Validation and integration of a rubber engine model into an MDO environment

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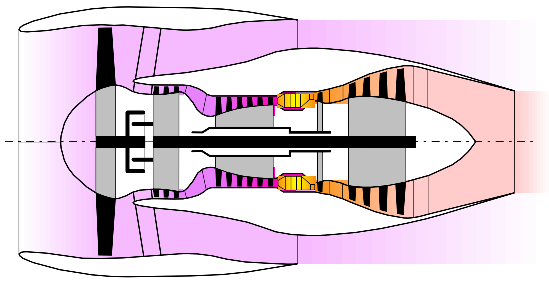

Master Thesis
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LIU-IEI-TEK-A--10/00959--SE

Linköping 2010

Abstract

Multidisciplinary design optimization (MDO) is a technique that has found use in the field of aerospace engineering for aircraft design. It uses optimization to simultaneously solve design problems with several disciplines involved. In order to predict aircraft performance an engine performance simulation model, also called “rubber engine”, is vital. The goal of this project is to validate and integrate a rubber engine model into an MDO environment.

A method for computer simulation of gas turbine aero engine performance was created. GasTurb v11, a commercial gas turbine performance simulation software, was selected for doing the simulation models. The method was validated by applying it to five different jet engines of different size, different type and different age. It was shown that the simulation engine model results are close to the engine manufacturer data in terms of SFC and net thrust during cruise, maximum climb (MCL) and take off (MTO) thrust ratings. The cruise, take off and climb SFC was in general predicted within 2% error when compared to engine manufacturer performance data. The take off and climb net thrust was in general predicted with less than 5% error. The integration of the rubber engine model with the MDO framework was started and it was demonstrated that the model can run within the MDO software. Four different jet engine models have been prepared for use within the optimization software.

The main conclusion is that GasTurb v11 can be used to make accurate jet engine performance simulation models and that it is possible to incorporate these models into an MDO environment.

Preface

This thesis work has been conducted at Bombardier Aerospace in Montreal, Canada. It will be submitted for my Master Thesis at the Institute of Technology, Linköping University, Sweden.

I would like to express my gratitude to the Advanced Design department at Bombardier Aerospace, especially Pat Piperni, Jean-Francois Viau, Ryan Henderson and Guenther Goritschnig who gave me guidance and support during the project.

I also would like to thank my academic supervisor Petter Krus at the Division of Fluid and Mechatronic Systems, Department of Management and Engineering, Linköping University.

October 2010
Linköping, Sweden
Hannes Wemming

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Nomenclature

Abbreviations and symbols

A_i	Area at station *i*
BPR	Bypass Ratio, $BPR = W_{13}/W_{21}$
D	Diameter
DLL	Dynamic Link Library
DPP	Pressure Loss
ECS	Environmental Control System
EIS	Entry Into Service
EPR	Engine Pressure Ratio, $EPR = P_5/P_2$
FHV	Fuel Heating Value
F_N	Net thrust
FPR	Fan Pressure Ratio, $FPR = P_{13}/P_2$
F_{RAM}	Ram Drag, $F_{RAM} = W_2 \cdot V_a$
GTF	Geared TurboFan
GUI	Graphical User Interface
HP	High Pressure
HPC	High Pressure Compressor
HPT	High Pressure Turbine
IPC	Intermediate Pressure Compressor
ISA	International Standard Atmosphere
LP	Low Pressure
LPC	Low Pressure Compressor
LPT	Low Pressure Turbine
MDO	Multidisciplinary Design Optimization
NGV	Nozzle Guide Vane
N	Rotor speed
N_H	High pressure spool speed
N_L	Low pressure spool speed
OGV	Outlet Guide Vane
OPR	Overall Pressure Ratio, $OPR = P_3/P_2$
P_i	Total pressure at station *i*
$P_{s,i}$	Static pressure at station *i*
PLA	Power Lever Angle
PR	Pressure Ratio
Q	Same as FHV
R	Specific gas constant
SFC	Specific Fuel Consumption
SL	Sea Level
SLS	Sea Level Static
SOT	Stator Outlet Temperature
T_i	Temperature at station *i*
TCDS	Type Certificate Data Sheet

TO	Take Off
TOC	Top Of Climb
V_i	Velocity at station *i*
V_a	Aircraft velocity
VAFN	Variable Area Fan Nozzle
W_F	Fuel mass flow
W_i	Mass flow at station *i*
γ	Heat capacity ratio
η	Efficiency
η_b	Burner Efficiency
η_O	Overall Efficiency
ρ	Density

Engine station definition

Two spool mixed flow turbofan

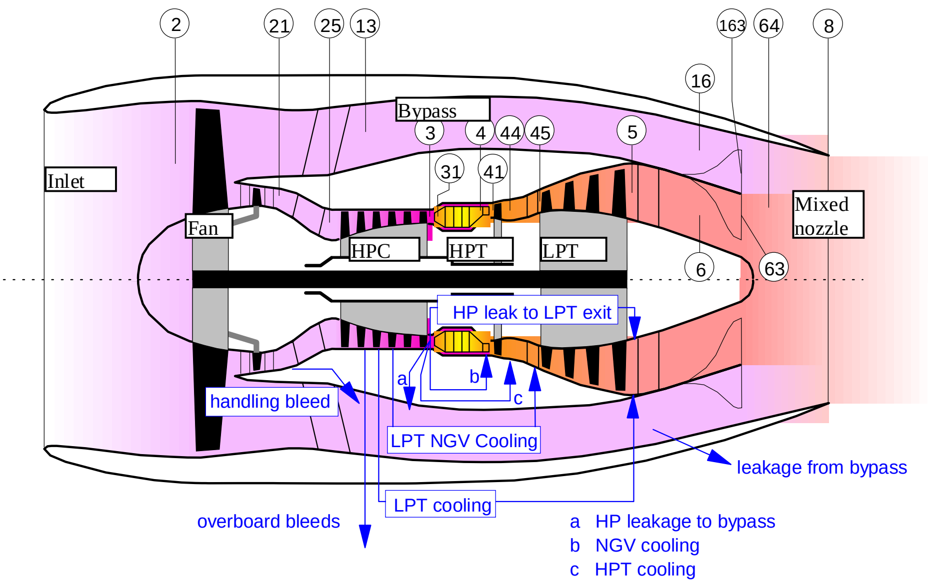

Figure 0.1 Engine stations, two spool mixed flow turbofan engine [1].

0	Ambient
1	Aircraft-engine interface
2	Fan inlet
13	Outer stream fan exit
16	Bypass exit
163	Cold side mixing plane
21	Inner stream fan exit
25	HPC inlet
3	HPC exit
31	Combustion chamber inlet
4	Combustion chamber exit

41 HPT NGV exit, identical to HPT rotor inlet
44 HPT exit after addition of cooling air
45 LPT inlet
5 LPT exit after addition of cooling air
6 Jet pipe inlet
63 Hot side mixing plane
64 Mixer exit
8 Nozzle throat

Two spool unmixed flow turbofan

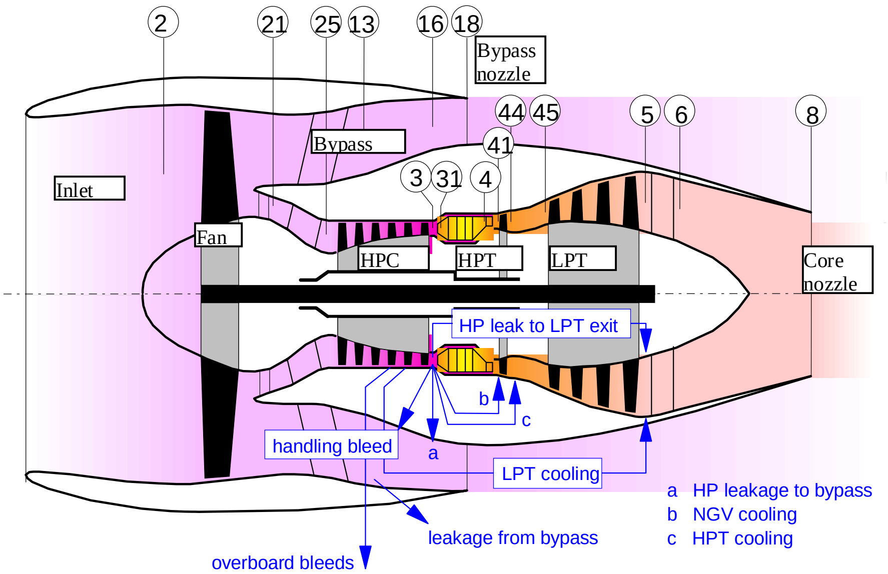

Figure 0.2 Engine stations, two spool unmixed flow turbofan engine [1].

0 Ambient
1 Aircraft-engine interface
2 Fan inlet
13 Outer stream fan exit
16 Bypass exit
18 Bypass nozzle throat
21 Inner stream fan exit
25 HPC inlet
3 HPC exit
31 Combustion chamber inlet
4 Combustion chamber exit
41 HPT NGV exit, identical to HPT rotor inlet
44 HPT exit after addition of cooling air
45 LPT inlet
5 LPT exit after addition of cooling air
6 Jet pipe inlet
8 Nozzle throat

Geared two spool unmixed flow turbofan with booster

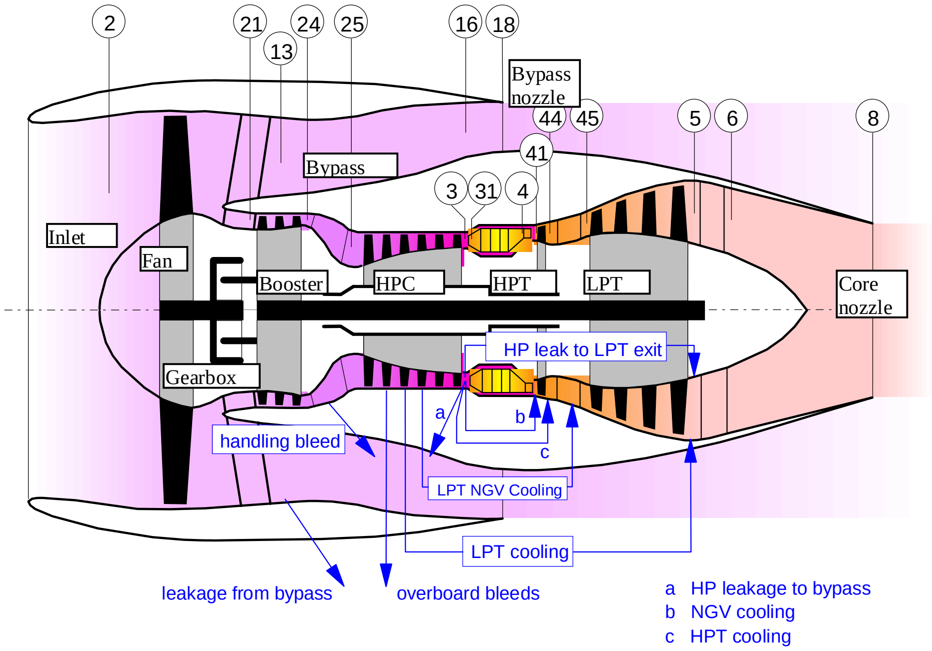

Figure 0.3 Engine stations, geared two spool unmixed flow turbofan engine with booster [1].

0	Ambient
1	Aircraft-engine interface
2	Fan inlet
13	Outer stream fan exit
16	Bypass exit
18	Bypass nozzle throat
21	Inner stream fan exit
24	Booster exit
25	HPC inlet
3	HPC exit
31	Combustion chamber inlet
4	Combustion chamber exit
41	HPT NGV exit, identical to HPT rotor inlet
44	HPT exit after addition of cooling air
45	LPT inlet
5	LPT exit after addition of cooling air
6	Jet pipe inlet
8	Nozzle throat

Geared two spool mixed flow turbofan with booster

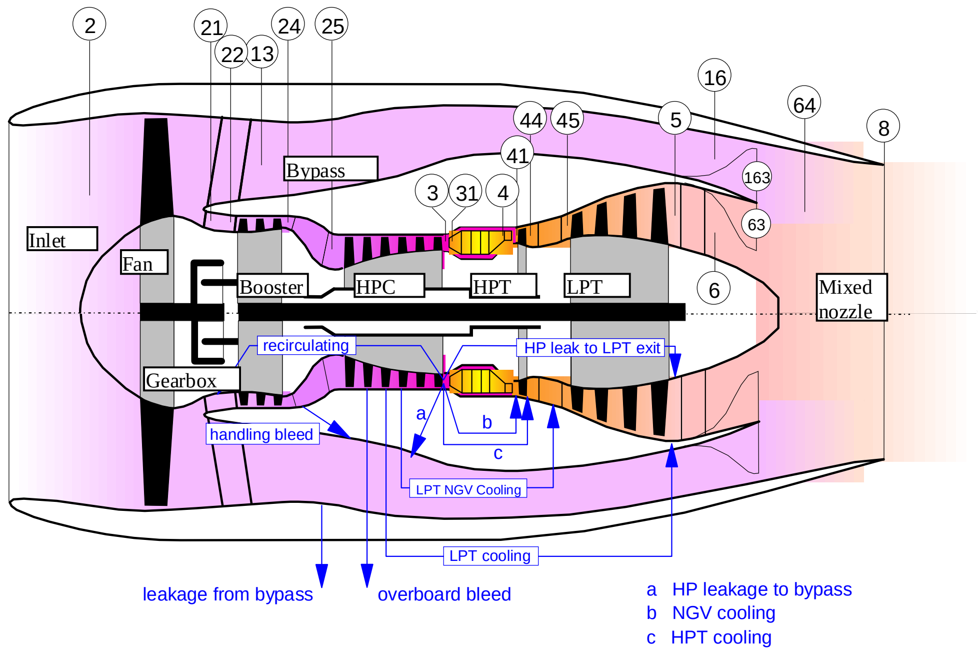

Figure 0.4 Engine stations, geared two spool mixed flow turbofan engine with booster [1].

0	Ambient
1	Aircraft-engine interface
2	Fan inlet
13	Outer stream fan exit
16	Bypass exit
163	Cold side mixing plane
21	Inner stream fan exit
22	Booster inlet
24	Booster exit
25	HPC inlet
3	HPC exit
31	Combustion chamber inlet
4	Combustion chamber exit
41	HPT NGV exit, identical to HPT rotor inlet
44	HPT exit after addition of cooling air
45	LPT inlet
5	LPT exit after addition of cooling air
6	Jet pipe inlet
63	Hot side mixing plane
64	Mixer exit
8	Nozzle throat

1 Introduction

Advanced simulation techniques have become increasingly important in the field of aerospace engineering for aircraft design. This thesis work deals with incorporating an existing commercial jet engine performance modeling software into a multidisciplinary design optimization software framework under development at Bombardier Aerospace.

1.1 Background

Multidisciplinary design optimization is used more and more for predicting aircraft performance in conceptual design of aircraft. In an MDO software environment, multiple disciplines, e.g. weight estimation, aerodynamics, engine performance, stability and control, are incorporated and optimized simultaneously. The advantage of this is that it captures the interactions between the disciplines and has the possibility of finding an optimum that is superior to the optimum found if each discipline is optimized separately.

Engine performance for a big range of engine operating conditions is vital for predicting aircraft performance. Traditionally, tables of engine performance called thrust tables were provided by engine manufacturers. The thrust tables contain thrust, fuel flow and other engine parameters for different flight conditions and engine thrust settings.

When an aircraft concept with a new engine is studied, the engine manufacturers need to be contacted in order to get engine performance data. In order to avoid this, there is a need for a generic engine model, called “rubber engine”, that can produce the performance parameters normally found in thrust tables.

1.2 Goal

The goal of this thesis is firstly to validate the capabilities of GasTurb, a commercial engine performance modeling software, concerning the prediction of engine performance by comparing it to engine performance data for existing engines. Secondly, investigate the possibility of using GasTurb as a rubber engine integrated into a multidisciplinary design optimization software framework.

1.3 Methodology

The first phase of the work consisted of reviewing gas turbine theory and application to aircraft propulsion as well as familiarization with GasTurb, the selected software for gas turbine performance modeling.

A strategy for generating the necessary output from GasTurb and formatting the output into the thrust table file format was identified. GasTurb can be run in batch mode where many points can be calculated in sequence with output to an Excel file. The data in the Excel file is formatted and exported to the thrust table file format with the help of Excel macros.

The next step was to validate the software against a number of turbofan engines. The turbofan engine type was selected because this is the primary selection for propulsion of civil aircraft. The selected engines were of different size, different type and different age in an attempt to validate GasTurb for a big range of engines useful for business jets, regional jets and commercial aircraft, see Table 1.1. Due to the sensitive nature of the information, the names of the investigated engines are not shown in this report. Because Advanced Design does conceptual design using new technologies, several of the investigated engines were proposed engine concepts and demonstrators that were not certified at the time of writing. The validation consisted of comparing the main performance parameters SFC and net thrust from the GasTurb model to manufacturer's data.

Table 1.1 Table of turbofan engines selected for validation of GasTurb.

Engine	Description	Take off thrust class	Aircraft Type
Engine 1 (E1)	2 spool mixed flow turbofan	< 10 klbf	Mid-size Business Jet
Engine Concept 1 (EC1)	2 spool mixed flow turbofan	~10 klbf	Mid-size Business Jet or small Regional Jet
Engine 2 (E2)	2 spool unmixed flow turbofan	~15 klbf	Regional Jet
Engine 3 (E3)	2 spool mixed flow turbofan	~15 klbf	Large Business Jet
Engine 3A (E3A)	Derivative of Engine 3	15-17 klbf	Large Business Jet
Engine 4 (E4)	2 spool geared unmixed flow turbofan	~25 klbf	Commercial Aircraft

A method of estimating engine parameters in order to create a GasTurb engine model was investigated. All GasTurb input parameters for a two spool mixed flow turbofan were listed and a sensitivity study performed.

Finally, GasTurb was run without the GUI in preparation for integrating it with the MDO framework. It was linked to Isight, the optimization software. The optimization software will optimize the objective function subjected to constraints on the design variables. The work consisted of defining the interface between GasTurb and the optimizer software and performing calculations to ensure that the same result is obtained when the engine model is run within the MDO framework. Figure 1.1 shows an overview of the MDO framework including the engine model.

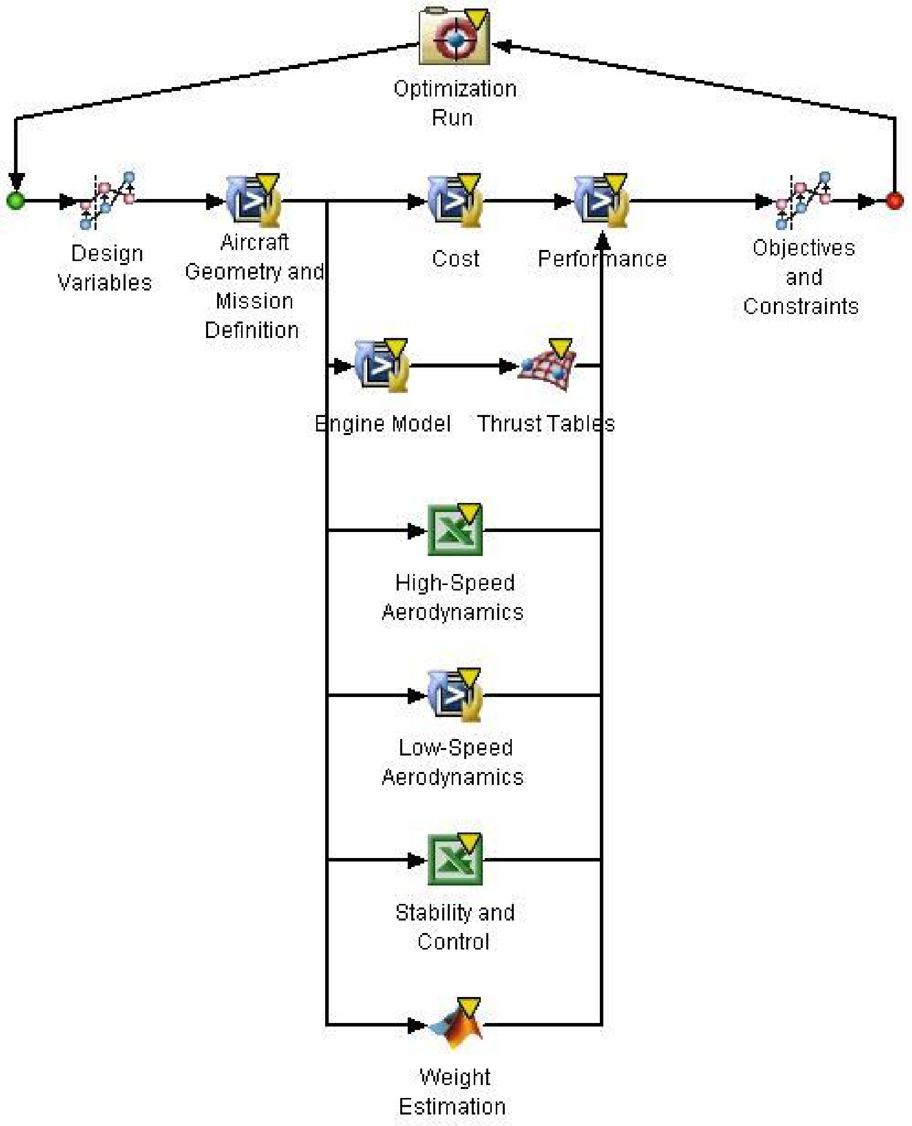

Figure 1.1 The different disciplines of the MDO software framework.

1.4 The company and department

The work was performed at the Advanced Design department at Bombardier Aerospace in Montreal, Quebec, Canada. Bombardier Aerospace is the third largest aircraft company in the world, designing and manufacturing business jets, commercial jets, commercial turboprops and special mission aircraft.

Advanced Design's mission is to lead the development of aircraft design concepts which enable the product portfolio strategy. This is achieved by performing conceptual design studies of new, derivative and special mission aircraft as well as investigating innovative designs and strategic technologies among other things.

2 Engine performance modeling

In this chapter, a strategy for creating a generic two spool turbofan engine model is presented. It is based on the assumption that the user has information about important cycle parameters and component characteristics. Some background theory is presented and explained, but for a more thorough understanding of the thermodynamic principles of a gas turbine the reader is referred to thermodynamic textbooks such as *Fundamentals of Thermal-Fluid Sciences* [2] or aircraft propulsion textbooks such as *Aircraft Engine Design* [3].

2.1 Thrust and SFC of a jet engine

In order to understand the following discussion, it is appropriate to introduce the concept of thrust and SFC of a jet engine that are the two main performance parameters considered in this work.

2.1.1 Thrust

The principle of creating a propulsive thrust is that air is accelerated. This follows from Newton's laws of motion that states that for every force acting on a body there is an opposite and equal reaction. Thrust can be generated from accelerating a lot of air a small amount, such as in a propeller/engine combination, or accelerating a small amount of air to a high velocity, such as in a turbojet engine.

Assuming that the jet is ideally expanded, Equation 2.1 is valid for a single nozzle jet engine.

$$F_N = W_8 V_8 - W_0 V_a \qquad \textbf{Equation 2.1}$$

If the jet is not fully expanded in the nozzle, there is an additional pressure term that generates thrust. This comes from the difference in static pressure between ambient and nozzle exit. For a single nozzle engine with the pressure term, Equation 2.2 is valid [3]. For a two nozzle engine, there will be one velocity term and one pressure term for each nozzle.

$$F_N = W_8 V_8 - W_0 V_a + A_8 \left(P_{s,8} - P_{s,amb} \right) \qquad \textbf{Equation 2.2}$$

The term specific thrust is defined as thrust divided by mass flow. This figure is important because it is closely related the mean jet speed, which determines the propulsive efficiency and jet noise. For a specified value of thrust it also determines the engine physical size [4].

2.1.2 SFC

The term specific fuel consumption, SFC, describes the fuel flow per unit of thrust, see Equation 2.3.

$$SFC = \frac{W_f}{F_N}$$ **Equation 2.3**

The overall efficiency η_O of the engine is defined as the useful work divided with the input energy from the fuel, Equation 2.4. It is seen that the overall efficiency is inversely proportional to the SFC for a constant FHV and aircraft velocity. Hence SFC is a measure of the overall efficiency and the mostly used means of comparing fuel efficiency between different engines.

Note that a decrease in SFC means reduced fuel consumption. A reduction is SFC is good, therefore the term improvement is used to describe a reduction in SFC.

$$\eta_O = \frac{F_N \cdot V_a}{W_f \cdot FHV} = \frac{V_a}{SFC \cdot FHV}$$ **Equation 2.4**

2.1.3 Non-dimensional and corrected quantities

The reason for using non-dimensional quantities is that the performance of different engine components can be described compactly and usefully. The non-dimensional quantities are often rewritten on a more compact form that applies for a given component with specified geometry and is related to a reference pressure P_{ref} = 101.325 kPa and reference temperature T_{ref} = 288 K. These quantities are called corrected and allow the results from a test to be related to other conditions of temperature and pressure. For example the performance of a compressor can be expressed on the form PR = *f* (corrected flow, corrected speed). The definition of non-dimensional and corrected flow and speed quantities are seen in Equation 2.5, Equation 2.6, Equation 2.7 and Equation 2.8 [5].

Non-dimensional mass flow $$\frac{W\sqrt{RT}}{PA\sqrt{\gamma}}$$ **Equation 2.5**

Non-dimensional rotor speed $$\frac{ND}{\sqrt{\gamma RT}}$$ **Equation 2.6**

Corrected mass flow $$\frac{W\sqrt{\frac{T}{T_{ref}}}}{\frac{P}{P_{ref}}}$$ **Equation 2.7**

Corrected rotor speed $$\frac{N}{\sqrt{\frac{T}{T_{ref}}}}$$ **Equation 2.8**

2.2 Two spool turbofan

The two spool turbofan is the prime choice of propulsion for civil aircraft. The main components of an engine of this type are inlet, fan, HPC, combustion chamber, HPT, LPT and nozzle.

The engine core consists of the HPC, combustion chamber and HPT. The core is used to drive the LP system with the fan and LPT in addition to producing thrust through the core nozzle. In a modern turbofan with high BPR, most of the thrust comes from the air going through the bypass duct.

The fan compresses the air from the intake and gives air mass flow through the bypass duct and to the engine core. The HPC compresses the portion of the air going through the core. The compressed air from the HPC exit is mixed with fuel and ignited in the combustion chamber. The combustion process is continuous and produces hot exhaust gases. The exhaust gases are expanded through the HPT and the extracted work is used to drive the HPC via the high speed shaft. The exhaust gases are expanded further in the LPT which drives the fan through the low speed shaft. Finally, the gases are expanded through the nozzle to produce thrust.

2.2.1 Mixed flow turbofan

In a mixed flow engine, the bypass air stream is internally mixed with the core air stream in a device called mixed and is expanded in a common nozzle. Figure 0.1 shows a two spool mixed flow engine. This engine type improves the SFC compared to an unmixed flow engine. The major drawback is the weight penalty from a larger engine nacelle.

2.2.2 Unmixed flow turbofan

In an unmixed flow engine, the bypass stream is not mixed with the core flow and there are two separate nozzles, the bypass nozzle and the core nozzle. Figure 0.2 shows an unmixed flow two spool turbofan engine.

2.2.3 Booster/IPC

In order to increase the OPR of an engine, a booster stage can be used. The booster stage is mounted on the low spool after the fan and before the HPC inlet. Sometimes the booster is referred to as IPC.

2.2.4 Geared turbofan

In a geared turbofan, the fan is connected to the LP spool through a gearbox. This has the advantage of decoupling the fan from the rest of the low spool turbomachinery, making it possible to run the fan at a slow speed and the booster and LPT at a high speed. A geared turbofan can be either mixed or unmixed flow. An unmixed flow two spool GTF with a booster is seen in Figure 0.3.

In order to increase the propulsive efficiency and improve the SFC of the engine, an engine with low specific thrust is desirable. This translates into high BPR and low FPR

[4]. In order to reach a specified level of thrust at high values of BPR, the fan diameter needs to be big and subsequently, the fan rotational speed has to be low in order to avoid too high tip speeds with the associated transonic losses and efficiency loss. High rotational speed of the LPT allows for higher component efficiency levels and larger work extraction per stage. This allows for fewer LPT stages and a considerable reduction in aerofoil sections and overall parts count, decreasing mass and cost [6]. These are the main reasons for connecting the LPT to the fan through a gearbox.

2.2.5 Installation losses

The different definitions of engine thrust are important. Equation 2.2 gives what is called the uninstalled thrust. When considering the whole propulsion system and the interactions between engine, nacelle and aircraft, a need for a more detailed definition of thrust arises.

The term net thrust is used to describe the summation of forces acting on the internal surfaces of the nacelle and engine, including forces exerted by the internal airflow on core cowling and nozzle plugs. The forces on the core cowling and nozzle plugs is called scrubbing drag, illustrated in Figure 2.1. The summation of forces acting on the outside of the engine nacelle is accounted for as drag and is not included in the engine model.

