the amount of mass flow that need to reach the T40 temperature. Therefore, a smaller fuel flow is required. The results in chapter 5.1.2 already showed that when the optimiser can't reduce the core size further, it reduces the T40 at EOR instead. This was already present in the larger fan diameter range due to the mixer, which already increases thrust at EOR compared to the unmixed cases. The mixer might also be limiting the reduction of core size due to the fact that a smaller core might reduce the theoretical thrust gain of the mixer, which might outweigh the benefit from a smaller core. This means that when the VGES is allowed to vary, the T40 is reduced even further. Another side-effect witnessed is the reduction of FPR when increasing the fan nozzle area at EOR. In regards to mixing, this might increase the theoretical thrust gain if the mixer inlet pressure ratio changes to a more favourable value in that operating point. Even with the reduction of T40 at EOR, this does not influence the SFC at cruise. Therefore, there is no performance benefit visible.

5.2.2 Individual variable area trends VGES

The following results focus on the cases where only one area changes at a time. These individual area variation cases are VAFN, VACN and VAM respectively. It is important to mention that the results in this figures reflect the general trend observed and that specific engines on the Pareto front may deviate from this general trend shown below. Area variation at EOR is shown in Figure 82. The second graph is the area variation trends at TOC, shown in Figure 83.

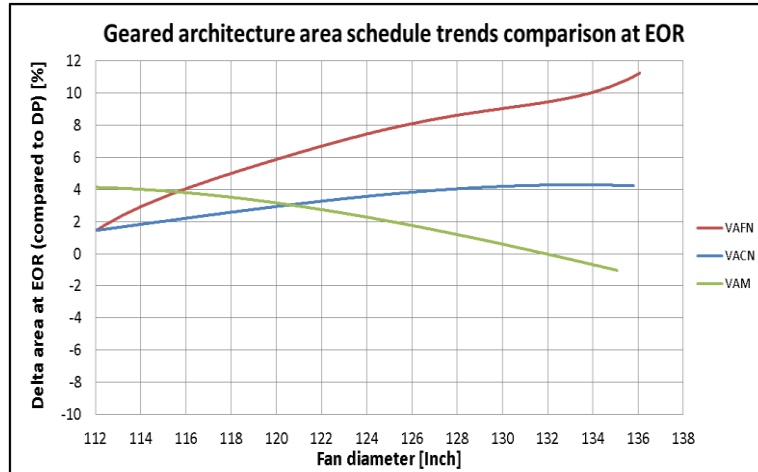


Figure 82 – Individual area variation trends for EOR operating point

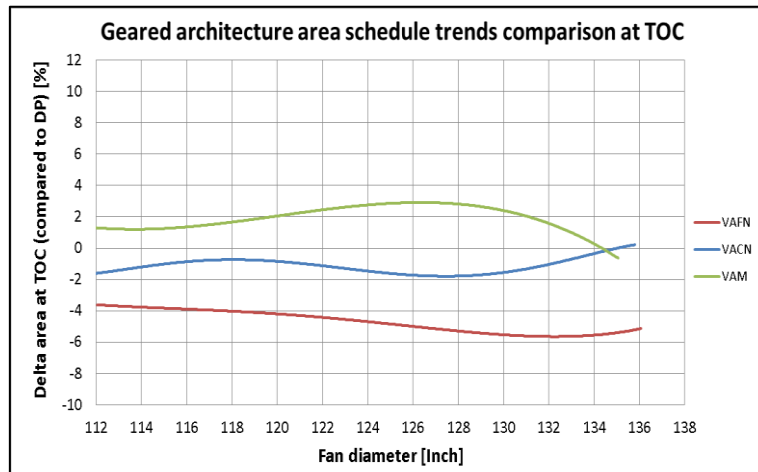


Figure 83 – Individual area variation trends for TOC operating point

From Figure 82 and Figure 83, the first observation to be made is that none of the individual area changes has hit the limit of variation. This was set at +/- 20%. This means that these are the optimum area variations, as further reduction or increase does not improve the performance of the engine. The next observation concerns the VAFN case. It can be seen that the nozzle area is increased at EOR and decreased at TOC. This means that the optimiser is following the correct design trends according to literature^[47]. The largest increase is 12% compared to cruise, and the maximum reduction is 6% at TOC. This results in a total variation

required of 18% for the VAFN. To accommodate an area variation of 18% a substantial amount of weight and complexity will be required for the system. This can only be justified by a significant improvement in performance, or if it is necessary from an operability view point.

According to literature^[47], opening the core area increases the expansion ratio of the LPT, therefore increasing the FPR, which would in turn increase the thrust of the fan nozzle. However, for the VACN of the VGES this is not the only effect that is occurring. This is because the core nozzle is also connected to the bypass duct. Consequently, this causes the same effect as increasing the VAFN area, which in turn reduces the FPR, reducing some of the increase in thrust from the fan nozzle. Furthermore, changing the core nozzle area also influences the stability of the mixer, as it changes the mixing ratio entering the mixer, but not the mixer inlet area. This seems to be a limiting factor for the amount of variability that is available for the core nozzle. The maximum increase that is desired is only 4%, and the maximum decrease is just 2%. This is not a significant amount but any variability will need to be justified by improvement in performance or operability.

When looking at the VAM, only the lower fan diameter range should be taken into account. This is due to the fact that, like the fixed case, the optimiser was trying to remove or reduce the impact of the mixer on the configuration for the larger fan diameter engines. This means that for a configuration that has a variable mixer, the trend is to increase the mixer inlet areas at the EOR and TOC. It can also be observed that the amount of area change required is small. The reason is due to stability; the mixer requires certain conditions for mixing to take place. When significant area change takes place in off-design, these criteria are not met. When increasing the mixer inlet areas at EOR and TOC the following occurs; there is a slight decrease in expansion ratio of the LPT, which results in a slight drop of FPR. Changing the FPR influences the mixer inlet pressure ratio. This influences the theoretical thrust that can be gained from the mixer. Furthermore,

by changing the cold chute area, the mixing ratio is also changed. This also influences the thrust gain. By increasing the thrust at EOR one hopes to reduce the core size of the engine and therefore reduce the SFC at cruise. This is the same reasoning for the use of a VAFN. The slight area change in off-design, would suggest that the optimum mixer design for cruise is very close to the optimum mixer configuration for EOR and TOC.

Even with all the beneficial effects due to the variability, there are no significant improvements in performance apart from a small benefit towards lifing, as the maximum T40 the engine will experience is lower than in the unmixed case. This is because the core size could not reduce further, therefore the T40 drops.

5.2.3 Multiple variable area trends VGES

The next sets of results illustrate the four cases which show the trends when allowing multiple areas of the VGES to change at the same time. The first combination of variable areas that will be looked at is the VAFN and VACN. The area variation trends for the EOR are shown in Figure 84 and the area variation trends for TOC are shown in Figure 85.

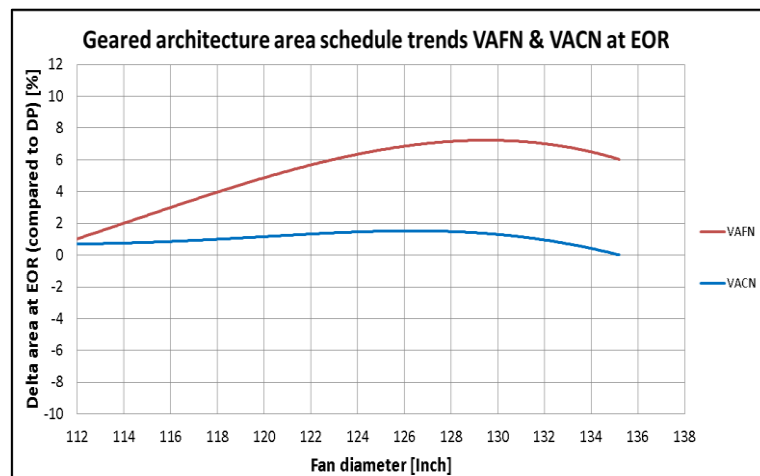


Figure 84 – Geared architecture area schedules VAFN & VACN at EOR

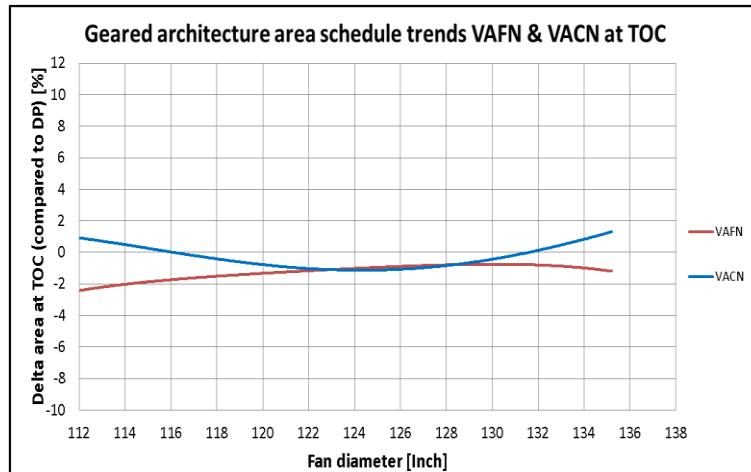


Figure 85 – Geared architecture area schedules VAFN & VACN at TOC

From these figures the following observations can be made. The variation trends that are shown show agreement with the literature. This means that the area increases at EOR. However, there is a difference between the individual variation and combined variation. When VAFN and VACN vary together the maximum variability that is needed is reduced. Where a total variation of 18% was required with individual variation of the VAFN, this has reduced to around 10%. When the VACN was only varied, it needed a maximum variation of 6%; this has been reduced down to only 2%. The reason for the reduction in variation is due to the fact that the core nozzle is connected through the mixer with the bypass duct. This results in the VAFN and VACN partially producing the same thrust gain at EOR. The individual variation already showed the optimum area variation that would produce the best performance. Consequently, by allowing the VACN to vary, the amount of variability for the VAFN is reduced. Besides the way in which areas vary, there is no real difference between the engine configurations of the individual VAFN variation or the combined VAFN and VACN variation.

The reduction of area required for the VAFN could be seen as an advantage. This is because variability in the VAFN is significantly more complex to achieve than variability in the VACN. For the VACN the actuator system could be placed within

the plug of the nozzle and be made to move longitudinally and parallel to the engine axis. Whereas with the VAFN, it would require a much larger and more complex system.

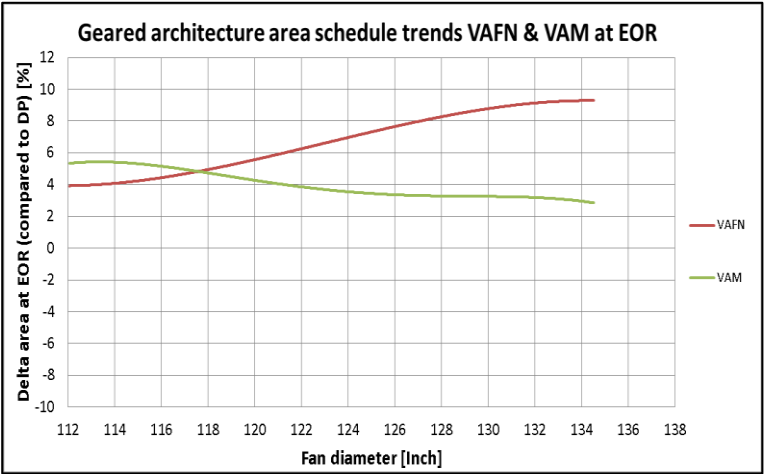


Figure 86 – Geared architecture area schedules VAFN & VAM at EOR

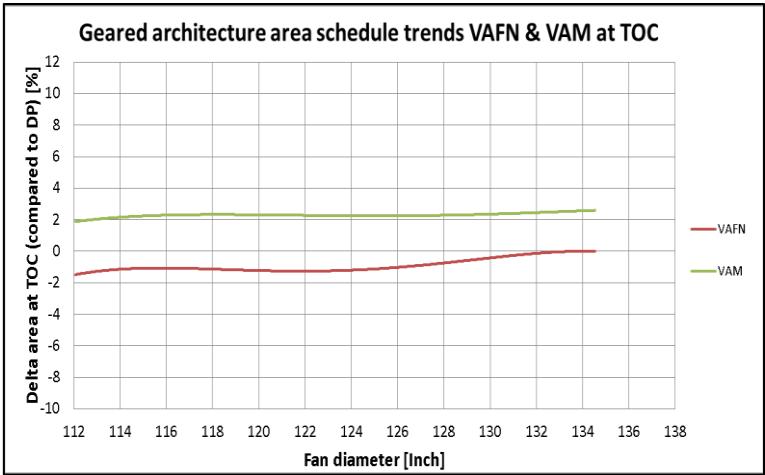


Figure 87 – Geared architecture area schedules VAFN & VAM at TOC

Results for a combined VAFN and VAM change are presented in Figure 86 for EOR, and Figure 87 for TOC. It can be noted that when the area variations of the VAFN and VAM are combined, results are very similar to the isolated cases

discussed before. However, there is a difference, which is when multiple areas are varied at the same time, it reduced the amount of variation needed for each of the variable areas to achieve the same result. The reason for this is that the VAFN area change influences the FPR. This in turn helps the mixer to achieve better stability when varying the VAM.

The third combination of variable areas is the VACN and the VAM. The EOR variability trends are shown in Figure 88 and the variability trends for TOC are shown in Figure 89. From these figures it can be seen that the effect of allowing both the VACN and the VAM to vary will help with the mixer stability issues that were present when varying the VAM individually. Additionally, it can be seen that both optimum areas of the VAM and VACN at EOR are larger than at cruise. But for the TOC the VACN is smaller than at cruise, and the VAM area is larger than at cruise. The trends of the combined VAM and VACN are still very similar to individual variable area trends.

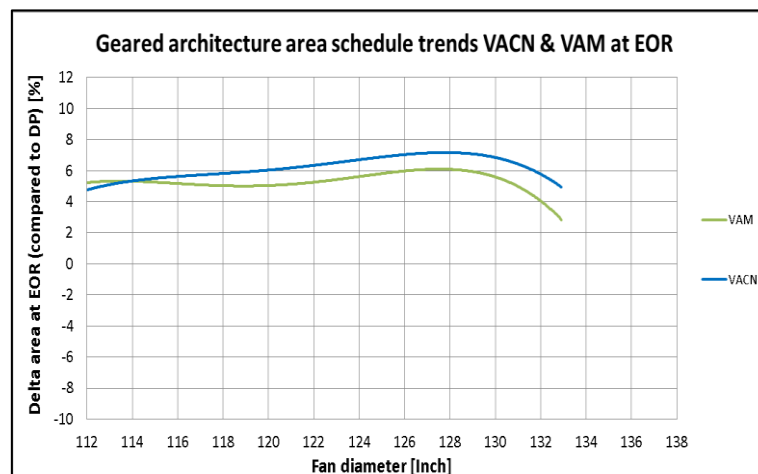


Figure 88 – Geared architecture area schedules VACN & VAM at EOR

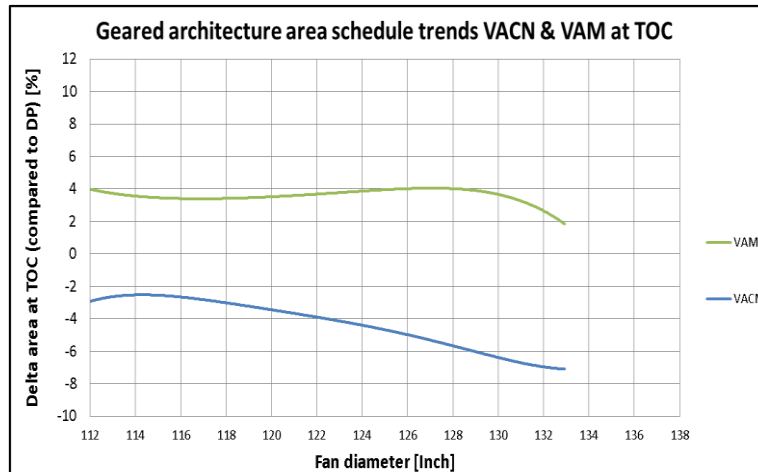


Figure 89 – Geared architecture area schedules VACN & VAM at TOC

The final results presented are with regards to the area schedule trend, when allowing all variable areas of the VGES to change at off-design conditions. The area schedule for the EOR is shown in Figure 90 and the results for the TOC are shown in Figure 91. From the EOR figure it can be seen that the optimum areas are always larger than at cruise. At TOC it seems that the VAFN does not require any variation as the variation which was required is now achieved by the VACN. With the addition of the VAM it seems that the area variation required for the VAFN is significantly reduced, and the total area variation is reduced to 6%. This means that compared to having the VAFN only, there is a total reduction of 12%. However, the area variation is now shifted from VAFN to VACN. This means that the VACN will require a greater change compared to when it was left to vary in isolation. As mention before this could be an advantage in a situation where variability is required.