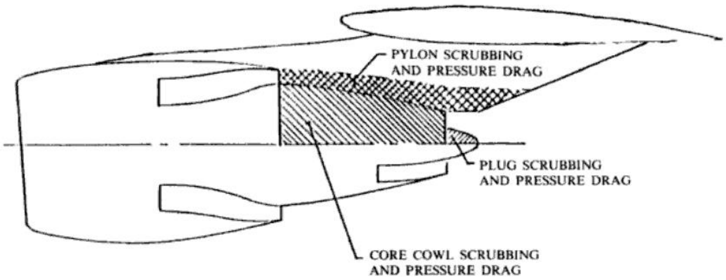

Figure 2.1 Scrubbing drag [7].

When comparing an engine in a testbed with an installed engine on an aircraft, there are a few differences in engine operation and behavior. The installation losses considered in this thesis are intake pressure loss, compressor bleed air, fan bleed air and power offtake. There are other installation losses, for example inlet flow distortion, that was neglected because their influence is small and it was deemed to be excessively detailed with too many hard to find input parameters for a conceptual design tool. The scrubbing drag is not modeled directly in GasTurb but is taken into account indirectly by the calibration of the generic engine model to available data. Also, the scrubbing drag only applies to unmixed flow engines and most of the investigated engines were of the mixed flow type.

The engine air intake is designed to have a pressure recovery close to one at design conditions. At off-design such as take off with low Mach numbers and high air mass flow rate through the inlet, the intake pressure recovery will be lower than one, which means that there is a pressure loss through the air intake.

Power is taken from the engine through a gearbox to drive accessory components such as generator and hydraulic pump. This is called the power offtake.

Compressor bleed air is typically used for wing and engine nacelle anti-icing and in the aircraft ECS for pressurization of the cabin and temperature control. There are often two compressor bleed ports at different locations in the HPC to adapt the compressor delivery pressure and temperature to the bleed air system need at different engine and aircraft operating conditions.

On some engine models, fan bleed air is used in the bleed air system. The fan bleed air goes through a heat exchanger (called precooler) where it cools the compressor bleed air to meet the bleed air system requirements.

The term installed net thrust is the net thrust the engine produces when all installation losses are considered. This is what is given from engine deck software.

2.2.6 Flight envelope

The engine flight envelope describes the combinations of altitude, Mach number and ambient temperature where the engine is certified to operate. The engine flight envelope must be bigger than the aircraft envelope in order to guarantee thrust throughout the whole aircraft flight envelope. The flight envelope of a commercial jet engine is seen in Figure 2.2.

2.2.7 Engine rating structure

The thrust of a modern engine is governed by a rating structure. The rating structure translates a power lever angle (PLA) setting in the aircraft cockpit to a certain level of thrust for a given ambient condition. For contemporary engines, this is attained by an electronic engine control system that is controlling engine parameters according to PLA, ambient conditions and thrust management tables.

2.2.7.1 Maximum Take Off (MTO)

The MTO rating is the maximum net thrust the engine can produce, used for take off, discontinued approach and baulked landing without exceeding permitted engine parameters values. The limiting engine parameters are normally one of the engine spool speeds, compressor delivery pressure or burner exit temperature. The MTO rating is only available in a part of the flight envelope consistent with take off conditions and the use of this rating is time limited.

The MTO rating is associated with the aircraft certification requirements as well as operational requirements. However, in reality the take offs are often done using a lower thrust than the MTO rating gives in order to reduce engine wear (increase engine life) and reduce noise.

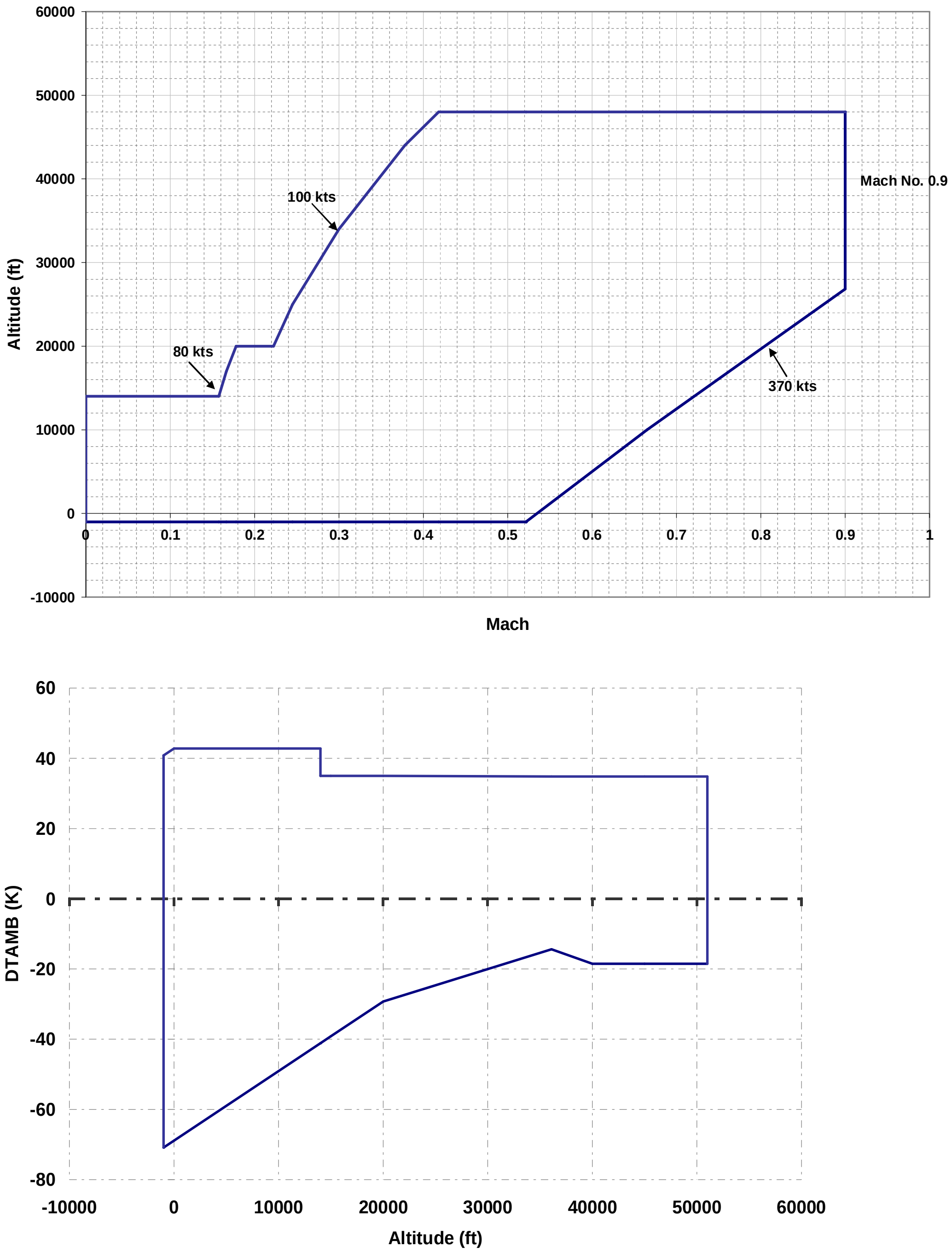

Figure 2.2 Flight envelope of a commercial jet engine. Source EC1 engine deck documentation.

Engine flat rating is an important concept for the MTO rating. Ideally, one would like to have a constant amount of thrust independent of ambient temperature. However, as ambient temperature increases, the temperatures in the hot section of the engine increase. To avoid damage to the engine the maximum temperatures must be limited because of material limitations. This means that at high ambient temperatures, the MTO rating thrust drops off. This can be seen in Figure 2.3 where Engine 1 engine deck F_N, ITT and P_3 is plotted as function of ambient temperature. The consequence of the reduced thrust is that the aircraft might be forced to leave payload behind in order to take off safely at hot days. The ambient temperature where the thrust starts to decrease is called the flat rating point or the kink point and is given in relation to ISA. An engine is typically flat rated to ISA+15K, which is equivalent to the temperature 30ºC.

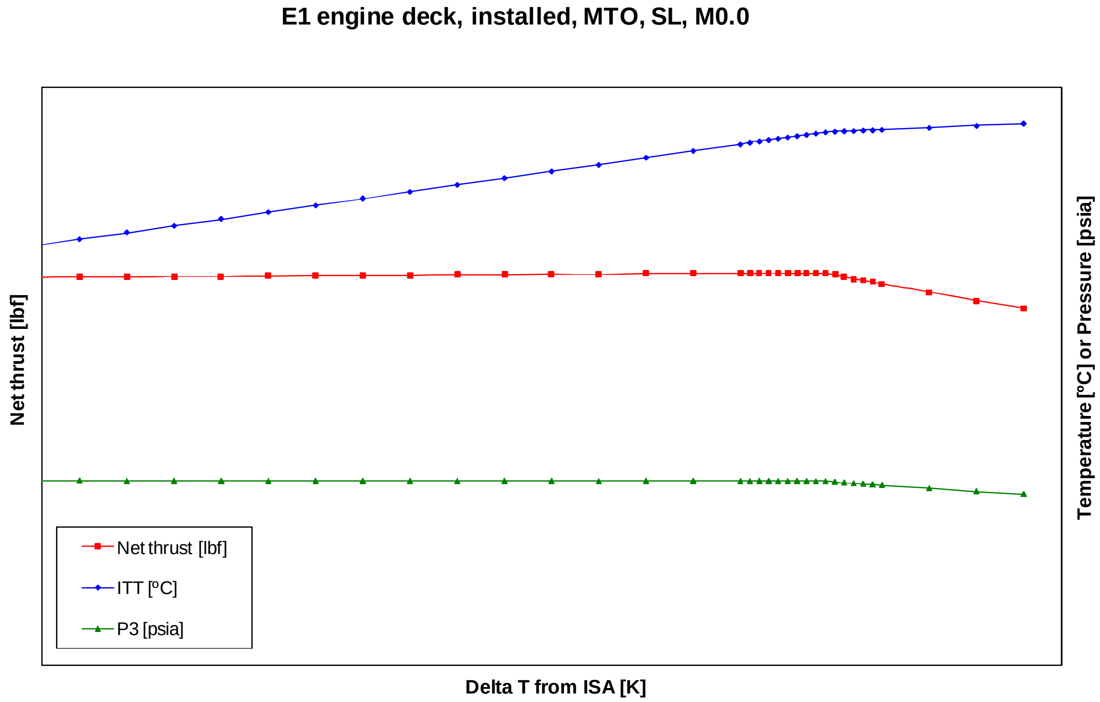

Figure 2.3 Engine 1 engine deck F_N, ITT and P_3 plotted against ambient temperature.

2.2.7.2 Maximum continuous (MCT)

The MCT rating is the maximum rating approved for unrestricted periods of use. It is intended for use in extreme circumstances such as engine failure on multi engine aircraft. It is also the rating that provides the highest amount of thrust outside of the TO envelope. This rating is associated with the aircraft certification requirements.

2.2.7.3 Maximum Climb (MCL)

The MCL rating is an operational rating that gives the maximum net thrust for use in climb.

2.2.7.4 Maximum Cruise (MCR)

The MCR rating is an operational rating that gives the maximum net thrust for use in cruise.

2.2.7.5 Flight Idle

The flight idle rating is the minimum thrust produced by the engine in flight conditions. It is determined by compressor and combustor stability as well as engine spool up time for missed approach conditions.

2.2.8 Engine control system

"Control systems must be designed to prevent aircraft engines from destroying themselves" [3]. The main functions of an engine control system are to produce the commanded amount of thrust in a repeatable way for both transient and steady operating conditions while maintaining stable engine operation within safe mechanical operating limits over the whole flight envelope. A modern jet engine has an electronic engine control system that is controlling engine parameters according to PLA, ambient conditions and thrust management tables. Because of the complexity of an electronic engine control system, it is difficult to model the thrust rating structure of a modern engine.

Going back to Equation 2.2, the net thrust consists of a difference in momentum and a difference in pressure. These properties are difficult to measure in flight and it is therefore not practical to use the engine net thrust directly in the control system. Instead, the thrust has to be related to another engine parameter that is measurable [3]. The most practical measure of the engine thrust is to relate it to the operating line in a fan map. Both rotational speed and pressure ratios are readily measured in the fan which has benign environmental conditions compared to the hot parts further back in the engine. The corrected fan speed correlates very well to engine mass flow and is therefore a useful parameter for controlling the thrust. A typical control system for a commercial engine attains the thrust ratings by controlling the corrected fan speed as function of ambient temperature, altitude and Mach number, with corrections applied to take the bleed air and power offtakes into account.

2.3 Engine data

In order to make reasonable engine performance prediction models, data about the engine parameters such as OPR, BPR, FPR, SOT, component efficiencies, pressure losses, internal cooling flows and nozzle coefficients are needed. In general, the more information available, the better the engine model.

In order to calibrate an engine simulation model and make comparisons of the predicted performance, reliable data for the engine performance is needed.

2.3.1 Openly available sources

The main openly available sources are engine manufacturer public reports and homepages, publications like *Jane's All the World's Aircraft* [8] and engine certification documents, e.g. *EASA TCDS* [9], *FAA TCDS* [10], *ICAO Engine Emissions Databank* [11]. The drawback is that there is limited information available and that only certified engines are in the databases which make it hard to find information on new engines.

2.3.2 Proprietary information from Bombardier and engine manufacturers

In the role as an intern at Advanced Design at Bombardier, there was proprietary information from engine manufacturers available. The information comprised of thrust tables, technical reports, presentations, engine manuals, engine deck software, engine deck software manuals and the knowledge of the experienced engineers at Bombardier.

An engine deck is an analytic steady state engine performance prediction software that Bombardier can obtain from engine manufacturers when a new or improved aircraft design is proposed. The engine deck provides the engine performance in the engine operating envelope.

The engine deck output variables vary from engine to engine, but in general, the engine manufacturers tend to hide the design parameters of their engines, especially burner exit temperature. Some engine decks output details like cooling flows, component efficiencies and nozzle coefficients but most engine decks hide those parameters for the user.

The engine decks are used for creating thrust tables. A thrust table contain the most important performance parameters i.e. fuel flow, net thrust, ram drag and T_{44} for different altitudes, Mach numbers, ISA values and engine thrust ratings.

The preferred source of information is an engine deck because the user can specify the flight conditions and installation losses. For some engines, there were no engine decks available and the information in the thrust tables was used. This can cause problems since the thrust tables do not always specify the assumptions regarding humidity, FHV and installation losses.

2.4 Design point

The engine design point is where all of the engine components are matched at their design condition and perform at their design pressure ratio, efficiency and flow. For subsonic transport type aircraft the design point is typically at TOC or TO [12]. All other conditions are called off-design. At off-design the component pressure ratio, efficiency and flow are different from the design point values.

The real engine design point is not known in general, therefore an assumed design point have to be guessed in order to make a model of a particular engine. The choice of assumed design point is dictated by the availability and quality of data about engine parameters.

To make an engine model, the model parameters should be matched to the known engine parameters at the assumed design point. The main parameters that describe a turbofan are OPR, burner exit temperature, component performance, FPR, engine total mass flow and BPR.

The OPR, the burner exit temperature and component efficiencies define the thermodynamic cycle and the thermodynamic efficiency of the engine core. The total engine mass flow gives the physical size of the engine together with the BPR which defines the relative size of the engine core to the overall engine. The BPR and FPR are important for the engine propulsive efficiency. In addition to the component efficiencies, there are other component performance parameters such as pressure losses, nozzle coefficients and shaft mechanical efficiencies. Also cooling flows and installation losses have to be defined in order to get a complete engine model.

The internal engine cooling is very important for the SFC. Unfortunately, it is very hard to find reliable data about the internal cooling flows. If better values not are available, a cooling flow model derived by *Shakariyants* [13] based on statistical data was used. This model gives the cooling flow as a function of the SOT and employs a technology level factor t and a model constant k. The model is seen in Equation 2.9 and Equation 2.10 and it is visualized in Figure 2.4.

$$\frac{W_{Cooling}^{NGV}}{W_{25}} \cdot 100 = t_{NGV} \cdot T_{SOT} + k_{NGV} \qquad \textbf{Equation 2.9}$$

$$t_{NGV} \in [1/60, 1/55];\ k_{NGV} = -20$$

$$\frac{W_{Cooling}^{blade}}{W_{25}} \cdot 100 = k_{blade} \cdot T_{SOT} + t_{blade} \qquad \textbf{Equation 2.10}$$

$$t_{blade} \in [-300/7, -324/7];\ k_{blade} = 6/175$$

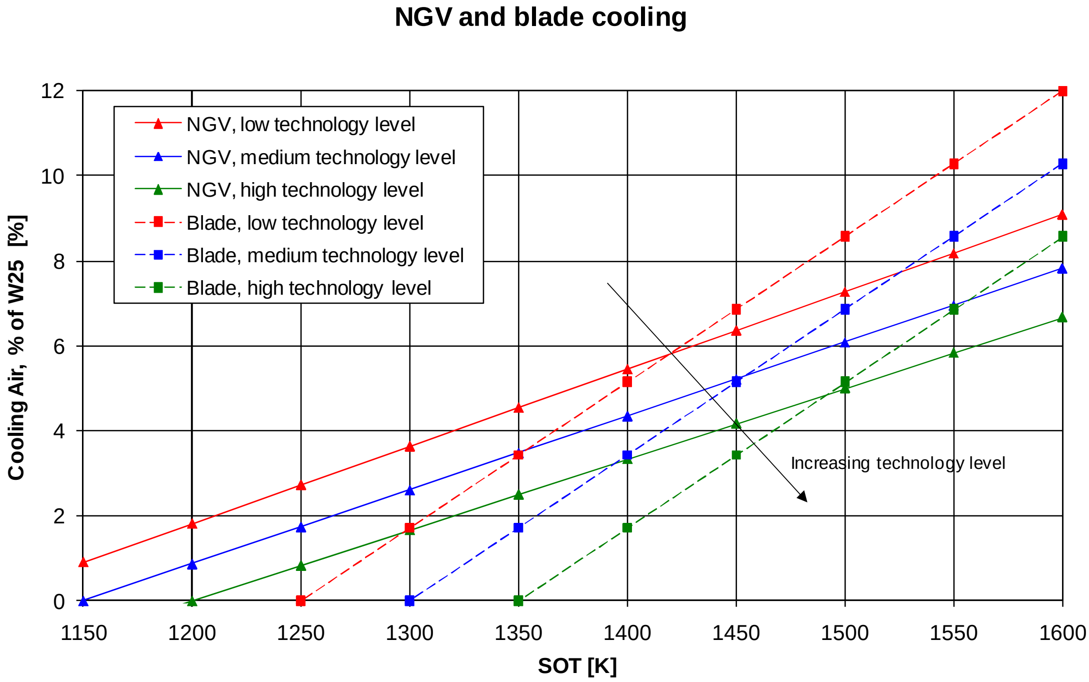

Figure 2.4 NGV and blade cooling as a function of SOT and technology level.

2.4.1 Calibration

If the engine model parameter values are unknown they must be approximated within reasonable limits decided by statistical data for other engines and engineering judgment. If the important engine model output variables not are in agreement with the available engine data for the assumed design point, the approximated parameters should be adjusted to obtain a reasonable engine model. The parameters considered important are F_N, F_{RAM} and W_F. Due to the definition of SFC and F_{RAM}, this ensures that the model predicts the correct SFC and inlet air mass flow. This process is effectively a calibration of the engine model to available data at the assumed design point and reduces the uncertainty in the unknown input parameters.

2.5 Off-design calculation

All conditions other than the design point are called off-design. At off-design the component pressure ratio, efficiency and flow are different from the design point values.

2.5.1 Compressor maps

A component map describes the characteristics in terms of mass flow, pressure ratio, efficiency and limits of stable operation of a component over its operating range of corrected speeds. Component maps are used at off-design conditions where the component performance is different from its design point value. GasTurb has a set of default component maps for compressors and turbines, but in order to predict the performance of a given engine accurately, correct component maps are needed. It is very difficult to obtain component maps as they are engine manufacturer proprietary information.

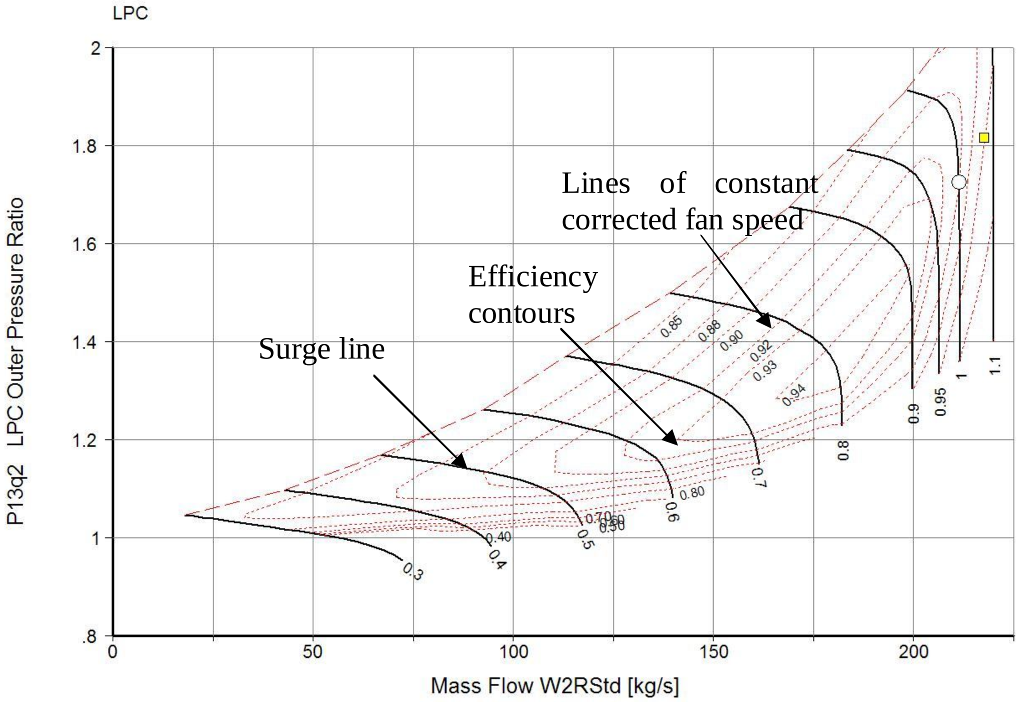

Figure 2.5 The default GasTurb fan map with scaling factors applied [1].

An example of a fan map is seen in Figure 2.5. The map shows pressure ratio as a function of corrected mass flow for lines of constant corrected speed. Each point on a specific corrected speed line is defined by a β-value in the interval [0, 1]. β = 0 is the point with the smallest pressure ratio, the point furthest down on the speed line. The surge line seen in the figure relates to the limit of stable operation. Surge means that the local angle of attack at the compressor blades is too high and the compressor blades stall. If surge occurs, the pressure ratio drops significantly.

2.5.1.1 Map scaling factors

In order to align the generic map to the assumed design point of a particular engine model, scaling factors have to be used for the mass flow, pressure ratio and efficiency.

As an example, assume that the engine assumed design point has a corrected mass flow of 100 kg/s and the default map scaling point in the unscaled map has a corrected mass flow of 35 kg/s. This gives a mass flow scaling factor of 100/35 = 2.9 if no correction is made for Reynolds number effects. A similar procedure is done for the pressure ratio and efficiency. A Reynolds number factor can be applied to the flow and efficiency scaling factors to take into account viscous effects affecting flow and efficiency. This procedure is described in *NATO RTO-TR-044* [14] or *GasTurb user manual* [1]. The details of the Reynolds number effects on a compressor map were deemed too detailed for the scope of this work and the default GasTurb Reynolds corrections were used in all calculations.

Ideally, the map scaling factors are close to one, meaning that the component map used is similar to the component that is modeled. For example, it is not a reasonable approach to scale a single stage fan map for use as a multi stage HPC map.

2.5.1.2 Calibration of map scaling point

Because a generic map that do not reflect the real compressor characteristics is used, the map scaling point should be calibrated in order to fit the engine model to available performance data. This is done by changing the corrected mass flow – efficiency correlation and the corrected mass flow – corrected speed correlation. This procedure is described by *Shakariyants* [13] as a procedure based on the work of *Kurzke*.

2.5.1.3 Corrected mass flow – efficiency correlation

The map scaling point is the cycle design point position in the map. This is illustrated in Figure 2.5 where the circle is the cycle design point of the scaled map and the square is the original cycle design point of the unscaled map. The map scaling point affects the corrected mass flow – efficiency correlation at off-design condition calculations. If the map scaling point lies at a high corrected speed, any further increase in speed will have quickly falling component efficiency. This is the case for the default GasTurb design point which is for a high power setting, i.e. high corrected speed. If the map scaling point is moved to a lower corrected speed, in the region of higher efficiency, an increase of corrected speed will have a smaller efficiency drop off. Due to the higher component efficiencies, the SFC would be lower with the new map scaling point when the engine is run at high power settings.

This corrected mass flow – efficiency correction can be readily seen in a SFC/FN chart, see Figure 2.6.

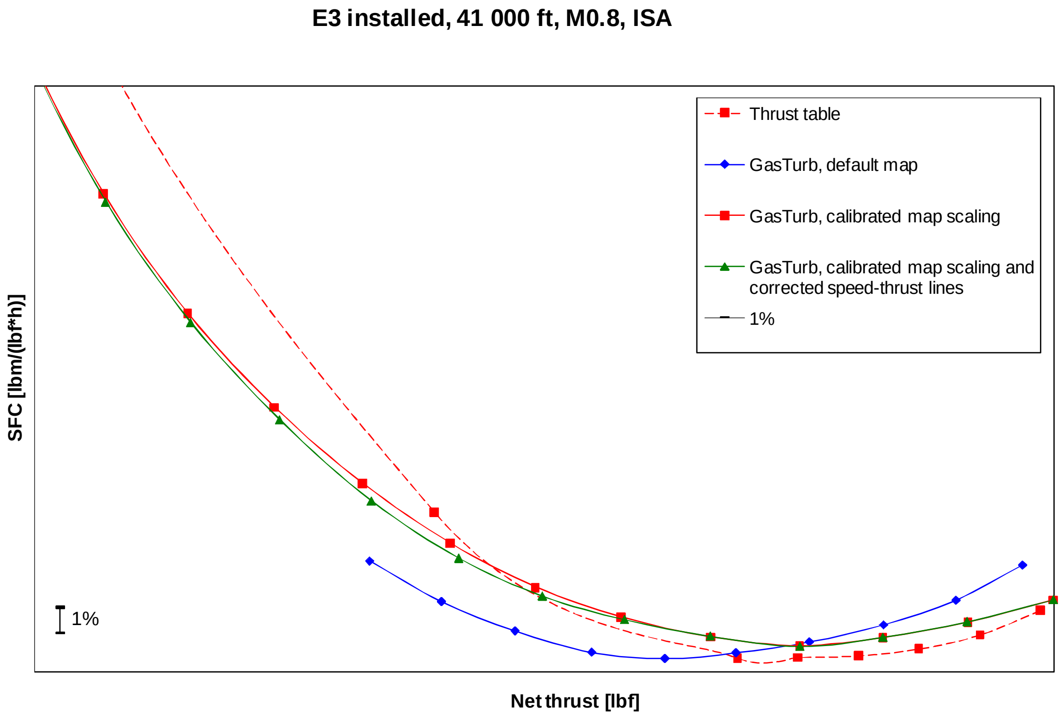

Figure 2.6 Engine 3 SFC plotted as a function of F_N. Thrust table data is compared to the default map scaling points and the calibrated, moved, map scaling points.

2.5.1.4 Corrected mass flow – corrected speed correlation

After the corrected mass flow – efficiency correlation is finished, the corrected mass flow – corrected speed correlation should be adjusted by re-labeling the speed lines in the component map. Because data on corrected mass flow is not generally available, net thrust can be used instead as it yields basically the same result. The component efficiency is not affected at a given corrected mass flow and pressure ratio. Hence the SFC/FN chart in Figure 2.6 is not affected. The effect on net thrust and corrected speed can be seen in Figure 2.7.

2.5.2 Turbine maps

Similarly as for the compressor component maps, there are turbine maps that describe the turbine characteristics. The operating range for a turbine is smaller than for a compressor and the turbine map scaling point has less impact on performance than the compressor map scaling point. Therefore, the turbine map scaling points are left at their default values.

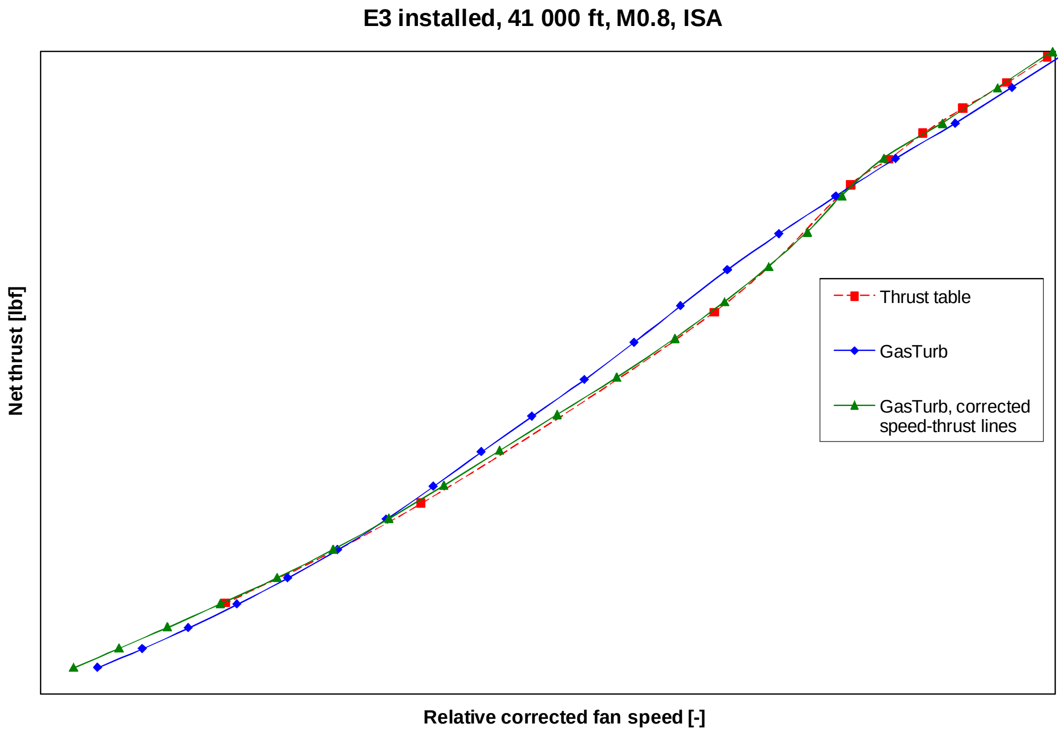

Figure 2.7 Engine 3 F_N plotted against corrected fan speed. Thrust table compared to GasTurb results showing the effect of the corrected F_N – fan speed correlation.

2.6 Engine control system limits

Intuitively, one might think that a given rating would correspond to a simple parameter, e.g. MTO is given by a high spool speed of 100% over the whole flight envelope. Unfortunately, this is not the case and in order to model the net thrust at the MTO and MCL ratings, a control system with limits on physical parameters was developed. This allows for running the GasTurb engine model by specifying a rating in a similar way as an engine deck program works.

2.6.1 Control schedule for MCL rating in whole flight envelope

Two important mechanical limits of the turbomachinery are OPR (or P_3) and T_{41}. As described in *Aircraft Engine Design* [3], at low ambient temperatures, the OPR limit will limit the thrust and at high ambient temperatures the T_{41} limit will limit the thrust. At the flat rating point, both the OPR limit and T_{41} limit will be reached simultaneously. Alternatively, as *Shakariyants* [13] describes, one could impose a limit on the maximum compressor delivery pressure P_3 and on the maximum HPT rotor inlet temperature T_{41}, derived from the MTO rating at TO at the flat rating point. In addition to this, there is a limit to the corrected fan speed in order to avoid that the compressor is operating at too high aerodynamic speeds. This limit will normally be reached at TOC. There are also other limits, such as a mechanical limit to the maximum absolute rotational speeds of the rotors and a maximum allowed temperature at the compressor exit and LPT rotor inlet.

By considering these limits as described by literature and comparing to data from engine decks, it is seen that the theory with a maximum OPR described in *Aircraft Engine Design* [3] does not apply to the two spool turbofans considered in this work. It is found that at the MTO rating, the OPR increases with increasing altitude. If a constant value of maximum OPR would be used, the predicted thrust will be wrong at some operating conditions. There is also an influence of Mach number on the OPR when considering the whole span from M0.0 to M0.9, the trend being that OPR is reduced with increasing Mach number. Figure 2.8 and Figure 2.9 show the OPR variation with altitude and Mach number for Engine 1 at MTO rating and EC1 at MCL rating. Judging from this figure, it would be necessary to schedule the maximum allowed OPR with both altitude and Mach number in order to use OPR as a thrust limiting parameter.

If a limit on maximum P_3 is considered as proposed by *Shakariyants* [13], it is found that for the investigated engines, P_3 at MTO rating will increase with Mach number. Again using Engine 1 as an example, this can be seen in Figure 2.10. To recreate the engine thrust accurately, the maximum P_3 should be scheduled with altitude and Mach number.

Looking at Figure 2.10 and Figure 2.11, it is seen that P_3 declines in a relatively smooth and linear fashion with altitude and Mach number. P_3 scheduled with altitude and Mach number was therefore chosen as the limiting parameter for the MCL control schedule. An example of a P_3 schedule for EC1 is given in Table 2.1.

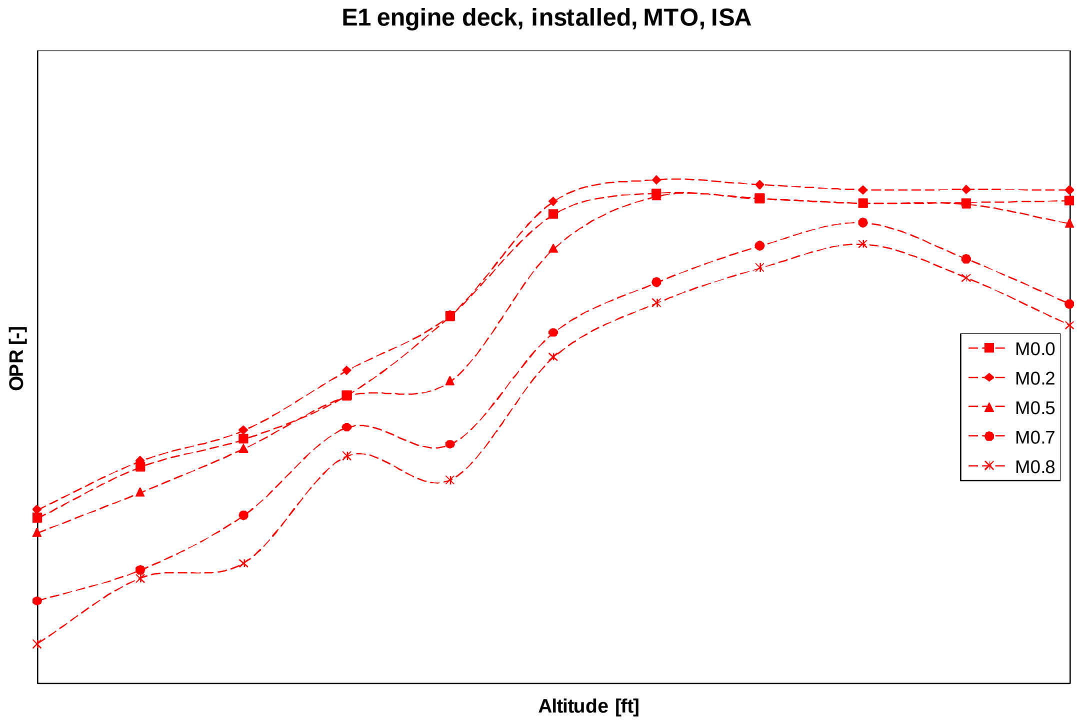

Figure 2.8 The variation of OPR with altitude and Mach number at MTO rating, E1.

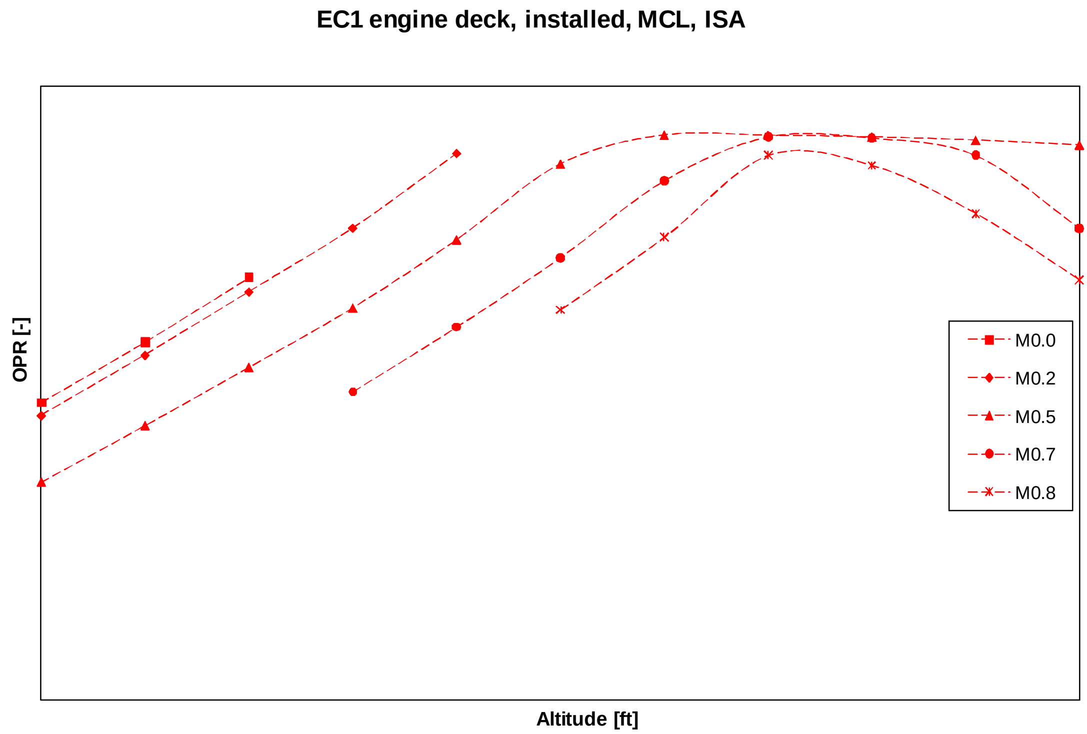

Figure 2.9 The variation of OPR with altitude and Mach number at MCL rating, EC1.

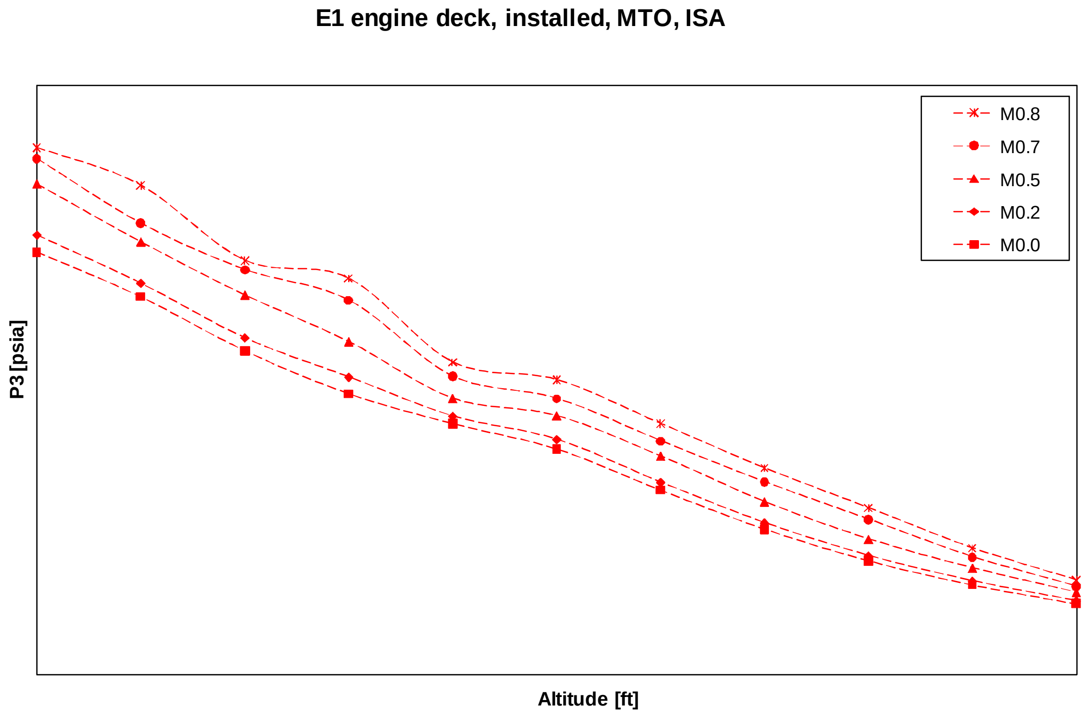

Figure 2.10 The variation of P_3 with altitude and Mach number at MTO rating, E1.

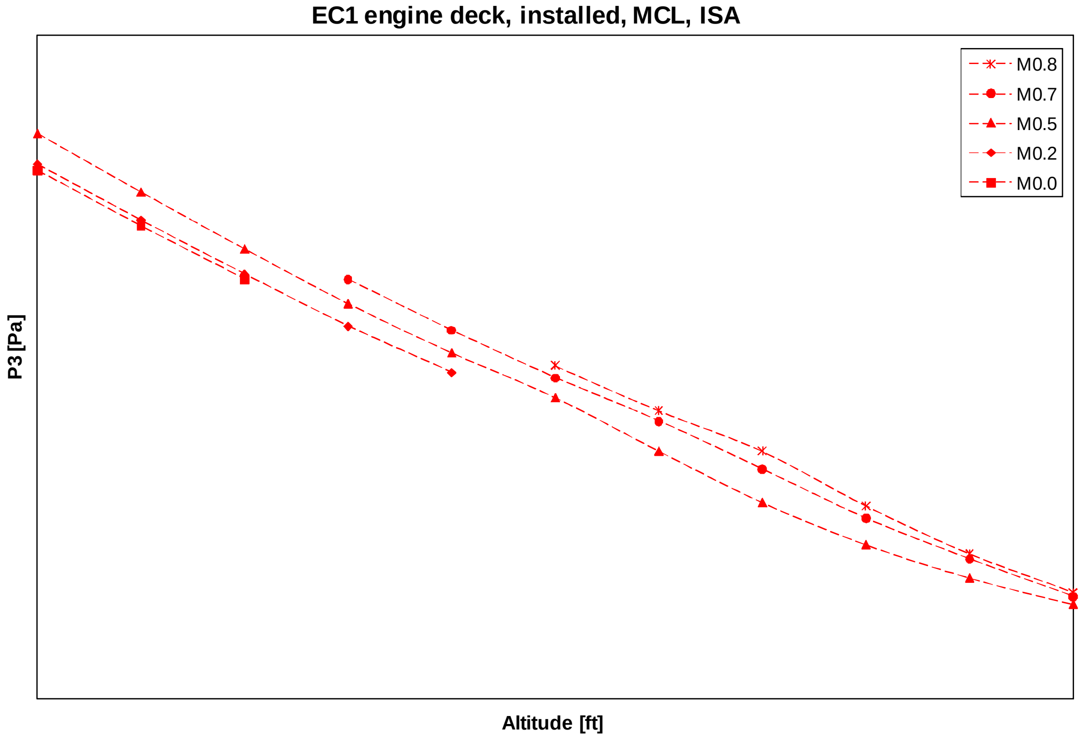

Figure 2.11 The variation of P_3 with altitude and Mach number at MCL rating, EC1.

Table 2.1 Points for deriving P_3 limits for the MCL control schedule.

DISA [K]	0	Altitude [ft]				
Max allowed P_3 [kPa]		0	10 000	20 000	30 000	50 000
Mach number [-]	0.00	P_3 increasing with Mach <=	P_3 decreasing with altitude =>			
	0.20					
	0.50					
	0.70					
	0.80					

At TOC conditions, the corrected fan speed is normally at its maximum allowed value. In addition to the P_3 schedule a limiting value of max corrected fan speed was used.

The limiting P_3 values are derived from the simulation engine model by running the engine model to the same thrust as the available data. The MCL control system is created by comparing a number of points in the flight envelope. At all other points, the limits will be linearly interpolated. It is assumed that the control system limits in the real engine are more or less linear and errors in predicted thrust will come from this assumption. This method calibrates the simulation engine model to the available data by the limits imposed by the control schedule.

The P_3 limit MCL control system does not take bleed air and power offtake into account like a real engine control system. The control system is derived at ISA for an installed engine with bleed air and power offtake but changes to the installation losses would reduce the accuracy of the net thrust prediction. Also, the control system does not take into account the flat rating of the engine at high ambient temperatures, potentially giving big errors in the predicted net thrust at high ambient temperatures.

2.6.2 Control schedule for MTO rating in TO envelope

The net thrust is of main importance at the MTO rating. In order to increase the accuracy of the net thrust prediction and catch the flat rating behavior of the engine, both P_3 and T_{41} limiting values are imposed (refer to Figure 2.3). P_3 is scheduled with altitude and Mach number, T_{41} is scheduled with ambient temperature and Mach number.

The TO envelope in terms of altitude and speed typically extends from 1 000 ft below SL to 15 000 ft and M0.0 to M0.5. Examples of the points used to derive the MTO control system limits are given in Table 2.2 and Table 2.3.

Table 2.2 Points for deriving T_{41} limits for the MTO control schedule.

Altitude [ft]	Delta T from ISA [K]	Mach number [-]
0	Engine flat rating point	0.0
0	Engine flat rating point	0.2
0	Engine flat rating point	0.4
0	30	0.0
0	30	0.2
0	30	0.4

Table 2.3 Points for deriving P_3 limits for the MTO control schedule.

Altitude [ft]	Delta T from ISA [K]	Mach number [-]
0	0	0
0	0	0.2
0	0	0.4
10 000	0	0
10 000	0	0.2
10 000	0	0.4

2.7 Method for comparison of results

In order to validate the simulation models of the investigated engines, the predicted performance was compared with engine manufacturer data from engine decks or thrust tables. The cruise, MTO and MCL flight conditions were considered. The Mach numbers, altitudes and ambient temperatures considered in the comparison varies from engine to engine due to availability of data.

2.7.1 Cruise

The SFC is of big importance at cruise condition as the airplane spends most of its airborne time here. This is especially true for long range aircraft, but not entirely true for short range aircraft, e.g. regional jets. The predicted SFC was plotted as a function of F_N

from a high power setting (MCL) down to a low power setting close to flight idle and compared to engine manufacturer data.

2.7.2 Take off

At TO, the maximum net thrust for the MTO rating is of main importance. The SFC is not as important because the aircraft spends only a small amount of time at this operating condition. The predicted net thrust at MTO rating was plotted as a function of ambient temperature for different Mach numbers and altitudes. In the same chart engine manufacturer data was plotted. Also the SFC was compared by calculating the error as a percentage of the real engine SFC.

2.7.3 Climb

In climb, both the net thrust and SFC are important, especially for short range aircraft that do many flights every day. The predicted net thrust at MCL rating was plotted against altitude for a few different Mach numbers and compared to engine manufacturer data. The SFC was also compared by calculating the error at each point.

3 Implementation in GasTurb v11

This section describes and discusses the implementation in GasTurb v11 of the method of creating an engine model presented in Chapter 2. It can be thought of as a user manual on how to create an engine model with the purpose of predicting SFC at cruise and net thrust for the MTO and MCL ratings. The resulting engine model file can be used in GasTurb with GUI or with the GasTurb dynamic link libraries (DLL's) without GUI for the MDO framework.

3.1 Description of GasTurb v11

GasTurb is a commercial gas turbine and aero engine modeling software developed by *Kurzke* [1]. It lets the user design both the thermodynamic cycle at design and off-design conditions as well as the geometry and disk stress calculations for turbojet, turbofan and turboprop engines. It runs under Windows and has a graphical user interface which makes it user friendly and easy to use. Along with the main program, there are programs to create and modify turbine and compressor maps and a big collection of non-proprietary compressor and turbine maps.

3.1.1 Reason for choosing GasTurb v11

The reason GasTurb was chosen as the engine modeling software was that it has been in use at the Advanced Design department at Bombardier for a few years. Furthermore standalone DLL's that can be run without the graphical user interface were recently developed. This makes it possible to run GasTurb in an MDO framework which was the goal of the project.

3.1.2 Description of DLL's

At the time of writing, Bombardier had access to one DLL for the mixed flow two spool geared turbofan with booster type of engine as illustrated by Figure 0.4. This engine type can also be used to model two spool turbofans without gearbox by setting the gear ratio to unity or a boosterless engine of either the geared or standard engine type. To model a boosterless engine, the inner FPR in GasTurb is set to unity and the real inner FPR is assigned to the booster PR. A DLL for the unmixed flow two spool geared turbofan (Figure 0.3) is planned to be bought by Bombardier but was not available during the project. With DLL's for these two engine types, it is possible to model all jet engine types considered for Bombardier aircraft. Engine model files for use with GasTurb DLL's were created for four different jet engines during this project.

The DLL contains a set of functions for calculating steady state performance in a similar way as an engine deck. In this implementation a C++ program was written to call the functions within the DLL. This C++ program reads an input file which specifies the ambient conditions, power setting and installation losses. Then it calls the DLL with a GasTurb v11 engine model file and calculates the point defined in the input file. Finally an output file is created with calculated properties such as net thrust, SFC, pressures and temperatures at different stations within the engine. Example input and output files are included in Appendix A. For a detailed description of the variables in these files, the reader is referred to the *DLL user manual* [15].

3.2 Choose engine cycle

The first choice the user must make is the engine cycle. There is a big selection of engine cycles in GasTurb v11 but only the geared mixed flow turbofan and the geared unmixed flow turbofan are considered because of the possibility to use these types of engine model files with the DLL's.

If the engine that is to be modeled has a centrifugal compressor stage a simplification has to be made and the centrifugal compressor is modeled as an axial compressor. This makes no difference at the design point calculation, but at off-design this could be a source of error because the maps used reflect an axial compressor. In the map collection, there are maps for axial-centrifugal HPC's. If such a map is used in the modeling, the behavior of an axial-centrifugal HPC would be captured. This was not done for the modeling of the engines with centrifugal compressor stages.

Pick an engine cycle in the drop down list as illustrated in Figure 3.1, chose "Performance" as the "Scope", "Design" as the "Calculation Mode", then press "Run". The user is prompted to choose an engine model file, chose the "Demo_xxxx.XXX" in the "GasTurb11" folder, where "xxxx.XXX" is dependent on the type of engine selected. Starting with the demo file gives a good starting point and reasonable values for all engine design parameters.

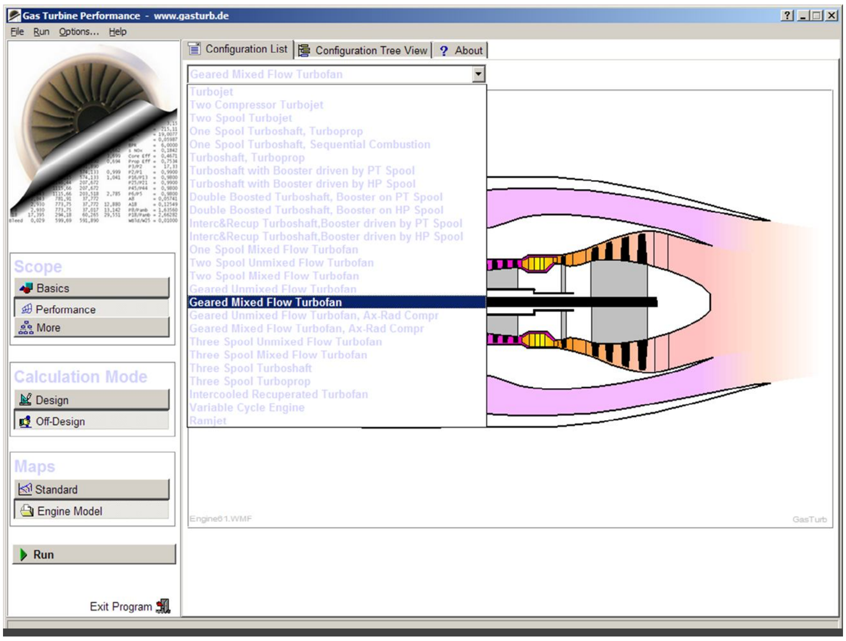

Figure 3.1 The GasTurb v11 home screen.

3.3 Choose assumed design point

An assumed design point in terms of Altitude, Delta T from ISA, humidity and Mach number must be chosen. From literature [12], this is typically at TO or TOC, according to sources within Bombardier it is probably TOC for the engines investigated in this project. Furthermore the user manual of GasTurb [1] suggests that the TOC is chosen as the design point because the default map scaling points in the component maps reflect this condition. This means that the map scaling point does not have to be moved as much in the off-design calibration procedure. However, the availability of data is more important and in practice, a point with as much engine parameter data as possible should be chosen as the design point.

Input the assumed design point in the “Basic Data” tab of GasTurb.

3.4 Choose basic engine parameters at the design point

There are approximately 50 different input parameters in 13 tabs for the design point calculation. If more detailed calculations such as calculation of component efficiencies or compressor design are to be performed, more parameters are needed.

Use the available engine data to fill in the values. If a value is unknown, use the default value or a value from literature together with engineering judgment to estimate the value.

Table 3.1 The pressure ratio input parameters in GasTurb v11 for a geared mixed flow two spool turbofan, Figure 0.4.

GasTurb parameter	Calculation using GasTurb station definition
Intake Pressure Ratio	P_2/P_{amb}
Inner Fan Pressure Ratio	P_{21}/P_2
Outer Fan Pressure Ratio	P_{13}/P_2
Core Inlet Duct Press. Ratio	P_{22}/P_{21}
IP Compressor Press. Ratio	P_{24}/P_{22}
Compr. Interduct Press. Ratio	P_{25}/P_{24}
HP Compressor Pressure Ratio	P_3/P_{25}
Bypass Duct Pressure Ratio	P_{16}/P_{13}
Turb. Interd. Ref. Press. Ratio	P_{45}/P_{44}
Burner Pressure Ratio	P_4/P_{31}
Turbine Exit Duct Press Ratio	P_6/P_5
Hot Stream Mixer Press Ratio	P_{63}/P_6
Cold Stream Mixer Press Ratio	P_{163}/P_{16}
Mixed Stream Pressure Ratio	P_8/P_{64}
Design Bypass Ratio	W_{13}/W_{21}

If an engine deck output is available and the pressure is given for different stations, some input parameter values can be calculated. Referring to Figure 0.4, the pressure ratios can be calculated with the help of Table 3.1. Take care that the engine deck output station definition is identical to the GasTurb station definition. Reasonable values for component total pressure losses are given in Table 3.2.

If T_3 and P_3 values are hidden from the user in the engine deck output, station 3 bleed air temperature and pressure can be used instead. If the engine mass flow is hidden it is in some cases possible to calculate the bypass and core mass flows if the bypass stream bleed and core bleed is given both as an absolute value and as a percentage of the bypass or core total mass flow respectively. Alternatively the engine mass flow can be calculated from the ram drag because $F_{RAM} = W_2V_a$.

3.4.1 Discussion of parameter values

The “No (0) or Average (1) Core dP/P” option can be used to model a radial pressure profile at the engine face, giving different pressure losses for the center and outer part of the inlet.

The “Intake Pressure Ratio” is typically around 0.995 at cruise conditions and less at low Mach numbers. An intake map can be used, but the generic intake map has too high losses compared to the investigated engines and was not used. There is also an option for calculation of shock losses for supersonic flight based on Mach number.

If an engine without booster is being modeled, input 1 as “Inner Fan Pressure Ratio” and the real inner FPR as “IP Compressor Press. Ratio” instead.

“Burner Exit Temperature” is normally unknown and a realistic value is most easily found by iteration.

It is recommended in the *GasTurb user manual* [1] to leave “Burner Partload Constant” at 1.6.

“Fuel Heating Value” can often be found in the header of a thrust table. “Overboard Bleed” and “Power Offtake” is normally varied during different flight conditions but was often 0 at the assumed design point for the investigated engines.

“HP Spool Mechanical Efficiency” and “LP Spool Mechanical Efficiency” are close to one, typically 0.99.

“Gear Ratio” can be set to 1 to model an engine without a gearbox.

“Mixer Efficiency” can be used as an iteration variable to adjust the net thrust in the calibration of the assumed design point.

“Design Mixer Mach Number” was set to 0 and instead “Design Mixer Area” was specified. The mixer area was assumed to be the same as the frontal area of the engine.

Under the “Air System” tab, the bleed and cooling flows are specified. “Rel. Handling Bleed to Bypass”, “Recirculating Bleed W_reci/W25” and “Rel. Enthalpy of Recirc Bleed” was not used and left at their default values.

“Rel HP Leakage to Bypass” was calculated from mass flow equilibrium in the bypass duct, $W_{16} - W_{13}$. “Rel. Overboard Bleed W_Bld/W25” was not used.

“Rel. Enthalpy of Overb. Bleed” was used to match the bleed port used for the core bleed air. If it is set to 1, bleed air is taken from the compressor exit. Otherwise it can be changed to match the bleed pressure and temperature to available data.

“HPT Cooling Air Pumping Dia” is used for the calculation of the work required for accelerating the rotor cooling air to blade velocity. It was assumed to be 0 because this work is small.

Cooling air flows might be possible to calculate from mass flow equilibrium if enough data is given by the engine deck output. “HPT NGV Cool Air W_Cl_NGV/W25” is calculated as $W_{41} - W_4$. “HPT Rotor Cooling Air W_Cl/W25” is calculated as $W_{44} - W_{43}$. “LPT NGV Cooling W_NGV_LPT/W25” is calculated as $W_{45} - W_{44}$. “LPT Rotor Cooling Air W_Cl/W25” is calculated as $W_5 - W_{49}$. If the cooling was not possible to calculate in this way, the cooling flow model from Figure 2.4 was used to estimate the cooling flows.