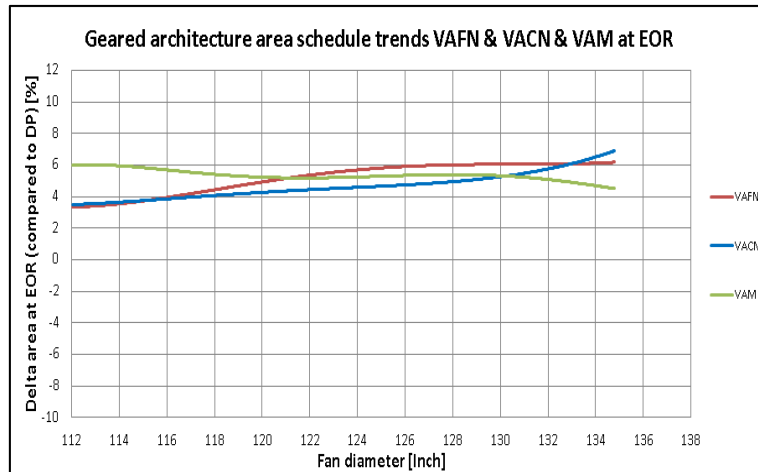


Figure 90 – Geared architecture area schedules VAFN & VACN & VAM at EOR

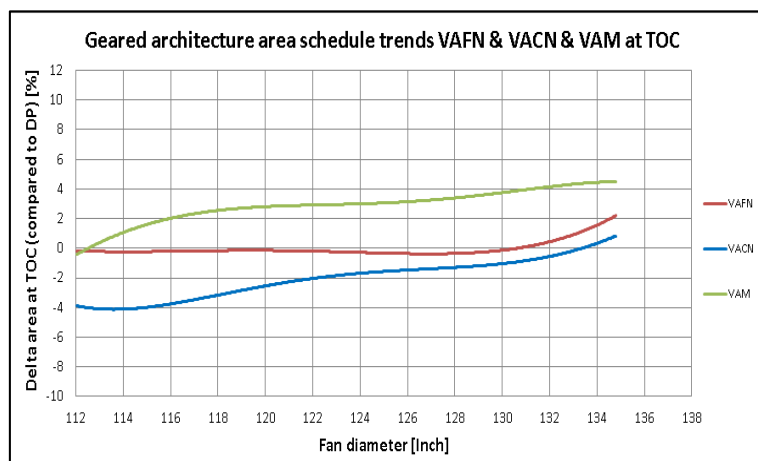


Figure 91 – Geared architecture area schedules VAFN & VACN & VAM at TOC

To summarise all the observations and conclusions made in this subchapter:

- There is no significant performance benefit when allowing any variability in the VGES
- The optimum fan nozzle area at EOR is larger than at cruise in all variable area cases
- The optimum fan nozzle area at TOC is always smaller than the area required at cruise

- When the core nozzle is allowed to vary, the optimum area variation for the fan nozzle is reduced, compared to the case where only the VAFN is varied
- When the mixer inlet area is allowed to vary, the optimum mixer inlet area at the EOR and TOC is larger than at cruise
- VAM works best in combination with a VACN, which allows a larger variation to be achieved
- Adding variability reduces maximum T40 experienced by the engine, which will increase the life of the turbine components

5.3 Engine operability results

From the previous subchapter it has been concluded that there is no performance benefit to be gained from using the variability of the VGES. However, in very low Specific Thrust engines it is thought that the fan is the critical component, due to very large fan diameter and low pressure ratio and is more like to surge at low flight speeds. To investigate if the designs the optimiser has provided are feasible, an engine operability study was conducted. This study will show if there is a necessity for area variability, as this can be used to move the fan operating line away from the surge margin. For the operability study a visual inspection of the engine's running lines was conducted, as described in subchapter 3.6.1. The compressor maps were plotted with their running lines, and any potential problems were highlighted. The maps were produced using TurboMatch.

5.3.1 Geared mixed fixed engine models

The first part of the operability study regards the small fan diameter cases. For the geared mixed fixed case with a fan diameter of 112 inches, the fan tip compressor map is shown in Figure 92. This map includes the EOR running line as well as the operating point for cruise and TOC. The conclusion that can be

made from the figure is that with a fan diameter of 112 inches there is no need for the variable areas in regards to operability. There is only a slight shift towards surge in the running line at the EOR compared to cruise and there is a sufficient surge margin present.

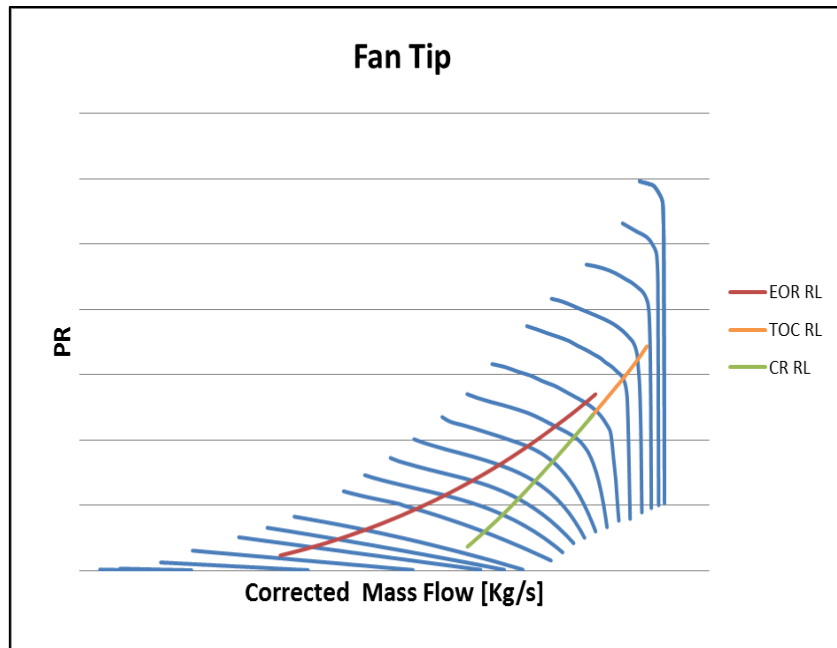


Figure 92 – Fan tip compressor map for a 112 inches fan diameter geared mixed exhaust turbofan with fixed exhaust areas

Figure 93, shows the fan tip compressor map for the 136-inch fan diameter geared mixed fixed case. This is considered to be a very low Specific Thrust turbofan engine. When comparing it against the smaller fan diameter compressor map the following can be observed; the cruise operating point is very similar, as well as the TOC operating point. Though the running line is slight less step as with the 112-inch fan diameter case. The EOR running line on the other hand has shifted considerably towards surge. However, even with the shift of the EOR running line towards surge, there is enough of a margin between the surge line and running line. Consequently, this graph establishes that in regards to fan operability there is no need for the variability of the VGES in regards to surge.

When comparing this with data out of the public domain like from Jackson, A. J. B^[43] of which a result is shown in Figure 24, it can be seen that this is a different conclusion. Where it was estimated a BPR 30 would require a massive 60% increase in fan nozzle area, it was expected that BPR close to 20 (like the current research) would also require an increase in fan nozzle area. The previous research always linked BPR to variable fan nozzle requirements, however, there is a key difference between the method use in that previous research and the current research that is in the previous research the core size was fixed. This is not the case for the current research where the core is reducing in size as the fan diameter increase. This means that when the previous research reached a BPR of 20 the fan diameter would be substantially larger than the fan diameter of the current research at a BPR 20. It would therefore be more prudent to link fan nozzle variability requirement to fan diameter instead of BPR.

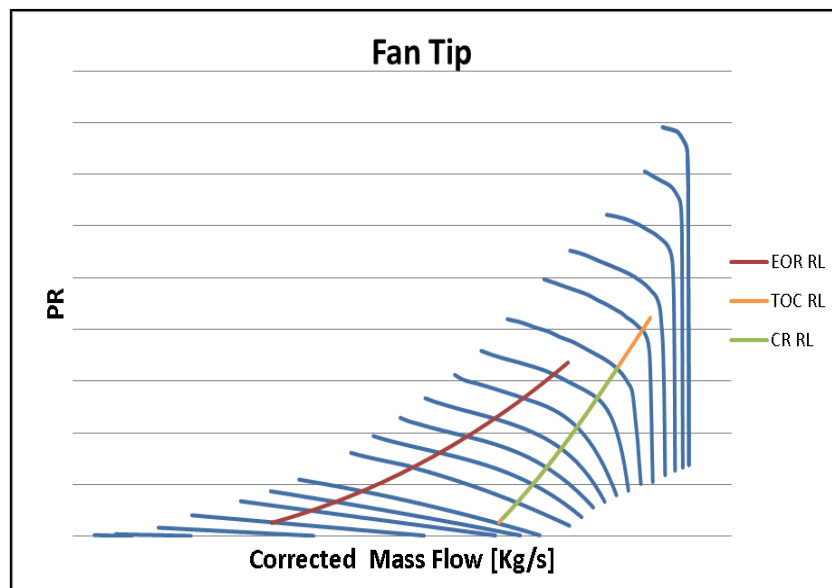


Figure 93 – Fan tip compressor map on 136 inch geared mixed fixed

To complete the operability study, the remaining compressor maps should be visually investigated. Figure 94 shows the IPC map of the 136 fan diameter geared mixed fixed engine model. This map includes the EOR running line as

well as VSV map changes. It should be pointed out that no BOVs were simulated in this model. It can be seen that the running line comes close to the surge line at low speeds. Even with the VSV the surge margin might not be sufficient enough at low speeds. This is considered to be tolerable as if the IPC would be simulated with BOV, the BOV would push the running line at low speeds away from the surge line.

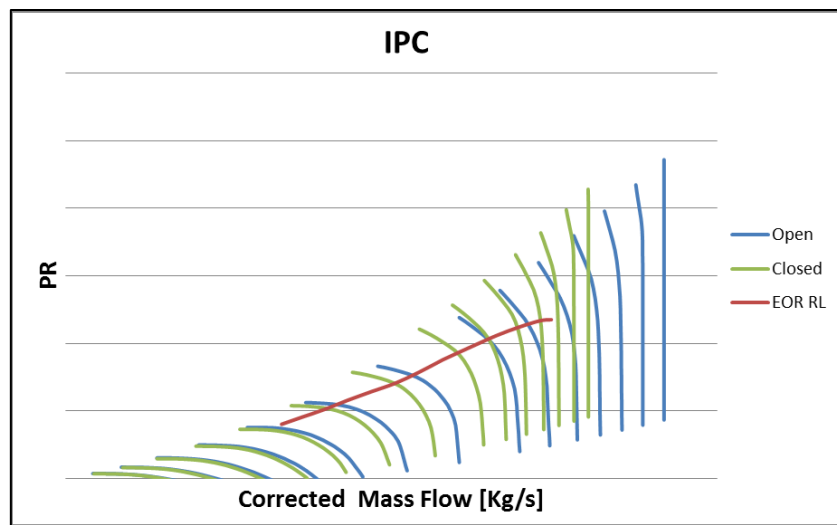


Figure 94 – Geared mixed fixed model IPC map with VSV

The final compressor map to visually investigate is the HPC. This is shown in Figure 95. However, due to the method describe in chapter 3.2.2 where bleed flows are taken from different location within the HPC, there is no single HPC map. There a total of five HPC maps, which gives an overview of the overall HPC behaviour. In this figure the top of each graph indicates which stages belong to that graph where 1 is the first stage and 10 in is the last stage. The first graph is 1 to 3, second graph is stage 4, 3rd graph is stage 5, 4th graph is stage 6 and 7 and the last graph is stage 8 to 10. From the figures it can be seen that the compressor is working as expected. The forward stages have a running line closer to surge, and the later stage have a running line further from surge. This figure also establishes that the engine model operates within satisfactory parameters. This concludes that in regards of the geared mixed fixed engine

model there is no need for the variability of the VGES in regards to performance improvement or operability reasons.

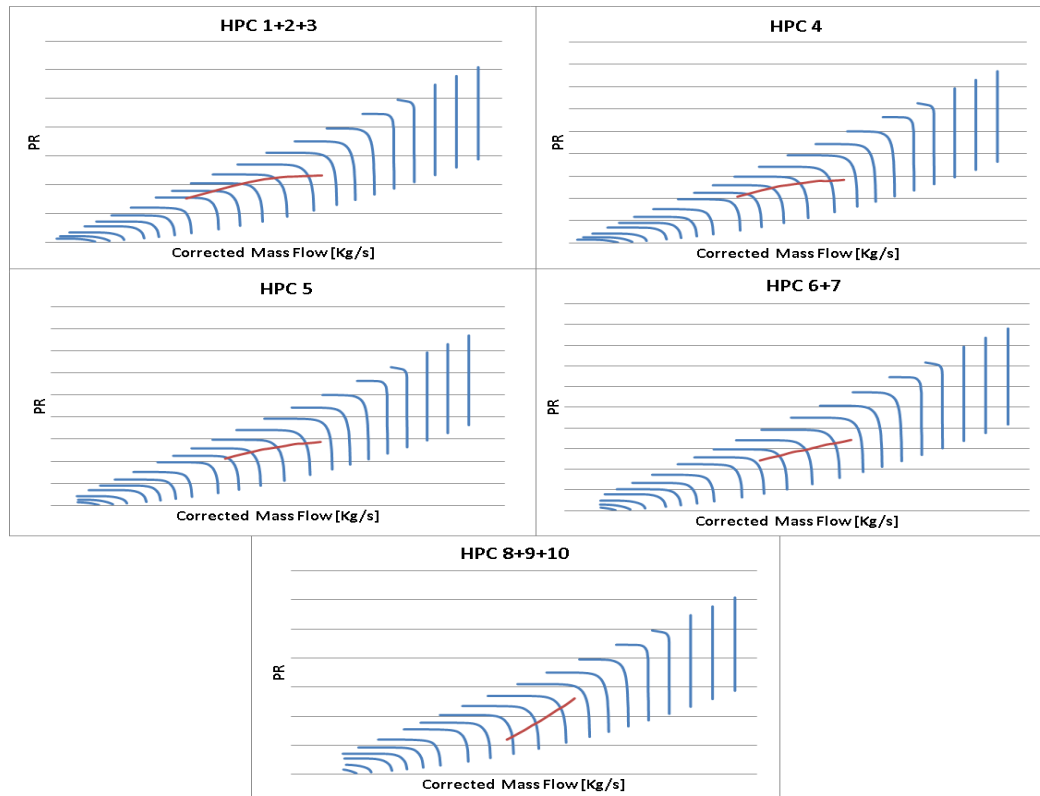


Figure 95 – Geared mixed fixed model HPC maps

5.3.2 Direct-drive mixed fixed engine models

Next to the geared mixed fixed engine model a similar investigation was conducted on the direct-drive mixed fixed engine models. A very low Specific Thrust with large fan diameter of 132 inches was investigated, in regards to engine operability at EOR. Figure 96 shows the fan tip compressor map with EOR running line and operating point of cruise and TOC.

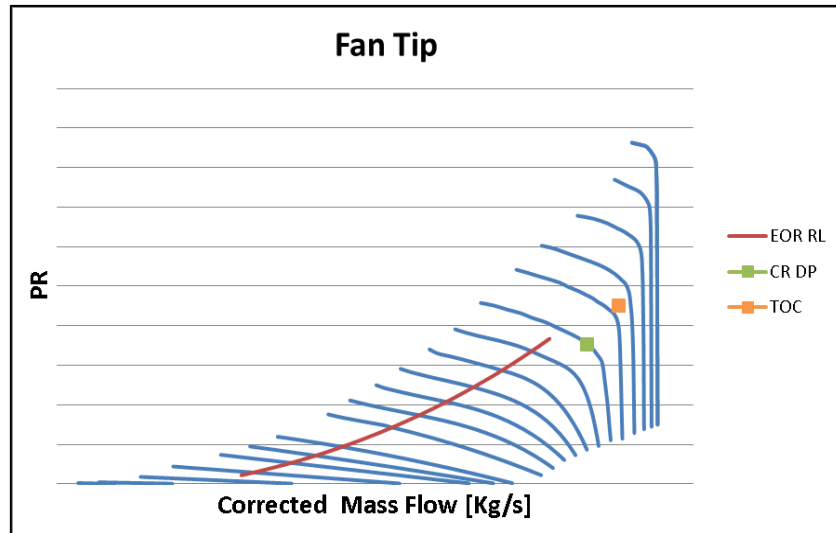


Figure 96 – Fan tip compressor map on 132 inch direct-drive mixed fixed

Figure 96, shows that the EOR running line still has sufficient surge margin and does not require variability in the VGES. Comparing this figure against the geared mixed fixed engine model, shows they are very similar. Further investigation into the other compressors revealed that all the other compressors demonstrate similar behaviour as the geared mixed fixed engine model.

From the complete operability study it can be stated that, without taking flutter into account, the operability of both the geared and direct-drive mixed fixed engine models does not require the variability offered by a VGES. Both engine architectures did not receive significant performance improvements from the variability of the VGES. It can now be concluded that the variability the VGES offers cannot be justified for the current architectures within the given design range. Therefore, no further investigation is required into the variability at this stage.

5.4 Engine weight and length estimation results

To estimate the bare-engine weight of the different engines on the Pareto fronts the Cranfield in-house software ATLAS was used. Alongside weight, this software also calculates the engine length. The description of the software is given in subchapter 3.5.1. ATLAS has been verified against data available in the public domain^[44]. Furthermore, it has also been verified against a Rolls-Royce Trent 1000^[5] engine architecture. The error in engine weight comparing against the Trent 1000 was only 0.29%, which is very small. Furthermore, the general model used in this study excluding the mixed exhaust system was verified in previous research^[5]. It should also be noted that ATLAS takes into account the thickness of the casing for Fan blade-off containment situations. Apart from the bare engine weight estimated by ATLAS, Matlab was used to estimate the nacelle weight and the mixer weight. The methodology for these components was described in 3.5.2. The weight estimation for the variable geometry was not conducted, as it had already been concluded that variable geometry was not beneficial to the performance of the turbofan currently investigated or needed in regards to preventing engine surge. Furthermore, ATLAS does not have the capability to perform weight estimation for variable geometry at the time investigation.

5.4.1 Engine weight and length estimation

The first results are the total engine weight comparison which is shown in Figure 97. The comparison is between the two engine architectures, with and without mini-mixer. These values have been compared against a baseline value of the direct-drive unmixed fixed engine with a 112-inch fan diameter.

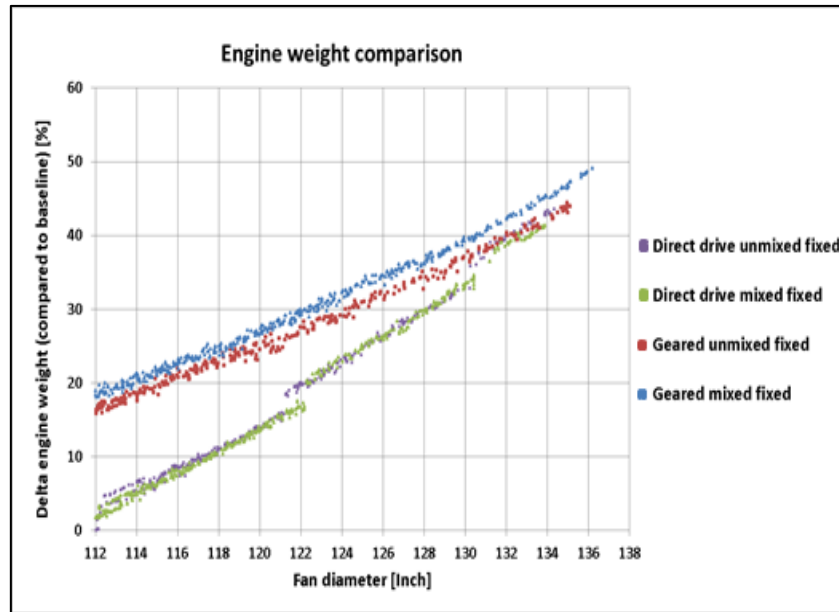


Figure 97 – Comparison of total engine weight for different engine architectures

The first thing that can be seen is that the geared mixed fixed Pareto front engine models are the heaviest engines. The increased weight of the mixed cases compared to the unmixed is due to the mixer. The difference in weight between the geared and direct-drive architectures is due to the gearbox. This adds a significant amount of weight to the configuration this is not only due to the weight of the gearbox but also the size of the gearbox, this will be explained when looking at the length of the engine. The difference in weight is especially clear in the lower fan diameter range compared to the direct-drive architecture. This is only present in the lower fan diameter range. Going to the larger fan diameter range, the difference in weight is reduced to a point where the unmixed geared and direct-drive weights are nearly identical. These results are consistent with results of previous research^[5] which also investigated the weight of and geared against direct drive engine architecture.

The second observation that can be made is the sudden increase in the weight of the direct-drive engine models. This is present in both the unmixed and mixed cases when going from small to large fan diameters. The reason for this is that

the LPT increases in stage count. The increase in stage count is to accommodate the high loading caused by the lower LP shaft speeds. It can also be observed that adding the VGES to the direct-drive case does not increase the weight for most of the fan diameter range. Only at the fan diameter of 116 inches and lower is any noticeable weight increase present. The reason for this is that the length of the direct-drive mixer is about half the length of the geared mixer. Furthermore, the benefit experienced due to a mixer, which tends to reduce bare engine weight, is sufficient enough to counteract the weight added from the mini-mixer.

The mixer weight is dependent on the geometry associated with the mixer. Figure 98, shows the percentage of weight associated to the mixer compared to total engine weight for the direct-drive and geared architecture. From this figure several observations can be made. First the mixer weight compared to total engine weight stays approximately constant in the direct-drive architecture at about 1%. It only increases when getting closer to small fan diameter of 112 inches. The second is that in the geared architecture case, the mixer starts to contribute more to the total weight once it reduces fan diameter. The geared configuration weight contribution curve seems to be very similar to the outer diameter curve shown in Figure 74, although the rate of change is not as aggressive in the outer diameter curve. However, the length also slightly increases. This, in combination with the increase in outer diameter, increased the weight contribution of the mixer. This means the length of the mixer begins to contribute more to the total engine length. This is also shown below in Figure 99.

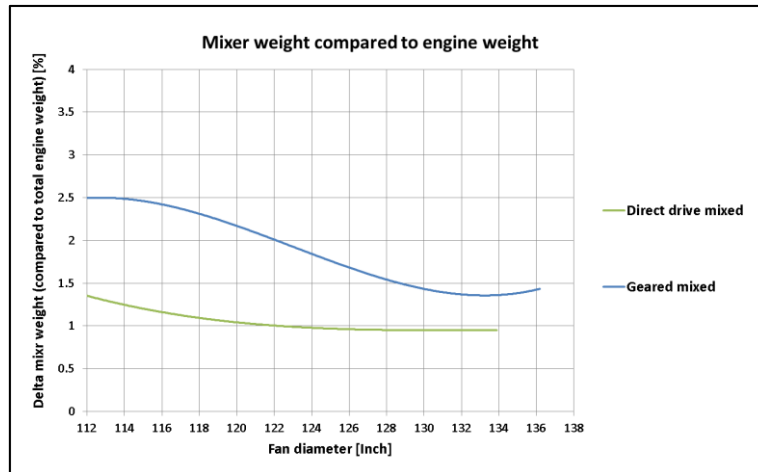


Figure 98 – Mixer weight compared to total engine weight

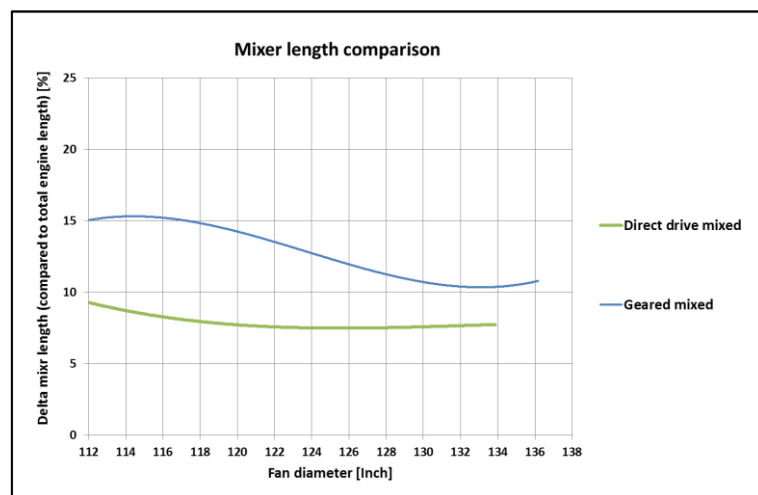


Figure 99 – Mixer length compared to total engine length

From Figure 99 it can be seen that once the mixer produces significant performance benefit at the lower fan diameter range, the mixer adds about 15% to the total engine length. Even with such a large contribution to total engine length the mixer only contributes 2.5% to the total weight of the engine. This is what is to be expected, as the mixer is in principle an empty duct with two different inlets. It can be concluded that the weight penalty of having a mixer would be minimal in both architectures over this fan diameter range.

The following results compare total engine length for all four configurations. The engine lengths produced by ATLAS do not account for mixer length. Thus, the mixer length was added manually to the resulting ATLAS length. The engine length comparison is presented in Figure 100.

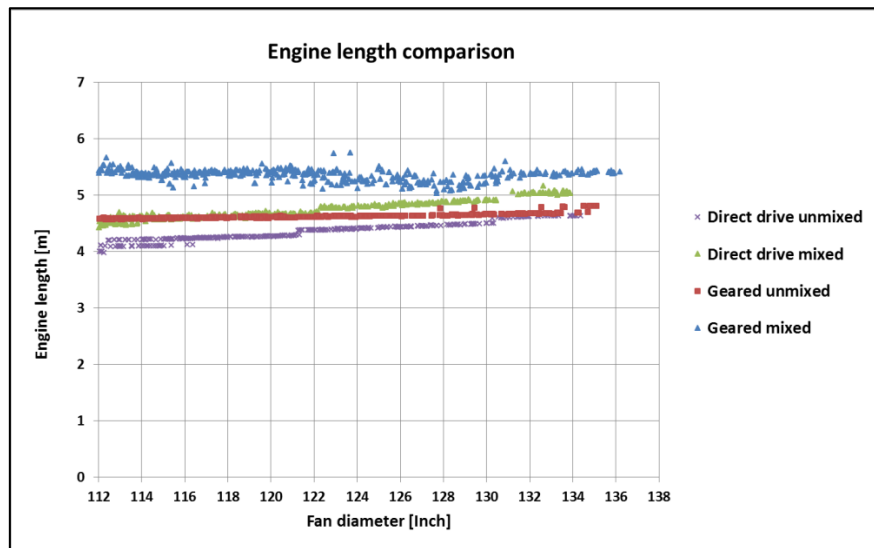


Figure 100 – Engine length comparison

Figure 100, shows that the geared configuration is always longer than the direct-drive configuration. This is due to the size of the gearbox which size has been optimised in previous research^[5], in this research an optimisation was conducted to find the a gearbox design which has the highest efficiencies and the lightest weight. This always pointed to very axially long and radially small gearboxes. The effect this would have on overall engine weight (due to longer nacelles etc.) was not taken into account but can now be seen. The gearbox even though is as efficient as possible with the given weight increases the engine length substantially, which when taking nacelle weight into account has increase the engine weight substantially as well as seen in Figure 97. Over the complete fan diameter range the geared unmixed engine length increases slightly. When looking at the length of the geared mixer it can be seen that at mixer length ranges

from around 0.5 to a meter in length this is due to the mixer efficiency which increases when going to lower fan diameter, which will result in a longer mixer. When looking at the geared mixed fixed it can be stated that the engine length is fairly constant. When looking at the length of the mixer there is a significant change in engine length over the fan diameter range for direct-drive. This is due to the increase in length experienced by adding more LPT stages. In the large fan diameter range it seems the two unmixed architectures are just as long. If the fan diameter could increase further than the direct-drive architecture, it would surpass the length of the geared architecture. The mixed cases do not cross each other, but are very close at the large fan diameters. Most likely these will cross at a larger fan diameter than was investigated. The mixer length for the direct drive case is shorter than for the gear case. This seems to concede with the lower mixer efficiency that the direct drive case has found which will automatically mean a short mixer due to how the mixer efficiency can be related to mixer length through Figure 17.

5.4.2 Effect of mixing efficiency on mixer weight and length

With the losses all taken into account, and the methodology described to estimate the weight and length, a study on the effect of mixing efficiency on these characteristics was conducted. Throughout the optimisation campaign mixing efficiency was limited to 45%, as this is a value which represents a reasonable efficiency for an annular mixer. The reason for setting this at 45% was a result of this study. By using the optimiser and setting the mixing efficiency at a fixed value instead of a variable, the optimiser will only produce engines with the set mixing efficiency. A range between 20% and 80% mixing efficiency was investigated and the effect this had on cruise performance is shown in Figure 101.

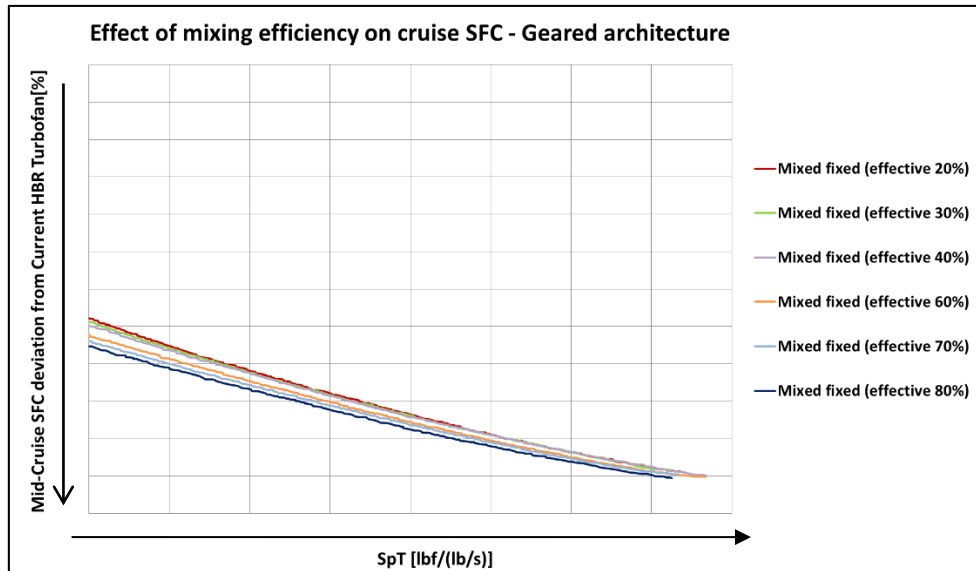


Figure 101 – Effect of mixing efficiency on Cruise SFC

From the figure it can be seen that, when increasing the mixing efficiency there is reduction in SFC at cruise. From this it can be concluded that the losses are not the limiting factor of the mixer regarding performance improvement at cruise, as there is still reduction of SFC at very low Specific Thrust levels. However, the amount of reduction at very low Specific Thrust values is very small. The amount of reduction present at the higher Specific Thrust range is more substantial.