“Rel. Enth. LPT NGV Cooling Air” and “Rel. Enth. of LPT Cooling Air” were unknown and often assumed to be 1 for simplicity. The influence of the LPT cooling air enthalpy is small so this assumption does not introduce large errors.

“Rel. HP Leakage to LPT exit” was assumed to be 0.

“Rel. Fan Overb.Bleed W_Bld/W13” was calculated from the engine mass flow equilibrium. It was used to take thrust reverser leakage into account. $W_1 + W_F = W_8 + W_{bleed} + W_{thrust\_reverser\_leakage}$.

The engine corrected mass flow is most easily iterated with the physical mass flow as target.

The efficiencies for Inner LPC, Outer LPC, IPC, HPC, HPT and LPT can be given as isentropic efficiency, polytropic efficiency or calculated by GasTurb. If the component efficiency is unknown but the pressure and temperature change over a component is known, it can be calculated from thermodynamic equations [3]. From a practical point of view it is easier and quicker to define iterations in GasTurb than calculating it. Reasonable values for component polytropic efficiencies are given in Table 3.2 .

Table 3.2 Component total pressure losses and polytropic efficiencies. Source *Aircraft Engine Design Table 4.4* [3]

	Level of technology			
Parameter	1	2	3	4
Intake total pressure loss	0.90	0.95	0.98	0.995
Burner total pressure loss	0.90	0.92	0.94	0.96
Fixed area convergent nozzle total pressure loss	0.95	0.97	0.98	0.995
Fan polytropic efficiency	0.78	0.82	0.86	0.89
Compressor polytropic efficiency	0.80	0.84	0.88	0.90
Burner efficiency	0.88	0.94	0.99	0.995
Cooled turbine polytropic efficiency	-	0.83	0.87	0.89
Uncooled turbine polytropic efficiency	0.80	0.85	0.89	0.91
Maximum T_4 [K]	1110	1390	1780	2000
The levels of technology can be thought of as representing the technical capability for 20-year increments in time beginning in 1945. Thus level 4 technology represents typical component design values for the time period 2005-2025.				

The LPC and HPC design options were not used and the “Nominal LP/HP Spool Speed” was entered. The spool speeds have no effect on the thermodynamic calculations.

No correction was made for “HPT Clearance” and the “Test Analysis” was not used.

Finally, the nozzle calculation was for the most cases set to “Standard”, but if the discharge and thrust coefficients are known, they can be entered directly if “Specify CFG and CD” is chosen. Normally, the nozzle coefficients were unknown and “Nozzle Thrust Coefficient” was iterated to obtain the correct net thrust as given by the data. In this case the default value of “Design Nozzle Angle [°]” 10° was used. The discharge coefficient is based on this angle as illustrated in Figure 3.2.

3.4.2 Input parameter estimation calculation

An excel spreadsheet was created that calculated the GasTurb model input parameters for a two spool mixed flow turbofan from the parameters given in an engine deck output. The input to the spreadsheet consisted of pressure, temperature and mass flow rate at different stations in the engine. The output was pressure ratios, component efficiencies and estimations of cooling and leakage flow rates.

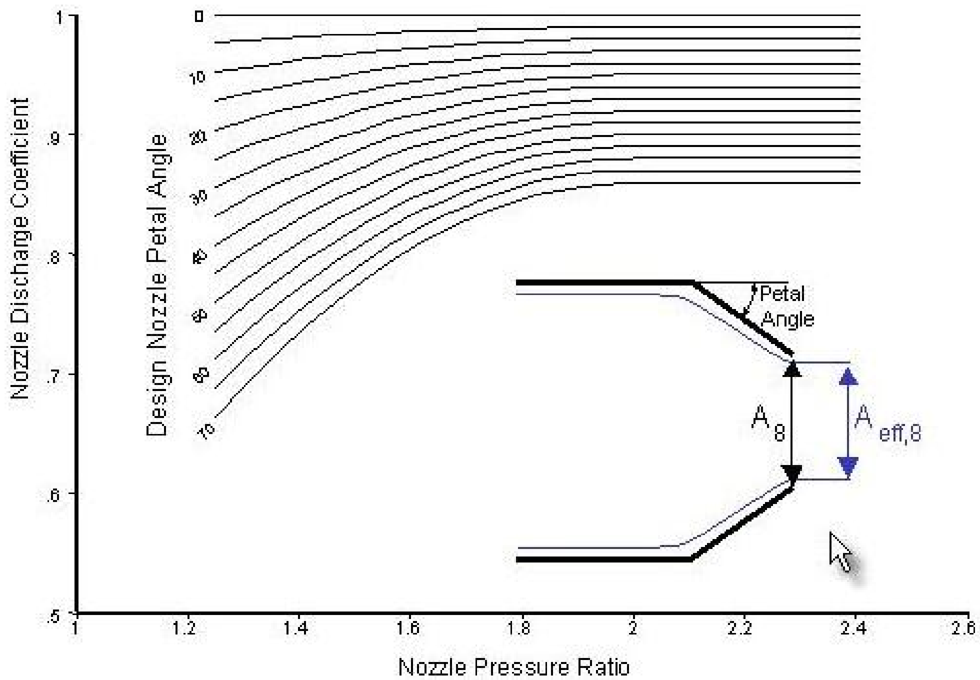

Figure 3.2 The GasTurb calculation of nozzle discharge coefficient [1].

The performed calculations follow Table 3.1 for the pressure ratios, equations from *Aircraft Engine Design* [3] for component efficiency calculations, cooling flow model from Figure 2.4, mass flow equilibrium for leakage calculations, T_4 is estimated with Equation 3.1. Equation 3.1 was derived by considering the energy equilibrium over the combustor. With all these equations, the spreadsheet contained the major parts of an engine modeling software for design point calculations. It was concluded that it is probably just as easy and just as accurate to calculate a few input parameters manually and/or use the *Iteration* capability of GasTurb to find others. The target values should be set to the known values in the available engine data.

The main problems are that each engine is different, the stations are defined differently for different engines and values are not known for all stations. This means that the spreadsheet must contain logics to check which input is given and depending on the input, calculate the parameters in the correct way. To illustrate this, the calculation of the outer FPR is demonstrated. Ideally, this should be calculated as P_{13}/P_2. If P_{13} is unknown and instead P_{16} is known, the FPR can be estimated as $P_{16}/P_2/PR_{bypass\ duct}$. If the bypass duct pressure ratio is unknown, the default GasTurb value is used.

Considering all possible combinations of input, a lot of work is needed to put this into a program. A strict definition of parameter names and file format of input and output files is necessary. Most of the parameters are the same for different engine types, but the set of input parameters for each engine cycle is different. Finally, the program would need to be validated to ensure that the input parameter estimations are reasonable. The program was not validated and was not developed further because of the reasons mentioned above.

3.5 Calibrate the engine model to match the design point

To reduce the uncertainties in the input parameters the GasTurb model assumed design point engine mass flow, net thrust and fuel flow is calibrated to the available data. This is most easily done by iteration within GasTurb.

To define iterations, click the “Iterations” option and select variable, reasonable min and max values and target value. After the iterations have been performed, make sure that the results are reasonable, e.g. by comparing to Table 3.2

The “Inlet Corr. Flow W2Rstd” is iterated with “Engine Mass Flow W2” as target.

If the burner inlet temperature, fuel flow and air mass flow is known, the “Burner Exit Temperature”, T_4, can be iterated with “Fuel Flow” as target. This follows from Equation 3.1, which is derived from energy equilibrium over the combustor. The fuel heating value, Q, is normally known and the design burner efficiency η_b is very close to 1. If for example T_{45} is known, an additional iteration can be defined where the burner efficiency is iterated to match this temperature.

$$\frac{W_{fuel}}{W_{air}} = \frac{\frac{T_4}{T_3} - \frac{C_{p3}}{C_{p4}}}{\frac{\eta_b Q}{C_{p4} T_3} - \frac{T_4}{T_3}}$$

Equation 3.1

Inner LPC, Outer LPC, IPC or HPC efficiency iterations can be set up if the pressure ratio and inlet and outlet temperature of the component is known. The efficiency should be set as variable with the compressor exit temperature as target.

HPT and LPT efficiency iterations can be set up if the cooling flows, temperature change and pressure change are known. Set the efficiency as variable and turbine exit pressure as target. This iteration is affected by several other parameters regarding the cooling flows, including amount of cooling flow and pressure and temperature of the cooling flow, and the results should therefore be judged and examined carefully.

When all input and iterations are defined, click “Run”. The assumed design point is calculated and the mass flow, pressure and temperature for the engine stations are shown together with many other parameters such as net thrust, fuel flow and SFC. Compare the GasTurb output with the available data and make corrections to the input data if necessary.

When the design point engine model is finished, save the model as a cycle data file by selecting “File – Save – Input”.

3.6 Calibrate the compressor maps to match off-design

The default GasTurb LPC, IPC and HPC maps are calibrated to available engine performance data. This must be done because the generic maps do not reflect a specific compressor in a specific engine. If compressor maps for the real components were available, this step would not be necessary.

3.6.1 Mass flow – efficiency (map scaling point)

The map scaling points of the LPC, IPC and HPC maps are moved in order to match the mass flow – efficiency behavior of the real engine. This is most easily done by plotting the available engine data SFC as function of net thrust at a given flight condition. Then the GasTurb engine model is run at the same flight condition with the same installation losses and plotted. This is illustrated in Figure 2.6. In this example, the SFC is too high at high thrust and too low at low thrust when compared to thrust table data. Referring to Figure 2.5, this means that the map scaling point should be moved down to the left from its current position, which will increase the component efficiency values at high thrust and decrease the component efficiency values at low thrust. The procedure used was to pick another point along the engine operating line in the compressor map, read the β-value and corrected speed value for each compressor, input this as the new map scaling point and then make a new comparison. This is an iterative approach which might require a few iteration loops to match the available data well.

To calculate off-design points in GasTurb, one must go back to the main screen and in "Calculation Mode" select the "Off-Design" option. Make sure that the correct cycle data file is used. It is possible to read in test data for plotting in GasTurb. Details on the test data file format are found in section 3.4.4 of the *GasTurb user manual* [1]. For this project, the data was plotted in Excel.

When the flight conditions and installation losses are given, calculate an off-design single point with the same amount of thrust as the point in the available data with maximum thrust. This is most easily done by use of "Limiters". Go to the limiter tab, select Net Thrust as a MAX limiter, give a value, turn it on, and then calculate the point by clicking "Run". Remember to activate the limiters by clicking the "Max" button.

Then select the "Operating Line" option and click "Run". The operating line is calculated by changing the high spool speed NH a small amount for each point. Give the step size, number of points, select "Decreasing speed" and then click "Run". Plot the SFC versus Net Thrust and compare with a loaded test data file or export the data to Excel. Using Figure 2.6 as an example, a point down the operating line should be selected as the new map scaling point. Close the "Operating Line" window and select a new ZXNH on the "Steady State" tab. As a first guess, decrease the NH by 0.02 and calculate this as a single point. Read the β-values and corrected speeds for the compressors from the "Oper.Point" tab in the output window. Close the results and go to "Maps – Special" to change the map scaling points. If this option is greyed out, set the input parameters to the same value as at the assumed design point. This is most easily done by re-opening the cycle data file.

In the map scaling window, input the β-values and corrected speed values for the new map scaling points of the LPC, IPC and HPC. The turbine map scaling points are left at their default values. Save the map scaling as a .SCL file for later use. Close this window, calculate a new operating line, compare the results, and if necessary make another adjustment of the map scaling points.

When the map scaling point calibration is finished, go back to the off-design window, set the input parameters to the assumed design point values and select "File – Write – Engine Model". This saves all the data, including the new map scaling points, as an engine model file. This engine model file can be used together with the DLL's.

3.6.2 Change corrected fan speed – net thrust relation

The corrected fan speed – F_N relation is altered by re-naming the speed lines in the fan map. This can be done within GasTurb, but unfortunately there seems to be a bug when a map is altered within GasTurb. A map is opened and then saved without doing any changes. If the saved map is opened again, it will be different from the original map and give different results at off-design calculations. Due to this problem the speed – F_N correction was not done for the investigated engines. The effect of this is that the engine model will not produce the correct amount of thrust for a given corrected fan speed. The thermodynamic model is not affected but the engine model does not entirely follow the real engine behavior in terms of fan rotational speed.

The method for correcting the speed lines is as follows. Net thrust is plotted as a function of corrected fan speed for the GasTurb engine model and the available data. It can be done within GasTurb with a test data file but for this project it was plotted in Excel. Open the "Operating Line" option in the off-design calculation, then select "File – Edit – LPC Map" and go to the "Speed" tab. There one can change the speed lines in accordance with the plot. After changing the speed lines, run an operating line and compare the results. If necessary, change the speed lines again. When the result is satisfactory, save the scaled map by clicking "File – Save Scaled Map – LPC Map". Then go back to the off-design window and write the result to an engine model file. This will save both cycle data, map scaling and the reference to the changed map. As mentioned earlier there is a problem with saving a map and therefore this method does not give good results.

3.7 Create an engine control system

The engine control system uses limiters, schedules and tables. A limiter is a single maximum or minimum value. A schedule is a parameter that is a function of one variable, e.g. T_{45} as a function of altitude. A table is a parameter that is a function of two variables, e.g. T_{45} as a function of altitude and Mach number. Tables and schedules can be assigned to limiters, e.g. so that the maximum allowed T_{45} can be expressed as a function of altitude and Mach number.

3.7.1 Requirements on engine model file for use with DLL

In the DLL documentation [15] a few restrictions regarding the engine model file are specified for steady state calculations.

- The engine model file must be created with GasTurb v11.
- The engine model file must use SI units.
- "rel NL for PLA = 0%" and "rel NL for PLA = 100%", found under the "Transient" tab in off-design calculation must be given reasonable values.
- Both maximum and minimum limiters must be defined and switched on.
- "ZXN given (1) or ZT4 given (2)" on the "Steady State" tab in the off-design window must be set to 1.
- There must be a composed value named "ZXX". It should be set to the engine pressure ratio EPR = P_5/P_2.

A useful minimum limiter for the control system is the composed value "Nidle=XN_HPC*100/(60+0.001*alt)". XN_HPC is the high spool speed and alt is the altitude. The minimum value of this limiter should be set to one to prevent convergence problems when the engine is run at low thrust settings.

3.7.2 Maximum take off rating

To model the maximum take off rating a table with T_{41} as a function of ambient temperature and Mach number and a table with P_3 as a function of altitude and Mach number is used. The limits are derived by using the net thrust limiter and running the engine model to a known value of F_N at a certain flight condition that corresponds to the MTO rating. Then the T_{41} or P_3 output from the simulation model is read and entered into the table. This is most easily done with a batch job because several points are going to be evaluated.

A batch job is defined by clicking "Batchjob" in the menu. To create a batch job, click "Edit – Job Input". Add input data, composed values, iterations, limiters and finally calculate to for each point to be calculated. Typical points in the TO envelope used to create the control schedule is seen in Table 2.2 and Table 2.3. The batch job output is most easily given to an Excel file. Due to the limit of 256 columns in the Excel 2003 spreadsheets, a maximum of 254 points can be calculated in a batch job if Excel 2003 is used to read the data. In newer versions of Excel this limit has been greatly increased. To output to Excel click "Excel – Initialize" select the desired output parameters (T_{41} and P_3) and close this window. Then run the batch job with output to Excel.

When the limiting values of T_{41} and P_3 have been derived, click "Control Schedules" in the off-design window, then click "define a general table". Create a table with Mach number as parameter and altitude as X-value and enter the P_3 values, Mach numbers and altitudes in the table. Save the table and create a T_{41} table in a similar way. Close the table window and go back to the Control Schedule window. Choose the parameter that should be limited, e.g. P_3 and select the newly created table from the drop down list. Repeat for T_{41} and save the created schedule. Activate the limiters in the off-design window and write to an engine model file. Both the tables and control system will be included when the engine model file is written.

3.7.3 Maximum climb rating

The control system for the MCL rating is created in a similar way as the MTO control system. The maximum allowed P_3 is expressed as a function of altitude and Mach number in a general table. It is not possible to have control systems for different ratings in a single engine model file, therefore it must be one engine model file for the MCL rating and one for the MTO rating.

In addition to this, there is a limit imposed on the maximum allowed corrected fan speed. This parameter typically limits the engine performance at TOC conditions. It can be derived by running the engine model to a known thrust at a typical TOC flight condition, e.g. M0.7, 30 000 ft. The corrected fan speed is read and set as a limiter.

3.8 Sensitivity study of input parameters

A sensitivity study was performed where input parameter influence on net thrust, fuel flow and SFC was studied. The purpose of the study was to determine the important parameters for net thrust, fuel flow and SFC when trying to model an engine in GasTurb at the design point. 42 input parameters for a two spool mixed flow turbofan were studied. The parameters consisted of nozzle coefficients, pressure ratios, pressure losses, component efficiencies, amount of cooling, bleed and leakage air mass flow. The Engine 3 engine model at its assumed design point was used as baseline. The reason is that most of the parameters were known from an engine deck output.

3.8.1 Method

The study was performed by changing the input parameters one by one at the assumed design condition at the design point calculation. Each parameter was reduced by 1% and increased by 1% and F_N, W_F and SFC was calculated. The change in output parameter was averaged from the upper and lower calculated points and was expressed as change in output parameter for each 1% increase in input parameter. Some parameters could not be increased by 1% because they were already at their maximum limit, e.g. a duct pressure ratio should never be higher than 1. A duct pressure ratio is defined as the exit pressure divided by inlet pressure.

The input parameter *Specified thrust coefficient CFG* is used in an example calculation to clarify the method used. Assume that the design point value of this input parameter is 0.96 and the resulting SFC is 0.70. CFG was decreased by one percent to 0.9504 and SFC calculated. The resulting SFC is assumed to be 0.71. Then CFG was increased by one percent from its design point value to 0.9696 and SFC calculated again. The resulting SFC is assumed to be 0.69. The influence of this input parameter on SFC is calculated as seen in Equation 3.2. The result -1.43% means that if the CFG is increased with 1% from its design point value, the SFC will decrease (improve) by 1.43%.

$$\frac{Percentage\ \ change\ in\ output}{Percentage\ \ change\ in\ input} = \frac{(0.69-0.71)/0.70\cdot 100}{2} = -1.43\%$$ **Equation 3.2**

3.8.2 Results

The results from the input parameter study are summarized in Table 3.3 where the change in W_F, F_N and SFC for 1% increase in input parameter value is given. The results are sorted with the most influential input parameters at the top. Many input parameters do not change W_F. All input parameters except four change F_N and SFC. Recall the definition of SFC which is W_F divided by F_N, Equation 2.3.

Table 3.3 The change in W_F, F_N, and SFC for 1% increase of input parameter value.

Input parameter	W_F	F_N	SFC
// Specified Thrust Coeff CFG	0.00%	2.11%	-2.11%
// HP Spool Mechanical Efficiency	0.00%	1.93%	-1.97%
// Intake Pressure Ratio	1.02%	2.29%	-1.29%
// Isentr. HPT Efficiency	0.00%	1.27%	-1.27%
// Burner Exit Temperature	2.16%	3.42%	-1.27%
// Mixed Stream Pressure Ratio	0.00%	1.24%	-1.25%
// Isentr. HPC Efficiency	0.49%	1.73%	-1.24%
// LP Spool Mechanical Efficiency	0.00%	1.02%	-1.03%
// Fuel Heating Value	-1.08%	-0.12%	-0.97%
// Burner Design Efficiency	-1.01%	-0.11%	-0.90%
// Isentr. LPT Efficiency	0.00%	0.85%	-0.85%
// Outer Fan Pressure Ratio	0.00%	-0.80%	0.80%
// Isentr.Outer LPC Efficiency	0.00%	0.70%	-0.70%
// Turb. Interd. Ref. Press. Ratio	0.00%	0.65%	-0.65%
// Hot Stream Mixer Press Ratio	0.00%	0.65%	-0.65%
// Compr. Interduct Press. Ratio	0.00%	0.63%	-0.64%
// Turbine Exit Duct Press Ratio	0.00%	0.63%	-0.64%
// Burner Pressure Ratio	0.00%	0.63%	-0.64%
// Cold Stream Mixer Press Ratio	0.00%	0.60%	-0.61%
// Bypass Duct Pressure Ratio	0.00%	0.59%	-0.59%
// Specified Discharge Coeff CD	0.00%	-0.33%	0.33%
// Isentr.Inner LPC Efficiency	0.11%	0.38%	-0.27%
// Design Bypass Ratio	-0.83%	-1.07%	0.24%
// HPT Cooling Air W_Cl/W25	-0.12%	-0.25%	0.14%
// NGV Cooling Air W_Cl_NGV/W25	-0.14%	-0.22%	0.08%
// Inlet Corr. Flow W2Rstd	1.02%	1.06%	-0.04%
// Mixer Efficiency	0.00%	0.04%	-0.04%
// Overboard Bleed	-0.02%	-0.05%	0.03%
// Design Mixer Mach Number	0.00%	0.03%	-0.03%
// Rel. Enthalpy of Overb. Bleed	0.00%	-0.02%	0.02%
// Inner Fan Pressure Ratio	-0.25%	-0.24%	-0.02%

// LPT Cooling Air W_Cl/W25	-0.01%	-0.02%	0.01%
// Power Offtake	0.00%	-0.01%	0.01%
// LPT NGV Cooling W_NGV_LPT/W25	-0.01%	-0.03%	0.01%
// Rel. Enth. LPT NGV Cooling Air	0.00%	-0.01%	0.01%
// Rel. Enth. of LPT Cooling Air	0.00%	-0.01%	0.01%
// HP Compressor Pressure Ratio	-0.25%	-0.26%	0.01%
// Rel. Overboard Bleed W_Bld/W25	0.00%	-0.01%	0.01%
// Rel. Fan Overb.Bleed W_Bld/W13	0.00%	0.00%	0.00%
// Rel. HP Leakage to Bypass	0.00%	0.00%	0.00%
// Nominal HP Spool Speed	0.00%	0.00%	0.00%
// Nominal LP Spool Speed	0.00%	0.00%	0.00%

3.8.3 Explanation of results

This section gives an explanation of the results presented in Table 3.3. The reasoning is based on standard thermodynamics and gas turbine theory.

// Specified Thrust Coeff CFG
No effect on W_F. Affects only the net thrust calculation, the remaining engine parameters are the same. It is a coefficient on the momentum term in the net thrust equation as seen in Equation 3.3 (compare to Equation 2.2).

$$F_N = W_8 V_8 CFG_8 - W_0 V_a + A_8 \left(P_{s,8} - P_{s,amb}\right) \qquad \textbf{Equation 3.3}$$

// HP Spool Mechanical Efficiency
No effect on W_F. The HPT pressure and temperature drop is less due to less work extraction, giving higher pressure and temperature at the exit, thus higher F_N.
// Intake Pressure Ratio
W_F is higher because there is more mass flow through the burner. Higher pressures throughout the engine and at the nozzle, both pressure and mass flow is higher, giving more net thrust.
// Isentr. HPT Efficiency
No effect on W_F. The turbine pressure and temperature drop is less, giving higher pressure and temperature at the exit, thus higher F_N.
// Burner Exit Temperature
Higher temperature means more fuel burnt, gives higher W_F. More fuel burnt gives higher temperatures and higher F_N.
// Mixed Stream Pressure Ratio
No effect on W_F. A pressure loss affecting the mixer, increased pressure ratio means higher pressure at the nozzle exit, thus higher F_N.
// Isentr. HPC Efficiency
Higher efficiency translates into lower temperature at the HPC exit which means higher temperature rise in the burner for a constant burner exit temperature, thus more W_F. Also means that the extracted work at the HPT needed to drive the HPC is lower, giving higher temperature and pressure at the nozzle, thus higher F_N.

// LP Spool Mechanical Efficiency
No effect on W_F. The LPT pressure and temperature drop is less due to less work extraction, giving higher pressure and temperature at the nozzle, thus higher F_N.
// Fuel Heating Value
Less W_F because fuel contains more energy and less fuel is needed to get the same burner exit temperature. Slightly lower F_N because lower mass flow.
// Burner Design Efficiency
Less W_F because the burner is more effective, meaning less fuel is needed to achieve the same temperature rise. Lower F_N because of the lower mass flow.
// Isentr. LPT Efficiency
No effect on W_F. The LPT pressure and temperature drop is less, giving higher pressure and temperature at the exit, thus higher F_N.
// Outer Fan Pressure Ratio
No effect on W_F. Higher bypass temperature. Less F_N because more work is needed to drive the fan. More F_N from bypass but less F_N from core. Cold/hot stream at mixer inlet pressure ratio changes. The sum is a lower pressure at the nozzle, thus lower F_N. According to theory, there is an optimum FPR which depends on BPR and component efficiencies among other things and the trend could be different if the FPR is at a non-optimum or the optimum value initially.
// Isentr.Outer LPC Efficiency
No effect on W_F. Slightly lower bypass temperature. Less work extracted at LPT to drive the fan, meaning higher temperature and pressure at exit, thus higher F_N.
// Turb. Interd. Ref. Press. Ratio
No effect on W_F. Pressure loss in core stream is smaller, thus higher pressure at nozzle and higher F_N.
// Hot Stream Mixer Press Ratio
No effect on W_F. Pressure loss is smaller, thus higher pressure at nozzle and higher F_N.
// Compr. Interduct Press. Ratio
No effect on W_F. Pressure loss is smaller, thus higher pressure at nozzle and higher F_N. Affects turbine operating conditions.
// Turbine Exit Duct Press Ratio
No effect on W_F. Pressure loss is smaller, thus higher pressure at nozzle and higher F_N.
// Burner Pressure Ratio
No effect on W_F. Pressure loss is smaller, thus higher pressure at nozzle and higher F_N. Affects turbine operating conditions.
// Cold Stream Mixer Press Ratio
No effect on W_F. Pressure loss is smaller, thus higher pressure at nozzle and higher F_N.
// Bypass Duct Pressure Ratio
No effect on W_F. Pressure loss is smaller, thus higher pressure at nozzle and higher F_N.
// Specified Discharge Coeff CD
No effect on W_F. Less F_N because the nozzle conditions are affected.
// Isentr.Inner LPC Efficiency
Less temperature increase in the LPC means higher temperature rise in the burner for a constant burner exit temperature, meaning higher W_F. Also means less work extracted at the turbine leading to higher temperature and pressure at the nozzle, thus higher F_N.

// Design Bypass Ratio
Less W_F and less F_N. The same inlet flow, but more flow is going through the bypass duct and less through the engine core, thus less mass flow through burner and less W_F. Lower F_N because more air is going through the bypass where it is accelerated less than in the core which means lower specific thrust. In this case, the loss of thrust is bigger than the decrease in fuel flow. Is also dependant on other engine parameters such as FPR. If the net thrust were to be held constant during this investigation, the SFC trend would be opposite.
// HPT Cooling Air W_Cl/W25
Less W_F because HPC air is extracted and there is less mass flow through the burner. Less F_N because there is less mass flow that does work in the HPT, meaning that the temperature and pressure is lower at the nozzle.
// NGV Cooling Air W_Cl_NGV/W25
Less W_F because HPC air is extracted and there is less mass flow through burner. Less F_N because the air that does work in the HPT has lower temperature and pressure, meaning that the temperature and pressure is lower at the nozzle.
// Inlet Corr. Flow W2Rstd
W_F and F_N increase about as much as the corrected flow increases. More mass flow through the burner means more W_F, more mass flow through the engine means more F_N.
// Mixer Efficiency
No effect on W_F. F_N is slightly increased. The definition of mixer efficiency is found in the manual of *GasTurb Details 5, pp. 33-34* [16].
// Overboard Bleed
Lower W_F and F_N. Air is extracted before the burner and there is less mass flow through the burner. Less air is doing work in the turbine, meaning lower temperature and pressure at the nozzle, thus lower F_N.
// Design Mixer Mach Number
No effect on W_F. F_N increases because the mixing is done slightly more efficient and the nozzle pressure is higher. Depending on the starting condition as there is an optimum value for this parameter.
// Rel. Enthalpy of Overb. Bleed
No effect on W_F. F_N decreases because the compressor has done more work on the high enthalpy air before it is extracted compared with low enthalpy air.
// Inner Fan Pressure Ratio
Lower W_F because higher temperature at fan exit, HPC exit and burner inlet. More LPT work must be extracted to drive the inner fan, meaning less F_N. Also affect the HPC operating conditions so that the HPC need more work for the compression.
// LPT Cooling Air W_Cl/W25
Less W_F because HPC air is extracted and there is less mass flow through burner. Less F_N because there is less mass flow that does work in the LPT, meaning that the temperature and pressure is lower at the nozzle.
// Power Offtake
No effect on W_F. Lower F_N because more work is extracted in the HPT.
// LPT NGV Cooling W_NGV_LPT/W25

Less W_F because HPC air is extracted and there is less mass flow through burner. Less F_N because the air that does work in the LPT has lower temperature and pressure, meaning that the temperature and pressure is lower at the nozzle.
// Rel. Enth. LPT NGV Cooling Air
No effect on W_F. F_N is decreased because more work is needed in the HPC, meaning more work must be extracted in the HPT and thus the temperature and pressure at the nozzle is lower.
// Rel. Enth. of LPT Cooling Air
No effect on W_F. F_N is decreased because more work is needed in the HPC, meaning more work must be extracted in the HPT and thus the temperature and pressure at the nozzle is lower.
// HP Compressor Pressure Ratio
Lower W_F and F_N. Higher HPC exit temperature gives less W_F. More HPC work means bigger temperature and pressure drop of the HPT, giving lower pressure and temperatures at nozzle, thus lower F_N.
// Rel. Overboard Bleed W_Bld/W25
Same as overboard bleed.
// Rel. Fan Overb.Bleed W_Bld/W13
Small influence.
// Rel. HP Leakage to Bypass
Small influence.
// Nominal HP Spool Speed
No influence.
// Nominal LP Spool Speed
No influence.