Having a very high mixing efficiency has a significant impact on the weight of the mixer, and this is shown in Figure 102. Also, having a higher mixing efficiency influences the mixer length, shown in Figure 103.

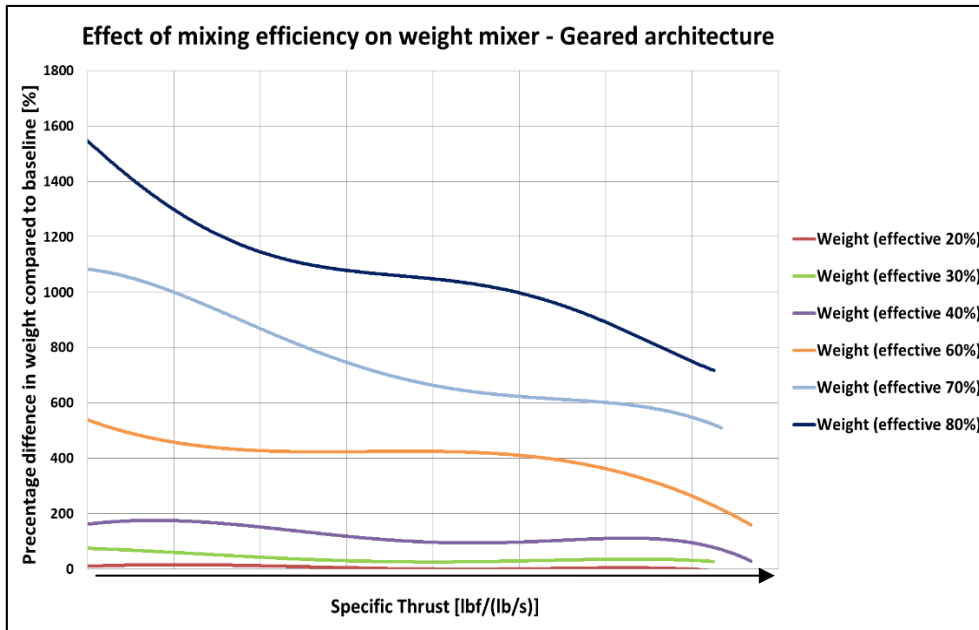


Figure 102 – Effect of mixing efficiency on mixer weight

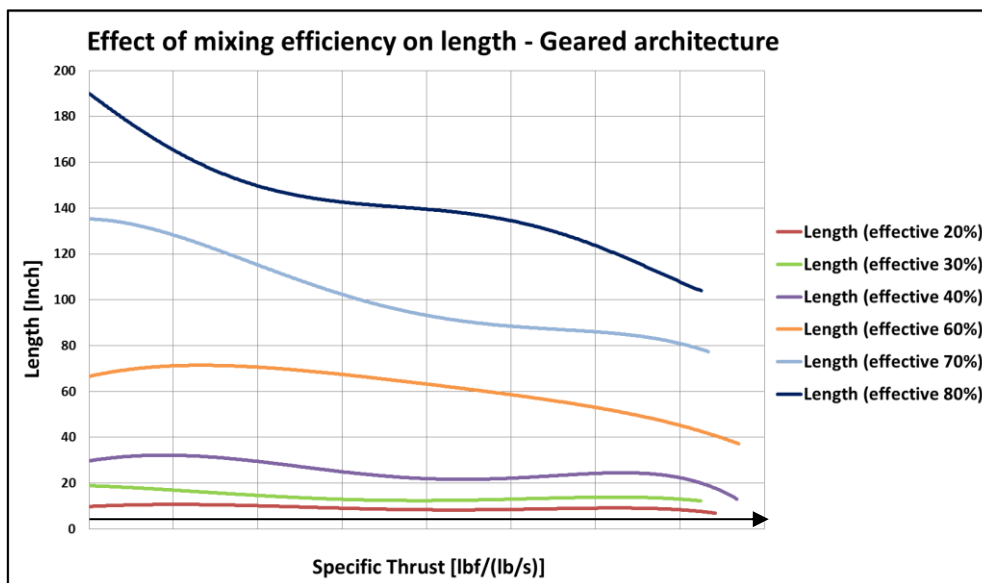


Figure 103 – Effect of mixing efficiency on mixer length

From the figures it can be seen that when setting the mixing efficiency higher, the weight and length increases significantly. This effect increases substantially when going to high values of mixing efficiency. The increase in mixer weight and length,

from the 20% to 40% mixing efficiency is still within acceptable values. Even the weight for a 60% mixing efficiency could be considered reasonable with respect to the total engine. However, any higher values of mixing efficiency, and hence mixer weight and length, would most likely increase mission fuel burn.

What can be concluded from this study is that there is potential for even a larger performance improvement than has already been demonstrated in this study, if a higher mixing efficiency is deemed practical. However, the higher mixing efficiency has significant influence on the mixer weight and length. Therefore, going to very high values of mixing efficiency, such as 80% would most likely not result in lower mission fuel burn, despite a reduction in SFC at cruise. Thus it was concluded that with the current mixer type, a maximum mixing efficiency of 45% should be set to prevent the optimiser from going to higher values in an attempt to improve cruise performance even further. Still if an alternative mixer type is chosen which allows very high mixing efficiency without increasing the weight, a useful performance increase could be achieved.

5.5 Mission fuel burn results

This section discusses whether the VGES shows any potential benefit in a very low Specific Thrust turbofan engine in terms of overall mission fuel burn. The mission fuel burn study was conducted using a mission profile described in section 3.7.1. The aircraft size that is investigated is an A330 and has a 3000nm mission range. The programme used to conduct this study is HERMES, which is described in section 0. To make comparability possible between the different configurations, all the results are compared against the baseline direct-drive engine with a fan diameter of 112 inches.

5.5.1 Direct-drive engine model results

The first results that are presented are the direct-drive fixed configuration, shown in Figure 104. This graph shows the SFC at cruise, drag, weight and mission fuel burn of the direct-drive fixed architecture. The drag values shown in all these results are the outcome of HERMES. This is 'cruise drag' and increases linearly as the fan diameter increases, due to the drag estimation method that is applied in HERMES (nacelle length scales linearly with fan diameter).

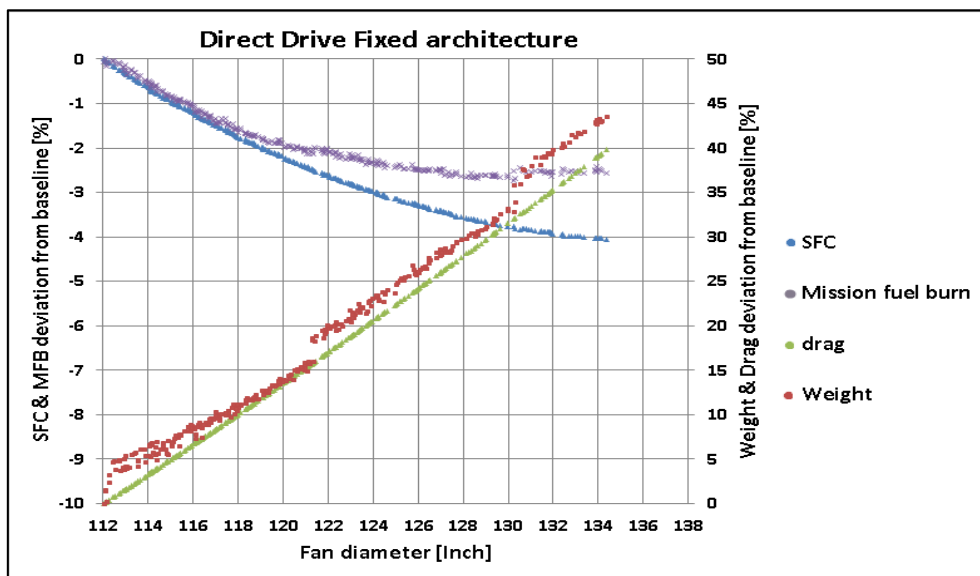


Figure 104 – Overall mission fuel burn performance of direct-drive, fixed exhaust architecture

From Figure 104 it can be seen that the weight and drag increase as the fan diameter increases. Furthermore, it can be seen that as the fan diameter increases the SFC decreases. The same applies for the mission fuel burn slope, but the mission fuel burn line flattens faster and more significantly than the SFC slope. This is due to the weight added on the engine. In this figure the effect of the increase in LPT stages diminishes the benefit of the SFC reduction, until a point where there is no more mission fuel burn reduction. It can be seen that by

going from a fan diameter size of 112 inches to fan diameter size of 134 inches, the total mission fuel burn reduction that can be achieved is about 2.5%.

Figure 105, shows the results of the direct-drive mixed fixed exhaust configuration. When comparing these results against those shown in Figure 104, a few differences can be observed. The first is that the weight of the engine is slightly higher. The drag of the mixed configuration is about the same as the unmixed configuration. From this it can clearly be seen that extra weight has increased the mission fuel burn slightly which now can only reduce by about 2.3% compared to the baseline engine.

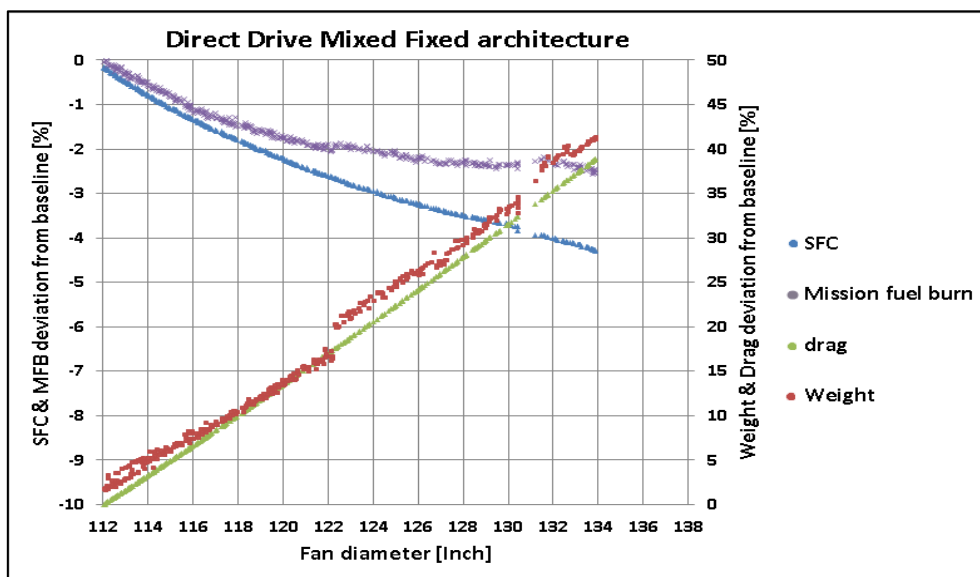


Figure 105 – Overall mission fuel burn performance of direct-drive, mixed, fixed exhaust architecture

5.5.2 Geared fixed engine model results

The results of the geared fixed configuration are shown in Figure 106. It can be seen that there is a significant reduction of SFC and mission fuel burn compared to the baseline engine. A total reduction of 6.1% is achieved for mission fuel burn

and 6.9% for SFC. The slope of the mission fuel burn curve does not flatten as much compared to the direct-drive case. The weight of the 112-inch geared case is already 15% larger than the baseline case. However, the weight increase is not as steep as with the direct-drive cases. It can be seen that the weight is very similar between the geared and direct-drive cases with a fan diameter of 134 inches.

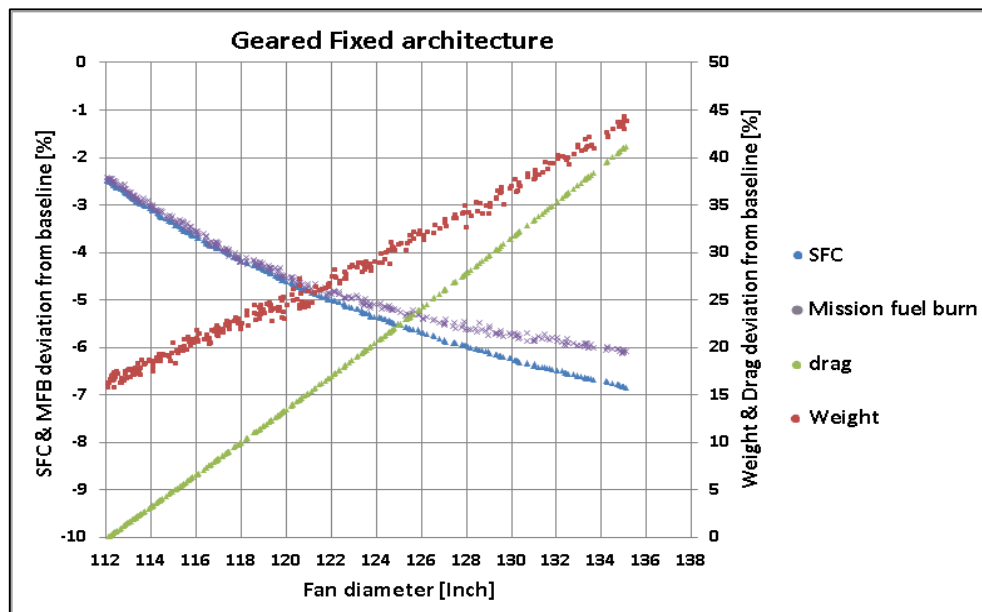


Figure 106 – Overall mission fuel burn performance of geared, fixed exhaust architecture

Figure 107 presents the results of the most promising configurations with VGES. In this figure the results of the geared mixed fixed configuration are shown. Comparing these results to the baseline engine and the unmixed geared fixed configuration, shown in Figure 106, will give more insight into the effect the mixer has on the engine configuration. The first thing that can be seen is that the drag of the engine is still very similar compared to the unmixed configuration. The weight added to the mixed configuration compared to the baseline engine increases by 5% at that large fan diameter, and 4% at the lower fan diameter. The mission fuel burn at the large fan diameter range is slightly worse in the

mixed fixed configuration, compared to the unmixed configuration. This is because there was nearly no performance benefit at that fan diameter, therefore the added weight will worsen the mission fuel burn. The mission fuel burn curve flattens out more for the mixed cases than the unmixed cases. This is a result of the reduction in benefit experienced by the mixer configuration when the fan diameter increases.

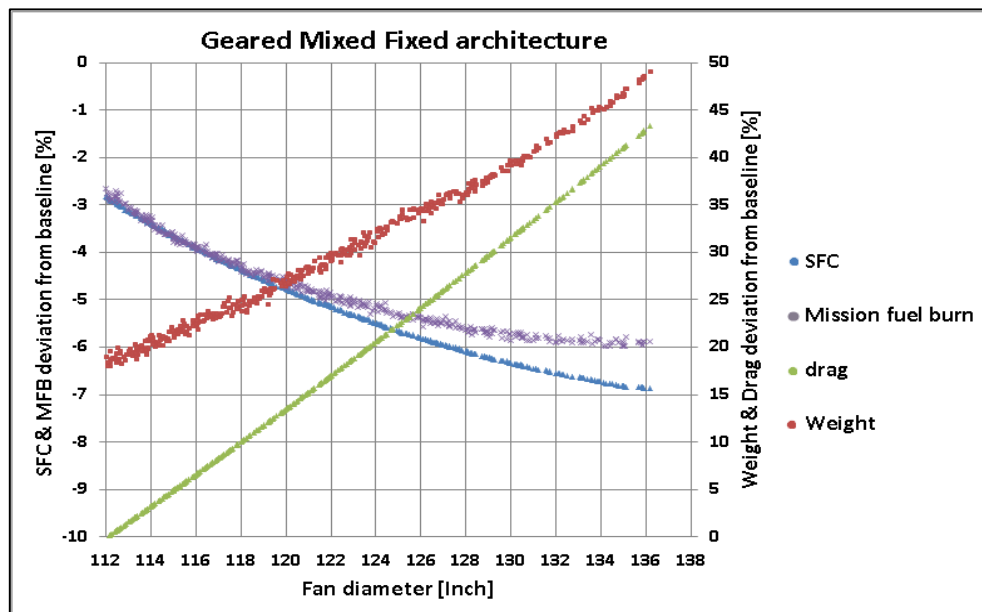


Figure 107 – Overall mission fuel burn performance of geared, mixed, fixed exhaust architecture

5.5.3 Mission fuel burn comparison

The final results which will conclude this research are the mission fuel burn comparisons. This graph will provide clear evidence of the potential benefit of using a VGES in the geared configuration. Figure 108 shows the mission fuel burn against fan diameter for the mixed and unmixed cases of both engine architectures.