3.8.4 Input parameter uncertainty

Some of the parameters that seem to have very little or no influence on the SFC will in fact be more important than they look. For example, the bleed air will typically be 0 during take off in order to have maximum thrust available. The amount of bleed air will be high at low altitude and at the design condition at high altitude it will be low, perhaps only half of the low altitude value. Therefore a change of 100% of the design point value is realistic.

This can be compared to the intake pressure ratio that rarely varies with more than 0.5% during a flight. Typical values are 0.997 at cruise and 0.993 at TO.

When the influence of the uncertainty of the input parameters is taken into account, the cooling flows will be prevailing, followed by component efficiencies and the influence from pressure ratios is relatively speaking small. This is presented in Figure 3.3. The results are grouped, each group has a comparable change in input parameter.

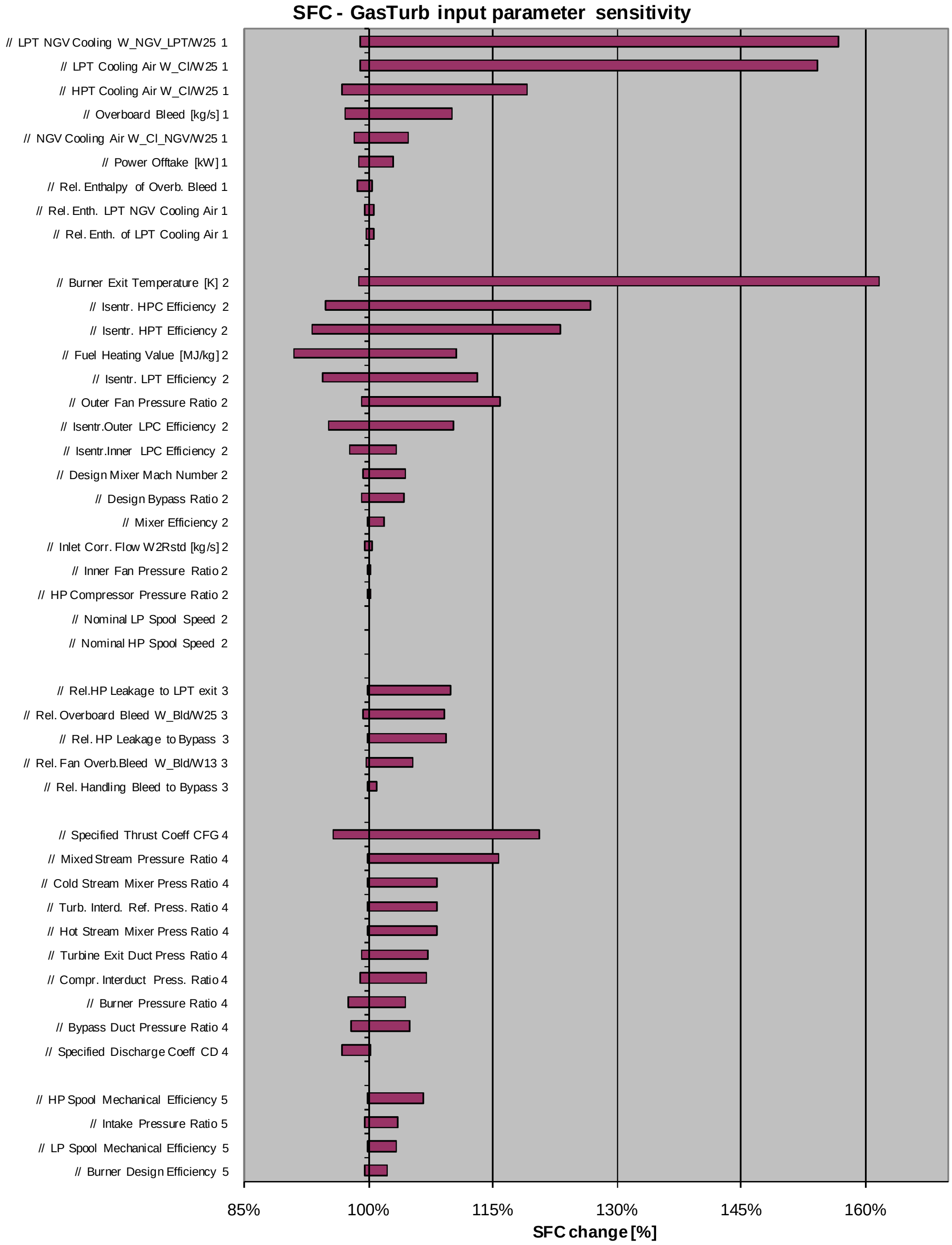

Figure 3.3 The effect of input parameter uncertainty on SFC of the E3 GasTurb engine model. The vertical line at 100% shows the baseline E3 SFC.

The boundaries set on the input parameters will greatly influence the result. The selected boundaries are given in Appendix B. The boundaries are based on the oldest technology in Table 3.2 and up to the maximum theoretical value for some parameters. The boundaries are large and too big when a modern engine is considered. When considering the purpose of this study, to set up an engine model, one has to keep in mind that many of the parameters are given or can be deduced from the available engine data. Furthermore, a calibration of the model to available data is made. This calibration effectively removes the uncertainties from the remaining unknown parameters and the SFC can be predicted accurately.

3.9 Geometry and weight prediction

GasTurb v11 has the possibility to predict the geometry and weight of a jet engine. It is based on a model where each component is modeled as a separate part with a material density, thickness and surface area. Not every component of a real jet engine is included in the model and therefore a “Net Mass Factor” is used to take this into account. More than a 100 additional input parameters are needed to accurately model the geometry and weight of an engine. The additional input includes the following parameters:

- Lengths, angles and entry/exit radii for inlet, inlet cone, burner, ducts, bypass, exhaust.
- Strut, guide vane and blade aspect ratio and pitch.
- Material properties, material thickness.
- Stress margins for disk stress calculations.

GasTurb has default values for all of these parameters and an attempt was made to model the investigated engines in the geometry and weight tool. The number of compressor and turbine stages was given as input, remaining parameters were left to their default values. The investigated engines that use a centrifugal compressor stage were modeled as axial flow compressors with approximately the same pressure ratio. This is because only axial flow turbomachinery is considered in the weight and geometry tool. This assumption could increase the predicted engine length and decrease the predicted engine diameter due to geometrical differences between an axial and a centrifugal compressor stage.

The results of the investigation are illustrated in Figure 3.4. The max diameter is underestimated for all engines but not close to the real engine value. Estimated length and weight is sometimes too high and sometimes too low. The poor result was expected given the lack of detailed input data.

The inputs that are needed are not readily available at the Advanced Design department. Furthermore, this tool is more suited for conceptual design of an engine than for engine performance and weight prediction within an MDO software framework. Therefore it was decided that other methods should be used to predict the engine weight and size. A reasonable method would be a statistical model based on existing engines where the weight is related to the thrust (thrust/weight ratio).

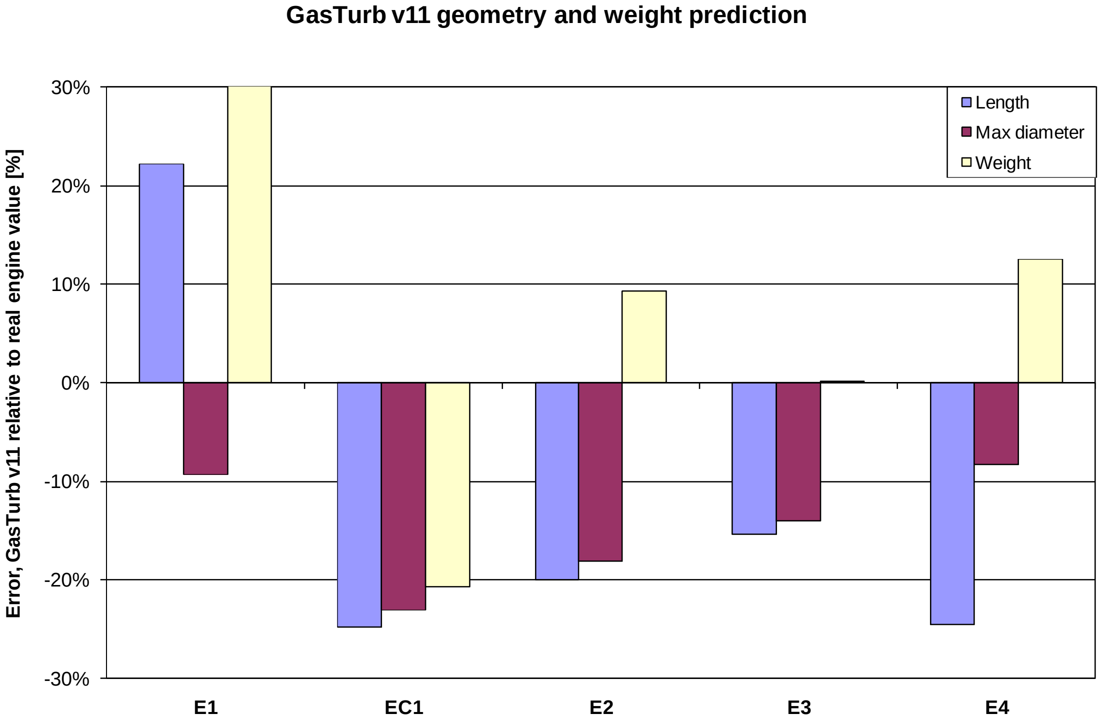

Figure 3.4 Predicted length, max diameter and weight from GasTurb v11 geometry and weight prediction tool for the investigated engines.

3.10 Integration with MDO framework

The software chosen for the MDO framework was Isight. Isight is a tool that connects different programs and executable files. To use the GasTurb engine model within Isight, a C++ executable program was written that calls the GasTurb DLL. This executable was connected to Isight and included in Isight batch jobs and optimization runs. The GasTurb engine model could not be used in a full scale multidisciplinary design optimization run because the MDO framework was still under development during this project. The engine model was run separately within the MDO framework and an example of this is illustrated in Figure 3.5. This figure shows that the EC1 engine model that was run within the MDO framework gives the same results as the EC1 engine model that is run from GasTurb with a GUI.

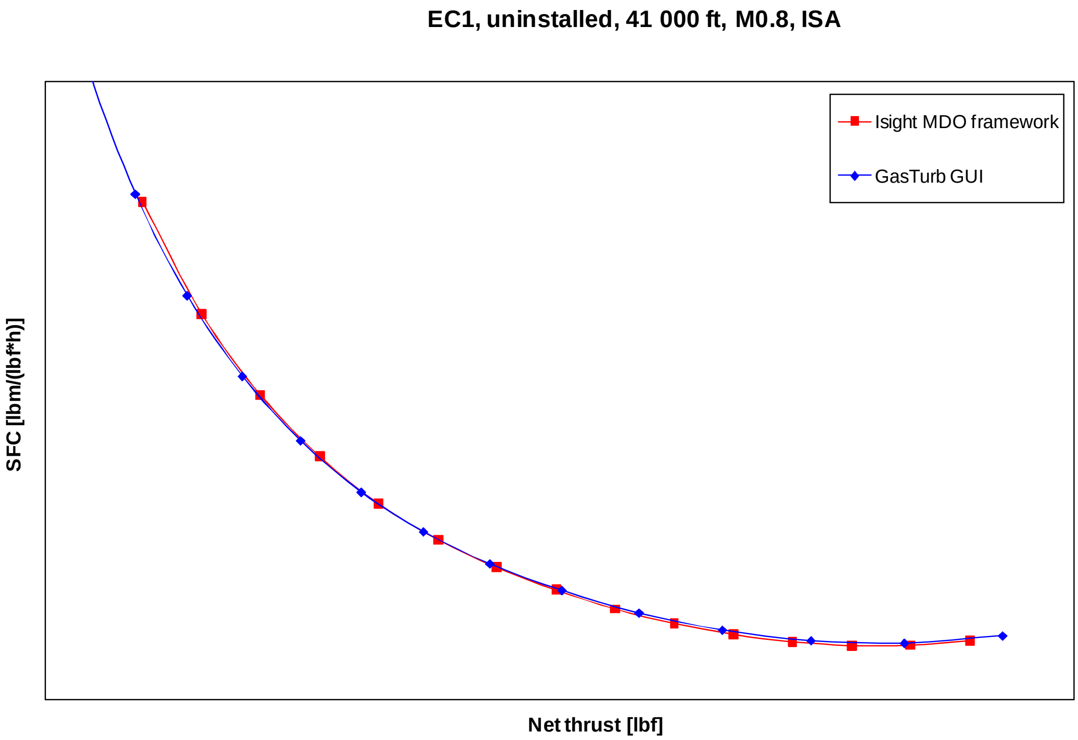

Figure 3.5 SFC of the Isight MDO framework engine model and GasTurb GUI engine model.

4 Case studies

The engine modeling method was applied to 5 different engines. The results are considered sensitive information in terms of actual numbers and no values or scales are given in text, figures and tables. For all results, the data labeled “GasTurb” is the data created by the GasTurb engine model. The “Thrust table” or “Engine deck” label represents engine data obtained from the engine manufacturer’s steady state performance program. Whenever an error is described as a percentage, it is calculated as seen in Equation 4.1. A negative error means that the value predicted by the GasTurb model was lower than the thrust table or engine deck value. A positive error means that the value predicted by the GasTurb model was higher than the thrust table or engine deck value.

$$Error\,[\%]:\left(\frac{GasTurb\ value}{Thrust\ table\ or\ Engine\ deck\ value}-1\right)\cdot 100 \qquad \textbf{Equation 4.1}$$

4.1 Engine 1

Engine 1 (E1) is a mixed flow two spool turbofan in the take off thrust class < 10 klbf intended for use on aircraft in the mid-size business jet segment.

4.1.1 GasTurb engine model setup

An engine deck was available for Engine 1. Therefore the uncertainties regarding certain input parameters, e.g. intake pressure loss, bleed air and power offtake were removed because the user is in control of them when running the engine deck. Many engine parameters such as efficiencies and nozzle coefficients were hidden from the user. E1 does not have a booster stage but was modeled as a geared mixed flow two spool turbofan with booster in order to use the engine model file with the DLL. Therefore the inner FPR was set to 1 and the real inner FPR set as the IPC PR.

E1 engine model estimated input parameters regarding the assumed design point:

- The assumed design point was selected as uninstalled 43 000 ft, M0.82 at 94% of maximum cruise rating.
- Fan isentropic efficiency was iterated to match T_{16}.
- HPC isentropic efficiency was iterated to match T_3. This is a bit low, comparing to Table 3.2.
- Iteration for correct inlet mass flow.
- Outer FPR calculated from engine deck output.
- Inner FPR 1 because engine without booster.
- IPC PR – is the inner FPR, value comes from Engine 1 documentation.
- HPC PR calculated from engine deck output.
- LPT efficiency and exhaust pressure losses adjusted to obtain correct P_6 pressure.
- Power offtake according to documentation.
- No HPC bleed.
- Assumed the relative enthalpy for the port 2.8 bleed to get the correct bleed pressure compared to the assumed design point engine deck output.

- Turbine cooling flow calculated from $W_2 - W_3$. Uncooled LPT NGV and LPT rotor was assumed, in line with the cooling flow model presented in Figure 2.4.
- Thrust reverser leakage flow calculated as total mass flow in – total mass flow out, $W_{leakage} = W_{12} + W_2 + W_F - W_8$.
- Leakage core to bypass is calculated as the difference between core in and out flow, $W_2 - (W_6 - W_F)$.
- The mixer efficiency was unknown and set to 1.
- The mixer area was assumed to be equal to the fan frontal area.
- The nozzle petal angle was left at its default value of 10º.
- Iterate nozzle thrust coefficient to match the net thrust.
- Iterate burner exit temperature for correct fuel flow. No data is given about T_{41}, but the resulting ITT is 24K too high. This could be due to an incorrect cooling flow model or because of the wrong temperature drop over the HPT. The HPT turbine temperature drop is decided by the HPT extracted work, which is determined by the HPC work and the HP spool mechanical efficiency.

4.1.2 Results

4.1.2.1 Cruise

The map scaling points of the compressors were calibrated to match the engine deck data, as illustrated in Figure 4.1. It is seen that the agreement between the GasTurb model and the engine deck for the uninstalled case is very good for a big range of net thrust. It is also seen that the effect of the bleed air and intake pressure loss is captured by GasTurb with little error. The shape of the engine deck curves is not smooth when compared to the GasTurb curves. This probably means that real engine effects, found by comparison to measured data, are taken into account by the engine deck while GasTurb represents a theoretical engine model. The SFC is predicted with about +-1.5% error for 30 000 ft to 49 000 ft, M0.70 to M0.85, uninstalled and installed. The biggest SFC error occurs at 30 000 ft, installed, M0.70. The smallest SFC error is at 49 000 ft, uninstalled, M0.85.

4.1.2.2 MCL

The P_3 table used for the E1 MCL control schedule comprised of 6 altitudes and 5 Mach numbers (30 values) similar to Table 2.1. In addition to the P_3 table, a maximum allowed value of corrected fan speed was also used. This value was derived at 30 000 ft, M0.5.

The results where GasTurb F_N is compared to thrust table F_N at MCL rating, ISA conditions, SL to 50 000 ft and five different Mach numbers are seen in Figure 4.2. Table 4.1 show F_N and SFC errors for Engine 1 MCL rating. As can be seen in the figure, the thrust does not decline smoothly with altitude. This behavior is very difficult to predict with such a simple control system that is implemented, therefore the MCL thrust will sometimes have a big error for a certain point. It is seen that the predicted net thrust is too low at low ambient temperatures and too high at high ambient temperatures. At higher temperatures ambient temperatures, the engine flat rating temperature limits will start to limit the thrust. This is not captured by the E1 P_3 control schedule, making the prediction worse as the ambient temperature increase.

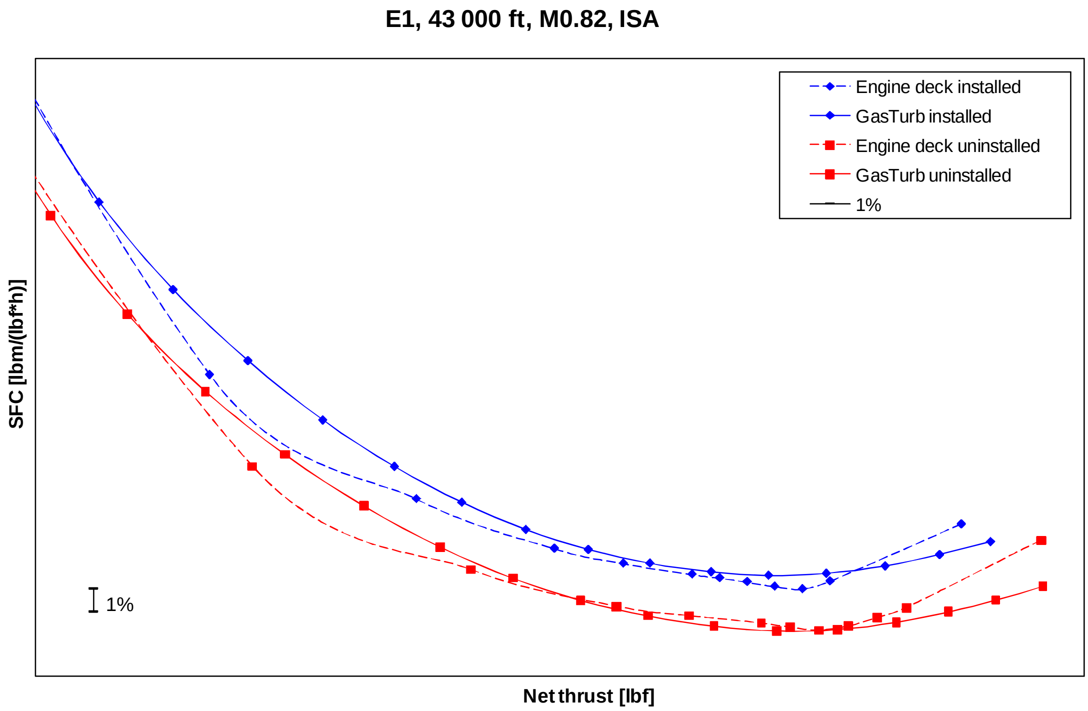

Figure 4.1 E1 installed and uninstalled SFC versus F_N at assumed design point.

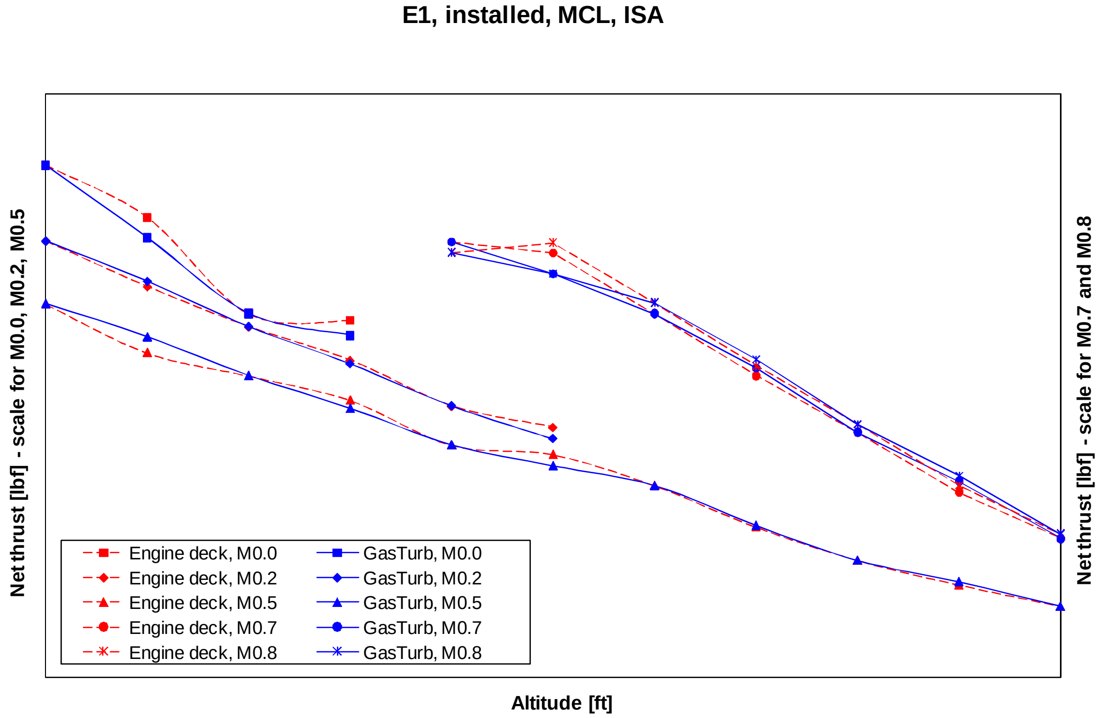

Figure 4.2 E1 MCL rating F_N plotted against altitude for different Mach numbers.

Table 4.1 F_N and SFC error for Engine 1 MCL rating.

F_N error	Minimum	Maximum	Average
ISA-15K	-12.2%	5.03%	-1.92%
ISA	-7.12%	5.96%	-0.24%
ISA+10K	-6.38%	8.17%	1.26%
SFC error	Minimum	Maximum	Average
ISA-15K	-3.14%	3.08%	0.04%
ISA	-2.84%	2.43%	-0.02%
ISA+10K	-2.73%	2.09%	0.02%

4.1.2.3 MTO

The MTO P_3 and T_{41} tables used were similar to the ones showed in Table 2.2 and Table 2.3. The P_3 table was derived for 2 altitudes and 5 Mach numbers (10 P_3 values) and the T_{41} table used 4 ambient temperatures and 3 Mach numbers (12 T_{41} values).

In Figure 4.3 GasTurb results for MTO, E1 are compared with engine deck data. It is seen that the results are good for most cases. This is expected because the control system limits were derived here. Note that the flat rating ambient temperature changes with Mach number which makes it more difficult to accurately predict the net thrust. This is included in the control schedule by having several ambient temperatures in the T_{41} table.

The results are good also for 5 000 ft and 10 000 ft at temperatures above ISA. At lower ambient temperatures, the net thrust characteristics of the engine deck do not follow the assumed model and the prediction is bad. At 15 000 ft, this control system does not capture the net thrust characteristics of this engine very well. This is because the net thrust of the engine deck does not change in a linear fashion with altitude and Mach number, as the assumption in the GasTurb model is. In order to predict the net thrust better, more points need to be included in the control schedule.

The error in the net thrust prediction increases with altitude, as seen in Table 4.2. The predicted SFC is too high for most points.

Table 4.2 E1 MTO rating F_N and SFC error.

F_N error	Minimum	Maximum	Average
Sea Level	-2.24%	2.26%	-0.08%
5 000 ft	-4.00%	4.59%	0.21%
10 000 ft	-1.81%	8.33%	1.64%
15 000 ft	-4.80%	8.15%	0.48%
SFC error	Minimum	Maximum	Average
Sea Level	0.08%	1.48%	0.73%
5 000 ft	1.39%	3.22%	1.98%
10 000 ft	1.19%	2.49%	1.70%
15 000 ft	-0.27%	1.16%	0.44%

Engine 1 is a derivative engine, therefore the control system is presumably adopted from the other engines in the E1 family. This probably makes the control system more specialized with more non-linearities than the control system of a completely new engine.

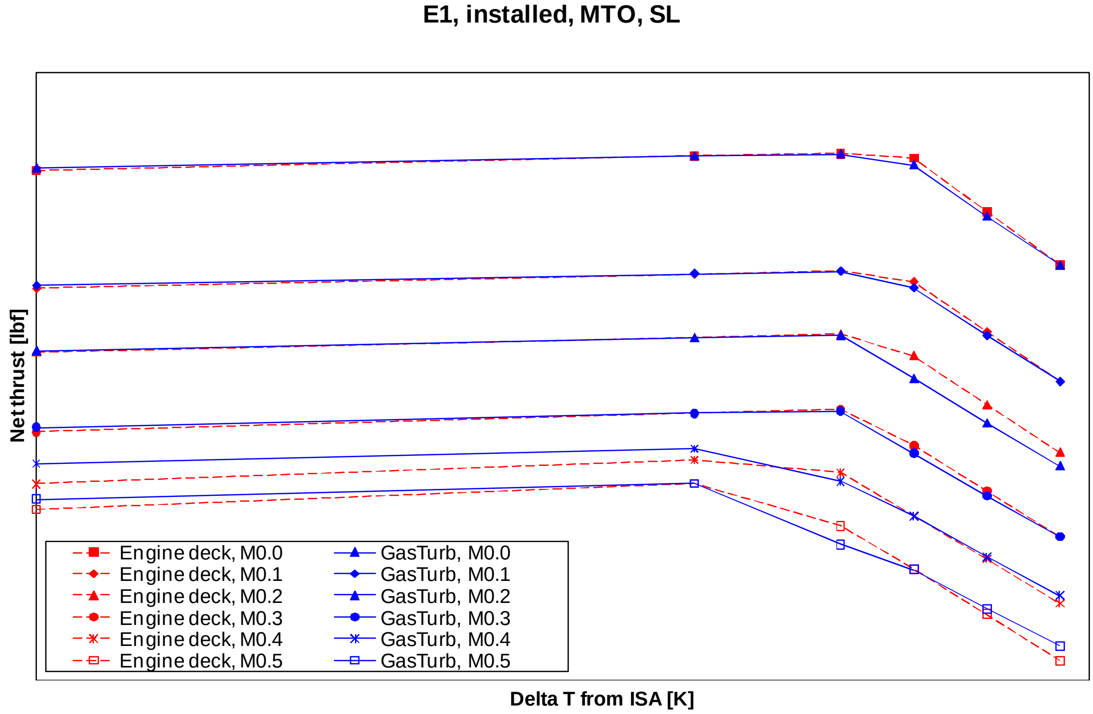

Figure 4.3 E1 MTO rating F_N plotted against ambient temperature and Mach number.

4.2 Engine Concept 1

Engine Concept 1 (EC1) is a two spool mixed flow turbofan in the 10 klbf take off thrust class that is under development.