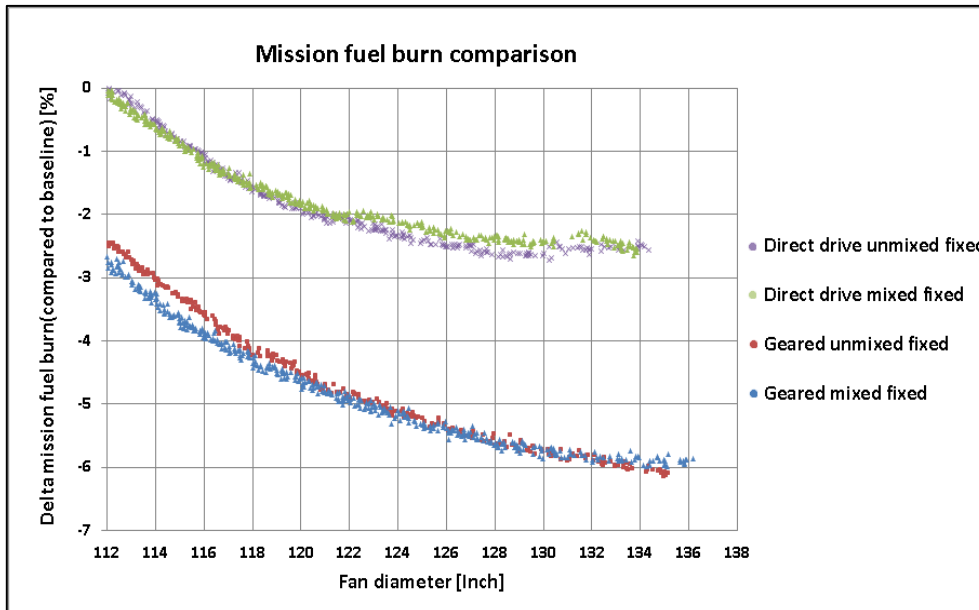


Figure 108 – Mission fuel burn comparison between mixed and unmixed configurations

From this figure it can be seen that from a fan diameter above 124 inches, there is no visible benefit of adding a mini-mixer to the configuration. Even when looking at the benefit between the 118- and 124-inch fan diameters, it can be seen that there is very little reduction in mission fuel burn. Only when going to fan diameters lower than 118 inches is the improvement in mission fuel burn approaching an acceptable value to be considered. The maximum reduction of mission fuel burn at 112-inch fan diameter is about 0.5%. When going to a higher fan diameter of 132 inches, the mini-mixer starts increasing the mission fuel burn compared to the case without mini-mixer. Looking at the direct-drive mixed fixed configuration, the maximum reduction in mission fuel burn that can be achieved is only 0.1%. This is only achieved at a fan diameter of 112 inches. When going to fan diameters higher than 118 inches the mini-mixer increases the mission fuel burn. This increases until the mission fuel burn is 0.1% worse compared to the unmixed case.

The mission fuel reduction for the geared case does not seem like a significant improvement, when taking into account the extra complexity the mini-mixer imposes. However, 0.5% reduction in mission fuel burn could still represent hundreds of tons of fuel saving over a full year in service. When looking at the complete life span of the engine, this quickly adds up to a significant amount.

Another observation to make is that even though there is a benefit at the lower fan diameter range, there isn't any at the larger fan diameters. However, the large fan diameter configuration has substantial less SFC and mission fuel burn compared to the smaller fan diameters anyway. This means there has to be another good reason for going with a smaller fan diameter, to be able to reap any benefit from the mini-mixer of the VGES. Reasons for this could be limited space under the wing, or fuselage mounted engines. In these circumstances, the mini-mixer from the VGES could provide an improvement in mission fuel burn.

The final conclusion regarding the VGES is that with regards to area variability, there is no requirement for them, as the performance and operability benefits are believed to be negligible. As mentioned above where the VGES with mini-mixer can have a benefit is at fan diameters lower than 118-inches. From this point there is a mission fuel burn reduction present. Further reduction in mission fuel burn might be achievable by changing the mission profile or thrust class.

6 Conclusions & recommendations

The main aim of this research has been to demonstrate the potential benefits of incorporating variable exhaust nozzle technology, and to investigate the various trade-offs between architectural and cycle parameters, in order to achieve an optimum solution in terms of mission fuel burn and/or operability. This research was part of the Rolls-Royce UTC in Aero System Design, Integration & Performance at Cranfield University. This study focussed on a specific patent from Rolls-Royce. The patent concerned a Variable Geometry Exhaust System (VGES). The novel technology in this patent is that the exhaust system mixes part of the bypass flow with the core exhaust, and features four variable areas. Two variable areas are nozzles; the fan and core, the other two variable areas are the inlets of the mini-mixer. The exhaust system could be applied to two different engine architectures, the direct-drive and the geared turbofan. Engine configurations investigated were considered to be very low Specific Thrust turbofans, to enter service between 2025-2030. For this research, a methodology was developed which focused mainly on the mini-mixer (mixing part of the bypass flow with core exhaust). The goal was to simulate as accurately as possible the engine with a mini-mixer, while taking all efficiencies and losses into account. The methodology developed was designed to be used in a computational framework, to be able to find the optimum engine configuration over a wide range of fan diameters. The outcomes of this study helped reveal potential benefits of the VGES and major limitations.

6.1 Major outcomes

In a nutshell, the overall methodology developed follows an engine cycle design optimisation, together with an estimation of engine weight, drag and mission fuel burn. It was important in the cycle design and engine weight calculations to

simulate the VGES as realistically as possible. The methodology was verified against public domain data, and / or was approved by the industrial partner. It describes how to accurately simulate mixing, while taking mixing efficiency and losses into account. To the best of the author's knowledge this method is presented for the first time in the public domain in such detail.

The optimisation tool was specifically developed with the VGES study in mind. After substantial amount of trial and error, it was concluded that the tool was robust enough. It was capable of producing optimum engine Pareto fronts, which could be used in trade-off studies. Furthermore, it was confirmed that the tool was able to investigate the entire design space. Stability issues within the tool were mainly caused by the mini-mixer. This was due to the fact that the mixer required a certain set of flow conditions to work. Stability issues were reduced to a minimum with the current setup. It needs to be pointed out that there are some fixed assumptions regarding axial Mach number in the LPT system, which can significantly affect the optimum design. These assumptions were verified by the industrial partner to be acceptable. A final point to make is that the accuracy of this tool is dependent on all the available input data that was used, and the specific range investigated.

The major conclusions of this doctoral research are listed below:

1. Mixing efficiency has a substantial effect on the engine configuration as it influences the length and the losses experienced by the mixer substantially. Furthermore, mixing efficiency influences the weight significantly due to the increase in mixer length. In regards to performance benefit, the mixer losses are not the dominant factor in limiting the mixing efficiency.
2. Pressure loss is a key parameter in the performance and hence viability of a mini-mixer design. The critical pressure loss within the mini-mixer system

is the skin friction pressure loss. Out of the two skin friction pressure losses, the cold-inlet was found to be the most critical. However, hot-inlet skin friction pressure losses have greater influence on the mixer configuration. Therefore, it is essential that losses are estimated as accurately as possible.

3. With regards to mid-cruise SFC it can clearly be seen that of the two engine architectures, the geared has a substantial reduction in SFC compared to the direct-drive. This was found to be the case across the full fan diameter range that was investigated. The reduction in SFC is due to the effect of the gearbox, which means that both the fan and the LPT can rotate at their optimum speeds, resulting in higher efficiencies.
4. When adding the mini-mixer of the VGES to the geared configuration, it can be seen that in the lower fan diameter range the mini-mixer does indeed reduce SFC at cruise. At the lowest fan diameter of 112 inches a 0.5% reduction in SFC was observed. However, when going to larger fan diameters, the reduction in SFC diminishes compared to the unmixed case. Investigation into why the SFC reduction diminished showed that the optimiser was trying to remove the mini-mixer from the design. Adding the mini-mixer of the VGES to the direct-drive configuration, also results in a small improvement in the performance at cruise for fan diameters of 112 inches. However, this improvement is only 0.1% compared to the configuration without mini-mixer. When increasing the fan diameter, the reduction in SFC diminishes. However, no clear benefit is visible from 118-inch fan diameter cycles, where the geared configuration still presents a slight benefit, up to 136 inches. The reason for the direct-drive configuration having less improvement is due to the limit the LPT lets the fan FPR be reduced, which influenced the thrust gain from the mixer and the mixer skin friction losses, compared to the geared configuration, to accommodate the mixer and maximise the benefit.

5. The main limiting factor in the optimisation is the T30 temperature. This maximum temperature limits the OPR that can be achieved. In the optimised engines, it can clearly be seen that T30 values were very close or exactly on the T30 limit. The T40 is also a factor which limits the optimisation. The T40 limitation is always hit at the EOR operating point. The T40 is a major consideration in setting the BPR. However, there are circumstances when setting T40 to the limit will not result in the best performance configuration. This is when the core size cannot reduce any further, or due to the fact that more performance benefit is achieved when using a mini-mixer when the core is slightly larger. Therefore, when going with the mini-mixer configuration or when allowing area variation in the VGES, it is seen that the T40 limit is not reached. This is an added benefit, as it will increase the life of some of the engine components.
6. From a performance view point, an investigation was conducted with regards the variability of the VGES. In this investigation a single or combination of variable areas of the VGES were allowed to change in off-design conditions. From this it was found that regardless of how many areas were allowed to vary, no significant performance benefit was present. Even though no performance benefit was discovered due to variability, a few other observations were made in regards to the variability.
- a) The optimum fan nozzle area at EOR is larger than at cruise in all variable cases
 - b) The optimum fan nozzle area at TOC is always smaller than the area required at cruise
 - c) When the core nozzle is allowed to vary, the optimum area variation for the fan nozzle is reduced, compared to the case where only the VAFN is varied
 - d) When mixer inlet area is allowed to vary, the optimum mixer inlet area at the EOR and TOC is larger than at cruise
 - e) Variability range was reduced when VAM was allowed to vary due to stability issues

- f) VAM works best in combination with VACN, which allows a larger variation range to be found
 - g) Adding variability reduces maximum T40 experienced by the configuration by about 20 Kelvin in the large fan diameter range. This could improve the turbine life.
7. From an operability view point, after extensive investigation it was found that no variability in the VGES is required. Normally when designing very low Specific Thrust engines, the likelihood of the fan to surge at EOR increases. However, in the cases examined and for the compressor characteristics used in this study, this was not the case.
8. From the mission fuel burn studies it can be concluded that there is a potential benefit for the VGES. This is in the smaller fan diameter range of the geared architecture. With a fan diameter of 112 inches a total mission fuel burn reduction of about 0.5% can be achieved. This does not sound like a significant amount, but over a complete year of operation this could result in a reduction of hundreds of tons of jet fuel. For the direct-drive architecture there is only a very small reduction in mission fuel burn of about 0.1%

To summarize, the major outcome is that there is a potential benefit for the VGES in the small fan diameter range and for the geared architecture. Nevertheless, going to larger fan diameters without VGES shows even further mission fuel burn reduction. Consequently, this means that there has to be a specific reason to go to smaller fan diameter to justify the use of the VGES. The benefit only arises from the implementation of the mini-mixer, as the variability does not improve the performance at cruise.

6.2 Contribution to knowledge

From the work presented in this research and the open literature survey that has been carried out by the author, the following can be considered to be the contribution to knowledge.

1. The methodology in chapter 3 describes in detail how to take into account the different aspects that are of importance to a design engineer, when designing a mixed exhaust turbofan engine. Some individual parts of this methodology have already been presented in the public domain. However, the critical areas; which are the mixing efficiency influence and the influence of individual mixer component pressure losses, and how to take them into account, have never been presented before. Additionally, a complete overview of how to take all the different aspects of the mixer design into account and how each aspect influences other parameters has not been published yet. This methodology can be adapted to any engine with a mixed exhaust configuration.
2. The investigation into the novel technology of VGES, which is derived from a patent, has shown that there might be engine configurations which could benefit from its use. However, this only applies to the mini-mixer, as the added variability has not been shown to provide any benefit in the design space investigated in this study. The literature states that exhaust mixing in gas turbines is not beneficial with higher BPRs; this research has shown that through the use of partial mixing there is still a potential for mission fuel burn benefit even in large BPR turbofans. The performance assessment of the geared and direct-drive, with and without VGES, has given insight into the influence of partial mixing on these configurations. This will increase the knowledge regarding the design of future mixed exhaust systems in gas turbines.

3. The variable area schedule study gave insight into the effects of having multiple variable areas. It also highlights the influence of the partial mixer when using multiple variable areas. It shows that when using partial mixing and variable areas, the core nozzle can be used to reduce the optimum variation required for the fan nozzle. Additionally, it reveals the need for a variable core nozzle, if the variable mixer inlet area is to be used. This increases the knowledge regarding the design of multiple variable nozzles.
4. The tool developed for the optimisation of the direct-drive and geared architecture with and without VGES can be used for educational purposes, as well as commercial. This tool can be adapted to investigate alternative thrust classes as well as alternative configurations. Furthermore, with minor alteration to certain equations, alternative mixer types can also be investigated.

6.3 Recommendations

Some recommendations for future work are listed below:

1. A conclusion of this research is that area variability in the VGES is not needed from a performance point of view. Without the variability requirement alternative mixer types can be considered. Consequently, an investigation into alternative mixer types, such as a lobed mini-mixer should be considered. If a lobed mini-mixer is used this could decrease the length of the mixer which is a major contribution to the pressure losses.
2. With the current requirement certain constraints were set that prevented searching a wider design range, which might show even further potential for the VGES. By allowing the following requirements to be altered a wider design space could be investigated.

- a. Changing the engine architectures
 - b. Changing the mission profile
 - c. Changing the thrust class
 - d. Changing technology limitations
3. Further upgrading of the optimisation tool to allow more variability to be taken into account. One of the variables that was not considered in this research is the plug size, which was assumed to be of a fixed size. The plug size influences the pressure losses significantly as it changes the required diameter for the mixer. Thus, by allowing the optimiser to find the optimum plug size it might widen the area of benefit for the VGES.
4. Alternative VGES designs might provide further benefit in mission fuel. A design that might be considered is the plug-less design. In this design the turbofan with VGES is designed to have no plug, which would reduce the pressure losses experienced by the mini-mixer.
5. The continuation of the development of complete mission fuel burn optimisation strategies. This can be achieved by connecting all the tools that are currently available into a single system. As the programmes are already designed to work together, and with basic input assumptions which have already been used in this research, this task should now be possible. This would reduce the time to conduct studies on different engine architectures significantly. Additionally, it could be used for teaching and educational purposes.

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APPENDIX A – Weight estimation

In this appendix the reader will find the tables regarding which information was needed to estimate the weight of each individual component when using Cranfield in-house software ATLAS. This includes Fan, compressor, turbine and combustor design.