4.2.1 GasTurb engine model setup

An engine deck was available for EC1. The uncertainties regarding certain input parameters, e.g. intake pressure loss, bleed air and power offtake were removed because the user is in control of them when running the engine deck. Many engine parameters such as efficiencies and nozzle coefficients were hidden from the user. Also important parameters like T_3, P_3, BPR and engine mass flow W_1 was hidden. T_3 and P_3 were assumed to be equal to the HPC exit bleed air properties. The mass flow of the bypass stream and the core stream could be derived from the bleed air properties. There is a provision for fan bleed air. This bleed can be defined both as a relative bleed in [%] and an absolute bleed in [kg/s]. When both these properties are known, the total bypass stream air mass flow can be calculated according to Equation 4.2. The same is true for the core bleed. Thus the total engine air mass flow and bypass ratio can be calculated because both the bypass and the core air mass flows are known.

$$W_i = \frac{W_{i,Bleed}}{W_{i,RelativeBleed}}$$ **Equation 4.2**

The assumed design point was chosen based on the amount of data that was available, in particular because T_{41} was known for this operating condition. Values for T_{41}, fan bleed, core bleed and power offtake were found in the engine deck documentation. The assumed design point is at 41 000 ft, M0.8, ISA+10K.

- The design intake pressure recovery was not known but was assumed to be 0.995.
- Fan isentropic efficiency iterated to match T_{13}.
- HPC isentropic efficiency iterated to match T_3.
- Outer FPR estimated to 1.8.
- Inner FPR estimated based on values from the other investigated engines.
- IPC PR estimated assuming 3 axial stages of PR 1.3 each.
- HPC PR iterated to match the engine deck OPR.
- Installation losses according to the assumed design point.
- The relative enthalpy for the overboard bleed was chosen to give approximately the same bleed temperature as the engine deck output.
- Bypass and core mass flows calculated from bleed properties, BPR calculated from bypass and core mass flow.
- P_3 and T_3 assumed to be equal to bleed PB_{30} and TB_{30} properties reported by the engine deck.
- The mixer efficiency was unknown and left at its default value.
- The mixer area was assumed to be equal to the fan frontal area.
- The nozzle petal angle was left at its default value of 10°.
- Iterate for correct mass flow.
- Iterate burner exit temperature for correct T_{41}.
- Iterate burner efficiency for correct fuel flow.
- Iterate nozzle thrust coefficient to match the net thrust.

4.2.2 Results

4.2.2.1 Cruise

The result of calibrating the generic maps to the EC1 engine deck data is illustrated in Figure 4.4. In this figure, the GasTurb model SFC is plotted against F_N and compared to the engine deck values for installed and uninstalled conditions at the assumed design point. The GasTurb model matches the engine deck data very well for high values of net thrust. The installed case shows a better match than the uninstalled, meaning that some error comes from the GasTurb way of modeling the installation losses. Both the engine deck and GasTurb curves are very smooth. This is probably because the EC1 is still under development and the engine deck reflects a theoretical model of the engine without any corrections to actual measurements.

Judging from plots for other altitudes, the GasTurb model gives very good results when compared to the engine deck, the maximum SFC error is approximately 1.5% for installed, 50 000 ft.

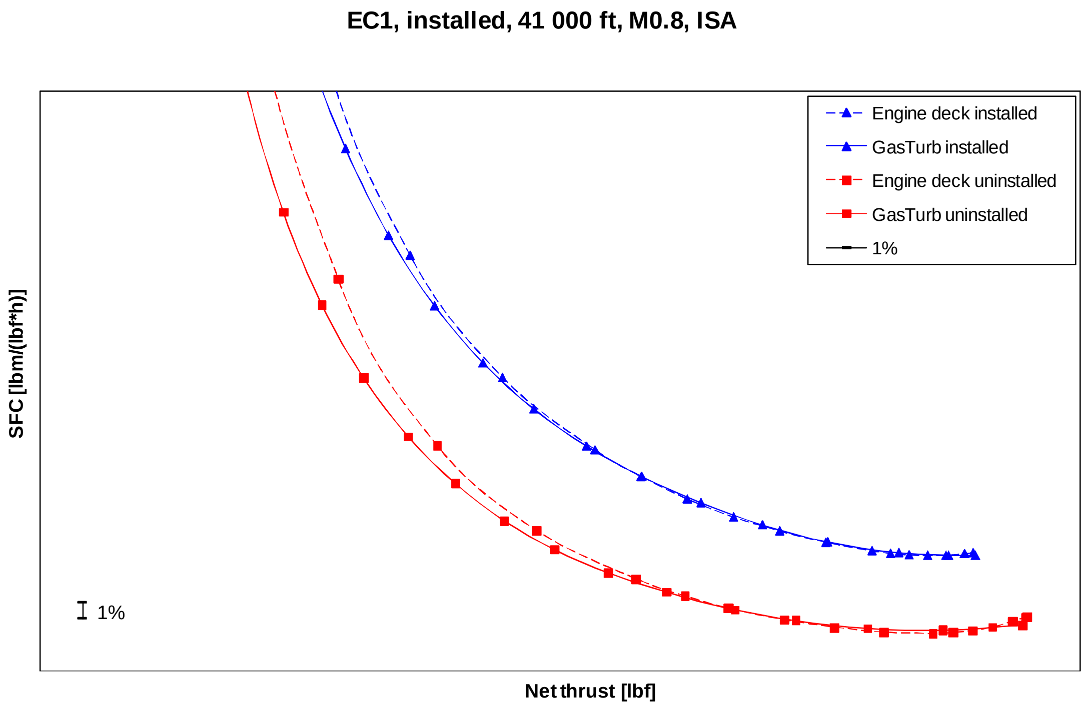

Figure 4.4 EC1 installed and uninstalled SFC versus F_N at assumed design point.

4.2.2.2 MCL

The P_3 table used for the EC1 MCL control schedule contains 5 altitudes and 5 Mach numbers (25 P_3 values), similar to Table 2.1. Some values in the table were extrapolated because they lie outside of the EC1 flight envelope. To avoid unrealistic values of F_N from the GasTurb model, the whole table should be filled. In addition to this table, a maximum allowed value of corrected fan speed was also used. This value was derived at 35 000 ft, M0.5.

Figure 4.5 shows GasTurb F_N compared to engine deck for the MCL rating, ISA, at different altitudes and speeds. It is seen that the error is small for most points. This is expected because this plot shows many of the points where the values in the MCL P_3 table were derived. When comparing other ambient temperatures in Table 4.3, it is seen that the net thrust errors are of a similar order of magnitude, between -3% and 5%. The SFC error is between -1% and 2%.

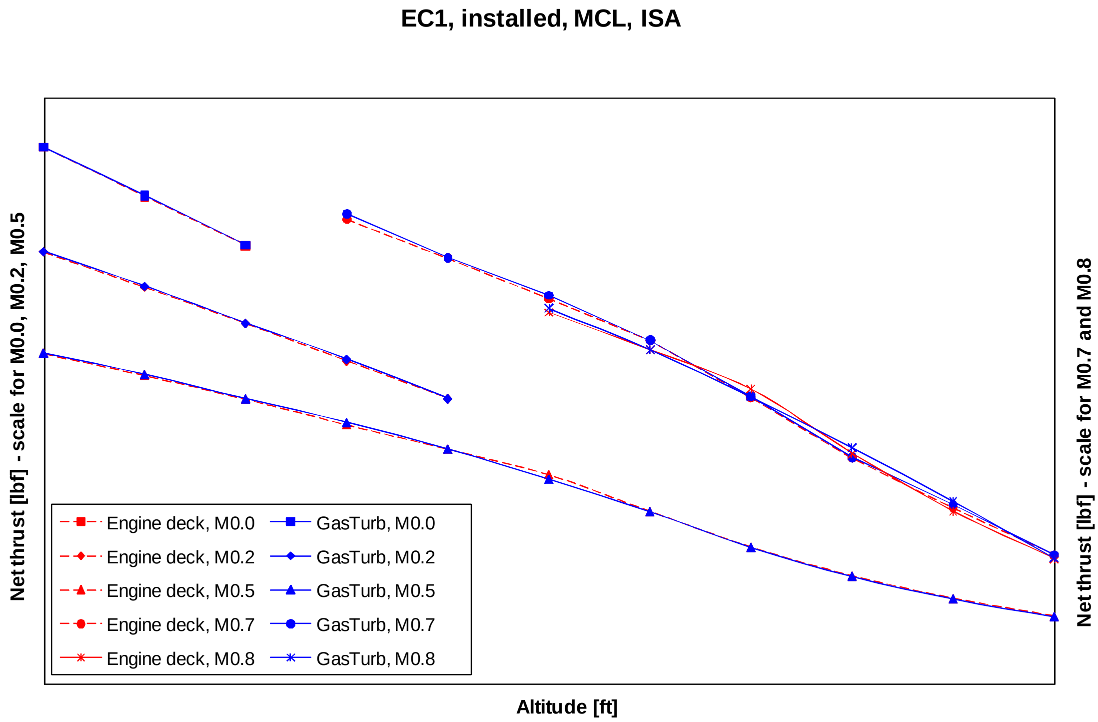

Figure 4.5 EC1 MCL rating F_N plotted against altitude for different Mach numbers.

Table 4.3 F_N and SFC error for the EC1 MCL rating.

F_N error	Minimum	Maximum	Average
ISA-15K	-3.31%	5.13%	0.00%
ISA	-2.60%	5.90%	0.38%
ISA+10K	-2.15%	6.38%	0.61%
SFC error	Minimum	Maximum	Average
ISA-15K	-0.86%	2.01%	0.17%
ISA	-0.70%	1.96%	0.24%
ISA+10K	-0.60%	1.92%	0.28%

4.2.2.3 MTO

The MTO P_3 and T_{41} tables used were similar to the ones showed in Table 2.2 and Table 2.3. The P_3 table was derived for 2 altitudes and 3 Mach numbers (6 P_3 values) and the T_{41} table used 3 ambient temperatures and 3 Mach numbers (9 T_{41} values).

The control system of this engine seems to be very linear and easy to predict. This could be because the engine and its control system are still under development. Figure 4.6 shows that the net thrust error between the GasTurb model and the engine deck is very small for MTO, SL and several different Mach numbers and ambient temperatures. Table 4.4 show that the F_N error is very small, between -2.64% and 4.04% also for other altitudes. The SFC error is between -0.99% and 2.08% for the investigated cases.

In most cases the SFC is somewhat overestimated. This suggests that the GasTurb engine model not completely captures the behavior of the engine deck model. The maximum allowed T_{41} according to the engine deck manual is lower than the model maximum allowed T_{41}. Another proof that there is something wrong in the GasTurb engine model is that if the T_{41} temperature and ambient conditions are matched, the net thrust is too low. The errors are however small and it is concluded that the GasTurb engine model gives a good representation of the EC1 engine deck performance.

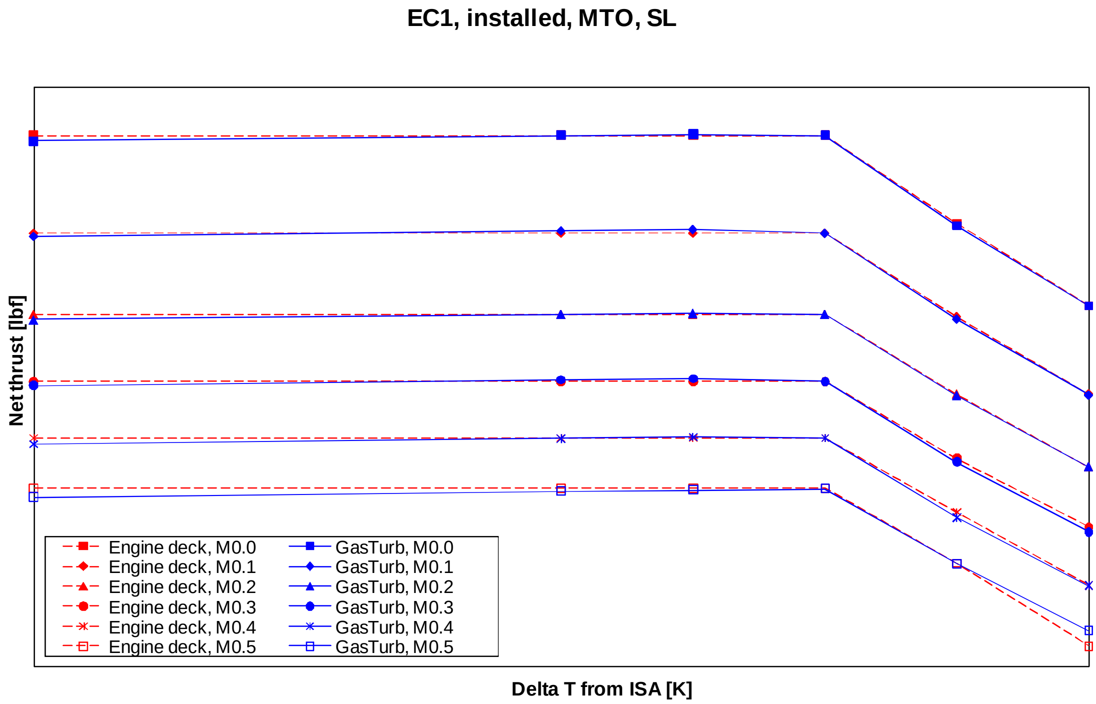

Figure 4.6 EC1 MTO rating F_N plotted against ambient temperature and Mach number.

Table 4.4 EC1 GasTurb engine model MTO rating F_N and SFC error.

F_N error	Minimum	Maximum	Average
Sea Level	-1.51%	3.51%	-0.19%
5 000 ft	-1.68%	1.23%	0.26%
10 000 ft	-2.06%	2.57%	0.43%
14 000 ft	-2.64%	4.04%	0.19%
SFC error	Minimum	Maximum	Average
Sea Level	-0.99%	2.08%	1.11%
5 000 ft	0.11%	1.84%	1.12%
10 000 ft	0.26%	1.98%	1.00%
14 000 ft	0.09%	1.90%	0.62%

4.3 Engine 2

Engine 2 (E2) is an unmixed flow, two spool turbofan with approximately 15 klbf take off thrust intended for use on regional jets.

4.3.1 GasTurb engine model setup

The E2 thrust tables contain a big uncertainty in the form of a "fuel flow factor". The purpose of the fuel flow factor is to adapt the engine performance program SFC to measured data from flight tests. Despite efforts to find the fuel flow factor for Engine 2, no one at Bombardier could provide this information. Therefore, the engine deck data was assumed to represent the E2 engine and engine deck output data was used to create the GasTurb model. This engine model was not converted to GasTurb v11 and only exists as a v10 cycle data file. Therefore all presented results were created by GasTurb v10.

In order to model the E2 unmixed flow engine in GasTurb some assumptions have to be made because of the differences in the two models. In the engine deck, the scrubbing drag as illustrated in Figure 2.1 is modeled and given as a separate term. The scrubbing drag is small but it cannot be modeled in GasTurb. There are terms generating thrust in addition to the core and bypass nozzle, including exhaust of core and core plug ventilation air. This thrust is similar in size to the scrubbing drag, so the net effect is small. These effects are taken into account by calibrating the GasTurb model net thrust to the engine deck data by iterating the turbine exit duct pressure ratio.

An intake pressure recovery map was made from data found in an engine manual. The map was adapted to the GasTurb file format. The map was also verified to give correct values at cruise by comparing to the available engine deck outputs. There were no engine deck data available for the TO condition so the validity of the map could not be verified for this flight condition. The map is seen in Figure 4.7. Note that the intake pressure recovery is essentially constant with regards to Mach number in the interval M0.4 to M0.75.

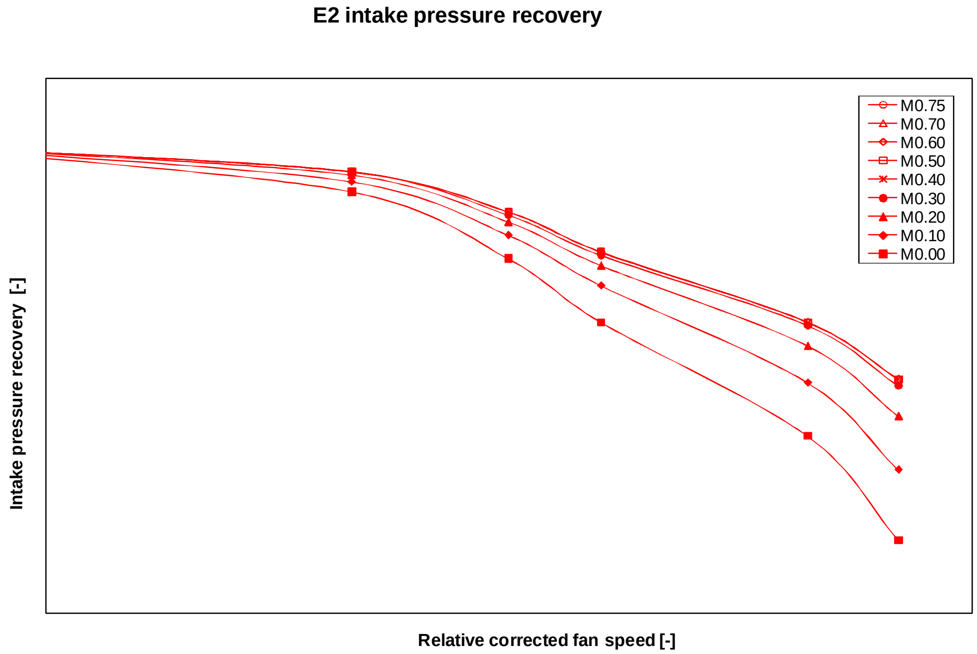

Figure 4.7 The E2 intake map used in GasTurb.

The design bleed flow rates that were used for the GasTurb model were found in engine documentation.

The assumed design point was chosen at 37 000 ft, M0.77, ISA. The engine deck output is limited so component efficiencies and pressure losses were to a big extent left at default GasTurb values. The bleed air properties TB_3 and PTB_3 for the compressor exit bleed port were assumed to be equal to T_3 and P_3 respectively. Thrust reverser leakage and leakage core to bypass were calculated by considering mass flow equilibrium. There is no data about the cooling flows in the engine deck output so similar values as for the E3 engine were used for the HPT and the LPT was assumed to be uncooled. The net thrust was matched by iterating the turbine exit duct pressure ratio. The burner efficiency and burner exit temperature were iterated in order to match the fuel flow and T_{45}.

4.3.2 Results

As the results in Figure 4.8 show, there is a big difference between the engine deck SFC and the thrust table SFC. The differences and its impact on the engine deck SFC were estimated by modeling in GasTurb.

- Different FHV – 0.27% decrease in engine deck SFC
- Include intake pressure loss – 0.57% increase in engine deck SFC
- Engine deck software version – small change in engine deck SFC
- Fuel flow factor – unknown effect on engine deck SFC
- The thrust table give 4.1% higher SFC than the engine deck when corrected for FHV and intake pressure loss.

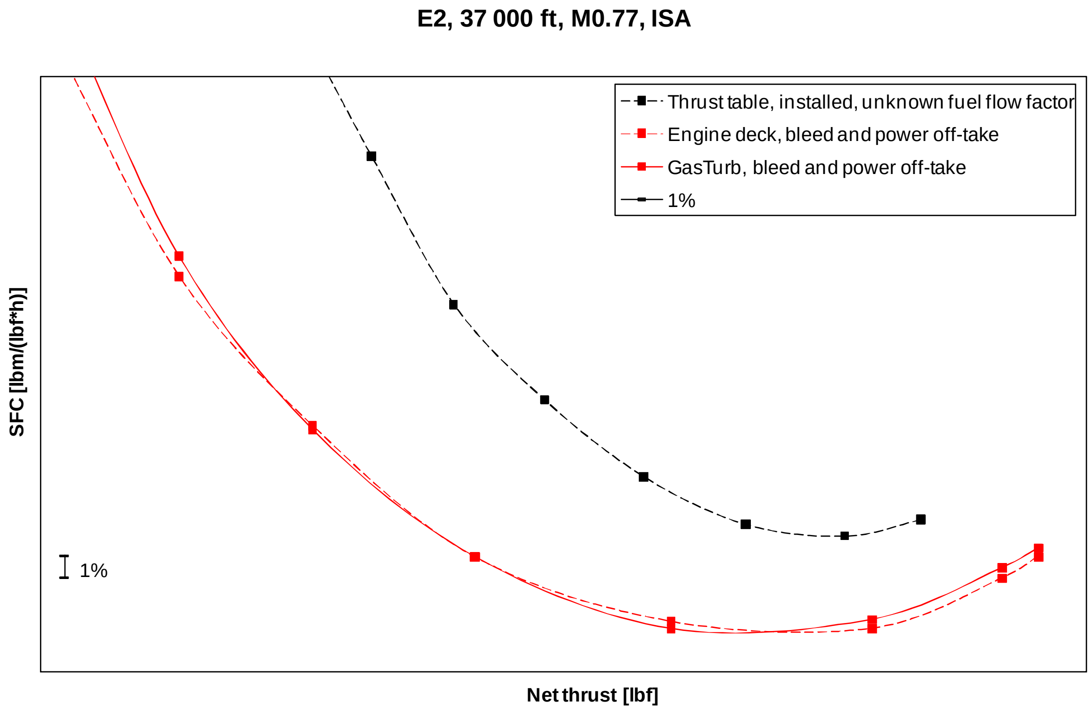

Figure 4.8 E2 SFC versus F_N at assumed design point.

The main reason for the difference is probably the unknown fuel flow factor as discussed earlier, but this could not be confirmed. Another difference between the engine deck data and the thrust tables was the version of the engine deck software used to create them. From sources within Bombardier, the difference between the engine deck versions should be small and not have an impact on SFC.

When comparing the GasTurb model predicted SFC to engine deck SFC at the assumed design point flight conditions, it is seen that the SFC correlation is excellent after the component maps have been calibrated. Because engine deck outputs without the fuel flow factor were available the installation losses are known and the results are directly comparable.

Because of the uncertainty in the thrust table data, only the assumed design point SFC as a function of F_N was investigated and plotted. No results were made for other cruise, MTO or MCL cases where only thrust table data was available.

4.4 Engine 3

Engine 3 (E3) is intended for use on a mixed flow two shaft turbofan engine with a take off thrust of approximately 15 klbf.

The available data for the assumed design point was an engine deck output and was very detailed. An engine deck user manual was also available and therefore the physical parameters of the engine model are close to the real engine. The documentation contains an engine station diagram and explanations of input and output parameters which helped to identify the values in the engine deck output file and to find the differences between the GasTurb and engine deck model.

4.4.1 Differences between GasTurb and Engine 3 engine deck

There are differences between the GasTurb model of a mixed flow two shaft turbofan engine and the engine deck model. The identified differences are discussed below and are summarized in Appendix C.

4.4.1.1 Intake

In the engine deck, the intake pressure loss is varying with Mach number and corrected inlet airflow. In the GasTurb model, it is possible to define a fixed value or an intake map. A fixed value is reasonable simplification for most cruise flight conditions. For TO conditions, where the Mach number is low and total corrected inlet airflow is high, the intake pressure loss can deviate considerably from the cruise value. In order to improve the performance prediction at TO, an E3 intake map was created with data from the engine deck documentation.

The intake map given in the engine deck documentation gives intake recovery factor as a function of total corrected inlet airflow for 4 different Mach numbers. The GasTurb format of an intake map is intake recovery factor as a function of relative corrected speed for different Mach numbers. The GasTurb user manual says that the corrected flow is the best choice, but the use of corrected speed makes the calculation simpler and does not affect the accuracy of the result very much. However, the correlation between corrected airflow and corrected speed had to be found to translate the engine deck intake map into GasTurb format. By running GasTurb for different values of corrected speed at cruise altitude and cruise Mach number and then using the Microsoft Excel "trendline" option to analyse the data, a 3rd degree polynomial equation that describes the corrected fan speed as a function of corrected airflow was found. This correlation is valid at cruise conditions but gives the correct intake pressure loss also at SLS at max TO rating. At SLS with lower thrust ratings than max TO, the inlet pressure loss according to the GasTurb intake map is somewhat higher than the intake loss according to the engine deck map. The intake map used in the GasTurb engine model is illustrated in Figure 4.9.

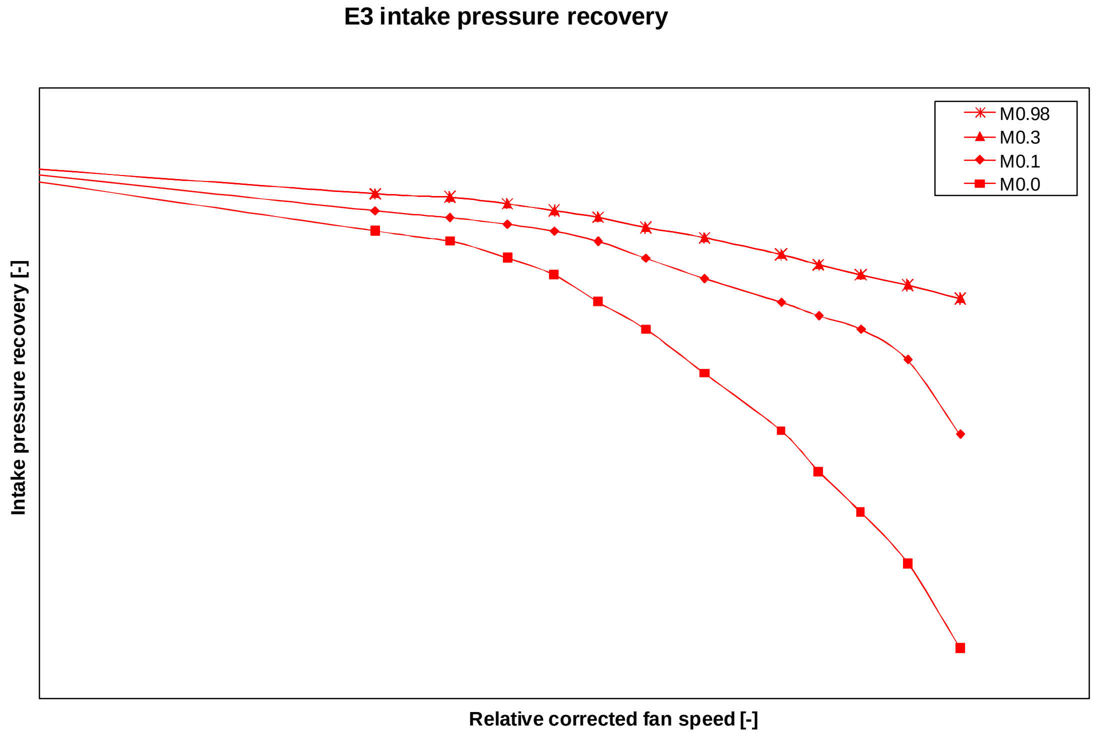

Figure 4.9 E3 intake map used in GasTurb.

4.4.1.2 Fan

In GasTurb, the fan tip and fan root areas are not differentiated from each other, they are called station 2. In the engine deck, the fan tip is called station 12 and the fan root is called station 2. It is therefore possible to have different conditions at the fan root and fan tip in the engine deck. GasTurb does not have the same station definitions, but it is possible to define a radial variation of the pressure loss to achieve a similar effect. This was not done because the radial pressure variation was very small.

The fan OGV pressure loss is specified in the engine deck output as DPP125 but it cannot be specified in GasTurb. This pressure loss should be included in the GasTurb bypass duct pressure loss and not in the FPR. The FPR must be correct because it determines the power needed to run the fan. If this is wrong, the LPT will not extract the correct amount of power from the core hot stream, which means that the temperatures and pressures at the LPT exit will be incorrect.

4.4.1.3 Mixer

The mixer inlet is similar in GasTurb and the engine deck with one cold stream and one hot stream. The hot stream mixer inlet is called station 6 in both GasTurb and the engine deck. The cold stream mixer inlet is called station 16 in GasTurb and station 164 in the engine deck.

The inside and outlet of the mixer are not modeled in the same way in GasTurb and the engine deck. In GasTurb, there is one common outlet, called station 64, with the intermediate stations station 163 at the cold mixing plane and station 63 at the hot mixing plane. In the engine deck, there are three mixer outlets, station 167 for the cold flow, station 967 for the mixed flow and station 67 for the hot flow.

Due to these differences it is not possible to compare temperatures, pressures and pressure losses between GasTurb and the engine deck.

4.4.1.4 Nozzle

The GasTurb model has one mixed stream nozzle, called station 8. The engine deck model has three streams, one with cold bypass air, station 19, one with hot air from the low pressure turbine exit, station 9, and one stream with mixed hot and cold air, called station 99. This difference makes it impossible to compare temperatures, pressures and pressure losses between GasTurb and the engine deck.

For the convergent nozzle calculation, GasTurb assumes an isentropic expansion to ambient pressure. If the nozzle Mach number is subsonic, the conditions at the nozzle has been found, else the nozzle Mach number is set to M1.0 and a new static pressure and temperature is calculated. The nozzle discharge coefficient $C_{D,8}$ describes the ratio between effective flow area $A_{eff,8}$ and geometric nozzle area A_8 and is a measure of the nozzle losses. In GasTurb, the discharge coefficient is calculated from the nozzle petal angle and the nozzle pressure ratio as seen in Figure 3.2. Alternatively, the nozzle coefficients can be defined as constants.

The engine deck documentation has a chart for the E3 nozzle discharge coefficient as a function of final nozzle pressure ratio P_{17}/P_{amb} and the pressure ratio between cold stream and hot stream P_{17}/P_7. The information in this chart cannot be used to model the E3 nozzle in GasTurb. Instead the GasTurb parameter nozzle petal angle is iterated in order to obtain the correct nozzle discharge coefficient at the assumed design point. At off-design conditions, the GasTurb nozzle model is used. Because the nozzle petal angle is small, 6.35º, the nozzle discharge coefficient is almost constant as can be seen in Figure 3.2.

4.4.1.5 Component maps

It is very difficult to obtain real component maps as they are engine manufacturer proprietary information. No component maps were available for Engine 3 so the default GasTurb v11 maps were used.

4.4.2 Setup the GasTurb model

Assumed design point data

Altitude	41 000 ft
Mach number	0.8
Delta T from ISA	0 K

4.4.2.1 Engine mass flow balance

By taking a mass flow balance where the mass flow in must equal mass flow out, it is found that there is a leakage flow from the engine. By examining the mass flow difference between station 13 and 16 in the bypass duct, W_{16}-W_{13}, it is found that there is a mass flow leaking from the core flow to the bypass flow. Combining these results, there must be a mass flow leaking overboard.

4.4.2.2 HPC exit mass flow balance

Air is extracted from the HPC and is used to cool the turbines and leakage overboard and leakage to bypass. By taking a mass flow balance, it is seen that the HPC extracted mass flow equals the cooling and leakage mass flow. It is not known if the overboard leakage is an actual leakage or an overboard bleed. Due to limitations in GasTurb, the overboard leakage was defined as overboard bleed and it is assumed that that the relative enthalpy of the HP overboard leakage is the same as for the overboard bleed.

4.4.3 Cooling flows

The cooling mass flows into the LPT and LPT NGV were determined in the previous section. The total amount of cooling mass flow to the HPT and HPT NGV were determined in the previous section, but neither the HPT cooling mass flow or the HPT NGV cooling mass flow can be calculated directly from the mass flow at different stations. It must be estimated, and the method used was to vary the ratio of HPT to HTP NGV mass flow while keeping the total amount of HPT and HPT NGV cooling mass flow constant.

The NGV cooling mass flow will mix with the flow from the burner and is assumed to do work in the turbine. The cooling air inserted in the rotating turbine is assumed to not do any work. Therefore, if all the cooling air would be inserted in the rotor, the pressure drop and temperature drop over the turbine would be larger than if all the cooling air is inserted at the NGV and allowed to do work in the turbine. Therefore, it is possible to maintain the same pressure drop P_{44}/P_{40} over the HPT as in the engine deck data by finding the correct ratio of HPT and HPT NGV cooling air mass flow. This was done by using the “iterate” feature in GasTurb, with the HPT exit pressure P_{44} as the iteration target. The P_{44} target value is calculated from the engine deck pressure drop P_{44}/P_{40} and the maximum pressure P_4 in the GasTurb engine model. This maintains the correct pressure drop even though the absolute pressure is not the same. The absolute pressure is not the same because the GasTurb inlet pressure is not exactly the same as the engine deck inlet pressure. Also, rounding errors in input values, which are calculated from rounded engine deck output values, and rounding errors within the GasTurb program and GasTurb output will affect the exact numbers in the GasTurb output.

The HP cooling flow is taken from the HPC exit. It is not clear from the engine deck data where the LP cooling flow is taken from. By changing the relative enthalpy of the LPT NGV cooling air, the HPT exit temperature T_{44} is matched to the value in the engine deck data. The principle is that if the cooling air has low enthalpy, less HPC work was done to compress it. This corresponds to less HPT turbine work, and less temperature and pressure drop in the HPT, thus raising the T_{44} temperature.

The relative enthalpy of the LPT rotor cooling air could not be found. A reasonable guess would be that it is taken from the same place as the LPT NGV cooling air.

4.4.3.1 Power offtake

The power offtake is defined in the header of the thrust tables. At the assumed design point, the power offtake is zero.

4.4.3.2 Bleed air

At the assumed design point, there is no bleed air extraction. Data for the bleed schedule for E3 at normal, installed, operation was found within the department. Both hot bleed air from the HPC flow and cold bleed air from the fan flow are extracted to the ECS. The fan bleed air is used to cool the high pressure air to a suitable temperature for the ECS system.

Engine 3 has two ports for bleed air extraction in the HPC. Bleed air is taken from the low pressure port until the pressure at this port falls below a predetermined threshold value and the high pressure port has to be used instead. This is not possible to model in GasTurb. It is possible to check the pressure for each calculated point and if necessary switch bleed port manually and calculate the point again with the correct bleed port. This was not done because the influence is small at most flight conditions and also due to practical reasons when hundreds of points are investigated.

In the heading of the thrust tables, the engine bleed air is defined, e.g. “`ECS in LOW/NORMAL mode`”. It was assumed that the bleed mass flow decreased linearly with altitude between sea level and 51 000 ft and that “`LOW/NORMAL mode`” in the thrust tables correspond to “`NORM bleed`” in the document describing the bleed schedule.

The mass flow of the fan bleed air depends on the cooling requirement of the hot bleed air. If the temperature and mass flow of the hot bleed air is high, more fan bleed air is required for cooling. The limit is set by the maximum allowed temperature at the ECS inlet.

From the bleed schedule documentation, it is possible to calculate the amount of fan bleed needed to cool the compressor bleed air. For practical purposes, one value of fan bleed air mass flow was calculated manually for the assumed design point. The ratio between fan bleed and HPC bleed at the assumed design point was assumed to be constant and was used to estimate the fan bleed for a few other altitudes.

In GasTurb, a value of fan overboard bleed air mass flow is defined as a percentage of the mass flow in the bypass stream at the fan exit. This means that when the engine is operating at a different mass flow than at the assumed design point, the fan bleed air mass flow will be incorrect. In addition to this, the temperature of both hot and cold bleed air will change when the engine operating conditions change, which requires a new computation to determine the correct amount of fan bleed mass flow. Neither of this is modeled as it is beyond the capabilities of GasTurb. It is possible to calculate this for every point that is examined but this was judged to be too detailed.

According to the engine deck documentation, there is a baseline thrust reverser leakage that is specified as as a percentage of the mass flow in the bypass stream. This leakage must be added to the fan bleed mass flow and entered as one value in GasTurb.

4.4.4 Design point calibration

It is important that the engine model is giving the correct values of fuel flow and net thrust at the assumed design point in an attempt to reduce the uncertainties in the input parameters.

The burner exit temperature was iterated to obtain the correct fuel flow. When comparing the predicted T_{45} to the design point data the difference was 0.3K, which suggests that the iterated T_4 and cooling flows are correctly estimated.

Due to the differences of the engine deck model and the GasTurb model after the mixer inlet, it is not possible to compare the mixer parameters in GasTurb to the assumed design point data. The mixer efficiency was iterated to get the correct net thrust. The design mixer area was estimated assuming that the mixer area is equal to the fan frontal area. All pressure losses, component efficiencies etc were input as described in the engine deck data. The remaining parameters were left at the ideal (no pressure loss, 100% efficiency) value.

4.4.5 Results

4.4.5.1 Cruise

The fan and HPC map scaling points were altered to obtain the result in Figure 4.10. The point furthest to the right show the maximum cruise thrust. In an actual flight, the engines will start to operate at a point close to the maximum cruise thrust. As the flight goes on and the aircraft burns fuel and becomes lighter, less net thrust is needed and the engines are throttled back. This means that the engines will operate at a point to the left of the point of minimum SFC. Therefore, the values on the far left of the curves are of little importance as the engine would not normally be operated here in cruise.

The difference between GasTurb and thrust table values is small in the region that is interesting for normal values of cruise thrust. In general, the maximum error between GasTurb and thrust table SFC is 1% for altitudes lower than 43 000 ft. Many points have an error less than 0.25%. At higher altitudes, the GasTurb predicted SFC is too low with a maximum error of about 1.5%.

Some thrust table curves have kinks, e.g at M0.9, 45 000 ft, which show a step in SFC. This is due to the bleed air system shifting between the low pressure port and the high pressure port. As mentioned earlier, the high pressure port is always used in the GasTurb model.

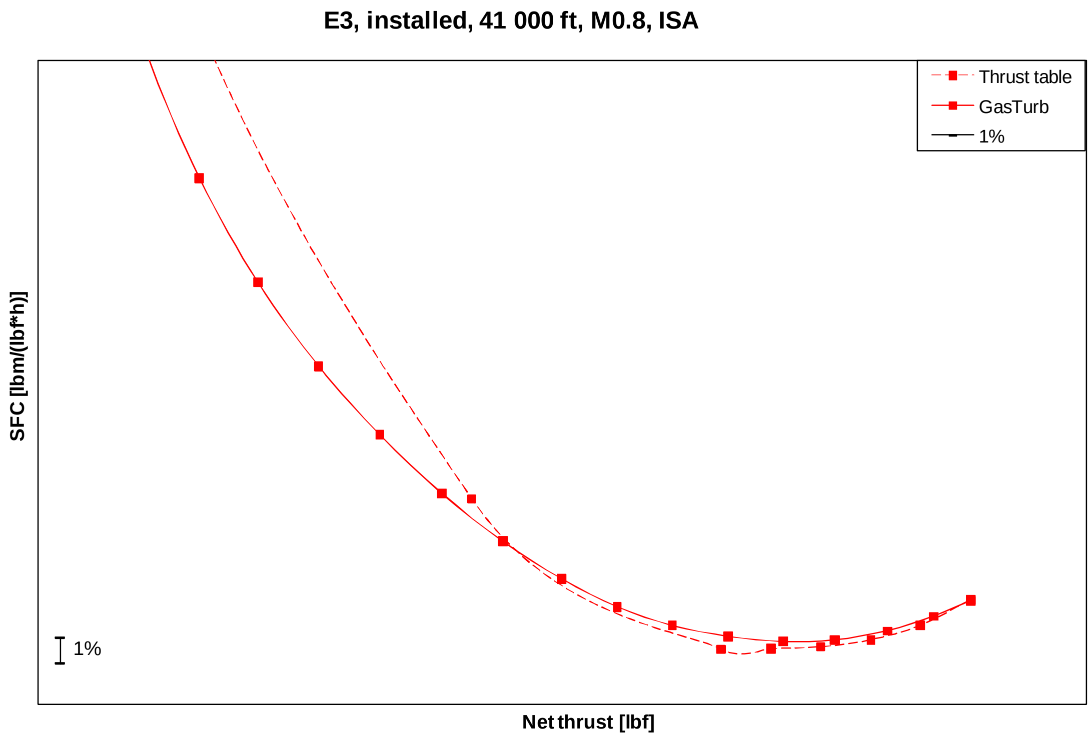

Figure 4.10 E3 installed SFC against F_N at the assumed design point.

4.4.5.2 MCL

The results where GasTurb F_N is compared to thrust table F_N at MCL rating, ISA conditions, SL to max cruise altitude and four different Mach numbers are seen in Figure 4.11. Table 4.5 show F_N and SFC errors for the E3 MCL rating for other ambient temperatures. It is seen that both F_N and SFC are too low at ISA-10K and too high at ISA+10. At high ambient temperatures, the engine might be limited by a temperature limit. This is not modeled in the GasTurb engine model and might be the reason for the high predicted F_N.

Table 4.5 F_N and SFC error for the E3 MCL rating.

F_N error	Minimum	Maximum	Average
ISA-10K	-5.48%	1.94%	-0.55%
ISA	-4.85%	2.75%	-0.03%
ISA+10K	-0.30%	8.54%	2.05%
SFC error	Minimum	Maximum	Average
ISA-10K	-1.96%	2.34%	0.13%
ISA	-2.08%	2.29%	0.23%
ISA+10K	-0.87%	2.25%	0.73%

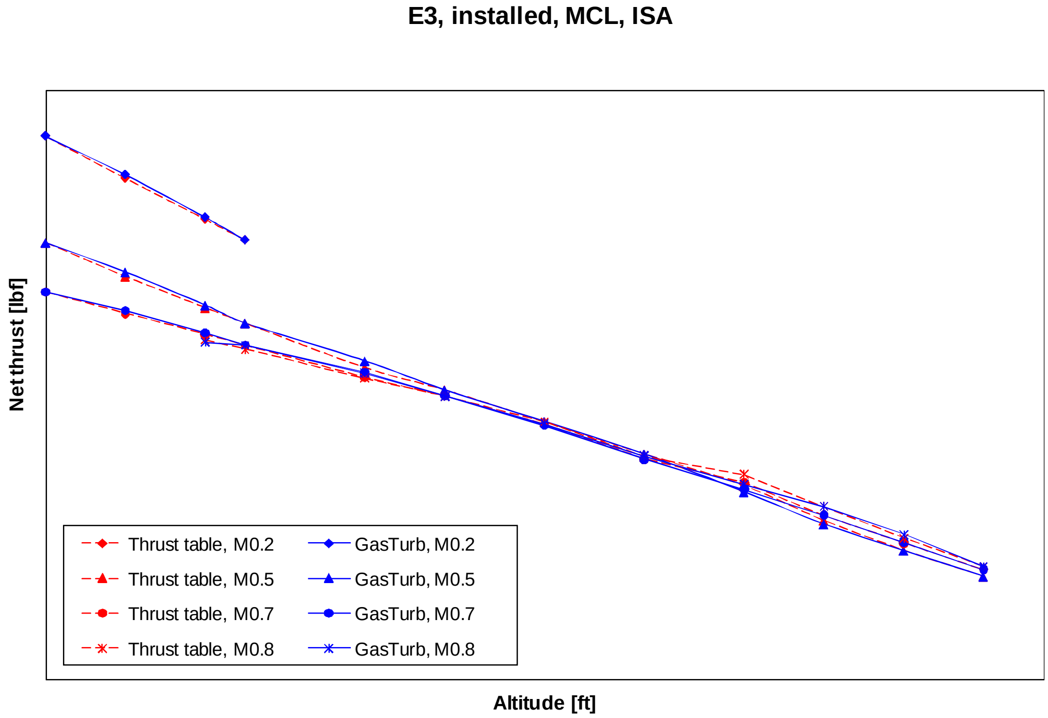

Figure 4.11 E3 MCL rating F_N plotted against altitude for different Mach numbers.

4.4.5.3 MTO

Results for the E3 GasTurb engine model compared to thrust table data at MTO rating, SL are seen in Figure 4.12. The errors are in general small except for 15 000 ft where the predicted F_N is too high. This is because the T_{41} schedule does not capture the flat rating point of the engine and the GasTurb result is somewhat offset from the thrust table data. The predicted SFC is about 1% too high.

Table 4.6 F_N and SFC error for the E3 MTO rating.

F_N error	Minimum	Maximum	Average
Sea Level	-1.56%	0.97%	-0.05%
5 000 ft	-0.36%	1.13%	0.37%
10 000 ft	-0.01%	2.80%	1.32%
15 000 ft	1.35%	9.06%	4.58%
SFC error	Minimum	Maximum	Average
Sea Level	0.56%	2.13%	1.03%
5 000 ft	0.49%	1.35%	0.86%
10 000 ft	0.39%	1.69%	0.88%
15 000 ft	-0.24%	2.74%	1.24%

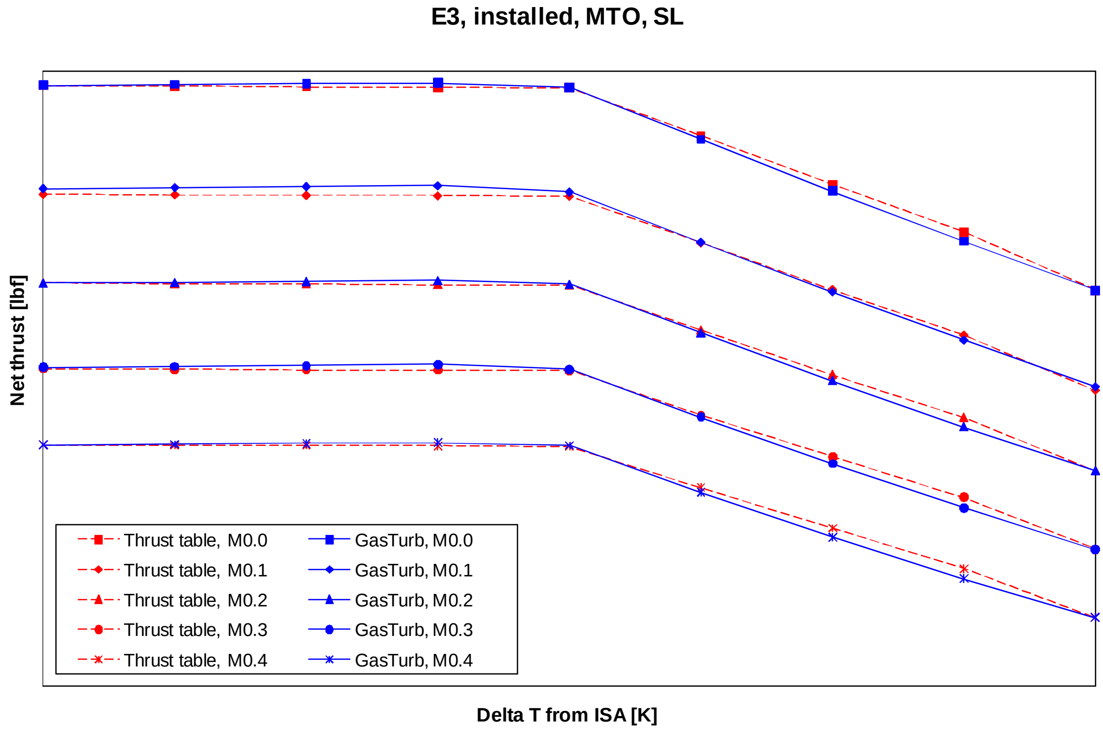

Figure 4.12 E3 MTO rating F_N plotted against ambient temperature and Mach number.

4.5 Engine 3A

Engine 3A (E3A) is a 15-17 klbf take off thrust derivative of Engine 3. The data available for E3A is a preliminary design study made by the manufacturer. The biggest change compared to E3 was the increase of fan diameter from the baseline and minor changes to the core. Other changes included optimized MTO rating, improvements to fan, HPC, combustor, HPT, and LPT. The reason for increasing the fan diameter is to increase the bypass ratio which in turn means lower specific thrust. This gives the engine a higher propulsive efficiency and improves the SFC.

An investigation of this type is included as an example of the investigations that could be made within the MDO environment. A baseline engine is chosen, a few parameters are varied and the impact on the SFC and F_N is given as the result.

The improvements were not known in terms of measurable quantities such as component efficiencies, but the following assumptions were made. The Engine 3 engine model is used as baseline.

- Assumed increase in component isentropic efficiencies:
    - +1% relative to baseline for fan, HPC, HPT.
    - +3% relative to baseline for LPT because of extra LPT stage.
- Reduced HPT cooling flow, assumed -20% relative to baseline.
- Assumed maximum T_{44} increased 15 K for MTO rating.
- Assumed constant hub diameter. This assumption puts higher demand on material strength as the fan diameter increases.
- Optimized Outer FPR and mixer area to maximize thrust. The mixer area upper limit was set equal to the fan frontal area.

The improved engine fan diameter was increased in steps of 2 inch. The expected performance of the enhanced engine comes from a preliminary study. The method used to estimate the performance in this study is unknown.

The way of implementing an increased fan diameter in GasTurb is to turn on the “LPC Design” option. The fan diameter is not given as an input parameter but it is calculated and can thus be set as a target for iteration. If the mass flow, velocity and density of the air going through the fan are known, the flow area can be calculated. This is found by considering the continuity through the fan as given by Equation 4.3. If also the LPC inlet radius ratio is known, the geometrical fan area can be calculated. The LPC inlet radius ratio is defined as the LPC hub diameter divided by the LPC tip diameter.

$$W = \rho A V \qquad \textbf{Equation 4.3}$$

The LPC inlet radius ratio of the baseline E3 was known. The inlet Mach number was not known but was found by iteration with the fan size as target. The bypass ratio was iterated with the fan diameter as target while keeping the inlet Mach number constant. When the new BPR was found, the net thrust was maximized by changing the outer FPR and mixer area with regards to constraints on maximum outer FPR and mixer area. The maximum mixer area was assumed to be equal to the fan frontal area, the maximum outer FPR was set to 1.8, according to *Guha* [4].

4.5.1 Results

4.5.1.1 Cruise

The improvement in SFC from increasing the fan size is seen in Figure 4.13 and Figure 4.14. It is expected according to the study that each 2 inch fan diameter increase gives roughly 1% improvement in SFC. The remaining SFC improvement comes from the enhancements made to the engine components. It is seen that the GasTurb model predicted SFC improvement is close to the expected, especially regarding the crude estimations of the engine enhancements. It is also seen that the point of minimum SFC moves to the right, to a value of higher net thrust, when the fan diameter is increased. This is consistent with the study made by the E3 manufacturer.

4.5.1.2 MCL

The change in MCL net thrust at top of climb conditions of the improved E3A engine with increased fan diameter is seen in Figure 4.15. It is expected that F_N increase with 6.5% for each 2 inch fan diameter increase. The prediction of the GasTurb model gives roughly 1.5% F_N increase for each 2 inch fan diameter increase. This is a very bad result and the reason for it is not known. The data about E3A is very limited, with more data about how the investigation was made and information about engine parameters such as engine mass flow and maximum temperature, a more detailed comparison of results could be made.

4.5.1.3 MTO

The change in MTO net thrust of E3A at sea level static, ISA+15K is seen in Figure 4.16. The expected increase in F_N is 6.5% for each 2 inch fan diameter increase. The GasTurb model gives 5.9% F_N for the first 2 inches of fan diameter increase. When the fan diameter is increased more, the F_N increase is less than expected. As for the MCL case, the reason for the bad result at large values of fan diameter is not known.

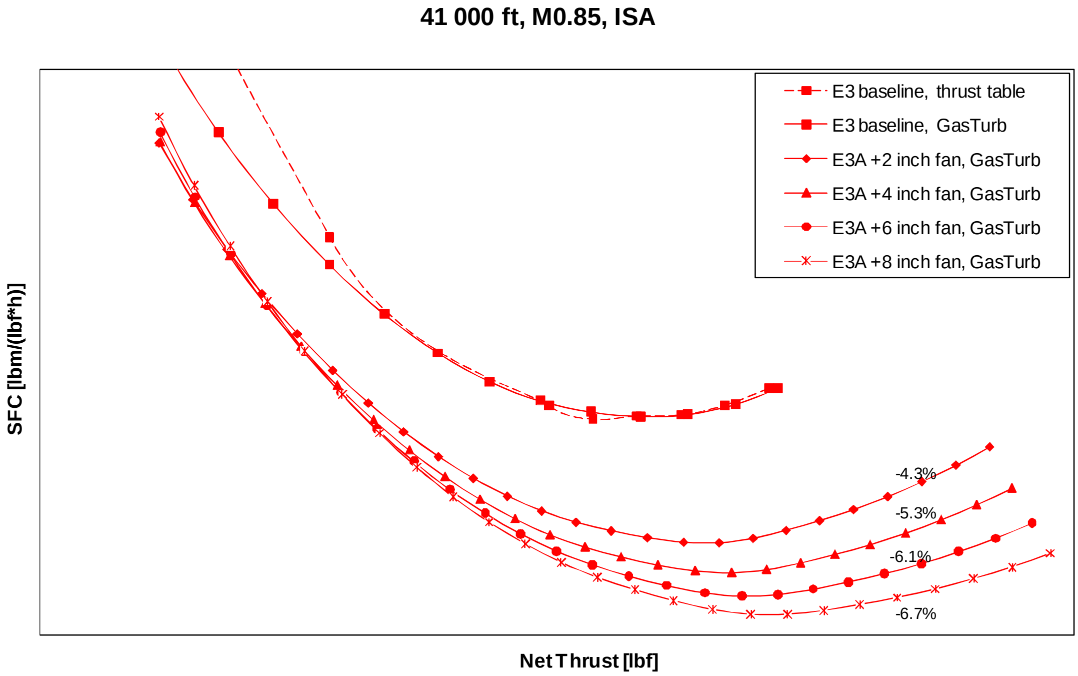

Figure 4.13 SFC plotted against F_N of E3 baseline and the improved E3A with increased fan diameter. The numbers in the figure show the change in cruise minimum SFC.

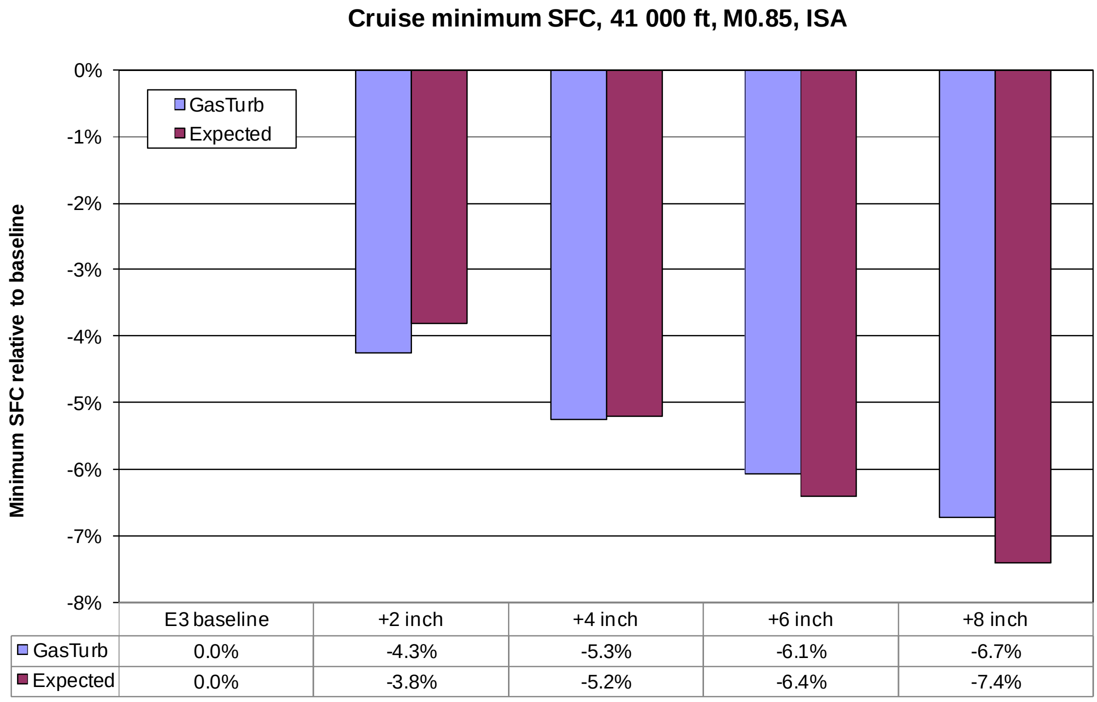

Figure 4.14 Cruise minimum SFC for E3A with increased fan diameter.

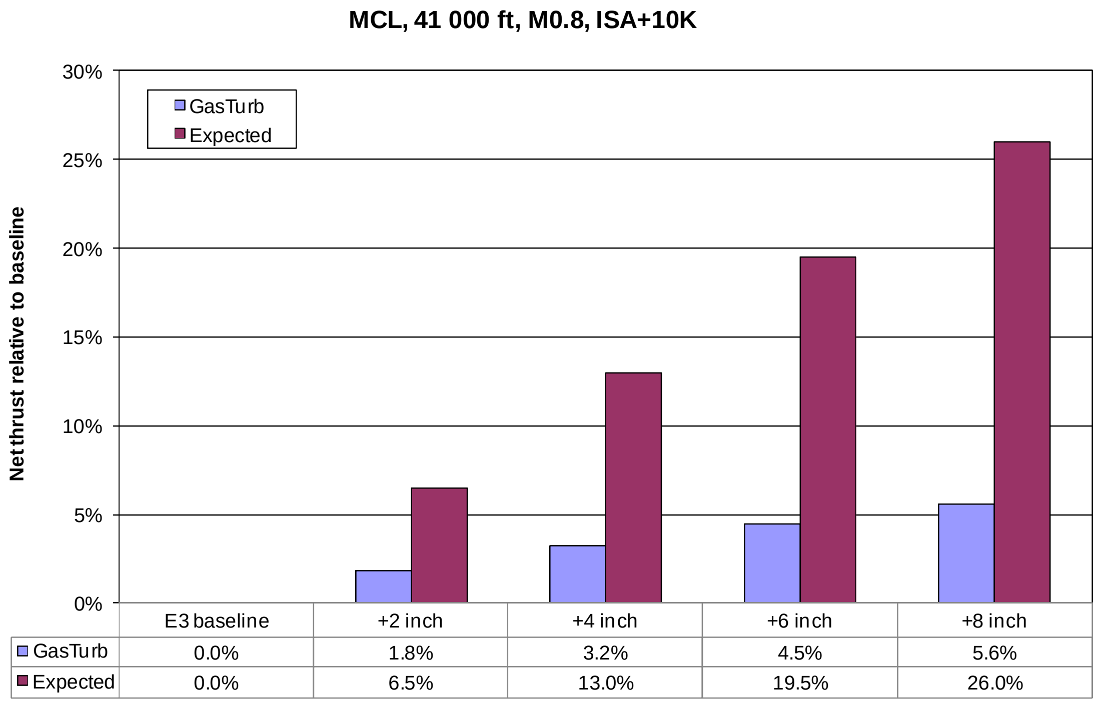

Figure 4.15 MCL net thrust at TOC for E3A with increased fan diameter.