Table 14 - Fan design input parameters

Preliminary Compressor Specifications	
OPR	Values relative to DP
BPR	
Inlet & Outlet Mass Flow	
Inlet & Outlet Pressure	
Inlet & Outlet Temperature	
Relative Rotational Speed	
Initial Design Considerations	
Gearbox Index	Switch for PGB existence
Annulus Configuration	Constant Outside/Inside/Mean Diameter
Core Configuration	Constant Outside/Inside/Mean Diameter
Core Diameter Variation	Fan hub outlet to ESS hub inlet
Rotor and Stator Space/Chord	
Rotor and Stator Profile	
De Haller Number Limit	
Diffusion Factor Limit	
Relative Tip Mach No. Limit	
Blade Tip Velocity Limit	
Rotor and Stator Solidity	
Rotor and Stator Aspect Ratio	
Rotor – OGV Gap	
Rotor – ESS Gap	
ESS Aspect Ratio	
ESS Profile	
Strut Switch	Switch for strut existence
Strut – OGV Gap	
Strut Aspect Ratio	
Strut Space/Chord	
Strut Profile	
Strut Solidity	
Inlet and Outlet Velocity or Mach No.	Absolute or Axial
Inlet and Outlet Flow Angles	Between Axial and Absolute Direction
Inlet and Outlet Hub/Tip	

Table 15 – Compressor design input parameters

Preliminary Compressor Specifications	
OPR	Values relative to DP
Inlet & Outlet Mass Flow	
Inlet & Outlet Pressure	
Inlet & Outlet Temperature	
Relative Rotational Speed	
Initial Design Considerations	
Compressor Type	IPC or HPC
Gearbox Index	Switch for PGB existence
IGV Switch	Switch for IGV existence
Annulus Configuration	Constant Outside/Inside/Mean Diameter
Rotor and Stator Space/Chord	
Rotor and Stator Aspect Ratio	
Rotor and Stator Profile	
De Haller Number Limit	
Diffusion Factor Limit	
Relative Tip Mach No. Limit	
Blade Tip Velocity Limit	
IGV Aspect Ratio	
Rotor – Stator Gap	
Inlet and Outlet Velocity or Mach No.	Absolute or Axial
Inlet and Outlet Flow Angles	Between Axial and Absolute Direction
Inlet and Outlet Flow Slope	Flow Slope Angle
Inlet and Outlet Hub/Tip	

Table 16 – Turbine design input parameters

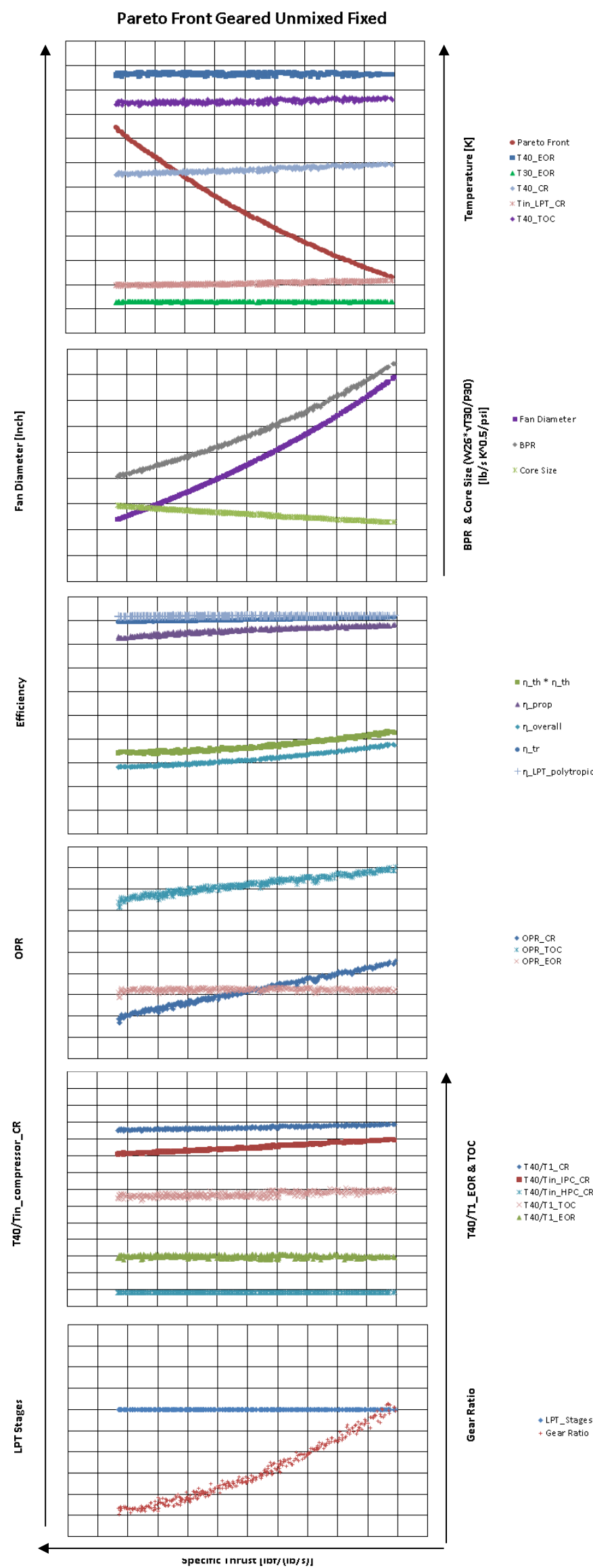
Preliminary Turbine Specifications	
Inlet & Outlet Mass Flow	Values relative to DP
Inlet & Outlet Pressure	
Inlet & Outlet Temperature	
Inlet and Outlet FAR	
Relative Rotational Speed	
Initial Chosen Values	
Turbine Type	HPT or IPT or LPT
Gearbox Index	Switch for PGB existence
OGV Switch	Switch for OGV existence
Shroud Switch	Indicates if the blades are shrouded
Rotor and Stator Aspect Ratio	
Rotor and Stator Profile	
OGV Aspect Ratio	
Outlet AN ₂ Limit	
Stage Loading Limit	
Tangential Loading Limit (Zweifel)	
Blade Tip Velocity Limit	
Reaction Target	At the blade mid-span
First Stage Cooling Fraction	Part of cooling used by the first stage
NGV Cooling Fraction	Part of cooling used by NGV in a stage
Cooling Feeding Point	Position of the bleed extraction point on the respective compressor
Inlet and Outlet Velocity or Mach No.	Absolute or Axial
Inlet and Outlet Flow Angles	Between Axial and Absolute Direction
Inlet and Outlet Flow Slope	Flow Slope Angle
Inlet and Outlet Hub/Tip	

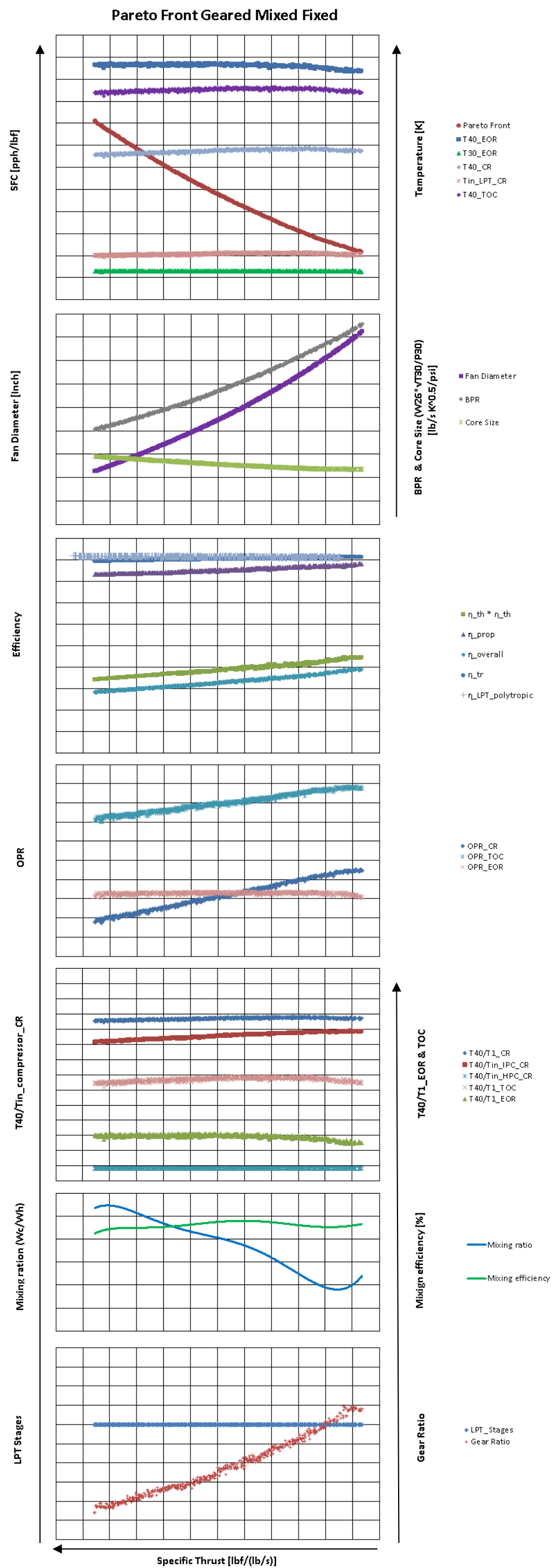
Table 17 – Combustor design input parameters

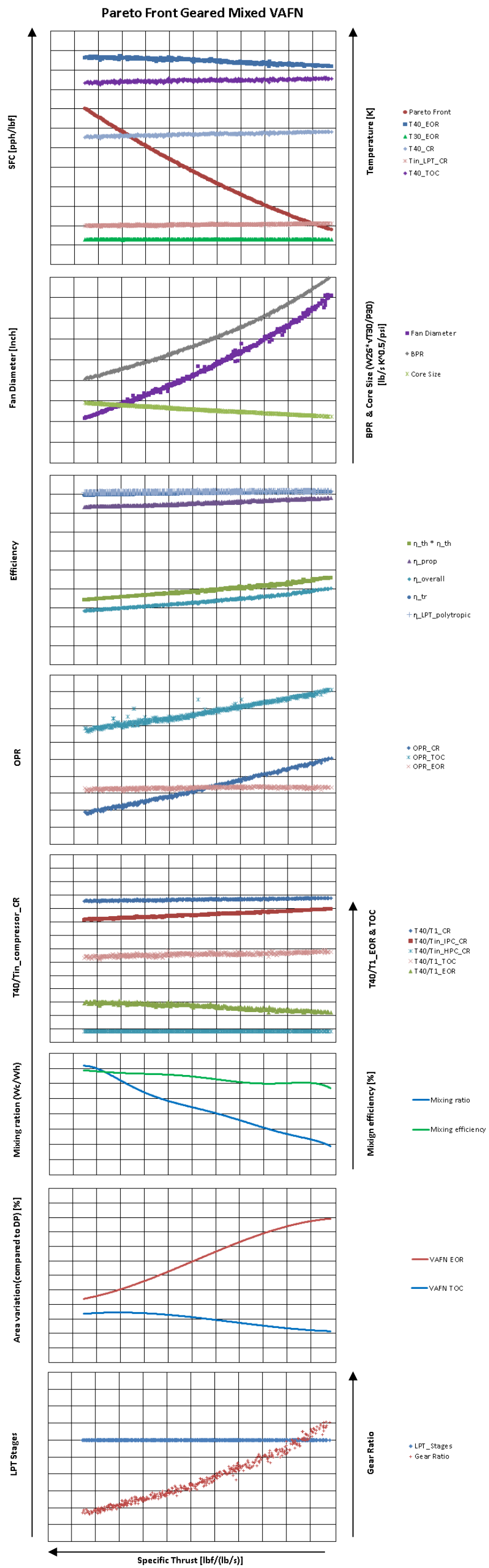
Input Design Considerations	
Combustor Type	Annular or Tubular or Turbo-Annular
Inlet and Outlet Mass Flow	
Fuel/Air Ratio	
Atomising Flow Ratio	
Dome Cooling Flow Rate	
Dome Gap Ratio	
Cooling Fraction	Percentage taken from the Compressor Exit
Primary Zone Equivalence Ratio	
Dome Axial Velocity	
Passage Axial Velocity	
Combustor Length to Height Ratio	
Stoichiometric FAR	Depending on fuel type
Inlet and Outlet Velocity or Mach No.	Absolute or Axial
Inlet and Outlet Flow Angles	Between Axial and Absolute Direction
Inlet and Outlet Hub/Tip	

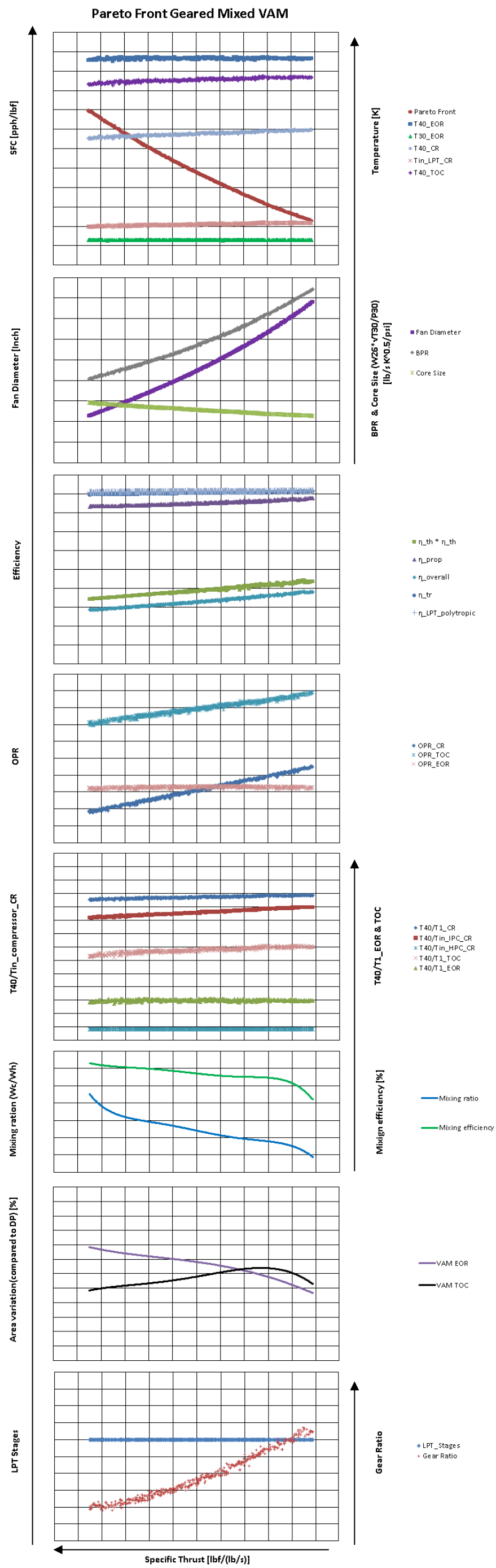
APPENDIX B – Optimised engine cycle parameters

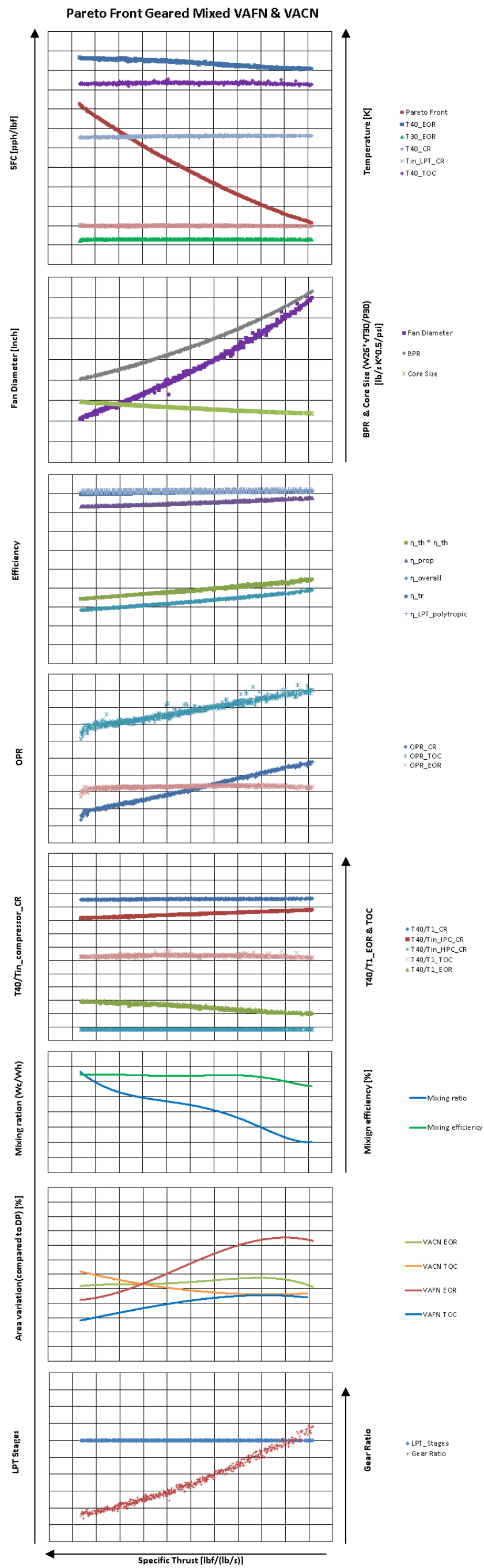
In this appendix the reader will find all the information regarding the optimised engine cycle parameters for each optimised case. This data is presented in the form of stacked graphs which all have the same x-axis of specific thrust and y-axis will be different for each graph