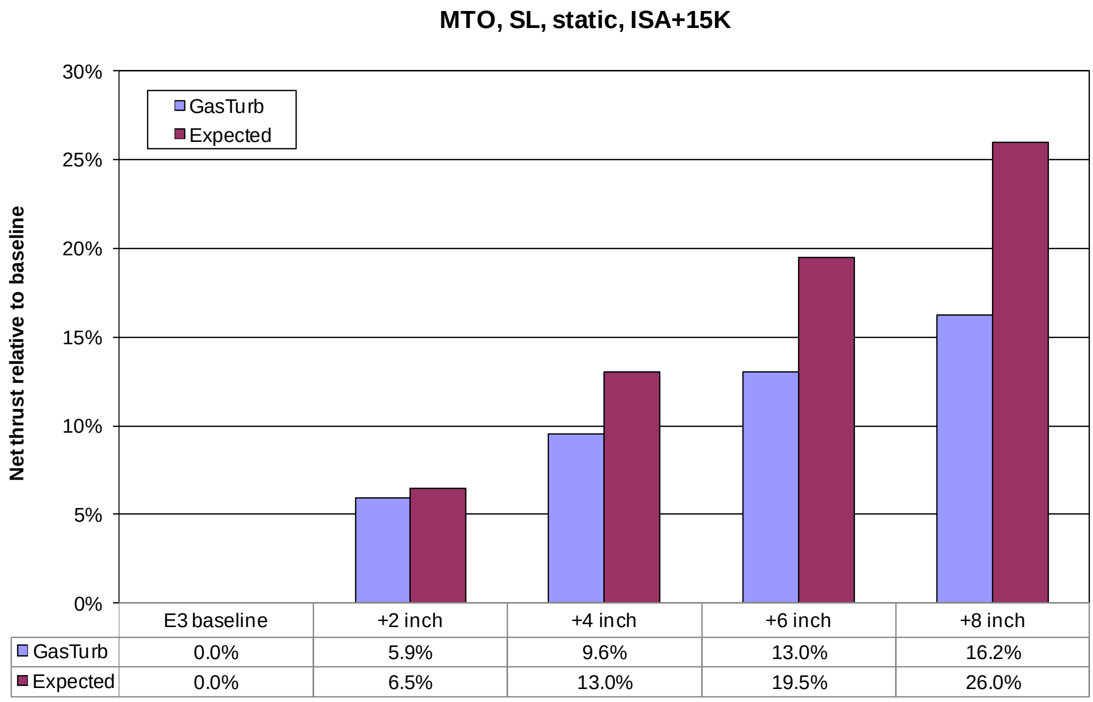

Figure 4.16 MTO net thrust at take off for E3A with increased fan diameter.

4.6 Engine 4

Engine 4 (E4) is a two spool unmixed flow geared turbofan with approximately 25 klbf take off net thrust intended for commercial aircraft. Because of the low pressure ratio fan this engine features a Variable Area Fan Nozzle (VAFN) in order to maintain fan operability. The gearbox makes a high bypass ratio possible.

4.6.1 GasTurb engine model setup

An engine deck output was available for Engine 4. It gives a detailed output of the engine parameters for most stations throughout the engine. Component efficiencies and nozzle coefficients are not given, but most component efficiencies can be found from the temperature and pressure change over each component.

The variable area fan nozzle is not modeled because there was not enough information available to make a reasonable model. The VAFN is fully closed during cruise so the cruise results should not be affected. The results for take off and climb could however be inaccurate due to the VAFN.

- The assumed design point was selected as a top of climb case, 39 000 ft, M0.78, intake pressure recovery according to engine deck data, no bleed, no power offtake, MCL rating.

- Outer fan isentropic efficiency iterated to match T_{13}. The iteration gave a very high efficiency but this value is required to match fan pressure ratio and inlet and outlet temperature to the engine deck data.
- IPC isentropic efficiency iterated to match T_{24}. The iteration gave a low value comparing to Table 3.2.
- HPC isentropic efficiency iterated to match T_3.
- Iteration for correct inlet mass flow.
- Outer FPR calculated from engine deck output.
- Inner FPR estimated.
- IPC PR iterated to match P_{24}.
- HPC PR iterated to match P_3.
- HPT and LPT efficiency and turbine exhaust pressure loss adjusted to obtain correct P_8 pressure.
- High and low spool mechanical efficiency estimated.
- Assumed relative enthalpy for the HPC low pressure port bleed to match bleed pressure and temperature according to the assumed design point engine deck output.
- Pressure losses calculated from engine deck output according to Table 3.1.
- Turbine cooling flow calculated from $W_2 – W_3$.
- Thrust reverser leakage flow calculated as total mass flow in – total mass flow out.
- Leakage core to bypass is calculated as the difference between core in and out flow.
- This matches the pressure, temperature and mass flow at most stations throughout the engine.
- The core nozzle petal angle was left at its default value of 10º.
- The bypass nozzle petal angle was left at its default value of 12º.
- Iterate nozzle thrust coefficient to match the net thrust.
- Iterate burner exit temperature for correct fuel flow.
- Iterate burner efficiency to match T_{45}.

4.6.2 Results

4.6.2.1 Cruise

Figure 4.17 shows SFC plotted as a function of net thrust for the engine deck installed and uninstalled, GasTurb installed and uninstalled and thrust table installed. It is seen that the engine deck installed and the thrust table installed are identical. By changing the default map scaling point, the error between GasTurb and engine deck at uninstalled conditions is close to perfect over a big thrust range. The error at installed condition is somewhat higher, but it is clearly seen that the thrust table installed SFC increases in a step. This is because the ECS bleed (core bleed) shifts between the high pressure port and the low pressure port. This is confirmed by comparing with engine deck outputs. The GasTurb engine model only models the low pressure port and does not switch to the high pressure port. This figure also shows that GasTurb models the installation losses accurately.

The investigation show that the SFC error varies with Mach number and with altitude. The variation with altitude can be explained from the assumption that the amount of bleed air is constant in the GasTurb model. In the thrust tables, the amount of bleed air reduces as the altitude is increased. This will give the GasTurb model higher SFC than the thrust table at high altitude and lower SFC than the thrust table at low altitude. The effect of bleed air on the E4 SFC is rather high because the baseline engine SFC is low. A small delta SFC in absolute terms will give a big percentage when expressed relative to the baseline SFC. The SFC error is between -1.5% and 1.5% for the investigated cases.

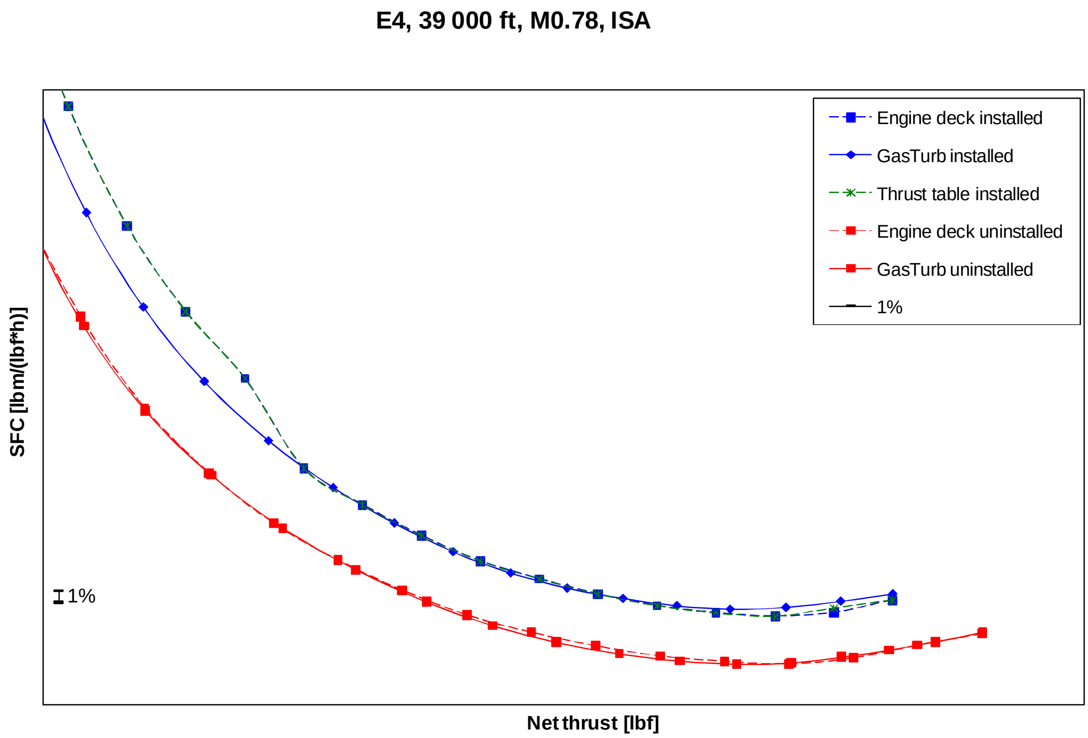

Figure 4.17 E4 installed and uninstalled SFC plotted against F_N at the assumed design point.

4.6.2.2 MCL

The P_3 table used for the E4 MCL control schedule comprised of 5 altitudes and 5 Mach numbers (25 values) similar to Table 2.1. Values for some combinations of altitude and Mach number were extrapolated because these points were not in the thrust table. To avoid unrealistic values of F_N from the GasTurb model, the whole table should be filled. For this engine, a limit on relative corrected fan speed was not necessary because the predicted thrust with the P_3 schedule was good.

Figure 4.18 shows GasTurb F_N compared to engine deck for the MCL rating, ISA, at different altitudes and speeds. It is seen that the error is small for most points. This is expected because this plot shows many of the points where the P_3 values were derived. When comparing other ambient temperatures, Table 4.7, it is seen that the net thrust error is between -4.8% and 2.2% for ISA and ISA+10K. The SFC error is between -5.5% and 1.9% for ISA and ISA+10. The flat rating is not taken into account by the P_3 control schedule and therefore the F_N errors are big at ISA+15K.

The SFC error is bigger than for the other engine models, which suggests that the GasTurb E4 engine model is not realistic. A more detailed investigation shows that the big SFC errors come from low Mach numbers. At typical cruise speeds, M0.7-M0.8, the SFC error is within -1.5% to 1.5%, comparable to the other engine models. It is thus concluded that the engine model is reasonably accurate at typical cruise Mach numbers, which is consistent with what was showed in section 4.6.2.1.

This difference in SFC could come from the variable area fan nozzle which is not included in the GasTurb model. According to engine technical description, the VAFN is fully closed at cruise but could be open at other flight conditions. An open VAFN would reduce the jet speed through the nozzle, thus reduce the net thrust. Assuming that the influence of changing the VAFN is small on the fuel flow, this means that the SFC will increase. Because this effect is not included in the GasTurb model, the GasTurb SFC will be lower than the thrust table SFC.

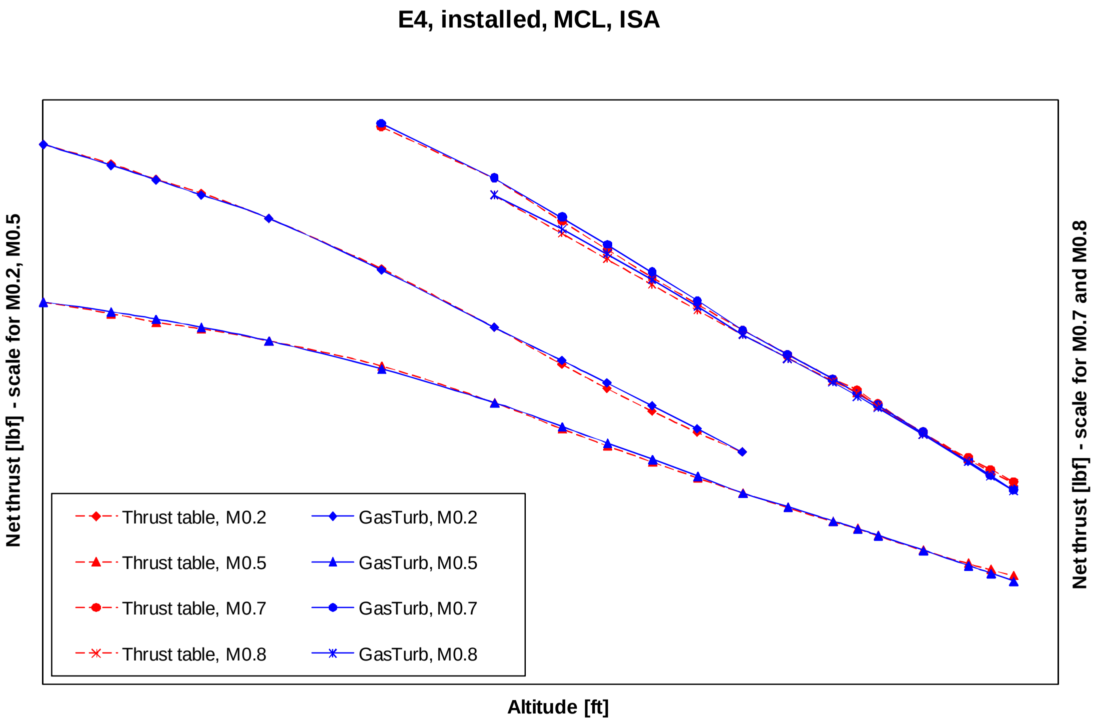

Figure 4.18 E4 MCL rating F_N plotted against altitude for different Mach numbers.

Table 4.7 F_N and SFC error for E4 at MCL rating.

F_N error	Minimum	Maximum	Average
ISA	-4.82%	1.97%	-0.13%
ISA+10K	-4.46%	2.23%	0.22%
ISA+15K	-0.89%	8.63%	4.99%
SFC error	Minimum	Maximum	Average
ISA	-5.50%	1.86%	-0.13%
ISA+10K	-5.44%	1.69%	-0.65%
ISA+15K	-4.22%	2.23%	-0.04%

4.6.2.3 MTO

The MTO P_3 and T_{41} tables used in Engine 4 MTO rating control schedules were similar to the ones showed in Table 2.2 and Table 2.3. The P_3 table was derived for 2 altitudes and 3 Mach numbers (6 P_3 values) and the T_{41} table used 3 ambient temperatures and 3 Mach numbers (9 T_{41} values).

Figure 4.19 illustrate the MTO rating net thrust as a function of ambient temperature for the thrust table and the GasTurb model at sea level for 5 different Mach number. The F_N error is small for most conditions, including above the flat rating ambient temperature. The biggest errors occur at low ambient temperatures and at M0.3. The control system P_3 limits were not derived at these conditions and the error comes from the assumption that they are linearly interpolated within the GasTurb model. Looking at other altitudes in Table 4.8, it is seen that the net thrust prediction gets worse with increasing altitude. The flat rating ambient temperature is the most difficult to predict and above this ambient temperature the errors are bigger. In general, the SFC error is between -3% and 2%. The GasTurb model net thrust is typically too low, the error increasing with altitude. Most points have F_N errors less than a few percent with single points having F_N errors between -8% and 4%.

Table 4.8 GasTurb E4 engine model, MTO rating, F_N and SFC error compared to thrust table.

F_N error	Minimum	Maximum	Average
Sea level	-1.71%	2.25%	0.35%
5 000 ft	-3.66%	0.19%	-1.65%
10 000 ft	-4.58%	2.70%	-1.27%
15 000 ft	-7.08%	3.03%	-1.46%
20 000 ft	-8.22%	3.84%	-1.21%
SFC error	Minimum	Maximum	Average
Sea level	-3.12%	1.94%	-0.48%
5 000 ft	-3.25%	2.08%	-0.62%
10 000 ft	-3.39%	1.57%	-0.68%
15 000 ft	-2.84%	1.80%	-0.11%
20 000 ft	-2.84%	1.93%	-0.27%

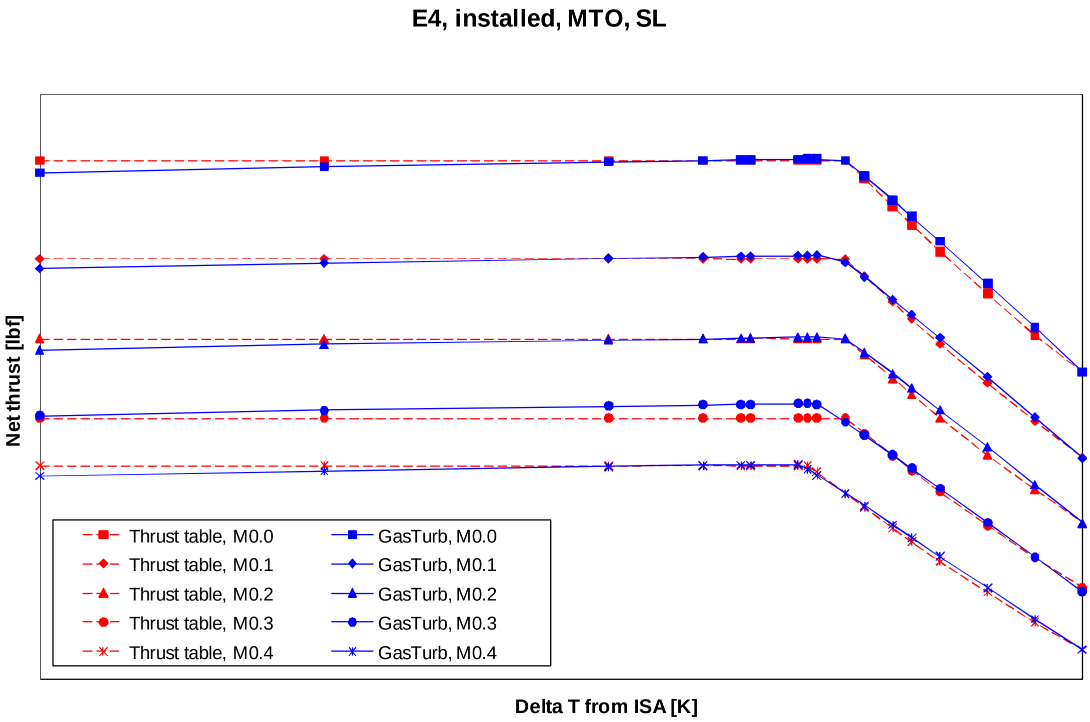

Figure 4.19 E4 MTO rating F_N plotted against ambient temperature and Mach number.

5 Difficulties and future work

From the first day at Bombardier, it was said that GasTurb v11 would soon arrive to the company. In the meantime, GasTurb v10 was used. Finally, v11 arrived only a few weeks before the project was supposed to finish. This meant that the planning did not hold and at the end of the project a lot of time was spent on upgrading engine model files to v11 and doing the validation again with the new software version.

A solution to the problem with saving component map files in GasTurb v11 (see section 3.6.2) was not found. This did not influence the results in terms of SFC or net thrust. It is recommended that a solution is found if future work involves changes to the component map files.

One problem with modeling of jet engine performance is the vast number of parameters needed to make a detailed model. Many parameters, e.g. component efficiencies, duct pressure losses and internal engine cooling, were estimated. This was overcome by calibrating the engine model to match the available engine performance data at the assumed design point. The calibration was done by manually tuning the unknown parameters. In the future, the calibration could be done automatically within the optimization software framework under development at Bombardier.

An accurate performance model of a jet engine requires component maps that describe the compressor and turbine characteristics. Component maps are engine manufacturer proprietary information and were not available during the project. The solution was to use generic component maps and calibrate them so that the predicted performance matched the engine manufacturer performance data. It is suggested that the calibration could be automated within the optimization software framework. There is also a possibility of using publicly available component maps that describe the component performance in a better way than the default GasTurb component maps.

A difficulty with the MTO and MCL rating net thrust prediction is to find reasonable limits to the control system. The method presented here relies heavily on the availability of detailed, perhaps proprietary, engine performance data. It would be desirable to develop a more generic method to find control system limits. This reduces the dependence on manufacturer data and makes it less of a data matching exercise. One conceivable example could be to relate the maximum allowed P_3 to the maximum OPR of the engine.

The rubber engine model could not be used in a full scale multidisciplinary design optimization run because the MDO framework was still under development during this project. The engine model was however run separately within the MDO framework. The setup of optimization loops and the details in how the engine model interacts with the other disciplines of the MDO software is left for future projects.

6 Conclusions

It is concluded that the goals set at the beginning of the project were fulfilled. The presented results prove that GasTurb v11 can be used to make accurate jet engine performance models. In addition, a method on how to create a model was developed and is presented in this work. Therefore the validation goal is fulfilled.

It was also proved that the GasTurb v11 engine models can be run without GUI and without user interaction in optimization loops within Isight, the software chosen for Bombardier's MDO software framework. Therefore the integration goal is fulfilled.

7 References

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8 Appendix

8.1 Appendix A: Input and output files for DLL

The following section shows an input file and the resulting output file for the C++ program that calls the GasTurb DLL and uses its functions to calculate the steady state performance of a GasTurb engine model file. For a detailed description of the variables, the reader is referred to the *DLL user manual* [15].

8.1.1 Input file *"Input.txt"*

```
ZCASE = 1
ZALT = 0
ZDTAMB = 25
ZERM1A = 1
ZPWXH = 50
ZPAMB = 0
ZPC = 20
ZPLA = 80
ZP1A = 0
ZRC = 0
SERAM = 2
SIM = 1
ZTAMB = 0
ZT1A = 0
ZWB3 = 0
ZWB3Q = 0
ZXM = 0
ZTIME = 0
ZHUMIDVI = 0
ZFHVVI = 43.124
ZFNVI = 0
ZWFVI = 0
ZXNLRPMVI = 0
ZXNHRPMVI = 0
ZWRCQ2VI = 0
ZXXVI = 1.3464
SESTVI = 0
ZBTAHCVI = 0.5
ZRXNHVI = 1
ZT4VI = 1800
ZBTALCVI = 0.5
ZRXNLVI = 1
ZBPRVI = 8
ZBTAICVI = 0.4
ZBTAHTVI = 0.5
ZBTALTVI = 0.5
ZDTRCVI = 0
STRANSVI = 0
ZCTRCPVI = 0.2
ZCTRCDVI = 0
ZCTRCIVI = 0
```

8.1.2 Output file *"Output.txt"*

```
NSIFO = 0
AE8FO = 2.50278
FRAMFO = 0
FGFO = 13.1618
FHVFO = 43.124
FNFO = 13.1618
PB3FO = 454.729
P7FO = 103.974
SFCFO = 12.7534
TB3FO = 522.562
T7FO = 348.038
WFEFO = 0.167858
WFTFO = 0.167858
W1AFO = 183.125
W7FO = 183.293
W2FO = 183.125
XNLFO = 1479.55
XNIFO = 1479.55
XNHFO = 9562.71
ALTFO = 0
PAMBFO = 101.325
PLAFO = 2.96815
P1AFO = 101.325
TAMBFO = 313.15
T1AFO = 313.15
XMFO = 0
SMLFO = 192.414
SMIFO = 54.133
SMHFO = 34.6855
TIMEFO = 0
ERAM1FO = 1
DTAMBFO = 25
PCFO = 20
RCFO = 20
WB3FO = 0
WB3QFO = 0
PWXHFO = 50
humidVO = 0
T13VO = 316.728
T25VO = 348.676
T3VO = 522.562
T4VO = 1041.39
T45VO = 825.952
T5VO = 732.211
P13VO = 104.574
P25VO = 138.21
P3VO = 454.729
Ps3VO = 442.928
P45VO = 173.177
P5VO = 102.903
NHDOTVO = 0
FAR4VO = 0.0152479
LIMCDVO = -3
BTAHCVO = 0.58708
RXNHVO = 0.6
BTALCVO = 0.509266
RXNLVO = 0.369887
BPRVO = 9.13032
BTAICVO = 0.657074
BTAHTVO = 0.235494
BTALTVO = 0.148432
DTRCVO = 0.000933188
```

8.2 Appendix B: Input parameter bounds

Parameters	Group	Min value	Max value
// Burner Design Efficiency	5	0.97	1
// Burner Exit Temperature [K]	2	1281	1566
// Burner Pressure Ratio	4	0.9	1
// Bypass Duct Pressure Ratio	4	0.9	1
// Cold Stream Mixer Press Ratio	4	0.9	1
// Compr. Interduct Press. Ratio	4	0.9	1
// Design Bypass Ratio	2	3.9	4.7
// Design Mixer Mach Number	2	0.1	0.4
// Fuel Heating Value [MJ/kg]	2	38.80	47.43
// Hot Stream Mixer Press Ratio	4	0.9	1
// HP Compressor Pressure Ratio	2	14.49	17.71
// HP Spool Mechanical Efficiency	5	0.97	1
// HPT Cooling Air W_Cl/W25	1	0	0.15
// Inlet Corr. Flow W2Rstd [kg/s]	2	190	232
// Inner Fan Pressure Ratio	2	1.43	1.75
// Intake Pressure Ratio	5	0.970	1
// Isentr. HPC Efficiency	2	0.75	0.91
// Isentr. HPT Efficiency	2	0.80	0.97
// Isentr. LPT Efficiency	2	0.80	0.98
// Isentr.Inner LPC Efficiency	2	0.82	1.00
// Isentr.Outer LPC Efficiency	2	0.81	0.99
// LP Spool Mechanical Efficiency	5	0.97	1
// LPT Cooling Air W_Cl/W25	1	0	0.15
// LPT NGV Cooling W_NGV_LPT/W25	1	0	0.15
// Mixed Stream Pressure Ratio	4	0.9	1
// Mixer Efficiency	2	0.5	1
// NGV Cooling Air W_Cl_NGV/W25	1	0	0.15
// Nominal HP Spool Speed	2	12483.4	15257.4
// Nominal LP Spool Speed	2	5744	7020
// Outer Fan Pressure Ratio	2	1.55	1.90
// Overboard Bleed [kg/s]	1	0	0.5
// Power Offtake [kW]	1	0	100
// Rel. Enth. LPT NGV Cooling Air	1	0.3	1
// Rel. Enth. of LPT Cooling Air	1	0.3	1
// Rel. Enthalpy of Overb. Bleed	1	0.3	1
// Rel. Fan Overb.Bleed W_Bld/W13	3	0	0.03
// Rel. Handling Bleed to Bypass	3	0	0.03
// Rel. HP Leakage to Bypass	3	0	0.03
// Rel. Overboard Bleed W_Bld/W25	3	0	0.03
// Rel.HP Leakage to LPT exit	3	0	0.03
// Specified Discharge Coeff CD	4	0.9	1
// Specified Thrust Coeff CFG	4	0.9	1
// Turb. Interd. Ref. Press. Ratio	4	0.9	1
// Turbine Exit Duct Press Ratio	4	0.9	1

8.3 *Appendix C: Engine 3*

The engine deck station definition is used in this section.

8.3.1 E3 Efficiencies compared to GasTurb values

E12	Outer fan.
E2	Inner fan.
E26	HP compressor.
E3	Burner.
E4	HP turbine.
E44	LP turbine.
Mixer efficiency	Not given in engine deck data.

8.3.2 E3 Pressure drops compared to GasTurb values

DPP125	Fan OGV, P13/P125.
GasTurb equivalent:	Not modeled in GasTurb, see discussion in section 4.4.1.2.
DPP13	Forward part of bypass duct with fan bleed port, P16/P13.
GasTurb equivalent:	DPP13*DPP16 = “Bypass Duct Pressure Ratio”.
DPP16	Bypass duct, P164/P16.
GasTurb equivalent:	DPP13*DPP16 = “Bypass Duct Pressure Ratio”.
DPP25	Swan Neck Duct, P26/P24.
GasTurb equivalent:	“Compr. Interduct Press. Ratio”
DPP31	Burner, P40/P30.
GasTurb equivalent:	“Burner Pressure Ratio”
DPP5	LPT EGV P6/P50.
GasTurb equivalent:	“Turbine Exit Duct Press Ratio”
DPP66	Hot mixer, P67/P66.
GasTurb equivalent:	“Hot Stream Mixer Press Ratio”
DPP166	Cold mixer, P17/P164.
GasTurb equivalent:	“Cold Stream Mixer Press Ratio”
DPP67	Hot nozzle.
GasTurb equivalent:	Only “Mixed Stream Pressure Ratio” exists – not modeled.
DPP967	Mixed nozzle.
GasTurb equivalent:	Only “Mixed Stream Pressure Ratio” exists – not modeled.
DPP167	Cold nozzle.
GasTurb equivalent:	Only “Mixed Stream Pressure Ratio” exists – not modeled.

8.3.3 E3 Pressure ratios

OPR	Calculated from P3/P1A.
Outer FPR	Calculated from P125/P12.
Inner FPR	Calculated from P24/P2.
HPC PR	Calculated from P3/P2.