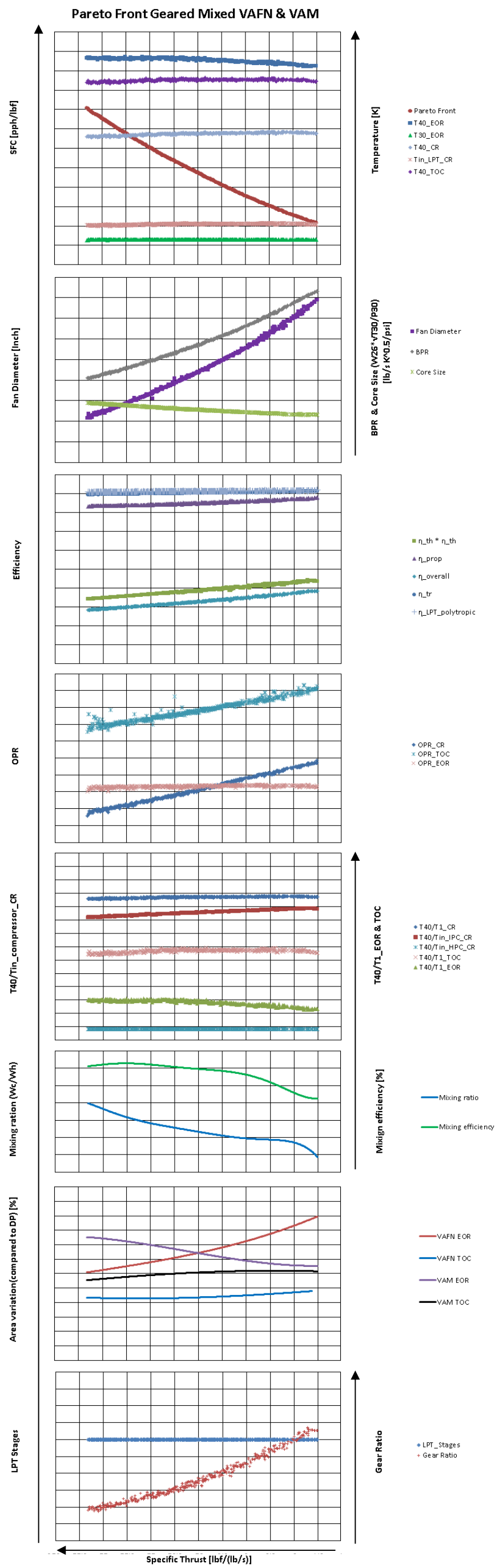


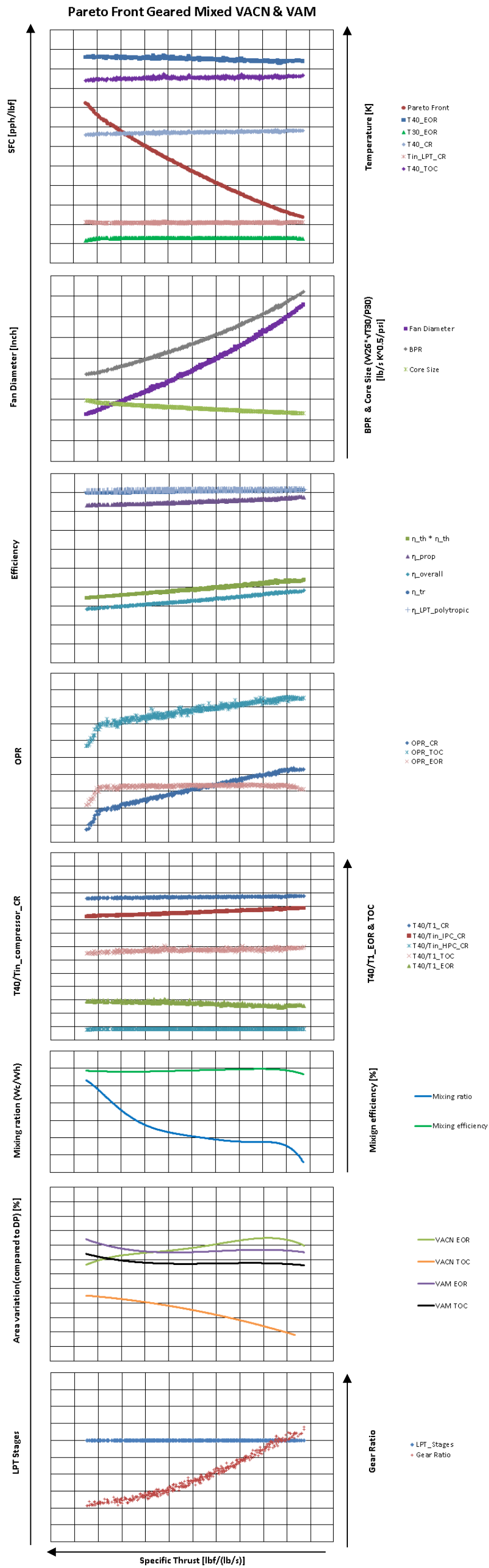


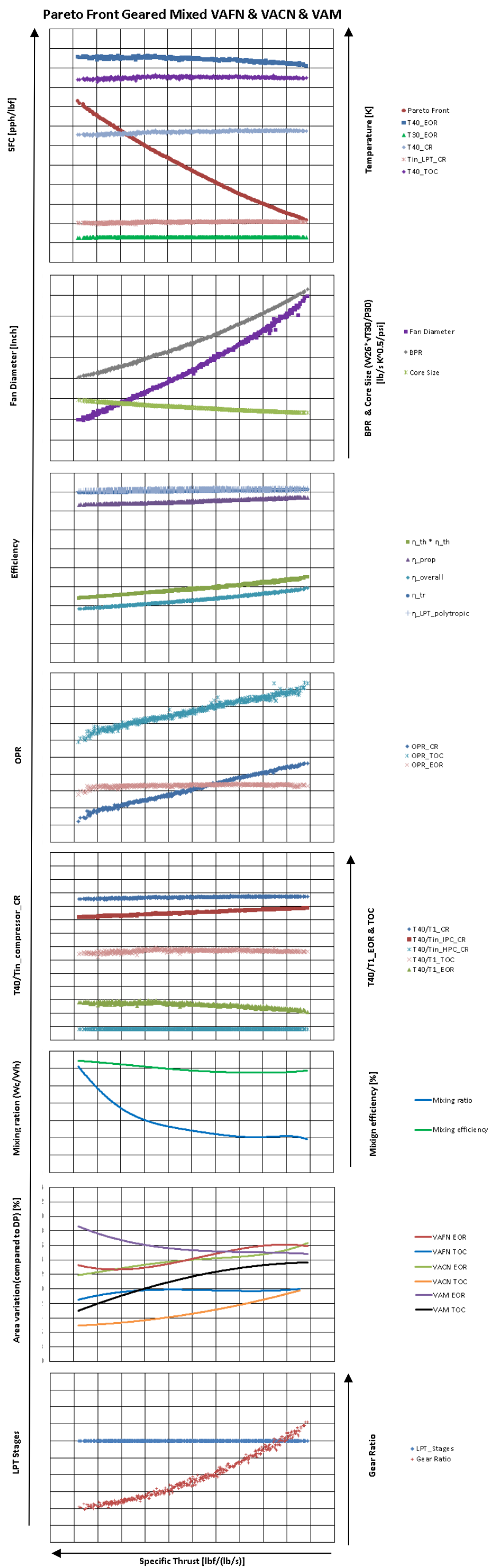


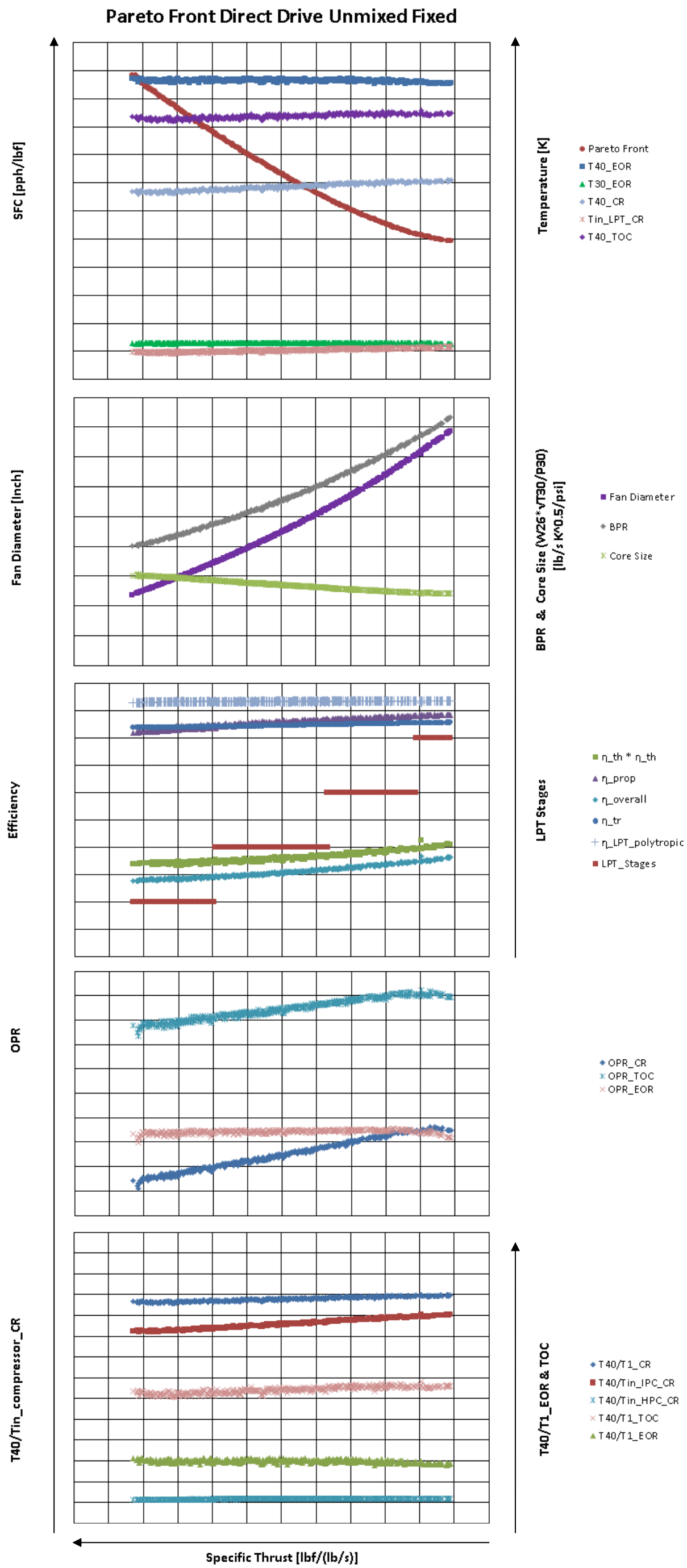


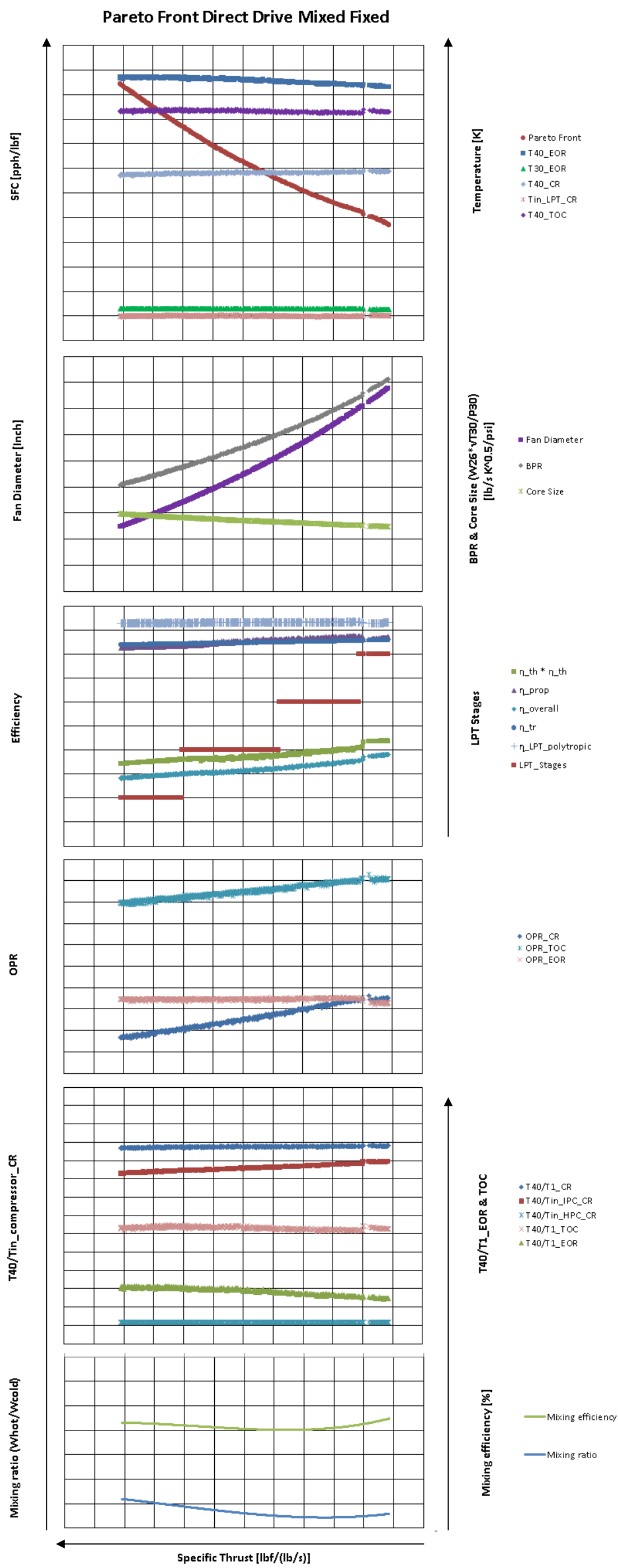




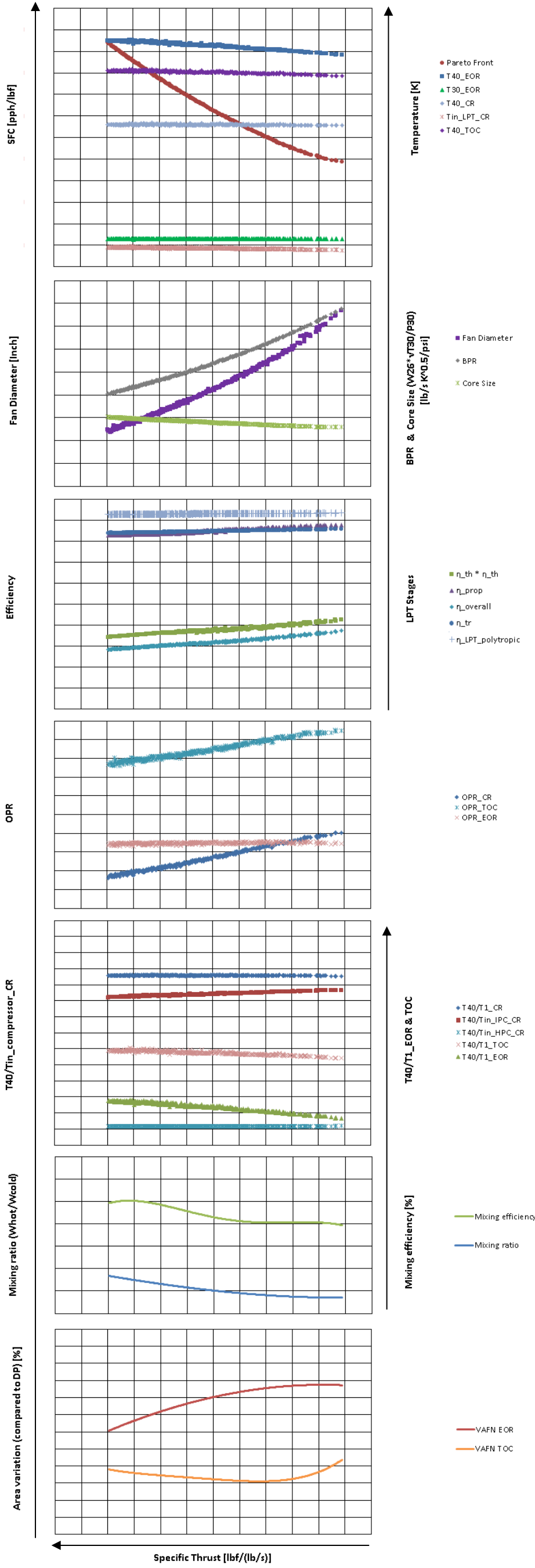




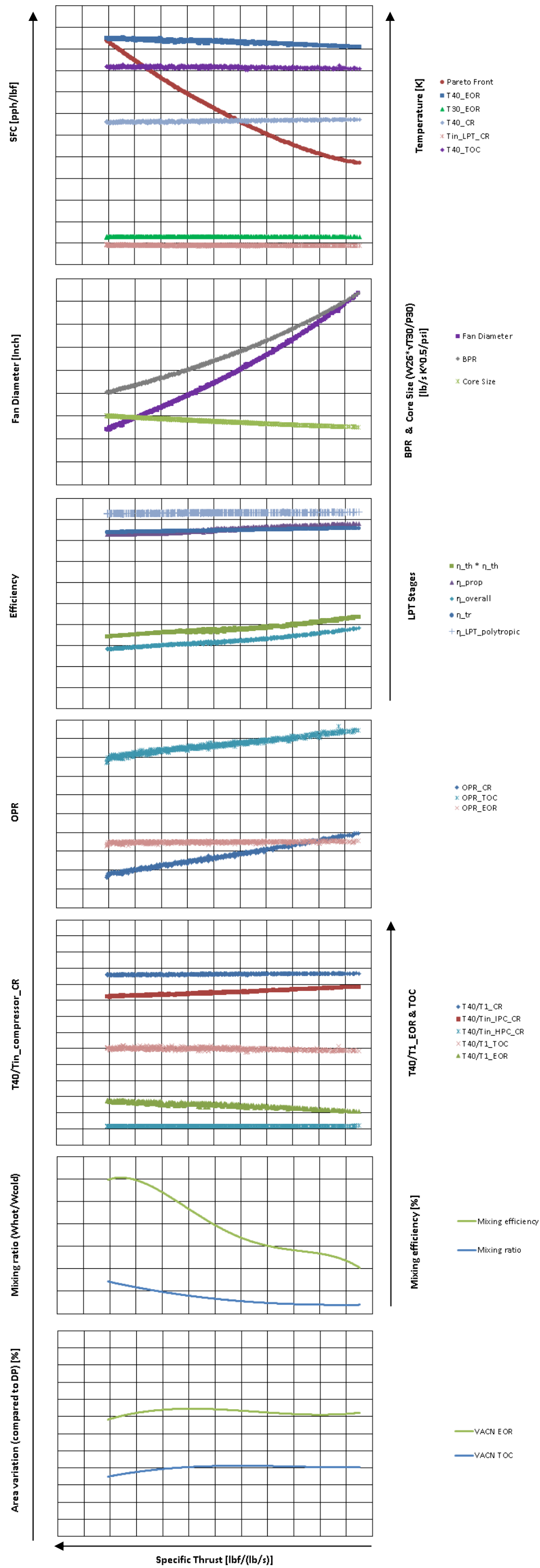


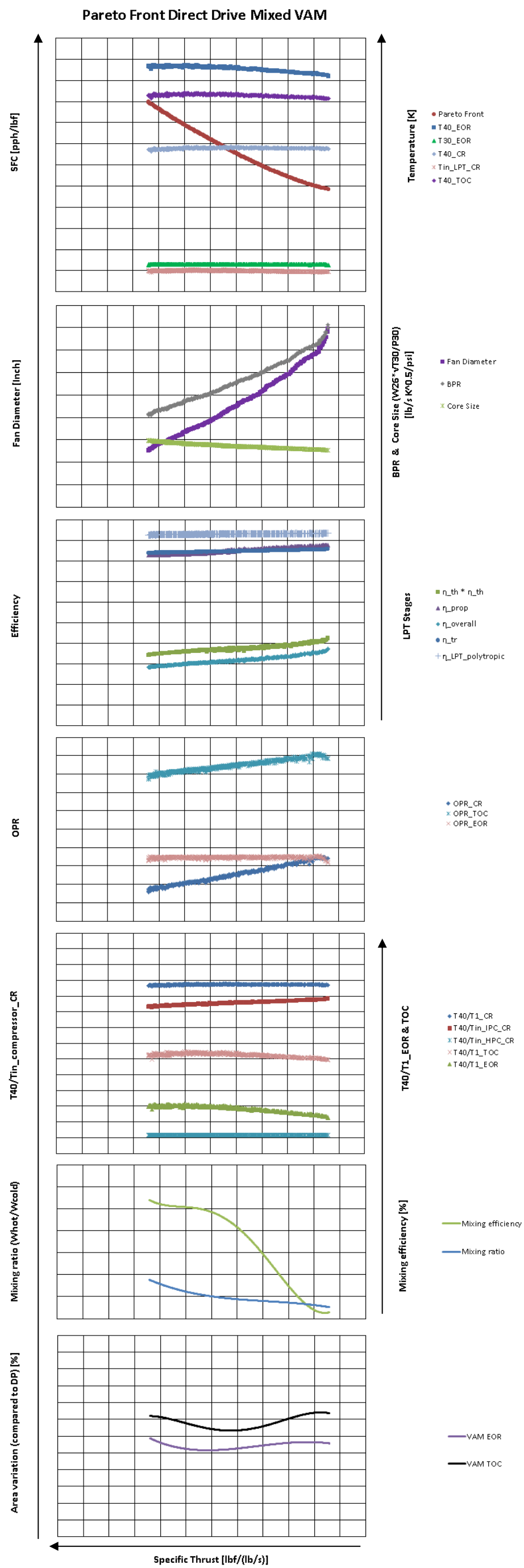


Pareto Front Direct Drive Mixed VAFN

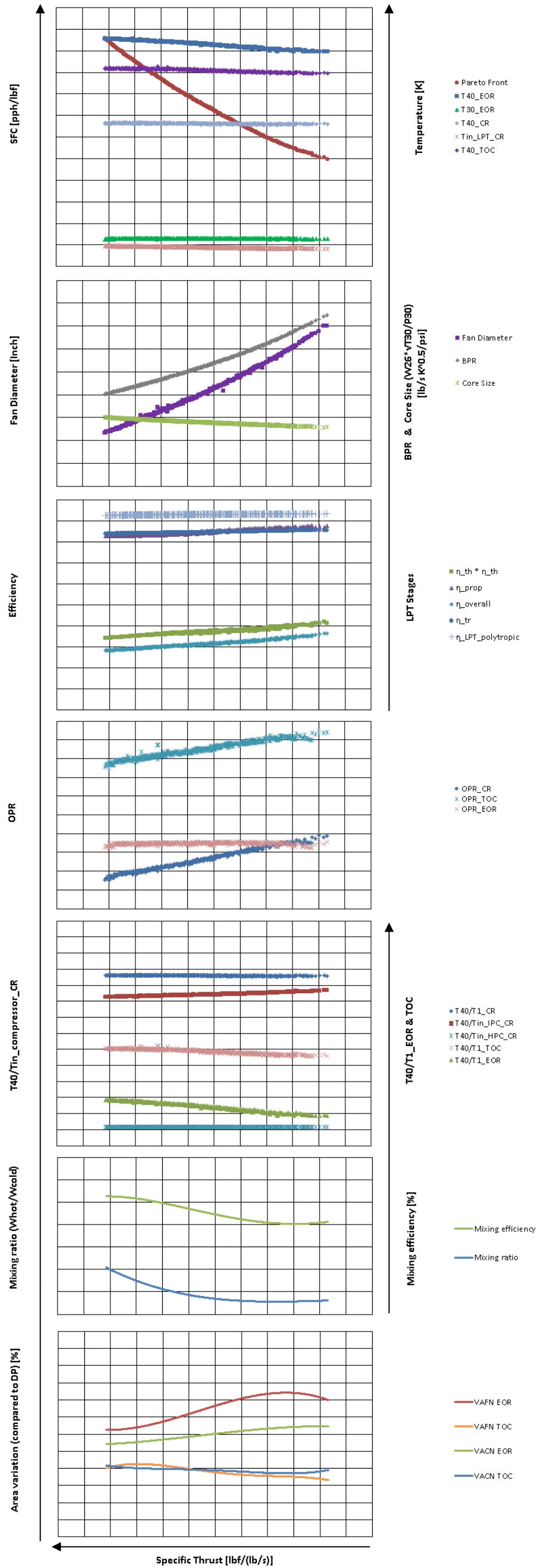


Pareto Front Direct Drive Mixed VACN

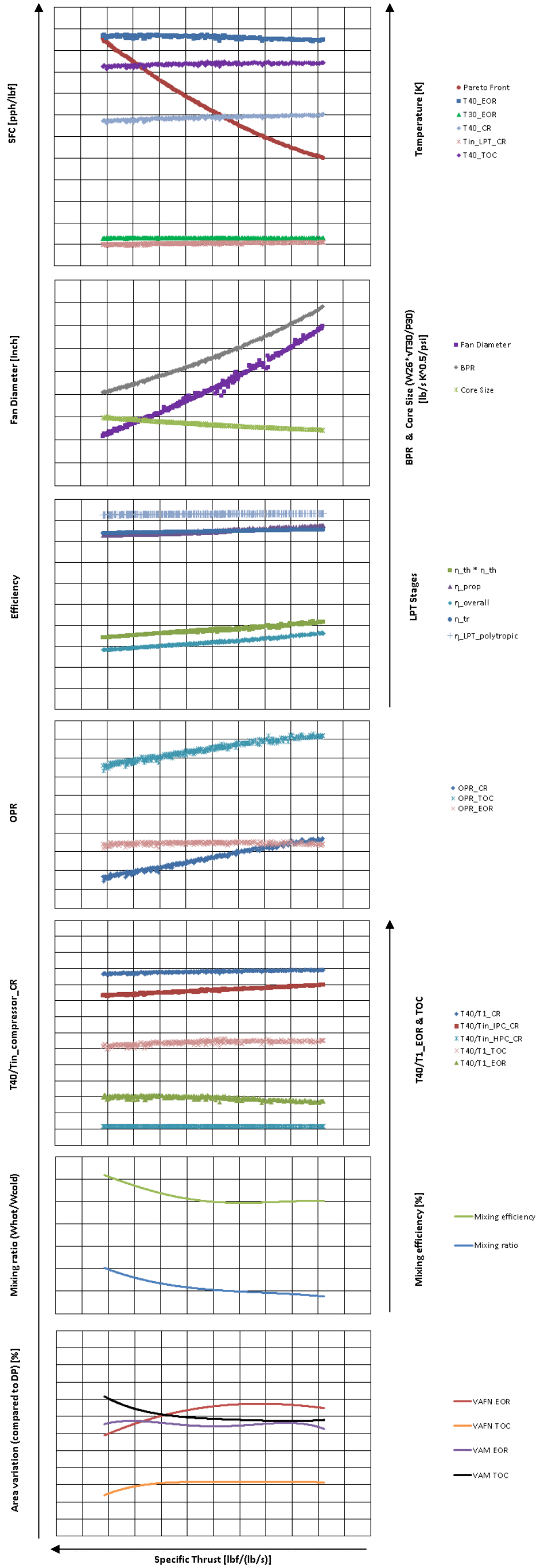




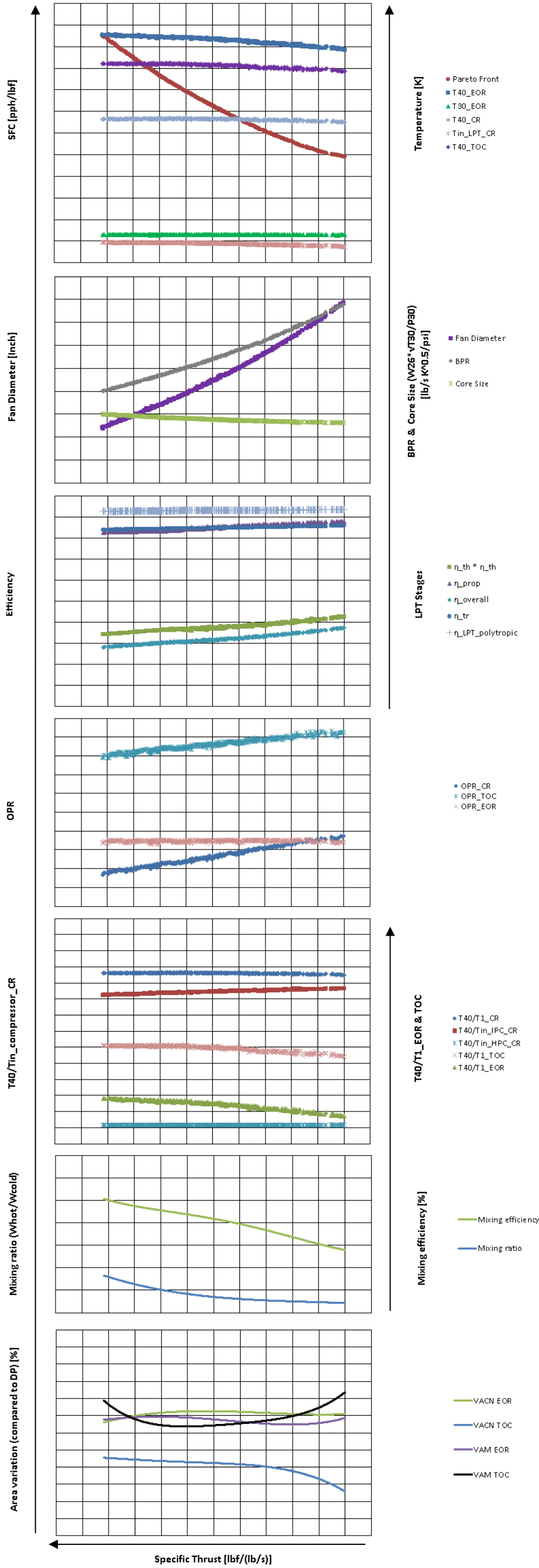
Pareto Front Direct Drive Mixed VAFN & VACN



Pareto Front Direct Drive Mixed VAFN & VAM



Pareto Front Direct Drive Mixed VACN & VAM



Pareto Front Direct Drive Mixed VAFN & VACN & VAM

